NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report 32-1265

Surveyor Project Final Report

Part I. Project Description and Performance
Volume I

Prepared by the Surveyor Project Staff

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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Prepared Under Contract No. NAS 7-100
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Preface

This two-part report documents activities carried out under the cognizance of the *Surveyor* Project, which was managed by the Jet Propulsion Laboratory for the National Aeronautics and Space Administration.

Part I, consisting of two volumes, contains a complete description of the *Surveyor* Project including the development and operation of each of its major systems: the *Surveyor* spacecraft, the *Atlas/Centaur* launch vehicle, and the *Surveyor* Mission Operations System. Part I was prepared from the contributions of many members of the *Surveyor* Project staff representing all supporting elements.

Part II presents the science data derived from the lunar soft-landing missions and the scientific analysis conducted by the *Surveyor* scientific evaluation team, the *Surveyor* investigation teams, and the associated working groups.

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Abstract

The *Surveyor* Project planned and conducted seven unmanned lunar missions for which spacecraft were launched between May 1966 and January 1968. Each of the spacecraft was successfully launched with the then newly developed *Atlas/Centaur* vehicle which utilized for the first time a high-specific-impulse, liquid hydrogen/liquid oxygen fueled stage. Five of the spacecraft successfully soft-landed and returned a great quantity of engineering and scientific data on extensive postlanding operations, accomplishing all mission and project objectives. Four of the spacecraft soft-landed at selected mare sites to provide data which were required to support the *Apollo* Program. The final spacecraft was then successfully used for scientific investigation of a contrasting site in the rugged lunar highlands.

Surveyor was a fully attitude-stabilized spacecraft designed to receive and execute a wide variety of earth commands, as well as to perform certain automatic functions including the critical terminal-descent and soft-landing sequences. Significant new and advanced subsystems that were developed and/or used in combination to enable *Surveyor* to execute the complex terminal phase of flight were: (1) a solid-propellant main retro motor, (2) throttlable liquid-propellant vernier engines (also used for midcourse velocity correction), (3) highly sensitive velocity- and altitude-sensing radars, and (4) an automatic closed-loop guidance and control system.

The first *Surveyor* spacecraft carried a survey television camera which, together with other engineering instrumentation, obtained in-flight and postlanding data. The complement of instruments carried on later missions included various combinations of the following additional devices: (1) a soil mechanics/surface sampler instrument for picking, digging, and handling lunar surface material; (2) an alpha scattering instrument for performing a chemical analysis of the lunar surface material; and (3) magnets attached to the spacecraft for determining magnetic properties of the soil.

I. Introduction

A. Project Objectives

The *Surveyor* Project was conducted by the National Aeronautics and Space Administration (NASA) to explore the moon with unmanned, automated, soft-landing spacecraft equipped to respond to earth commands and to transmit scientific and engineering data from the lunar surface. The overall objectives of the Project were:

(1) To accomplish successful soft landings on the moon as demonstrated by operations of the spacecraft subsequent to landing.

(2) To provide basic data in support of the *Apollo* Program.

(3) To perform lunar operations designed to contribute new scientific knowledge about the moon and to provide further information in support of the *Apollo* Program.

These objectives reflect the important role of *Surveyor* in demonstrating the lunar soft-landing technique as a precursor to the *Apollo* Program, which will utilize similar final descent and landing system technology, and in providing the only lunar surface surveys and measurements to be made before the manned missions.

The *Surveyor* series consisted of seven spacecraft, *Surveyors I* through *VII,* which were launched during the period May 1966 to January 1968. Five of the seven spacecraft successfully soft-landed on the moon, and each performed extensive engineering and scientific post-landing operations to satisfy completely all objectives. *Surveyors I, III, V,* and *VI* soft-landed at selected lunar mare sites, separated from each other by at least 600 km, within the *Apollo* zone of interest (see Fig. I-1). Data derived from these four missions indicated the suitability of the mare sites for *Apollo* and completely fulfilled the *Surveyor* Project obligations to obtain data for *Apollo.* The final mission, *Surveyor VII,* was used exclusively for scientific investigations of a site basically different from those previously explored—the ejecta blanket of the crater Tycho in the rugged lunar highlands.

B. Project Elements

The *Surveyor* Project was managed by the Jet Propulsion Laboratory (JPL) for the NASA Office of Space Science and Applications. The major administrative and functional elements supporting the project were the Launch Vehicle System; Spacecraft System; and Mission Operations System (MOS), which included the Tracking

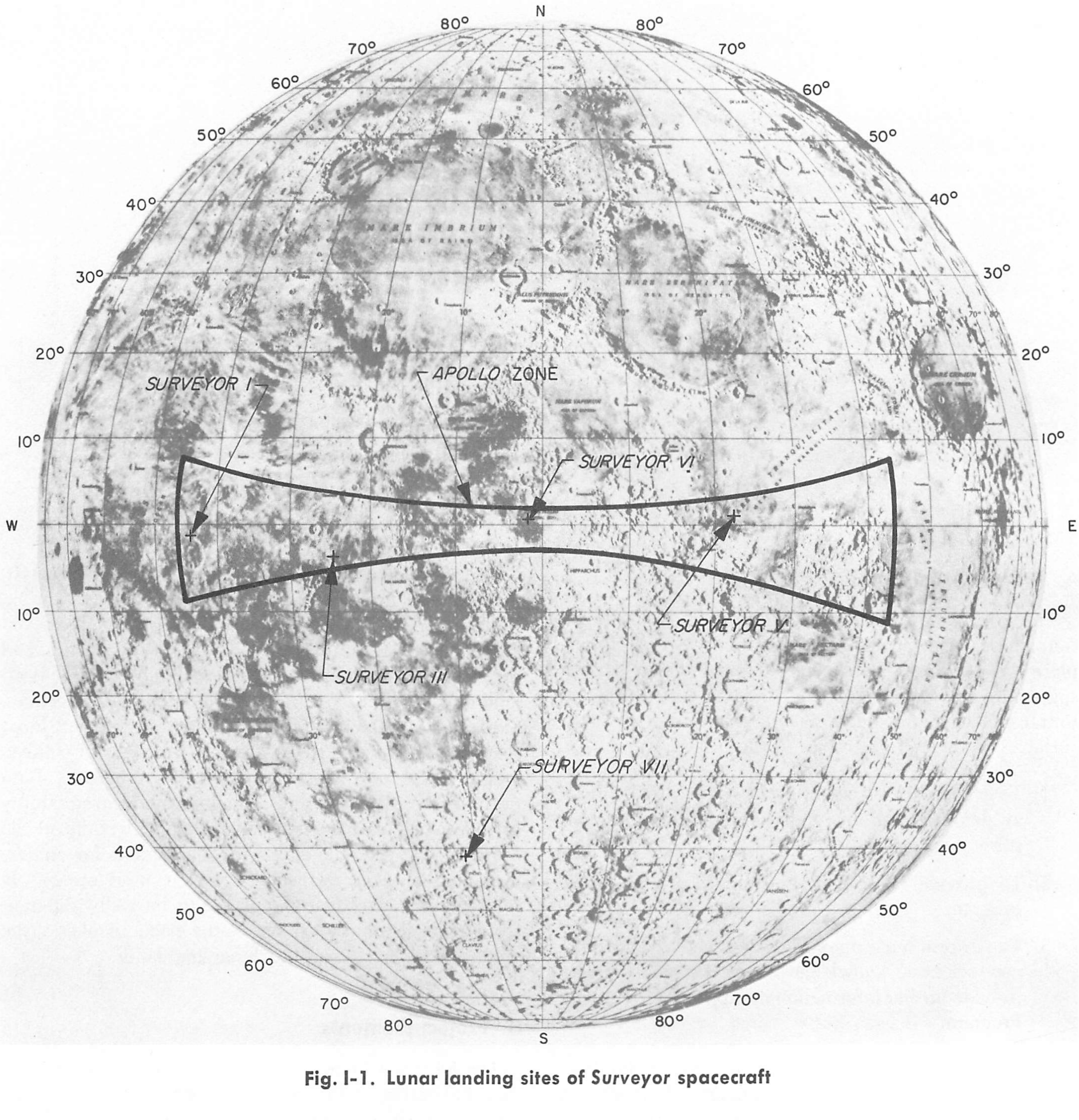

Fig. I-1. Lunar landing sites of *Surveyor* spacecraft

and Data System (T&DS). In addition to overall project management, JPL was assigned the management responsibility for the Spacecraft System and MOS/T&DS. The Lewis Research Center (LeRC) was assigned responsibility for the *Atlas/Centaur* Launch Vehicle System.

Surveyor was the first project to use the *Atlas/Centaur* launch vehicle (Fig. I-2), which was developed and flight-tested in parallel with the spacecraft development. The primary contractors for the launch vehicle were: General Dynamics/Convair (GD/C) for modification of the *Atlas* first stage and development of the new *Centaur* upper stage; Pratt and Whitney for development of the engines for the *Centaur* stage, which was the first to use liquid hydrogen/liquid oxygen, high-specific-impulse propellants; and Minneapolis-Honeywell for the all-inertial *Centaur* guidance system. The Kennedy Space Center, Unmanned Launch Operations Branch, working with LeRC, was responsible for *Centaur* launch operations. Eight flight tests were conducted in the *Centaur* development program to qualify the vehicle for *Surveyor* direct-ascent as well as for parking-orbit missions.

Surveyor spacecraft design, fabrication, and test operations were performed by Hughes Aircraft Company (HAC) under JPL contract and technical cognizance. *Surveyor* was a fully attitude-stabilized spacecraft designed to receive and execute a wide variety of earth commands, as well as to perform certain automatic functions including the critical terminal-descent and soft-landing sequences. To permit *Surveyor* to execute the complex terminal phase of flight, significant new and advanced subsystems, developed and/or employed in combination, were used. These subsystems were: a solid-propellant main retro motor, throttlable liquid propellant vernier engines (also used for midcourse velocity correction), extremely sensitive velocity- and altitude-sensing radars, and an automatic closed-loop guidance and control system (see Fig. I-3).

All *Surveyor* spacecraft carried a survey television camera and engineering instrumentation for obtaining in-flight and postlanding data. As an additional scientific instrument, *Surveyors III* and *IV* carried a soil mechanics/surface sampler to provide data based on picking, digging, and handling of lunar surface material. On *Surveyors IV* through *VII*, a magnet was attached to one of the spacecraft footpads to determine the magnetic properties of the soil. On *Surveyors V* and *VI*, an alpha scattering instrument was substituted for the surface sampler in order to obtain data from which a chemical analysis of the lunar surface material could be made.

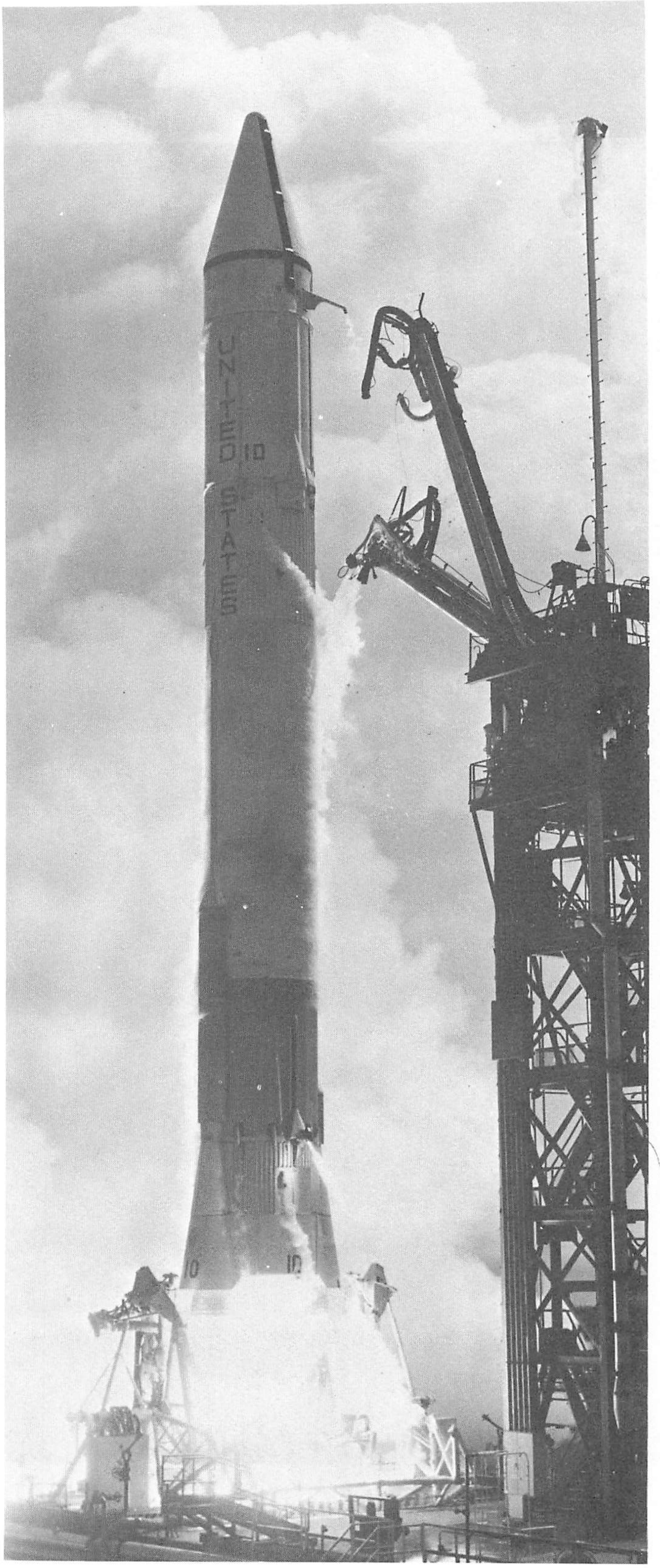

Fig. I-2. *Atlas/Centaur* (AC-10) launching *Surveyor I*

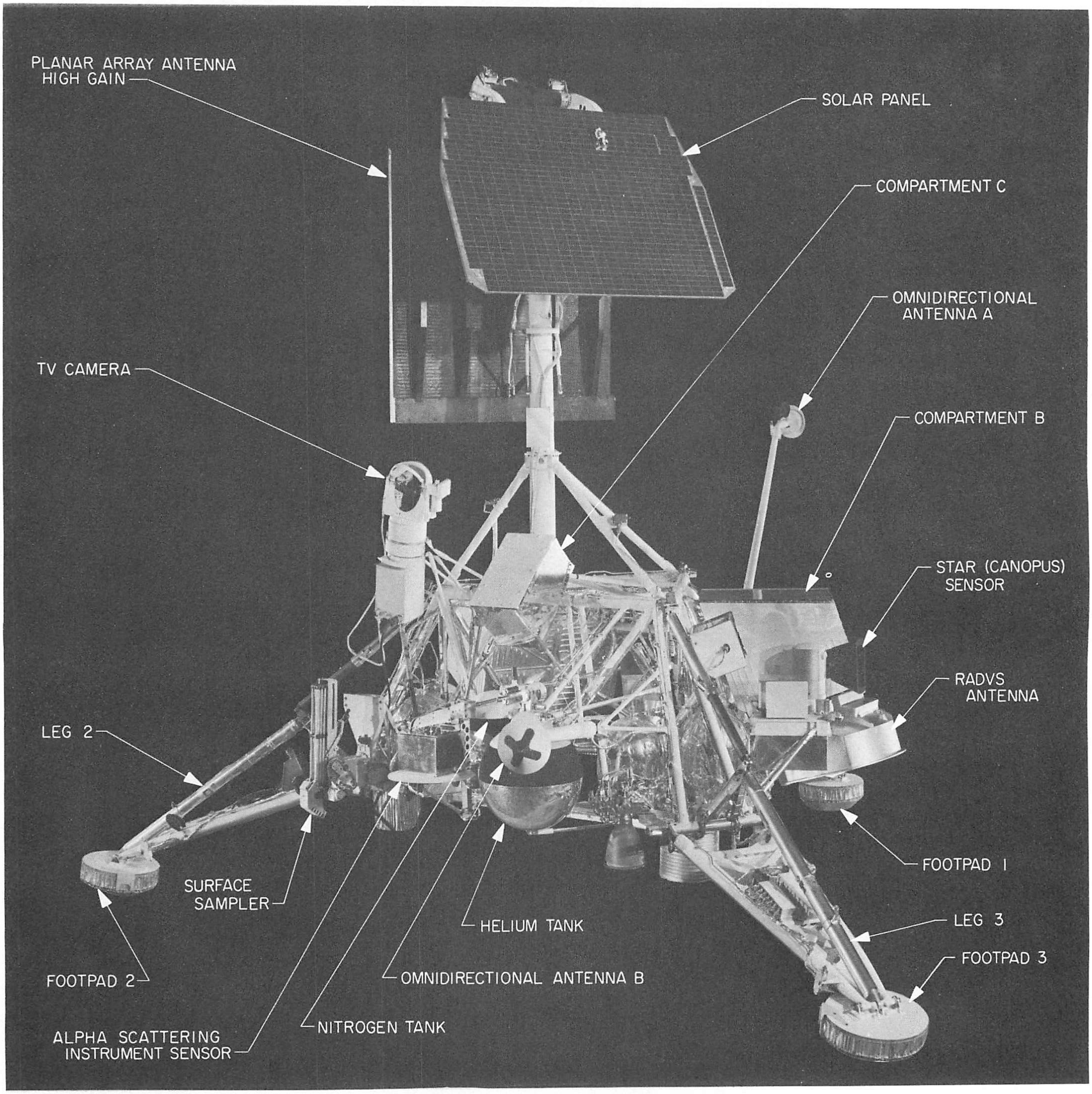

Fig. I-3. ***Surveyor VII*** **spacecraft model**

Surveyor VII carried both the surface sampler and the alpha scattering instrument, as well as additional magnets attached to a second footpad and to the surface sampler.

The MOS controlled the spacecraft from launch through termination of each mission. In this function, the MOS communicated with the spacecraft, assessed spacecraft performance (based on telemetry data), and prepared and issued appropriate commands. The T&DS provided the tracking and communications link between the spacecraft and the MOS and processed spacecraft data in accordance with MOS requirements. For *Surveyor* missions, the T&DS used the facilities of:

(1) Stations, ships, and aircraft of the Air Force Eastern Test Range for tracking and telemetry of the spacecraft and vehicle during the launch phase.

(2) JPL Deep Space Network (DSN) for continuous precision tracking communications, data transmission and processing, and computing.

(3) Manned Space Flight Network and the NASA Communications Network (world-wide), both operated by Goddard Space Flight Center.

The critical *Surveyor* flight maneuvers and most television and surface-sampler operations on *Surveyor* missions were commanded and recorded by the Deep Space Station (DSS) at Goldstone, California (DSS 11, see Fig. I-4), during its view period. Other Deep Space Stations that provided a continuous view of the spacecraft and prime support from their strategic positions around the world were near Canberra, Australia (DSS 42), Johannesburg, South Africa (DSS 51), and near Madrid, Spain (Robledo, DSS 61). Additional support, on a limited basis, was provided by other Deep Space Stations such as Cape Kennedy, Florida (DSS 71), used for prelaunch verification of DSN/spacecraft compatibility, Ascension Island (DSS 72), used along with DSS 71 for near-earth support, and another Goldstone Station (DSS 14), with its large 210-ft antenna. DSS 14 was used as a prime alternate to DSS 11 during the midcourse and terminal descent sequences.

The MOS conducted the *Surveyor* missions from the DSN Space Flight Operations Facility (SFOF) at Pasadena, which served as the center for data processing, performance analysis, and command operations. Special mission-dependent equipment was developed and provided by the *Surveyor* Project, including command data handling consoles (CDCs), installed at the Deep Space Stations, and a television ground data handling system (TV-GDHS), installed at DSS 11 (TV-11) and at the SFOF (TV-1) for processing of *Surveyor* pictures.

C. Project Test Program

The *Surveyor* Project included a comprehensive test program in which each of the supporting systems and interfaces was qualified, beginning at a unit level and progressing to combined systems tests, to ensure overall flight readiness before each launch. Many tests of the spacecraft subsystem and system development were conducted; these tests required the construction of several special spacecraft test models, as well as special test equipment and facilities. The concept used in qualification testing of the spacecraft was to qualify the flight design of representative hardware in type approval tests and to verify the actual flight hardware in flight acceptance tests. Spacecraft type approval and flight acceptance tests were performed at both the unit control item level and at the system level. The system level type approval tests were conducted on a prototype system-test spacecraft (T-21); flight acceptance tests were performed on each flight spacecraft (SC-1 through SC-7)* as soon as initial assembly was completed at the HAC El Segundo facility. Specially developed system test equipment assemblies and spacecraft checkout computer facilities were used to provide real-time processing and evaluation of spacecraft data during tests and operations.

A combined systems test stand (CSTS) was constructed at GD/C for demonstration of electrical and mechanical compatibility of the *Surveyor* spacecraft, the *Atlas/Centaur* launch vehicle, and the aerospace ground equipment during simulated countdown and launch. The *Atlas* and *Centaur* stages of each vehicle used for *Surveyor* missions were tested together in the CSTS, and combined tests (including the spacecraft) were performed in the CSTS with the T-21 and SC-1, SC-4, and SC-5. The other flight spacecraft were not tested in the CSTS because of schedule limitations; however, for each mission, the sequence of tests and operations at AFETR included a joint flight acceptance composite test after initial mating of the spacecraft to the launch vehicle.

The MOS/T&DS tests included extensive equipment and compatibility tests and functional compatibility tests of the various facilities culminating in combined operational readiness tests to validate the entire system prior to each mission.

*Spacecraft serial designations.

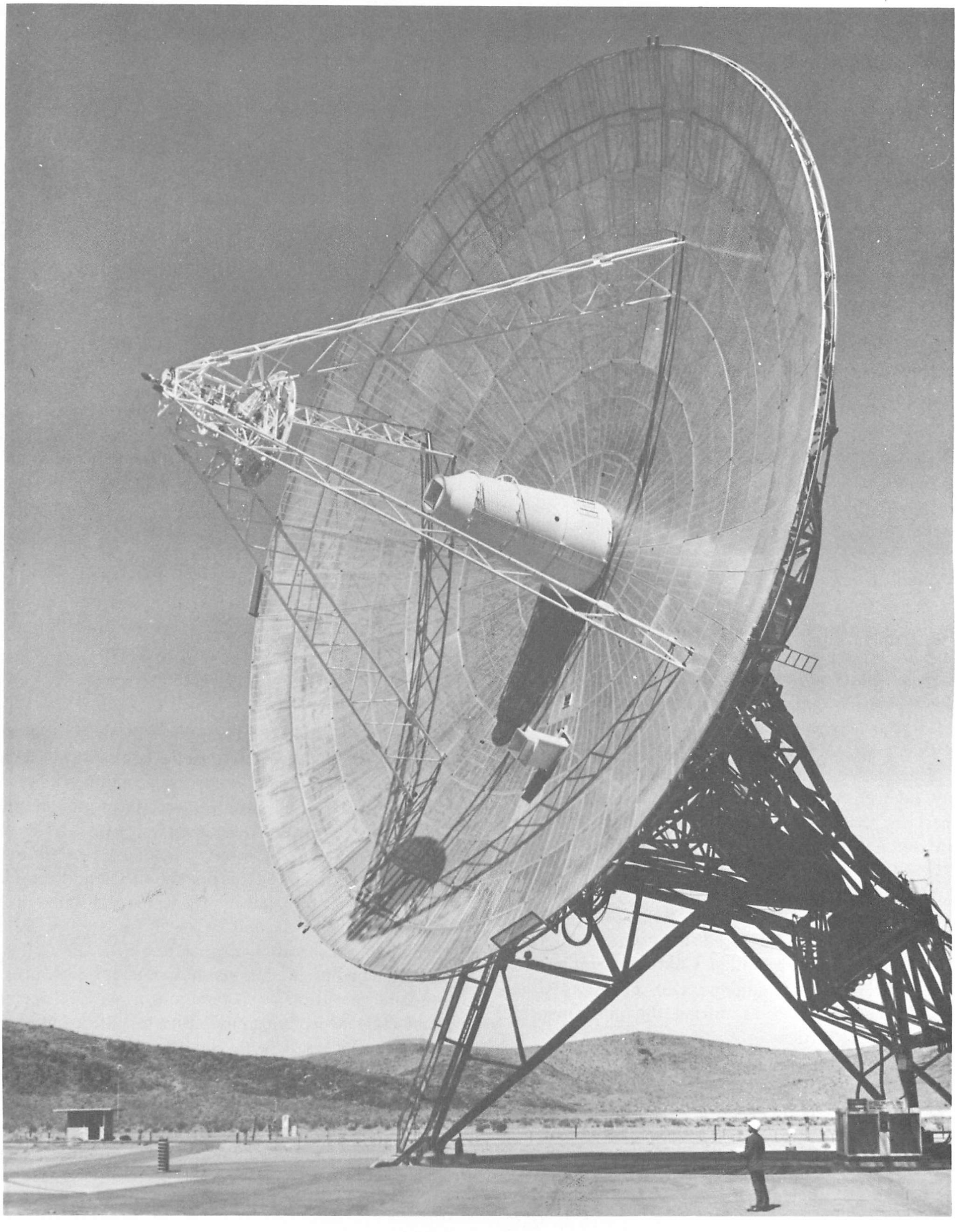

Fig. I-4. The 210-ft antenna at DSS 11

D. The Mission

The transit, or flight, phase of the *Surveyor* spacecraft was from 62 to 66 hr long (see Fig. I-5). All *Surveyor* spacecraft were launched from Launch Complex 36 at Cape Kennedy, Florida; pad A or B, connected to a common blockhouse, with separate control consoles for each of the pads, was used. *Surveyors I, II,* and *IV* were launched via the direct-ascent mode, where the *Centaur* second stage provided only one continuous burn to achieve injection into the desired lunar transfer trajectory. The remaining *Surveyor* spacecraft were launched via the parking-orbit mode, which involved two burns of the *Centaur* stage. The first burn injected the vehicle into a parking orbit with a nominal altitude of 90 nmi. After a coast-period. which could be up to 25 min long, the *Centaur* reignited and provided the additional impulse necessary to achieve a lunar intercept trajectory. The use of the parking-orbit ascent mode permitted the launching of *Surveyor* spacecraft for all values of lunar declinations, thereby allowing the design of launch periods compatible with favorable postlanding lunar lighting.

During a short coast period between *Centaur* main engine cutoff and spacecraft separation, the spacecraft responded to *Centaur* commands to unfold the landing legs, deploy omnidirectional antennas, and turn on spacecraft transmitter high power to facilitate DSN initial acquisition. After separation, the *Centaur* performed a retro maneuver sequence to provide increased separation distance and to miss the moon, and the spacecraft executed automatic antenna/solar panel positioning and sun acquisition sequences to establish the desired attitude of the spacecraft roll axis and to ensure an adequate supply of solar energy for the coast period.

Initial two-way acquisition by a Deep Space Station was accomplished within 1 hr of launch, permitting the MOS to exercise control of the spacecraft by command. Thereafter, the Deep Space Stations received and recorded all desired spacecraft data and transmitted necessary commands. Nearly continuous, two-way tracking coverage was provided during the transit phase.

At a suitable time, between 3½ to 9½ hr after liftoff, spacecraft lockon with the star Canopus was achieved following a spacecraft roll maneuver initiated by earth command. Canopus lockon, by means of a Canopus sensor, provided three-axis attitude reference, which was required before the midcourse and terminal maneuvers could be executed.

Midcourse correction was normally performed during the first pass over DSS 11. After the spacecraft thrust axis was pointed in the desired direction by a combination of roll, yaw, and pitch maneuvers, the three liquid propellant vernier engines were fired by command from earth for a preset time interval. In addition to correcting the trajectory to achieve the desired aim point, the midcourse correction was computed to optimize terminal-descent conditions including main retro burnout velocity, vernier system propellant margin, and arrival time. Following the velocity correction, the spacecraft was returned to coast orientation with sun and Canopus lock.

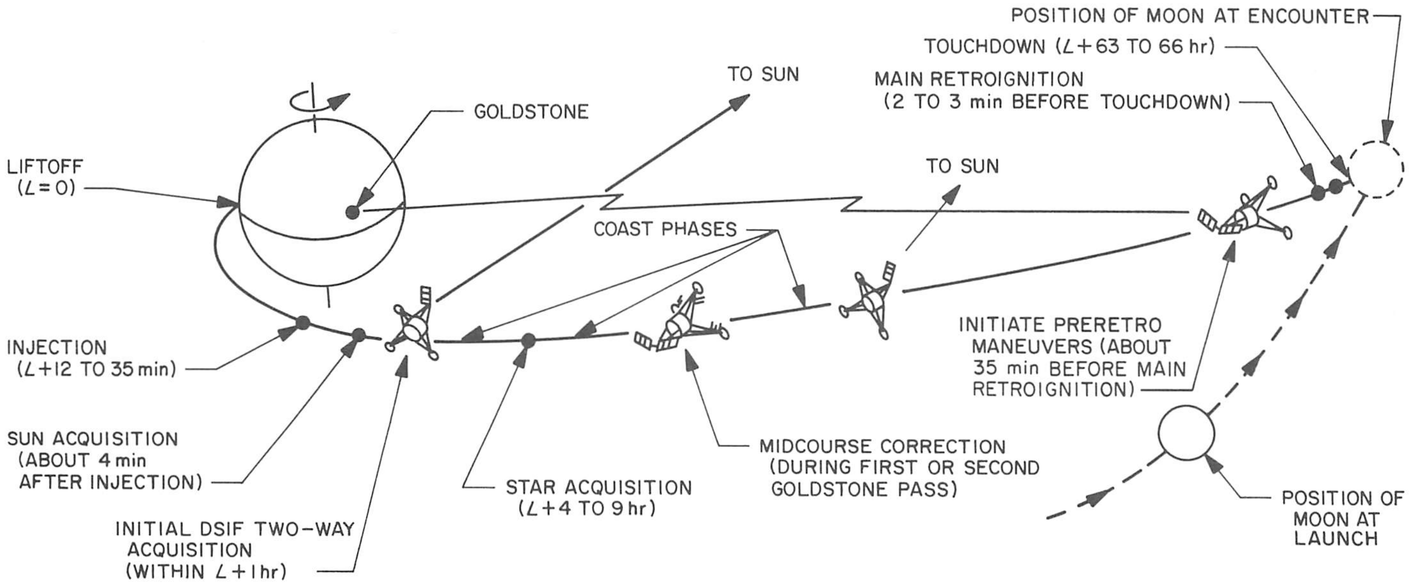

Fig. I-5. Earth-to-moon trajectory—showing major events

In preparation for terminal descent, a combination of roll, yaw, and pitch maneuvers was executed about 35 min before retroignition, to properly align the retro motor nozzle in the direction of the velocity vector. After the attitude orientation maneuver and other preparations commanded from earth, the actual terminal descent sequence was performed automatically by the spacecraft.

The terminal-descent sequence began with a *mark* signal from the spacecraft altitude marking radar (AMR) when the spacecraft was 60-mi slant range from the lunar surface. After a timed delay of a few seconds, which had been preset in the flight control programmer by earth command, the vernier engines ignited, followed by ignition of the solid propellant main retro motor. The main retro motor provided 8,000- to 10,000-lb thrust for about 40 sec, with the vernier engines maintaining the spacecraft in an inertially fixed attitude. Following retro case ejection, which occurred 12 sec after main retro burnout, the radar altimeter and doppler velocity sensor (RADVS) became operational and differentially throttled the vernier engines to point the spacecraft's longitudinal axis along the flight path, while maintaining a total thrust level of about 0.9 lunar *g*. When the spacecraft reached velocity and slant range approximately equal to conditions specified by a "descent contour" that had been programmed in the spacecraft before launch, the RADVS controlled the spacecraft to closely follow the velocity/slant range contour down to the 14-ft altitude mark. At this point, where the spacecraft velocity was about 5 ft/sec, the vernier engines were cut off, causing the spacecraft to free fall and touch down on the lunar surface at about 12 ft/sec.

Landing was planned to occur during the lunar morning to permit early television pictures to be taken under favorable lighting conditions and to provide maximum time for lunar operations before sunset. After an initial engineering assessment of the spacecraft condition, the first television pictures were taken in a 200-line mode using the spacecraft omnidirectional antennas. After the initial 200-line pictures, nearly all television operations were performed in the 600-line mode, which required reconfiguration of the spacecraft, including positioning of the solar panel to receive maximum solar power and precise pointing of the planar array antenna toward earth to provide maximum signal strength. Although most television operations were performed during Goldstone view periods, when the special TV-GDHS could be used to relay the pictures from Goldstone to the SFOF for evaluation in real-time, the overseas Deep Space Stations also participated actively in television operations.

Extensive operations were conducted throughout the lunar day until spacecraft shutdown during the lunar night. This permitted the performance of many engineering and scientific experiments, including those conducted with the alpha scattering instrument and surface sampler. Only near lunar noon was it necessary to restrict the spacecraft duty cycle because of high temperatures.

Spacecraft engineering data were received frequently throughout the lunar day to enable repeated assessment of the spacecraft condition. In addition, two-way doppler tracking data were obtained whenever possible to aid in the determination of the site location.

Before lunar night shutdown, the spacecraft operations were conducted so that chances for spacecraft survival, while operating on successive lunar days, would be increased.

II. Mission Performance

Of the seven *Surveyor* missions, the five soft-landed spacecraft completely fulfilled the overall project objectives, as noted in Section I of this report. After the capability for performing the complicated *Surveyor* missions was proved by the successful launch of *Surveyor I*, maximum use was made of the remaining spacecraft to explore other landing sites and to conduct additional engineering and scientific experiments on the lunar surface. This section presents a summary of the *Surveyor* missions; Section III presents a summary of the science results from the missions. Each *Surveyor* mission is described in detail in the *Surveyor* Mission Reports (Refs. II-1 through II-7).

A. Transit Phase

Information on the transit (flight) phases of the *Surveyor* missions is given in Table II-1. The times listed in this table and throughout this report are GMT, as recorded by the DSN, unless otherwise noted.

Launch occurred on the first day of the scheduled launch period for each mission except for *Surveyor IV*, which was launched on the second day. As shown in Table II-1, on four of the missions, liftoff occurred within less than 1 min of the opening of the launch window. *Surveyor II* liftoff occurred just before closing of a 36-min-long window. Maximum liftoff delay after window opening occurred on *Surveyor III*, with a window penetration of 51 min.

Launch phase performance was very satisfactory on all *Surveyor* missions. The *Atlas/Centaur* launch vehicles performed reliably and demonstrated remarkable injection accuracy. The injection error mapped to the lunar surface was less than 100 nmi on five of the missions, and about 250 nmi on the other two. All spacecraft preseparation events occurred correctly except that deployment of one of the omnidirectional antennas was not indicated until after landing on the *Surveyor I* mission. This malfunction did not cause any degradation of the mission.

Spacecraft separation was normal in all cases, and the small angular rates were quickly reduced to the deadband by the spacecraft flight control system. Following separation, automatic spacecraft sun acquisition and solar panel positioning occurred properly, as did *Centaur* retro maneuver execution. The separation distance between the spacecraft and *Centaur* 5 hr after separation was always well in excess of the specified minimum of 336 km.

Table II-1. Transit (flight) phase summary

Parameter	Mission						
	Surveyor I	*Surveyor II*	*Surveyor III*	*Surveyor IV*	*Surveyor V*	*Surveyor VI*	*Surveyor VII*
Serial number of spacecraft	SC-1	SC-2	SC-3	SC-4	SC-5	SC-6	SC-7
Serial number of launch vehicle	AC-10 (LV-3C)	AC-7 (LV-3C)	AC-12 (LV-3C)	AC-11 (LV-3C)	AC-13 (SLV-3C)	AC-14 (SLV-3C)	AC-15 (SLV-3C)
Event times							
Liftoff L, GMT	May 30, 1966, 14:41:01	September 20, 1966, 12:32:00	April 17, 1967, 07:05:01	July 14, 1967, 11:53:29	September 8, 1967, 07:57:01	November 7, 1967, 07:39:01	January 7, 1968, 06:30:01
Time after scheduled window opening	1 sec	36 min, 0 sec	51 min, 1 sec	29 sec	18 min, 1 sec	1 sec[a]	0.5 sec[a]
Spacecraft separation	L + 00:12:37	L + 00:12:33	L + 00:34:53	L + 00:12:37	L + 00:19:26	L + 00:25:29	L + 00:35:15
Initial two-way acquisition completed	L + 00:27:30	L + 00:32:58	L + 00:56:49	L + 00:28:17	L + 00:33:10	L + 00:35:14	L + 00:58:03
Initial Canopus lockon	L + 04:32:19	L + 06:39:57	L + 09:22:50	L + 06:16:53	L + 06:30:51	L + 08:49:19	L + 08:17:31
Midcourse thrust command	L + 16:04:02	L + 16:28:02[b]	L + 21:55:01	L + 38:36:33	L + 17:48:01[b]	L + 18:41:01	L + 17:00:09
Main retro motor ignition	T − 00:02:47	L + 45:02:09[c]	T − 00:02:59	T − 00:03:13[d]	T − 00:01:51	T − 00:03:01	T − 00:03:22
Initial touchdown T, GMT	June 2, 1966, 06:17:37 (L + 63:36:36)	–[c]	April 20, 1967, 00:04:18 (L + 64:59:17)	–[e]	September 11, 1967, 00:46:45 (L + 64:49:43)	November 10, 1967, 01:01:05 (L + 65:22:04)	January 10, 1968, 01:05:38 (L + 66:35:37)
Ascent mode	Direct	Direct	Parking orbit (22.1-min coast)	Direct	Parking orbit (6.7-min coast)	Parking orbit (12.9-min coast)	Parking orbit (22.4-min coast)
Spacecraft weight, lb							
At liftoff	2193	2204	2280	2294	2216	2219	2288
At touchdown	649	–[c]	659[f]	–[e]	670	662	674
Injection accuracy (miss distance on lunar surface from prelaunch target point), km	460	142	466	176	46	126	77
Midcourse correction, m/sec	20.4	9.6[b]	4.2	10.3	14.0[b]	10.1	11.1
Midcourse correction accuracy (estimated actual landing site error), km	19	–[c]	2.9	5.6	32	10.5	1.7
Terminal unbraked approach angle (from vertical), deg	6.1	–[c]	23.6	31.5	46.4	24.5	35.9
Retro-ignition velocity, ft/sec	8570	–[c]	8620	8600	8490	8490	8590
Conditions at start of RADVS controlled descent							
Altitude, ft	27,800	–[c]	32,900	–[e]	4150	36,600	41,500
Longitudinal velocity, ft/sec	425	–[c]	460	–[e]	45	460	430
Maximum shock-absorber axial load during landing, lb	1600	–[c]	930	–[e]	1660	1810	1700
Total commands sent during flight	288	1579[c]	345	322[g]	725	360	335

[a]Window opening rescheduled after start of countdown to permit improved T&DS average.
[b]Initial midcourse firing.
[c]*Surveyor II* mission was terminated before lunar encounter.
[d]Based on planned time of soft landing.
[e]*Surveyor IV* signal was lost during terminal descent.
[f]At final touchdown.
[g]Until loss of signal.

The premidcourse coast phase was normal on all missions. Initial two-way acquisition was completed satisfactorily and, with few exceptions, excellent deep space communications were provided throughout each mission. Use was made of doppler resolver counters on the *Surveyor VI* and *VII* missions to provide smaller residuals in the tracking data used for orbit determination.

Canopus acquisition was accomplished as desired after completion of a customary star-mapping roll maneuver which permitted verification of Canopus position. In some cases, Canopus acquisition was delayed to avoid possible interference of bright sources, such as the earth, with the Canopus sensor. Canopus acquisition was achieved automatically by the spacecraft on four of the missions, while on *Surveyors I, II,* and *IV*, the Canopus tracker sensitivity was such that a usable lockon signal was not assured, and lockon was achieved by sending a command from earth when Canopus was in the field of view. This mode of initial acquisition was a planned option and presented no difficulty; the Canopus tracker maintained lockon in all cases after initial establishment.

All midcourse corrections were performed during the first Goldstone pass except for the *Surveyor IV* mission, when it was performed, as planned, during the second pass. With the exception of the *Surveyor II* and *V* missions, the midcourse maneuvers were achieved with good results, including trajectory correction to permit soft-landing well within the desired landing site area. The magnitude of the executed midcourse corrections shown in Table II-1 does not reflect actual injection error, since, for each mission, the maneuver also provided for adjustments in aim point and arrival time made after launch to achieve optimum landing and postlanding conditions.

During *Surveyor II* midcourse thrusting, one of the three vernier engines did not fire and spacecraft attitude control was lost. A spacecraft tumbling condition resulted, and could not be corrected either by activation of the cold-gas jet system or by repeated attempts to fire the vernier engines. The spacecraft condition precluded completing the transit phase and achieving a lunar soft landing. Before depletion of spacecraft battery power, the main retro motor was fired. Loss of the spacecraft signal occurred about 30 sec after retroignition.

After the *Surveyor V* midcourse thrusting was completed, the helium regulator in the vernier propulsion system failed to reseat properly, permitting the loss of helium that was required for normal execution of the terminal-descent sequence. Attempts were made to clear the regulator by repeated firing of the vernier engines, but the leak persisted. After careful analysis, the decision was made to redesign the terminal-descent sequence in an attempt to accomplish a soft landing despite very low helium pressure, which finally stabilized at the downstream manifold relief valve setting of about 825 psi. Additional vernier engine firings were made to reduce the spacecraft weight and increase the helium ullage, thereby ensuring a minimum helium pressure drop during terminal descent. Main retroignition was delayed to achieve main retro burnout at a much lower than nominal altitude, and other terminal descent event times were altered to minimize the required total vernier engine impulse; however, the revised sequence would not have resulted in a soft landing if the spacecraft had not performed in a predictable manner during the descent. The success of *Surveyor V* can be attributed to the intensive and resourceful efforts of project personnel responding to the emergency.

Of the six *Surveyor* spacecraft that entered into the critical terminal-descent sequence, all but *Surveyor IV* achieved successful soft landings. The performance of *Surveyor IV* was flawless until the abrupt loss of spacecraft signal occurred just before the expected main retro burnout. Although a detailed examination of the *Surveyor IV* mission was conducted by a formally appointed Technical Review Board, no evidence was found of any single or multiple cause for the failure. On the other missions, spacecraft performance was excellent, and except for the unique landing sequence on *Surveyor III* all events occurred as planned.

Surveyor III terminal descent was completely normal until a few seconds before touchdown, when the radar altimeter and doppler velocity sensor (RADVS) lost lock at an altitude of about 37 ft. Loss of lock, which probably was due to lunar surface irregularities, prevented normal cutoff of the vernier engines at 14-ft altitude. With the engines still thrusting at a level equal to about 90% of the spacecraft lunar weight, the spacecraft lifted off twice after initial contact with the lunar surface. The engines were turned off by an earth command about 1 sec before the third touchdown; the spacecraft came to rest in an upright position, with attitude control maintained throughout the landing sequence.

Briefly, the *Surveyor* soft landings occurred as described in the following paragraphs. Although the unbraked approach angle was as high as 46.4 deg (*Surveyor V*, as shown in Table II-1), the final approach angle and spacecraft attitude before touchdown were nearly vertical on all flights. With the exception of

Surveyor III, each spacecraft succeeded in maintaining the landing parameters well within the design limits. Surface slopes up to about 20 deg (*Surveyor V*) were encountered.

Surveyor I. Touchdown of *Surveyor I* occurred, with a velocity of about 12 ft/sec, on a relatively smooth surface with a slope of less than 3 deg. The three footpads contacted the surface nearly simultaneously, penetrated a few inches, and rebounded about 2½ in. before coming to rest. At least one of the three crushable blocks under the frame also contacted the surface as the shock absorbers compressed. Maximum shock-absorber force was about 1600 lb, which was below the maximum predicted for a hard surface landing.

Surveyor III. The *Surveyor III* spacecraft landed in a crater about 650 ft in diameter and 50 ft deep. As explained previously, three touchdowns occurred. The initial touchdown occurred near the rim of the crater, and interaction with the sloping surface caused the spacecraft to rebound in the downhill direction, although the flight control system acted during each liftoff to correct the spacecraft attitude to the pre-touchdown state. The spacecraft traversed about 50 to 70 ft during the first liftoff and 35 to 45 ft during the second (Fig. II-1). A hop of about 1 ft took place in connection with the third touchdown event. Each of the *Surveyor III* touchdowns was more gentle than the landing of *Surveyor I.* Although no mechanical damage occurred, some subsystem electronics damage was sustained coincident with the second touchdown. There is speculation that this damage, which principally affected signal processing, may have resulted from induced damage caused by the arcing of radar high voltage in the presence of the ionized plasma, which enveloped the lower part of the spacecraft at touchdown with the vernier engines thrusting.

Surveyor V. The landing of *Surveyor V* occurred on a 17-deg slope inside a small, 30-ft-diameter crater. Although the slope exceeded the design limit of 15 deg, no stability problem was encountered. One spacecraft leg touched down approximately 0.2 sec before the nearly simultaneous contacts of the other two. The spacecraft slid about 32 in. downslope and rotated about 6 deg

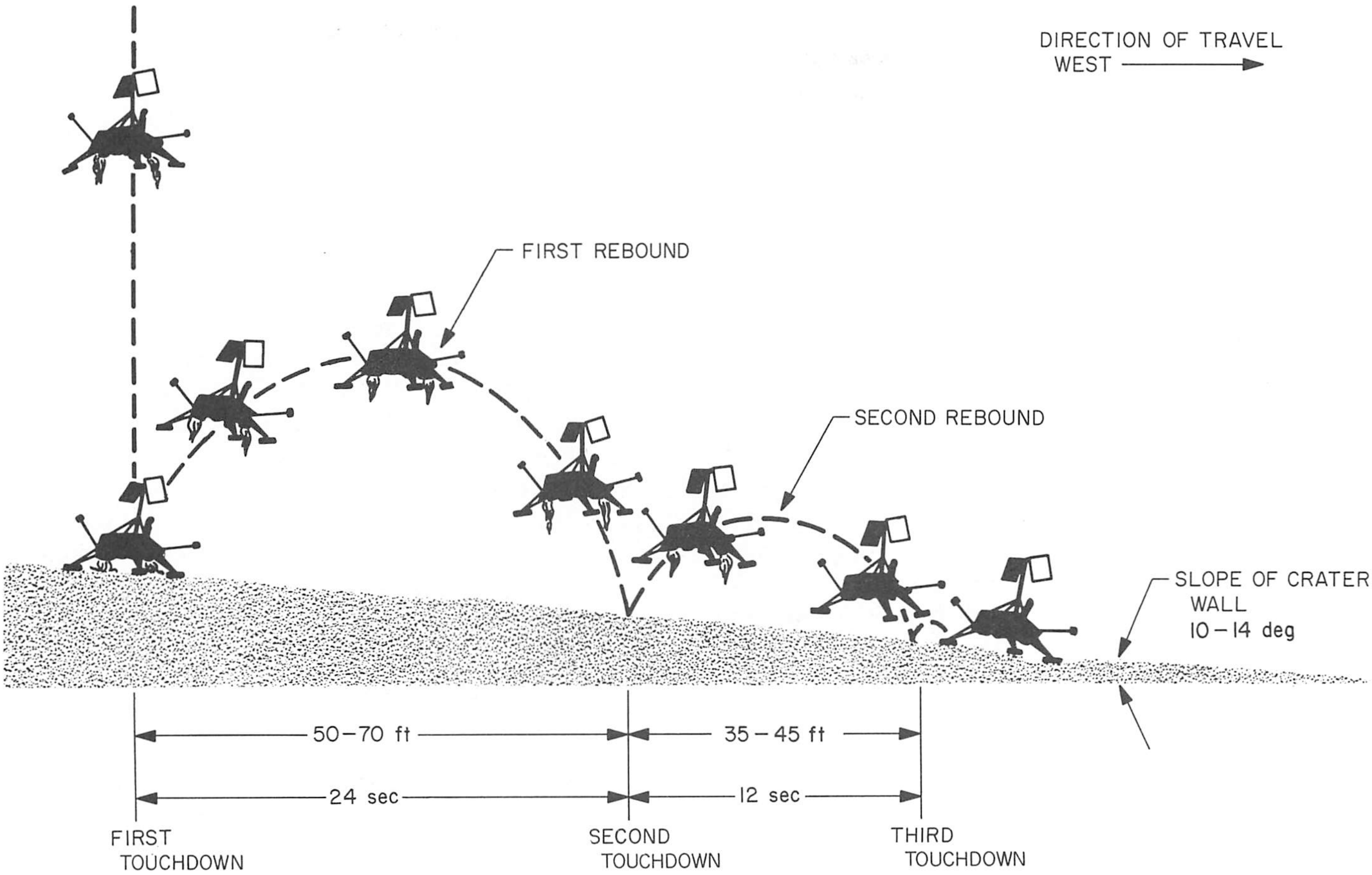

Fig. II-1. Sketch showing events during *Surveyor III* lunar landing

before finally coming to rest with a tilt of about 20 deg. The sliding motion during landing caused the footpads to dig trenches in the lunar surface (Fig. II-2).

Surveyor VI. The *Surveyor VI* landing was on a relatively smooth and essentially level surface. The shock-absorber loads, as indicated by telemetered strain-gage readings, were similar to those of *Surveyor I.* It is estimated that the spacecraft rebounded 8 to 11 in. before finally coming to rest.

Surveyor VII. The landing of the last *Surveyor* occurred on a relatively flat surface located on the ejecta, or flow, blanket near the large crater Tycho. The spacecraft missed hitting any of the large rocks and boulders that litter the area; one large rock was close to footpad 2. Spacecraft rebound was approximately 9 in.

B. Postlanding Phase

With the exception of *Surveyor III,* the engineering assessments conducted immediately following touchdown confirmed that each of the soft-landed spacecraft was in excellent condition, permitting television surveys and other operations to proceed. On *Surveyor III,* special interrogation sequences were conducted to investigate the telemetry failure which occurred during landing. This failure appeared to be shorting caused by high-voltage puncture of telemetry signal switches, and primarily affected analog data obtained in the higher rate modes. *Surveyor III* data obtained in the lowest rate mode were less seriously degraded. All landed spacecraft performed extensive television operations and other lunar surface experiments and operations. Summary information on the lunar surface phase of the *Surveyor* missions in contained in Table II-2.

A large quantity of television pictures was obtained on each of the successful missions, beginning shortly after landing and extending until after sunset. *Surveyors I, V,* and *VII* also obtained pictures on subsequent lunar days. Television performance was generally good, and the quality of the picture was excellent on each mission when favorable lighting conditions existed. The 600-line, high-resolution pictures show details as small as 1/50 in. The vast quantity of television pictures returned by the *Surveyor* spacecraft was obtained during a variety of operations and experiments and included:

(1) Wide- and narrow-angle panoramas and special area surveys of the visible lunar surface out to the horizon, repeated at different sun angles during the lunar day. A few pictures also were obtained after sunset with the surface illuminated by earth light.

(2) Images obtained on *Surveyors I, III,* and *V* with different color filters, which were reconstructed

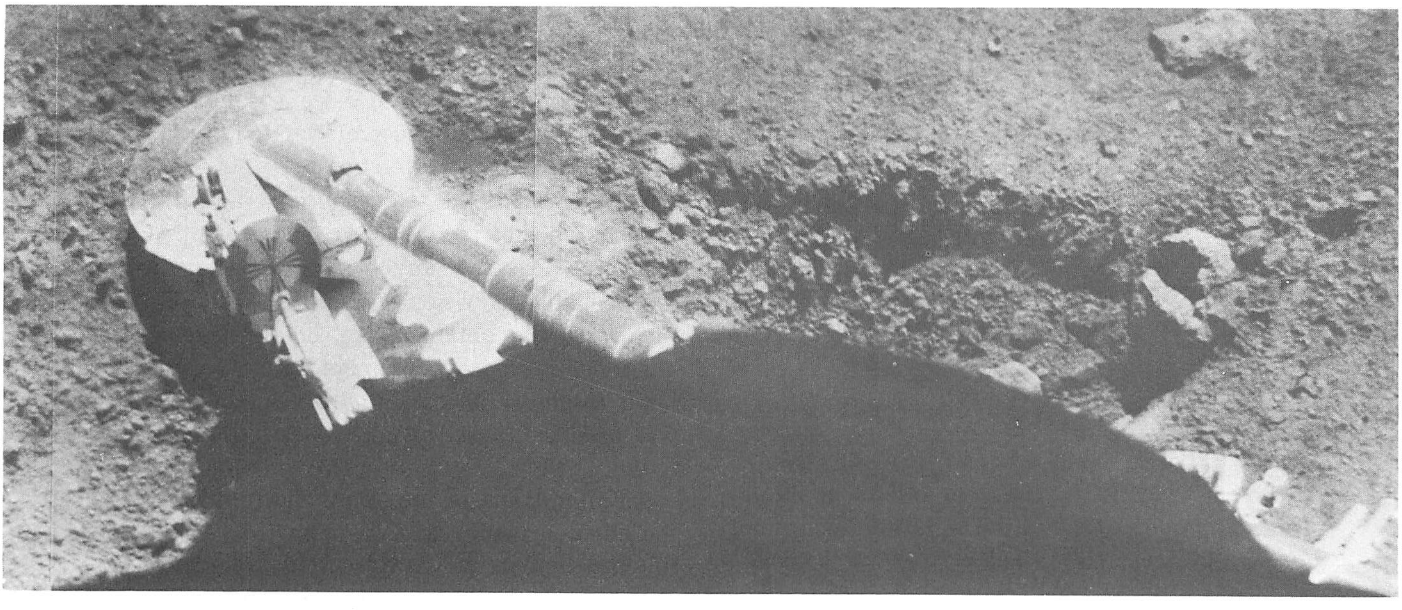

Fig. II-2. Wide-angle mosaic of footpad 2 and the trench formed during *Surveyor V* landing. The depression formed during the first impact of footpad 2 can be seen at the right-hand end of the trench (September 14, 1967, between 04:00 and 06:00 GMT; Catalog 5-MP-19)

Table II-2. Lunar surface phase summary

Parameter	Mission				
	Surveyor I	*Surveyor III*	*Surveyor V*	*Surveyor VI*	*Surveyor VII*
Landing site (best estimate)					
Latitude	2.46°S[a]	2.99°S[b]	1.42°N	0.51°N[b]	40.88°S[b]
Longitude	43.23°W[a]	23.34°W[b]	23.20°E	1.39°W[b]	11.45°W[b]
Landed roll attitude (spacecraft +X axis relative to lunar north), deg clockwise	179	134	114.5	149[c]	290
Lunar surface slope, deg	<1	12.5	19.5	<1	3
Landing after local lunar sunrise, hr	57	23	35	9	35
Operational period, lunar days	1, 2, 5, 6, 8	1	1, 2, 4	1, 2	1, 2
Number of commands sent to spacecraft after landing	121,700	57,000	122,000	163,600	150,000
Number of television pictures taken	11,240	6,326	19,118	29,952	21,038
Spacecraft shutdown after first lunar sunset, hr	53	6	115	41	80
GMT of final loss of spacecraft signal	January 7, 1967, 07:30	May 4, 1967, 00:04	December 17, 1967, 04:30	December 14, 1967, 19:14	February 21, 1968, 00:24

[a]Based on *Lunar Orbiter III* photograph of landed *Surveyor I*.
[b]Based on correlation of *Surveyor* pictures and *Lunar Orbiter* photographs.
[c]Before hop.

on earth in true color, and images obtained on *Surveyors VI* and *VII* with different polarizing filters.

(3) Images obtained with different focus settings (and using a special stereo viewing mirror on *Surveyor VII*) to provide a means of determining the distance of objects from the camera.

(4) Pictures of the visible parts of the spacecraft (directly or through viewing mirrors) including deployment and operation of the surface sampler and alpha scattering instrument and lunar surface material that adhered to the soil magnets.

(5) Pictures showing lunar surface disturbances that resulted from initial landing and subsequent experiments.

(6) Pictures of stars to aid in spacecraft attitude determination.

(7) Pictures of the earth at different phases of illumination (*Surveyor VII*) and during a total eclipse of the sun (*Surveyor III*, in color).

(8) Pictures of the solor corona after sunset.

(9) Pictures of laser beams directed at *Surveyor VII* from two locations on the earth.

Although the *Surveyor III* camera was equipped with a modified sunshade to minimize glare, some of the *Surveyor III* pictures contained more glare than pictures obtained on the other missions. This was, in part, due to the unfavorable landed roll orientation to which *Surveyor III* was restricted, because of a RADVS beam cross-coupling characteristic, and partly to contamination, or pitting, of the upper part of the camera mirror, which probably occurred because the vernier engines were on when the spacecraft landed. The *Surveyor III* camera also experienced some problems in stepping, which restricted the television coverage and limited the total number of pictures taken on that mission to about 6300, the least for any of the landed spacecraft. The most reliable television operation and the highest-quality pictures were obtained on the *Surveyor VI* and *VII* missions; an improved camera with a redesigned mirror-hood assembly and an improved dynamic light range was used on these missions.

An abundance of important data on the properties of the lunar surface was also obtained by the special scientific instruments which were selectively included on the later *Surveyor* missions. The surface sampler performed very extensive operations on the two successful missions on which it was flown (*Surveyors III* and *VII*), and the alpha scattering instrument was used successfully on all three of the missions on which it was carried (*Surveyors V, VI,* and *VII*).

Operations of the surface sampler were guided almost entirely by television pictures taken at selected intervals between stepping commands. For this reason, surface-sampler operations were conducted during Goldstone view periods when the TV-GDHS could be used for real-time monitoring of the instrument at the Space Flight Operations Facility (SFOF) in Pasadena. The basic experiments performed by the surface sampler are summarized in Table II-3 and included: (1) bearing tests in which the surface sampler scoop was pushed down against the surface; (2) impact tests in which the scoop was dropped from different heights; (3) the digging of trenches, involving one or more passes with the scoop door open; and (4) the handling and dumping of various surface materials, including rocks. Data from the surface sampler operations consisted primarily of the television pictures obtained. Measurement of surface sampler motor currents, compared with prelaunch calibrations, were also a means of deducing strength and density of lunar surface material. As noted below, the surface sampler was also used effectively in support of the alpha scattering instrument on *Surveyor VII,* the only mission that carried the surface sampler and the alpha scattering instrument.

Data for determination of chemical composition of the lunar surface were accumulated after the sensor head of the alpha scattering instrument was deployed to the lunar surface in a two-step operation. Before deployment, calibration data were obtained in the stowed position with the sensor head in contact with a standard sample. In the first step of deployment, the instrument was released to a position above the lunar surface where background radiation was detected. The deployment mechanism of the alpha scattering instrument operated correctly on *Surveyors V* and *VI*; however, the mechanism jammed on the *Surveyor VII* mission. It was finally freed and pushed to the lunar surface by the surface sampler.

Table II-3. Summary of basic surface-sampler tests

Test	Surveyor III	Surveyor VII
Bearing	7	16
Trenches (single or multi-pass)	4	7
Impact	13	2
Rocks moved or handled	1	4

Table II-4 is a summary of the alpha scattering instrument data accumulation time in each position, based on the spectra data assembled by the on-site computers at the prime Deep Space Stations and transmitted by teletype to the SFOF.

On the *Surveyor V* mission, the alpha scattering instrument sensor head was moved a few inches downslope as the result of a vernier engine static firing. Sufficient alpha scattering data were accumulated, before and after the firing, to permit a chemical analysis of both samples. After sufficient on-surface data were obtained on *Surveyor VI,* a spacecraft hop was performed, which resulted in the sensor head turned upside down.

The *Surveyor VII* mission was extremely productive from the chemical analysis standpoint as the result of coordinated support by the surface sampler. In addition to aiding in the initial deployment of the sensor head, the surface sampler was used to relocate the instrument on the surface. Alpha scattering data were obtained at these locations, each representing a different type of local sample: relatively smooth and undisturbed lunar surface, a lunar rock, and a subsurface area extensively trenched by the surface sampler.

As noted previously, a vernier engine static firing and a spacecraft hop were conducted on the *Surveyor V* and *VI* missions, respectively; both engine tests were successful. About 53 hr after the landing of *Surveyor V,* the vernier engines were fired briefly (0.55 sec) to determine lunar surface erosion effects. Erosion of the lunar surface was clearly evident in the postfiring television pictures. There was no apparent spacecraft degradation caused by

Table II-4. Science-data accumulation times for alpha scattering instrument

Operation	Time, min		
	Surveyor V	Surveyor VI	Surveyor VII
First lunar day			
Stowed position (standard sample)	75	320	312
Deployed above surface (background radiation)	170	370	720
On lunar surface			
Sample I	1056	1800	1860
Sample II	4005	—	618
Sample III	—	—	402
Second lunar day	1475[a]	—	2070

[a]As reported by Deep Space Stations.

the firing, and only a very slight shift in spacecraft attitude occurred, along with the movement of the alpha scattering instrument.

On *Surveyor VI,* the vernier propulsion system was monitored closely, beginning immediately after landing, in order to prepare for the hop. A time shortly after lunar noon was selected, thus permitting sufficient time to obtain substantial results from the other tests. Also, during the lunar noon period, it was possible to shade critical propulsion system elements to effectively reduce the temperatures below maximum recommended limits, before firing the engines.

In performing the hop, the engines were operated 2.5 sec, being shut down by a backup earth command transmitted 0.5 sec after the primary command, which was not acted upon by the spacecraft. The spacecraft rose about 12 ft above the lunar surface and landed again about 8 ft from its initial location. The time from liftoff to touchdown was approximately 6.1 sec. Although the hop was greater than expected because of the 0.5-sec-longer burn time, the spacecraft made a completely successful landing, and no spacecraft damage resulted, other than the upside-down position of the sensor head. From the spacecraft's new lunar surface location, excellent pictures were obtained of the original landing site, including the erosion caused by the vernier engine firing.

In another unique test, the attitude control gas jets were fired twice, for durations of 4 and 60 sec, to determine the effects on the lunar surface. Television pictures showed the lunar surface disturbances caused by the firings.

Spacecraft thermal control functioned well on the surface during the lunar day, although the spacecraft was not designed to maintain the temperatures of all elements within operational or survival limits during the lunar phase. On the missions for which landing occurred in the *Apollo* zone, the solar panel and planar array antenna were used to maximum advantage by positioning them to shade critical elements such as the television camera and electronics compartments. This permitted operation to be conducted throughout most of the lunar day with a minimum of low-duty cycle periods required near lunar noon when temperatures were the highest. The latitude of the *Surveyor VII* site made shading more difficult; nevertheless, by carefully scheduling operations and experiments, a high level of activity was possible most of the time.

A total eclipse of the sun was encountered during the first lunar day of *Surveyor III* and also during the second lunar day of *Surveyor V*. Valuable thermal data were obtained. Also, as a result of the high tilt of the *Surveyor III* spacecraft, it was possible to point the camera toward the earth and obtain pictures of the event.

Generally, the condition of each of the spacecraft remained excellent throughout the first lunar day, until shutdown during the lunar night. The spacecraft electronic compartment thermal switches, which were designed to open after sunset, did not always open when temperatures dropped to the established set point. This forced earlier shutdown of the spacecraft during the lunar night than otherwise would have been possible.

The spacecraft suffered varying degrees of degradation during the extremely cold lunar night; however, all but *Surveyor III* responded to commands on the second lunar day. Operations were conducted with *Surveyors I* and *V* as late as the eighth and fourth lunar days, respectively.

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III. Scientific Results Summary

Surveyor spacecraft have been successfully landed in four maria and one highland site, as follows:

(1) *Surveyor I:* A flat surface inside a 100-km crater in Oceanus Procellarum.

(2) *Surveyor III:* Interior of a subdued 200-m crater, probably of impact origin, in Oceanus Procellarum.

(3) *Surveyor V:* A 10-m crater, which may be a subsidence feature, in Mare Tranquillitatis.

(4) *Surveyor VI:* A flat surface near a mare ridge in Sinus Medii.

(5) *Surveyor VII:* A flat surface in the ejecta or flow blanket close to Tycho.

All of these sites are strikingly similar. The resemblances apply to topographic details, the structure of the surface layer, mechanical, thermal, and electrical properties, and, for those sites where such measurements were made, to chemical composition and content of magnetic material. It is unlikely that sites on earth, thousands of kilometers apart and selected in a manner similar to the *Surveyor* landing sites, would be so much alike. The highland site near Tycho differs most from the other four, but even for this site, the resemblances to the other sites are more notable than the differences. It is clear that the same general processes are responsible for the formation of the outer layers of the moon both in the maria and highlands.

In all of the sites the visible surface consists primarily of fine particles. The bulk of the material is smaller than 100 μm diameter and is probably primarily between 10 and 50 μm. Lying on and in this fine matrix are coarser blocks in sizes up to several meters across. There is a continuous distribution of particle sizes from the coarse blocks down to the fines. Many of the resolvable particles, larger than 1 mm, are weak aggregates of fines. Some, however, including most of the large blocks, are rocks.

The thickness of the fragmental layer in the maria sites apparently varies from about 1 to 20 m, being deepest in flat areas and on crater floors, especially in regions which appear older on the basis of the frequency of craters around 1 km in diameter. It is shallower on crater rims and mare ridges. In the one highland site so far examined, where the surface is believed to consist of rather recent ejecta or flows from Tycho, the layer of fine fragments may be only a few centimeters deep.

The distribution of craters extends down to diameters of a few centimeters. Below a centimeter or so, clear crater shapes are not identifiable; smaller craters, if present, blend into the general texture of the surface. The shapes of the small craters (1 cm–10 m) are not markedly different from those of larger craters. Coarse blocks, presumably ejected during crater formation, are concentrated around the rims of some of the craters larger than a few meters. The crater size, below which these coarse blocks are not produced, appears to vary from place to place and is believed related to the depth of the fragmental layer: coarse blocks are produced when the crater extends into the denser material below. Indeed, this relation has been used as the primary basis for estimating the thickness of the layer of weakly bound fragments.

The shapes and positions of the blocks associated with a crater vary with the apparent age of the crater, as indicated by its sharpness and contour. Blocks associated with fresher craters are more angular and tend to lie on the fine material, rather than being partially buried. The density of one rock was 2.8 ± 0.4 g/cm^3.

In the mare areas, the density of the fine-grained matrix at the surface appears to be between 0.7 and 1.2 g/cm^3; about 5 cm below the surface the density is about 1.6 g/cm^3.

The dielectric constant, averaged over areas a few meters to a few tens of meters in diameter and measured by radar reflectivity at 2–3 cm wavelength, is about 2.5 in the maria and 3.8 at the highland site. The radar return comes from within ½ m of the surface. Local increases in radar reflectivity are associated with rock concentrations at crater rims and with ridges.

The thermal parameter j^* for areas within 20 m of the spacecraft was measured at about 500 cm^2 sec$^{1/2}$ °K/cal for the maria at sunset and during eclipses. This is about half that observed for 20-km areas from earth. The difference may be due to experimental error or to the increased view angle for rocks from a landed spacecraft as compared with that from earth. For the highland, a considerably lower value, 240 cm^2 sec$^{1/2}$ °K/cal for the thermal parameter was observed in one direction and was presumably associated with a large nearby group of rocks. During daytime, with most of the spacecraft, variations in brightness temperature were found that appeared attributable to directional thermal emission.

The static bearing capacity of the top few millimeters of surface is less than 10^4 dyn/cm^2. At a depth of 2 cm it is about 2×10^5 dyn/cm^2. At a depth of 5 cm it averaged 5×10^5 dyn/cm^2. These values relate to bearing diameters 5–10 times the depth. In the small crater where *Surveyor V* landed, the bearing capacity was slightly lower than for the other mare sites. At least the upper 10 mm compressed under a bearing load at the mare sites to a density of about 1.5 g/cm^3.

The cohesion of the fine material is about 10^4 dyn/cm^2. The adhesion of material impacting painted spacecraft parts is 10^2 to 10^3 dyn/cm^4; lunar material pressed against the spacecraft generally did not adhere.

The permeability to gases of the top few centimeters of mare material is 10^{-8} to 10^{-7} cm^2. The shear wave velocity, in the maria, is estimated to be about 20 m/sec and the compressional wave velocity approximately 60 m/sec.

The composition of the surface material at the *Surveyor V, VI,* and *VII* sites is, in atomic percent, presented in Table III-1.

The composition of the mare surface, therefore, is basaltic in character, and for the major elements resembles that of common terrestrial basalts. The composition of the highland material, which is believed to have been ejected from many kilometers below the surface, is similar except for the lower iron content.

Table III-1. Surface material in atomic percent

Element	Mare	Highland
C	< 2	< 2
O	58 ±5	58 ±5
Na	< 2	< 2
Mg	3 ±3	4 ±3
Al	6.5 ±2	8 ±3
Si	20 ±2	18 ±4
Ca, K, P, S	6 ±2	6 ±2
Fe, Co, Ni, Ti, Mn	5 ±2	2 ±1

$^*j = (k\rho c)^{-1/2}$ where: k = thermal conductivity, ρ = density, c = specific heat.

These compositions differ considerably from those believed representative of condensed solar material, as indicated by, for example, the composition of the solar atmosphere. They strongly suggest that the lunar surface material solidified from a melt, with considerable chemical differentiation occurring during the process of melting and solidification. This in turn suggests, though it does not prove, that major portions of the moon have been molten and perhaps may still be; also, that the composition of portions of the lunar interior differs significantly from that of the surface.

The quantity of ferromagnetic particles found at the sites of *Surveyors V, VI,* and *VII* was relatively low. Less than 1/4% of the particles appear to be magnetic. These results are consistent with a basaltic composition with little addition of meteoritic iron.

The albedo of the exposed fine material is about 20% lower than that of the great majority of exposed rock surfaces. It is about 20% higher than that of the fine material a millimeter or so beneath the surface. Since the fine material must be stirred by meteorite impact, there must then be some process that whitens the exposed surface and presumably a process that darkens the fine material a millimeter or more down so as to maintain a sharp interface between the lighter and darker material.

The photometric function of the exposed matrix is similar to that for the whole lunar disk as seen from earth. The fine material, a few millimeters beneath the surface, also has a photometric function similar to that of the whole disk. When the surface is compressed without stirring, the photometric function changes toward a more Lambertian relation; presumably this change arises from a change in the surface structure toward one of a higher density more closely resembling that typical of terrestrial soils.

The color of the maria surface material was found to be gray, with no obvious color differences. Some of the highland material, especially the rocks, produces noticeable polarization of reflected sunlight.

Under the various landing conditions the footpads and blocks of the spacecraft landing gear penetrated up to 10 cm into the mare surface. When a horizontal velocity component was present, a trench was plowed in the surface. Even with no horizontal velocity, surface material was thrown out, pushed aside, and compressed downward by the landing gear. Some small particles were moved along the surface by the action of cold nitrogen gas directed against the surface through the attitude-control jets of the spacecraft. When the vernier engines were fired against the surface, erosion was produced in two ways. First, with a short (0.5-sec) firing that was abruptly cut off, the gas entered the pores of the lunar material and then blew off at shutdown, taking lunar material with it. Second, with a longer (2.5-sec) firing, during which the engines gradually lifted away from the surface, surface material gradually sheared away by viscous erosion. Most of this erosion consisted of the near-horizontal blowing of particles and aggregates a few millimeters to a few centimeters in diameter. These particles were blown several meters horizontally and in some cases a meter vertically. When the exhaust passed across a surface previously disturbed by the landing gear, some excavation of the matrix occurred by viscous erosion; elsewhere, excavation was associated only with gas blowoff at shutdown. The only degradation to the function of the spacecraft produced by firings against the surface was a fogging of some mirrors which had a direct line-of-sight to the surface directly below the vernier engine exhausts.

On flat areas, the fine matrix surrounding large blocks generally contacts the blocks at the level of the surrounding matrix. On slopes, however, fillets of fine material are observed on the uphill side of blocks. This strongly suggests that the fine material has gradually moved downhill, faster than have the larger blocks.

Two-way doppler tracking of the landed *Surveyors* from earth stations provided data which identified errors in the best available lunar ephemerides and should permit a corresponding improvement, as well as improvement of knoweldge of lunar physical librations. Also, by providing information on positions of the landed *Surveyors* in selenocentric inertial coordinates which can be compared with positions determined with respect to lunar features visible from *Lunar Orbiter* photographs, the doppler results should permit a determination of the position of the lunar center of figure relative to the center of mass.

Besides providing a great deal of data on the moon, the *Surveyors* made some useful observations of earth and sun. Pictures of the earth were taken through polarizing filters at frequent intervals near quarter-earth, and occasionally of the crescent. Pictures were taken through color filters of the crescent and of the earth totally eclipsing the sun. The eclipse pictures provided data on the transmission of light through the earth's atmosphere as a function of wavelength, cloud distribution, and altitude.

In a test of procedures for pointing laser beams at a lunar object, pictures were taken by a *Surveyor* of laser beams pointed at it from earth. The transmitted beam power was 1–2 W, and the emitter collimation was 0.5–3 sec of arc.

Observations of the solar corona revealed appreciable luminance out to about 50 solar radii, thus indicating that the K-corona is continuous with the zodiacal light. Polarization data were obtained in these observations.

A much more complete account of the scientific results of the *Surveyor* Project is given in Part II of this report. Still greater detail is included in Part II of the reports on the individual *Surveyor* missions (Refs. III-1 through III-6). Part III of these reports (Refs. III-7 through III-11) presents pictures taken by *Surveyors*.

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IV. Project History

A. Project Origin

The *Surveyor* Project was conceived by the National Aeronautics and Space Administration, in conjunction with the Jet Propulsion Laboratory, as a program to soft-land a series of unmanned spacecraft on the moon. This program was to utilize the *Atlas/Centaur* as a launch vehicle. The program objectives were to:

(1) Demonstrate the technical feasibility of unmanned lunar exploration by soft-landing equipment at given sites on the surface and by performing useful experiments with that equipment.

(2) Obtain lunar and cislunar scientific information to increase understanding of the origin of the moon and solar system.

(3) Gather information and develop technology needed to place a man safely on the moon.

These objectives were integral to NASA's long-range unmanned lunar program, which also projected lunar orbiting vehicles, as well as follow-on *Surveyors* with extended payloads and roving capabilities.

1. Preliminary Activities

The actual implementation of the *Surveyor* Project was preceded by a number of key studies, decisions, and proposals which emphasized the feasibility of soft landings and provided the basis for a program plan. Briefly, these preliminary steps were:

(1) In the fall of 1958, a NASA Working Group on Lunar and Planetary Surfaces initiated a study of feasible lunar scientific exploration programs. Studies at Marshall Space Flight Center (MSFC) pertaining to use of the *Saturn* vehicle established overall technical feasibility of lunar soft landing.

(2) *Centaur* development was initiated in October 1958 by the Advance Research Projects Agency, which originally conceived the *Centaur* vehicle as a second-stage, multiple-burn vehicle, capable of placing a payload into a circular, 24-hr (synchronous) earth orbit. General Dynamics/Convair, already building and supplying the *Atlas,* was also designated to develop and supply the *Centaur.* Pratt and Whitney was named as the developer/supplier of the liquid hydrogen/liquid

oxygen, high-impulse engine required for *Centaur* applications.

(3) In July 1959, responsibility for the *Centaur* vehicle was transferred to NASA, and, under NASA, in December 1959, to MSFC in Huntsville, Alabama.

(4) In February 1960, a study was initiated by JPL to determine the application of the *Centaur* to lunar and planetary missions. Results of this study were issued in June 1960 as the *Centaur* Study Report. The section on lunar missions and spacecraft contained the conclusion that soft landing a significant scientific payload on the lunar surface could be accomplished with an *Atlas/Centaur* vehicle.

(5) Early in the lunar program, a decision was made by NASA and JPL to rely on industry for the development of lunar spacecraft systems following *Ranger*. In May 1960, consistent with this decision, JPL transmitted bidder's packages to 32 selected firms, soliciting proposals for the design of a lunar soft-landing spacecraft, compatible with *Centaur* size and weight limitations. The bidder's package consisted of Bidder's Instructions and Contract General Provisions; also included was a copy of the JPL lunar-soft-landing design study requirements document, which set forth the requirements, objectives, and constraints pertaining to the spacecraft design.

(6) A bidder's conference attended by 39 firms was held by JPL in May 1960; in June, 24 firms submitted proposals (some representing team efforts) in response. Following a detailed screening of the proposals and management-oriented visits to several of the concerns, JPL's Source Selection Board selected four of the proposing firms as suitable candidates for design study contracts.

(7) In July 1960, partially funded contracts were awarded to Hughes Aircraft Company (HAC), McDonnell Aircraft, North American Aviation, and Space Technology Laboratories for the procurement of preliminary spacecraft system designs and a detailed proposal for program implementation. The projected scope of effort was for the design, development, test, and operation of seven lunar soft-landing spacecraft; the first soft landing was to occur in August 1963.

(8) In December 1960, after 5 mo of study effort and two interim oral presentations by each contractor, the study reports and proposals were submitted to JPL. Evaluation of the proposals covering all program aspects (i.e., technical, managerial, cost, schedule, etc.) was performed by a joint NASA/JPL Source Evaluation Board (SEB) augmented by special JPL technical and management assistance as required.

(9) In January 1961, following a briefing of SEB conclusions, the NASA administrator selected HAC as the spacecraft system contractor. Project management for *Surveyor* was vested in JPL.

2. Design Study

As mentioned previously, the basic objectives that governed the *Surveyor* design study were to:

(1) Soft-land a cargo of scientific instruments on the lunar surface.

(2) Provide for operation of the instruments for at least a 30-day period on the surface (through the light and dark cycle).

(3) Telemeter the scientific data back to earth for retrieval and reduction.

Basic constraints on the spacecraft initial design were:

(1) Total spacecraft vehicle weight and dimensions to be consistent with the expected injection capability (2500 lb) of the *Atlas/Centaur* booster.

(2) Launch from AFETR with specified fixed ascent trajectory characteristics to injection.

(3) Trajectory transit times limited to about 40, 66, or 90 hr to provide view of the spacecraft by Goldstone at arrival and after landing. Probable daylight landings for television operation, expected lunar surface physical and thermal parameters, and the specific list of science instruments provided additional constraints.

Summary conclusions of the HAC design study were that:

(1) The overall mission was feasible within the current technical state of the art and the seven-vehicle schedule specified.

(2) An instrument payload of 315 lb could be landed at touchdown velocities less than 10 ft/sec at any position on the moon, and up to 365 lb under favorable calendaric conditions corresponding to the complete set of listed experiments.

(3) A 90-day operating life on the lunar surface could be achieved.

(4) A maximum data rate of 4400 bits/sec could be achieved.

(5) A nominal 66-hr trajectory represented the best tradeoff relative to performance risks, guidance, and total payload.

(6) The system design would permit landings at practically any point on the western half of the visible face of the moon (that portion illuminated in the last quarter) within an accuracy of 60 km, 3 σ.

The design study report submitted by HAC consisted of three volumes, which described the management plan and the overall *Surveyor* System, including the preliminary design of the spacecraft and the operations necessary to accomplish lunar soft landings during the period August 1963 to August 1965. Detailed contractual and cost data for the proposed program were also provided.

B. Project Implementation and Evolution

1. *Centaur* Definition

Centaur development initially induced major perturbations to the spacecraft development program as originally projected and had a major impact on the *Surveyor* Project as a whole. The sensitivity of the overall *Surveyor* spacecraft design to *Centaur* performance led to the need, early in the project, for discussions with the *Centaur* Project Office at MSFC in order to obtain a better definition of the configuration and probable performance of the "operational" *Atlas/Centaur* vehicle to be developed for *Surveyor*. This was the first formal attempt to distinguish between the configuration and performance of the research and development vehicles, not yet in flight test, and the "operational" *Centaur* expected to evolve.

To provide a better definition of the operational launch vehicle, MSFC initiated a study and preliminary design effort with GD/C under an Air Force contract. Definitive descriptions of the launch vehicle, its performance, and of the nose fairing to be used for the *Surveyor* and *Mariner* Projects were to result. Meanwhile, the absence of adequate and validated launch vehicle performance information necessitated continued dependence on estimates of those parameters that constrained or influenced the design of *Surveyor*, so that the spacecraft effort could proceed.

In the spacecraft preliminary design study, the maximum separated weight of the spacecraft was specified as 2500 lb. This weight, coupled with the dynamic envelope to which the stowed spacecraft was restricted on board *Centaur*, was a major factor in determining the spacecraft configuration and the amount of science payload that could be carried.

Because of the initial uncertainty in the estimates of injection performance that would eventually be achieved by *Centaur*, it was necessary to incorporate an element of flexibility in the spacecraft design. Thus, the spacecraft was configured so that its total weight could be reduced to as low as 2400 lb, if necessary. Conceptually, this was to be accomplished with relatively short lead time by the off-loading of up to 70 lb of selected scientific instruments and the resulting reduction of about 30 lb of main retro propellant.

Other parameters that affected final spacecraft design were the composite boost environment that the spacecraft would experience and the guidance errors that would result at injection. Vibration, temperature, vehicle induced electromagnetic interference, and acoustic levels to be encountered during boost were major factors considered in the design, as well as the design proofing, of the spacecraft hardware. The lunar landing accuracies required and expected errors at injection due to *Centaur* guidance were primary factors in determining nominal time, number, and magnitude of trajectory corrections to be made. These influenced, in turn, such items as the size of the vernier propulsion system tankage, attitude and thrust control techniques, and estimates of propellant utilization.

Because no specific *Atlas/Centaur* flight test data were available on these parameters, it was necessary to extrapolate from previous experience, such as that encountered with the *Atlas/Agena* and used on *Ranger* and to design and to test conservatively. Because of the weight limitation imposed on the spacecraft, there was a clear motivation not to become over-conservative or to overdesign.

Subsequent, periodic appraisal of *Centaur*-predicted performance indicated that the original 2500-lb estimate was optimistic. Studies ensued within the Project to determine configurations for "lighter weight" *Surveyors*, which ranged from 1900 to 2100 lb, without clear indication as to what the final performance within this range might be.

This uncertainty persisted until April 1962, when the decision was made that at least the first several spacecraft would be limited to 2100 lb. At the same time, a *Centaur* upgrade was proposed to permit the launching of the last two of the seven spacecraft in the 2500-lb configuration with the original payload complement.

In January 1963, after *Centaur* Project Management had been transferred to LeRC and a fresh assessment of *Centaur* performance had been made, it was concluded that all of the seven spacecraft should be of the 2100-lb configuration. The reality of a 2500-lb capability on the *Surveyor* schedule projected at that time seemed entirely too speculative. Not until November 1963, when AC-2 was successfully flown, was the 2100-lb capability finally confirmed.

One final effort was made early in 1965 by the project to upgrade the total spacecraft weight in order to increase the spacecraft payload complement to be flown after the first four missions. This followed several successful *Centaur* performances with *Surveyor* dynamic and mass models and the identification of potential improvements to the vehicle that would ensure the 2500-lb capability.

This plan was formally abandoned in the fall of 1965 because of serious difficulties encountered in putting the first flight spacecraft (SC-1) through test and because of the schedule, dictated by *Apollo* Program needs, which called for the completion of all seven *Surveyor* flights by the end of 1967.

2. Spacecraft Development

A letter contract issued by JPL on March 1, 1961, formally initiated the spacecraft development effort. This contract called for eight 2500-lb spacecraft (one designated as a spare) with a science payload of about 340 lb and a midcourse capability of about 30 m/sec. The contract also called for developmental test vehicles, system test equipment assemblies, and command and data consoles necessary for design validation, system tests, and operation of the spacecraft.

Prior to the issuance of this contract and following its issuance, discussions and negotiations were conducted between the JPL and HAC project organizations directed to activities such as:

(1) Review of the HAC proposed spacecraft preliminary design.

(2) Identification and ratification of design changes and associated implementation schedules.

(3) Review and confirmation of major problem milestones.

(4) Review and formalization of HAC/JPL interfaces and relative roles and responsibilities.

(5) Refining the statement of work and development of a governing *Surveyor* spacecraft system specification.

The original system specification and the initial spacecraft development efforts closely followed the HAC study proposal. No changes of programmatic impact that would influence either the initial schedule or dollar projection, were introduced in the early JPL/HAC discussions. Minor design changes were instigated such as the substitution of a Canopus sensor for the earth sensor, originally proposed as one of the inflight attitude reference sensors, and revisions to the terminal guidance sequence to eliminate the use of radar guidance during the main retro burn phase. Uncertainties as to the effect of the main retro exhaust plume on radar performance were the primary motives for the latter change.

The main spacecraft system development effort was centered at HAC under JPL technical direction and with the support of several subcontractors to supply specific spacecraft hardware items. Major subcontracted items were:

(1) Radar altimeter and doppler velocity sensor (RADVS) to be supplied by Ryan Electronics.

(2) Solid propellant main retro engine to be supplied by Thiokol Chemical Corp.

(3) Canopus sensor to be supplied by the Santa Barbara Research Center.

(4) Liquid propellant vernier engines (throttlable over a range of 30 to 100 lb each) to be supplied by Reaction Motors Division (RMD) of Thiokol.

(5) Landing gear shock absorbers to be supplied by the National Waterlift Corp.

In addition to these major items and large-scale component parts procurement, a number of lesser items were also vendor-supplied. These included pressure vessels for the vernier propulsion and attitude control systems, vernier propulsion system valves and gas pressure regulators, silver–zinc batteries, and pyrotechnic-actuated devices, each fabricated to *Surveyor* requirements and specifications. All procurements were in accordance with a formal make-or-buy list generated by HAC and approved by JPL. Major procurements were subject to JPL review and formal approval before final commitment.

Procurement of scientific payload items was somewhat hybrid in that a number of the instruments were to be designed and/or processed by HAC; others were to be government-furnished equipment through the joint collaboration of JPL and the Principal Investigators assigned by NASA to conduct the experiments.

Beyond the level of vendor dependence indicated, total spacecraft system hardware development, manufacture, and test were committed to HAC for execution. This was in accord with the original HAC proposal and with NASA/JPL intent to vest the spacecraft effort, insofar as practical, with a single contractor to ensure an orderly, integrated effort and to centralize the responsibility for the design, manufacture, test, and operation of the spacecraft.

The scope and magnitude of the spacecraft effort, as conducted by HAC are best reflected in the original task descriptions, which collectively defined the effort for which HAC was contractually responsible. In May 1966, those tasks not yet completed were converted to a work item format to provide HAC and JPL with better technical visibility and cost control for the remaining effort. Detailed work assignments and the total work to be completed at that time were not altered greatly, if at all, by the conversion.

The major items of hardware produced by HAC are divided into three main categories: (1) test vehicles manufactured and used in the formulation and/or validation of flight hardware designs (Table IV-1); (2) flight spacecraft; (3) an extensive amount of *Surveyor* unique ground support and operational support equipment required to assemble, test, service, and fly the spacecraft. (A detailed summary of this equipment is provided in Section IX of this report.) The overall scope and complexity of this Project is further disclosed by the design, documentation, quality control, test, operation, and management efforts consigned to the development and application of each of the above items for its intended purpose.

The *Surveyor I* (SC-1) launch on May 30, 1966, its successful lunar landing, and its extended survival on the lunar surface were of individual significance to the *Surveyor* Project in proving the designs of each of the major systems, (Launch Vehicle, Spacecraft, Mission Operations) and their mutual compatibility in a complex mission profile. *Surveyor I* not only validated the *Surveyor* lunar soft-landing concept, but also the spacecraft configuration and subsystem mechanizations, which were closely representative of the original proposed designs.

The original plan projected for spacecraft system development was directed to a first launch in August 1963. This was predicated on the original *Centaur* development schedule and an allowable 2500-lb separated weight capability for the spacecraft. When it became clear, early in the Project, that the 2500-lb capability figure could not be achieved by *Centaur*, a study effort was initiated to determine what changes in spacecraft configuration and payload complement would provide the best "compromise" capability. This became an iterative effort, since a final *Centaur* performance number was not immediately available. Spacecraft weights ranging from 1900 to 2300 lb and a number of possible payload combinations were investigated during this period to be ready for the eventual declaration. With this declaration came the need for formal revisions to the spacecraft development schedule and to the entire *Surveyor* Project plan.

The *Centaur's* failure to meet initial performance objectives was clearly a major contributor to lagging spacecraft development, but progress also was impeded by other factors. It had become evident by this time that several major items of hardware, originally thought to be near off-the-shelf, would require considerable additional development and testing to meet the stringent requirements of *Surveyor*. Outstanding among these were the RADVS and the vernier engines, each a basic flight control element.

Some of the early design and later manufacturing difficulties arose from tight weight restriction on basic spacecraft hardware common to both the 2500- and 2100-lb spacecraft configurations. State-of-the-art improvements to reduce weight were demanded in all areas to meet the weight allocations. The improvements were accomplished, but not without a major expenditure of effort, which was not originally anticipated.

The early requirement for rigorous heat sterilization of *Surveyor* flight hardware where possible in its manufacturing cycle and for the capability to handle and process the hardware thereafter in a near sterile environment also were more severe design constraints than originally expected. Latent surface contamination was to be handled by extended exposure to a sterilizing gas after encapsulation of the spacecraft for launch. The

Table IV-1. *Surveyor* test vehicles

Model designation	Description
M-12	Full-scale engineering mockup of the *Surveyor* spacecraft
MA-1	One-fourth scale spacecraft model for antenna tests. Simulated RF interference silhouette of spacecraft
MA-2	Full-scale spacecraft model for antenna tests. Reflective surfaces similar to those of actual spacecraft
MT-1	Full-scale spacecraft model (built in three separate sections) for thermal tests. Equipped with thermally simulated components for evaluation of thermal control provisions
S-1	Laboratory spaceframe fitted with point-mass simulated components for static and vibration structural tests
S-2	Laboratory spaceframe for vibration, shock, and static structural tests. Employed flight-type, inert retro-rocket motor. Had simulated components and attachments closer to flight type than those of S-1
S-2A	S-2 laboratory spaceframe modified and updated. Used for vibration, shock, static structural, drop, and flight control tests
S-4, S-5	Laboratory spaceframes for vernier propulsion system tests at RMD
S-6	Spaceframe for use in vernier propulsion system tests. Met same specifications as spaceframe for T-1 prototype vehicle
S-7	Spaceframe for testing of vernier propulsion system. Met T-1 spaceframe requirements and had complement of dummy masses (like those on S-2A) simulating spacecraft components
S-8	Laboratory spaceframe intended for flight control/propulsion tethered (buzz) tests. The requirement for this vehicle was deleted and S-8 was
S-8 (contd)	later designated S-10, the A-21 thermal control model
S-9	Structural test model, previously designated STM-62, was composed of a spaceframe and substructure of the A-21 configuration. Many units and nonfunctioning prototype units from S-2A were retained for S-9
S-10	Thermal control model for the A-21 configuration, previously designated S-8. A full-scale spacecraft model for thermal evaluation
S-15	Basic T-1 vehicle upgraded to meet the requirements for the T-1/62 drop test program. Minor changes were made to the spaceframe and ballast racks to accommodate the new gross weight. New flight-quality landing gear, crushable blocks footpads, and instrumentation units were installed
SD-1 through SD-4	Spacecraft dynamic models for use in *Centaur* tests
T-1	Dynamic stability test vehicle, for landing gear drop tests under simulated lunar gravity, spacecraft/*Centaur* separation tests, and retro motor separation tests
T-2	Test vehicle for descent dynamic test program terminated in October 1964
T-2N-1, T-2N-2	Test vehicles for descent dynamics test, which included operation of the vernier propulsion system and RADVS in drops from a tethered balloon
T-2N-R	Test vehicle for evaluating performance of the recovery system for the T-2N vehicle
T-21	Prototype 2100 lb spacecraft of flight (A-21) configuration for system type approval and mission simulation tests
X-1, X-2	RADVS development models supplied to HAC by Ryan Aeronautical Company

requirement to sterilize the *Surveyor* spacecraft was eventually removed by NASA as being unnecessary and unrealistic, but not before a significant amount of effort, and attendant cost in time and dollars, had been expended in attempting to meet the requirements.

As the spacecraft development proceeded, difficulties were encountered in manufacturing and proofing, and, consequently, in the delivering of flight hardware of the quality necessary to hold spacecraft assembly and test schedules. Problems developed with the T-2 and T-21 test programs. T-2 testing was directed to the validation of the spacecraft terminal-descent system; T-21, as the type approval unit, was the first flight-configured spacecraft to enter ambient and environmental testing.

Finally, in September 1964, in view of these collective difficulties and others pending in the processing of SC-1, the decision was made that JPL should significantly increase its technical involvement in the spacecraft effort to augment the HAC management and technical team. In the months that followed, JPL and HAC together sought to correct the difficulties existing in various hardware development areas. Organizational changes were made to strengthen design, test, and manufacturing teams and to bring them under tighter project control. In this process, hardware assembly procedures, test specifications and procedures at unit and system levels, test equipment, and performance analysis procedures underwent joint critical examination and, as necessary, upgrading.

This technical and management enriched environment provided the background for the successful conclusion to the T-2N test program at Holloman Air Force Base, demonstrating, as a mandatory prerequisite to the *Surveyor I* launch, the validity of the terminal descent system design. It resulted in the installation of real-time spacecraft checkout computer facilities (SCCF) at HAC, El Segundo, and AFETR to support spacecraft test and launch operations and in the later application of the SCCF in flight operations support. This environment produced the periodic, rigorous evaluation of each spacecraft at various stages of testing, formalized consent-to-ship and consent-to-launch reviews, improved configuration control techniques and improved failure reporting and failure analysis. Finally this environment prevailed through the final stages of testing, launching of *Surveyor I*, and the processing of subsequent spacecraft.

3. Mission Operations Development

The requirements for the *Surveyor* mission, as given to the study contractors, indicated that JPL would furnish the Deep Space Network, Central Computing Facility, tracking data evaluation, and would supply acquisition data to the tracking stations. HAC, who became the successful bidder, recommended in their proposal that mission operations should include both launch operations and the lunar operations, and that mission operations should be the responsibility of the Spacecraft Mission Operations Department, composed of three sections, as follows:

(1) The Launch Operations Section would operate the HAC facility at Cape Canaveral (now Cape Kennedy), check out spacecraft, perform on-stand spacecraft prelaunch operations, control and monitor spacecraft performance during the countdown, and prepare the final launch operations report. Operationally, this section would support the firing director, who, in turn, would report to the lunar mission director (JPL).

(2) The Transit and Lunar Operations Section would design, implement, install, and test the command and data handling consoles to be installed at the Deep Space Stations; prepare the detailed specifications and procedures required for flight and for lunar operations; supervise all operations at each station when carrying out *Surveyor* operations; and reduce and analyze engineering and science results.

(3) The Mission Planning and Control Section would coordinate the activities of the launch vehicle contractors, the spacecraft contractor, the AFETR, and the Deep Space Stations under the general direction of JPL. They would also be responsible for ensuring complete and timely documentation for launch operations and lunar operations.

In early 1961, JPL indicated that a data, operation, and control facility would be provided at JPL to be used as a centralized facility for computation and postinjection mission control. This became part of the spacecraft development plan in June 1961. This centralized control concept was continued to the end of the project.

In late 1962, a Space Flight Operations Group, composed of JPL and HAC personnel, was organized to produce the Space Flight Operations Plan. The Space Flight Operations Facility (SFOF, formerly the data, operation, and control facility) was also formed, and its planned existence was reflected in the *Surveyor* Spacecraft System Design Specification as revised in December 1962.

In early 1963, JPL appointed managers for the Mission Operations and Spacecraft Systems. About this time, the HAC spacecraft development plan was modified to separate launch operations from mission operations and to reflect the change in prime responsibility for mission operations from HAC to JPL. A short time later, JPL appointed a Space Flight Operations Director.

The Space Flight Operations Group continued in existence until December 1963, providing advice and assistance to the Space Flight Operations Director and to the Mission Operations System Manager. At that time the first design review of space flight operations was held. This review indicated awareness of the complexity of the operation, and resulted in an endorsement of the centralized control concept, and a recommendation for simplifying the operations by incorporating some "on-board" automatic sequencing on the spacecraft, such as automatic sun acquisition. Recommendations for improved documentation of mission objectives and tradeoffs, of spacecraft operations interfaces, and of responsibilities and schedule were also included.

A second design review was held in August 1964. All recommendations of the first review were either completed or well underway. The second review also disclosed the need for a Project Policy Document for Mission Operations, a DSN commitment document, including test support, a firm space flight operations schedule, a defined list of constraints and a computer programming plan and schedule, including required manpower. The centralized control concept again was endorsed, with the recommendation that nonstandard procedures be established as a backup, in the event communications with the SFOF became inadequate. Other minor improvement recommendations were made. All of these recommendations were regarding details of implementation, not concept. In general, they argued for increased rigor and definition.

In August 1965, the Space Flight Operations Group was actively training and testing in preparation for a launch late in the calendar year. During a functional compatibility test, some problems and deficiencies, relative to commanding, were noted in computer programs. Since all manual modes of commanding worked well, a decision was made by the Mission Operation System Manager to delete, as *Surveyor I* mission requirements, the command tape search mode of the on-site data processing system, the ability to produce, transmit, confirm, retransmit, and verify command tapes during space flight operations, to perform command confirmation except by voice or teletype, and the power/thermal program. Command tapes were pre-prepared and mailed to the Deep Space Stations. Special tests and alignments verified and ensured adequate reliability of the tape readers on-site. Efforts continued to restore these capabilities for the *Surveyor II* mission. Further changes in the launch schedule made some of these capabilities available and certified in time for the *Surveyor I* mission.

Before the *Surveyor I* mission, a spacecraft analysis team (SCAT) was organized to augment *Surveyor* spacecraft performance analysis and command (SPAC) operations. This team, following an operational pattern used on *Ranger*, was composed of selected spacecraft design and test team personnel and was to provide in-depth, real-time evaluation of spacecraft performance and to report to SPAC any anomalous trends or failures. The formation and somewhat informal chartering of this initial SCAT activity was the first of a series of steps taken by the Project during the flight program to develop closer working relationships between operations and design team personnel to ensure that operations during flight would be adequately supported by design personnel who could help to solve possible inflight difficulties.

For the *Surveyor I* and *II* missions, SCAT activities were conducted outside the SFOF using flight data processed by the SCCF and with necessary information exchanged by telephone. Following the difficulties encountered with *Surveyor II* in flight, the SCAT activity was strengthened and formally incorporated as an integral part of space flight operations. Renamed the Trend and Failure Analysis Group, it was installed in the SFOF in an area adjacent to SPAC and placed under SPAC direction to ensure its optimum utilization. Provisions were also made for more extensive use of the *Surveyor* SCCF for flight data processing and display as an alternate and/or backup data source for the Trend and Failure Analysis Group and SPAC support during flight and lunar operations.

These actions, together with other changes to the mission operations organization, were accomplished before *Surveyor III* mission to develop maximum operational proficiency. They were consistent with the extensive attention given to the spacecraft (SC-3) in preparing it for launch.

The high level of proficiency developed by operations personnel was displayed repeatedly in *Surveyor* flight and lunar operations. This is seen in the near "on-target" landing of *Surveyor V* after failure of a helium regulator necessitated repeated midcourse corrections and a redesign of the terminal-descent profile. A wide range of operations management techniques was developed. These techniques were applied to spacecraft operations on the lunar surface to preserve the spacecraft and its instruments in order to ensure the return of the maximum amount of useful data from each mission. These techniques were instrumental in obtaining the large quantity of data from *Surveyor III* after the spacecraft was damaged during landing and almost all spacecraft telemetry had been lost. The "keep-alive" action directed to *Surveyor VII* for an extended period during the second lunar day was instrumental in obtaining additional data from the alpha scattering experiment, considered of prime importance to the *Surveyor VII* mission.

The efforts required to develop and maintain a high level of competence within the Mission Operations System and the Tracking and Data System were no less demanding or rewarding throughout the Project than those directed to achieving the excellence of performance demonstrated by the launch vehicles and spacecraft. Each major system represents the development and demonstration of important advances in space technology.

V. Project Organization and Management Controls

A. Management Structure

The organization for the *Surveyor* Project (Fig. V-1) was developed in accordance with NASA's General Management Instructions 4-1-1, and consisted of the Project Manager and his administrative and technical staff with System Managers designated for the four major elements of the Project, i.e.:

(1) Launch Vehicle System.

(2) Spacecraft System.

(3) Mission Operations System (MOS).

(4) Tracking and Data System (T&DS).

In designating the responsibility along system management lines, the Spacecraft, Tracking and Data, and Mission Operations Systems were assigned to JPL. Launch Vehicle System management was originally assigned to MSFC, but was transferred to LeRC in October 1962.

The responsibilities of the *Surveyor* System Managers are given in Fig. V-2. Within the framework of the Project, the accomplishment of these tasks remained the overall responsibility of the Project Manager and his parent organization. However, the detailed execution of, and the day-to-day support for, system management effort was the responsibility of the designated system manager and his parent organization.

Additional JPL functional elements of the Project (Fig. V-3) established for support to Project Management, were:

(1) Mission Analysis and Engineering.

(2) Reliability and Quality Assurance.

(3) Project Science.

(4) Project Control and Administration.

(5) Contracts Admnistration.

The Project Science organization consisted of a Project Scientist, who reported directly to the Project Manager, and a group of Principal Investigators, selected by NASA and JPL to represent each scientific experiment on the spacecraft. The work of this Science Team, in summary, was to participate in defining the science experiments and instruments for *Surveyor*, to assist in instrument development and validation, to participate in operations, and to evaluate data and report results. The formal definition of, and commitment to these efforts was through written contract between JPL and the Principal Investigators designated.

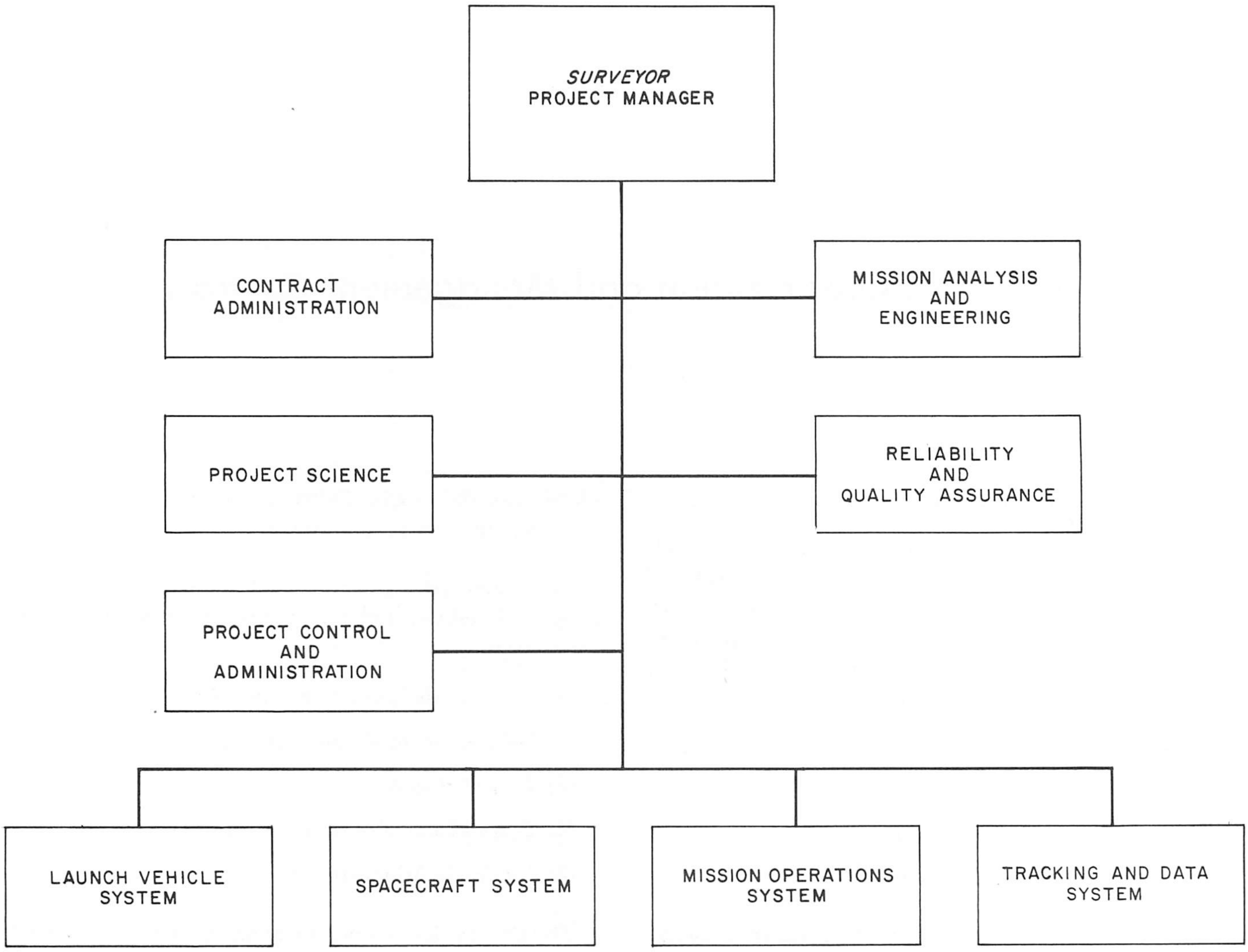


Fig. V-1. *Surveyor* Project organization

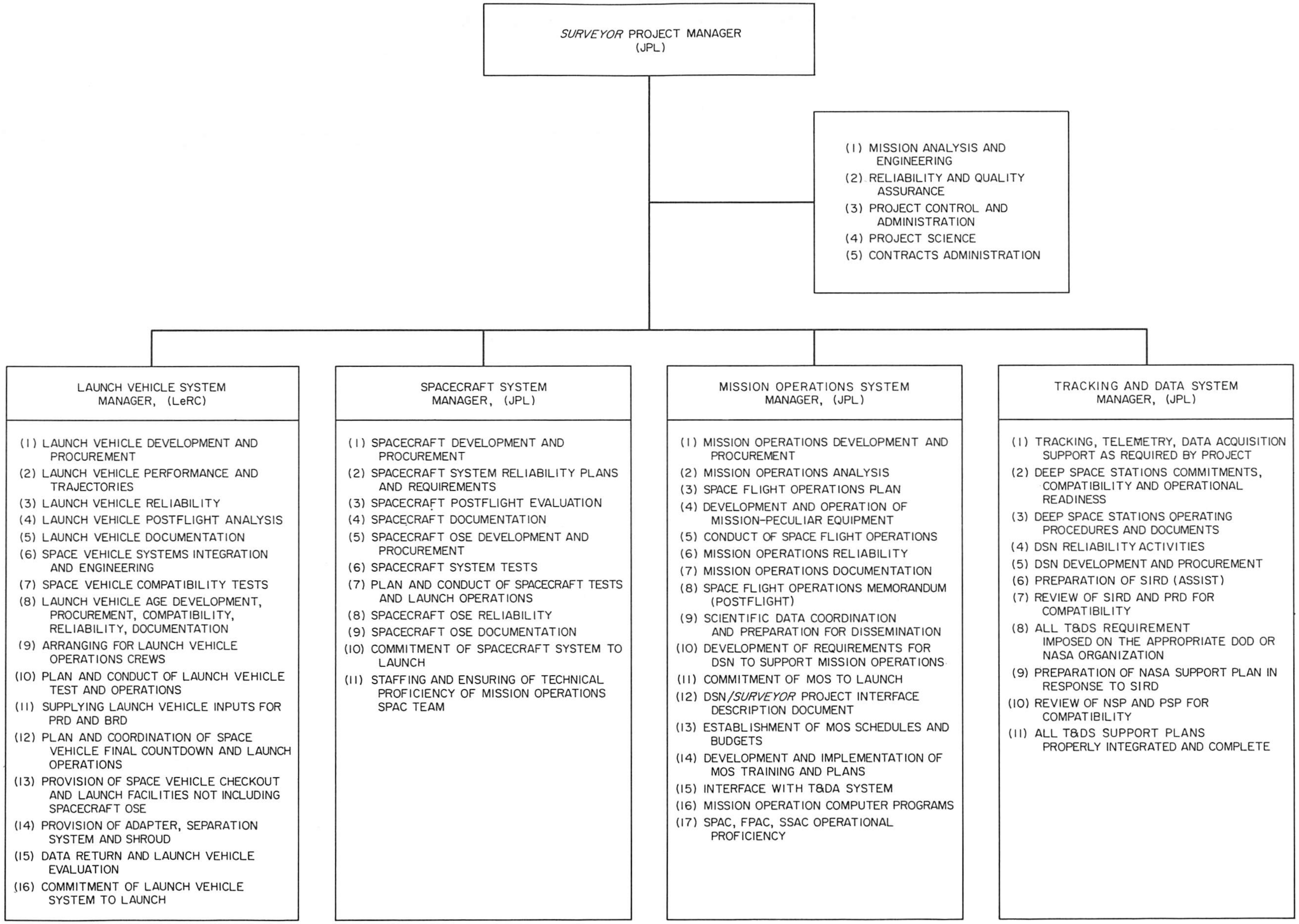


Fig. V-2. Responsibilities of *Surveyor* System managers

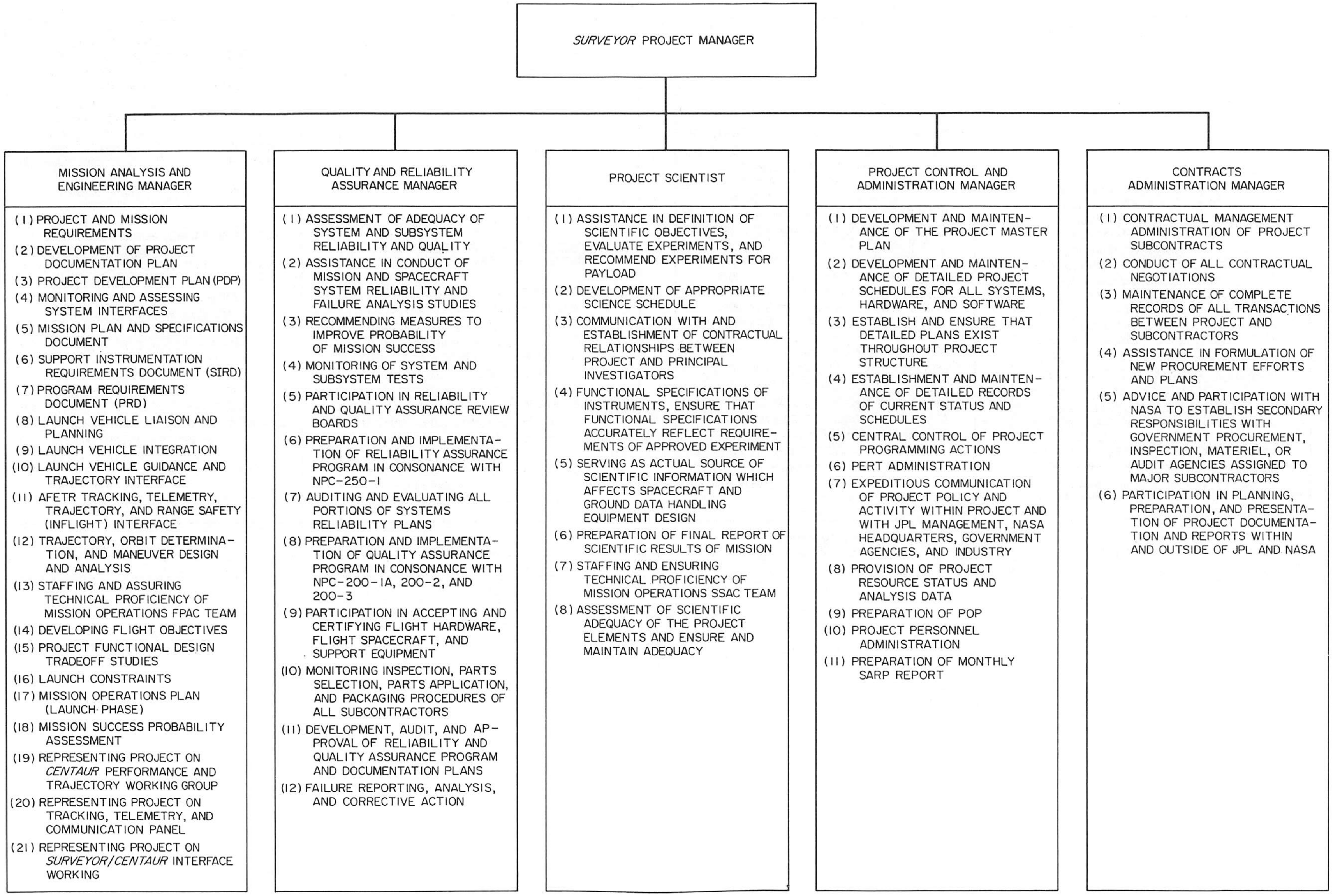


Fig. V-3. Responsibilities of *Surveyor* Project staff

The *Centaur* Project Office (Fig. V-4) at LeRC provided the Launch Vehicle System Manager, reporting in a line sense to his center director. Within JPL, the T&DS Manager reported to the Assistant Laboratory Director for Tracking and Data Acquisition (T&DA), and the Spacecraft and Mission Operations Systems Managers were organized integral to the *Surveyor* Project and reported directly to the Project Manager. Each JPL system manager had appropriate staffing to administer to the on-going effort, but drew supplemental technical support from the JPL technical organizations, as required.

Organizationally, LeRC maintained primary responsibility over the launch vehicle activity, which included management of the contracts with the vehicle contractor, GD/C.

The Launch Vehicle System effort encompassed all facets of the procurement of the launch vehicle. This work included the manufacturing, modifying, and testing of the *Atlas* vehicle, development and testing of the *Centaur*, and the launching of the total vehicle.

Close coordination was maintained with the *Surveyor* Project Manager concerning schedule and performance. Targeting considerations and launch window requirements and constraints were under constant surveillance to ensure maximum launch opportunity and success.

Finally, functional and operational compatibility between the Spacecraft and Launch Vehicle Systems was of prime concern. The combined systems test (CST) between launch vehicle and spacecraft was instigated to provide control over this interface. Initial assembly was, therefore, scheduled at San Diego to permit confirmation of compatibility between the systems before shipment to AFETR.

The management of the Spacecraft System was the responsibility of JPL; at the start of the Project, HAC was selected to provide the spacecraft and various support effort. In the early part of the Project, a similarity between the JPL and HAC organizations was apparent; JPL personnel in a certain technical area worked directly with their HAC counterparts. Thus, a specialist in thermal control from JPL worked closely with the HAC thermal group to understand the problems and proposed solutions, to assess the response to such problems, and, if desirable, to proffer assistance.

JPL management of the Spacecraft System was established with seven sections, each identified with a unique technology (Spacecraft System, System Test and Launch Operations, Power and Guidance, Instrument Development, Telecommunications, Mechanics, and Propulsion; see Fig. V-5). Each major technical section was divided into various groups of a more specialized nature such as in Telecommunications, which included RF and Radar as one group and Command and Data Processing as another group. These technical sections and their respective groups were organized from personnel or technical divisions within JPL. Special support effort was obtained from these technical divisions, as required by the Project.

The HAC support was derived within the HAC organization from the various technical elements (Fig. V-6), reported to the HAC *Surveyor* Program Office. The initial design and engineering effort was coordinated between the Project and various technical disciplines that carried over into the manufacturing divisions as well. Therefore, specific technical disciplines including Propulsion, Flight Control, Electrical or Telecommunications, Mechanics and Payload, are again noted. The Guidance and Trajectory Analysis, and Space Flight Operations Department, which were additional tasks under the HAC contracts, are shown. These were matched organizationally at JPL (Figs. V-5 and V-7).

Although the discussion thus far has stressed technical organizations within the HAC structure, other support activities were also part of the effort. Thus, all activity required was basically contained in, or was at least controlled from, the Project.

A comprehensive manufacturing effort was required for the *Surveyor* which was not large in quantity but rather of a complex nature and which extended over a considerable period of time. The Project established strict controls with the manufacturing division, which provided direct access for monitoring and supervision of the *Surveyor* work. Thus, checks and balances were maintained on hardware performance, quality, schedule, and cost.

In addition to the engineering and manufacturing of the spacecraft, there was also a HAC effort on the design and manufacturing of ground support equipment (GSE), and a significant effort on system testing and launch operations support of the spacecraft, all under the Test and Operations Laboratory. HAC mission support for the transit and lunar operations phases was

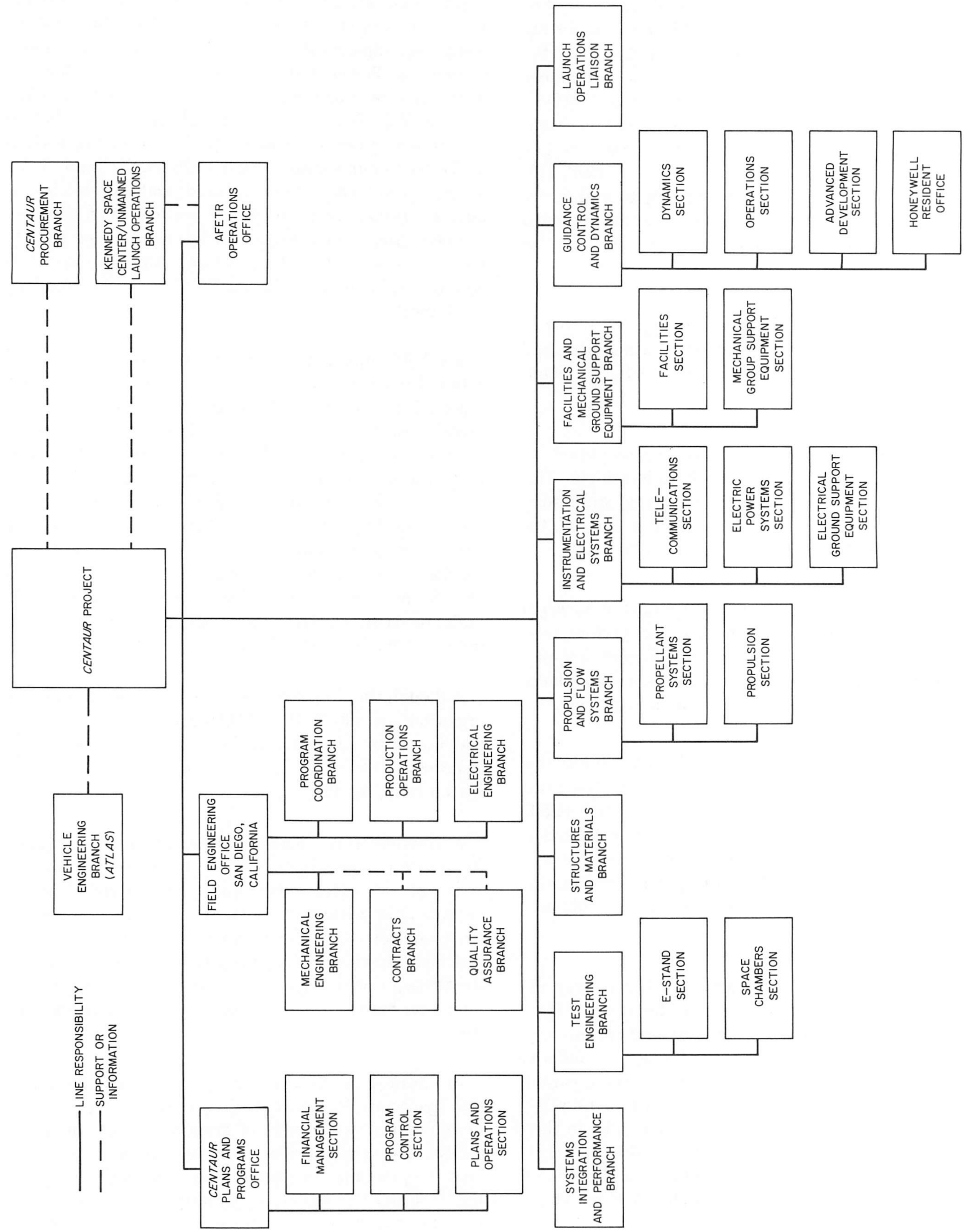


Fig. V-4. *Centaur* Project organization

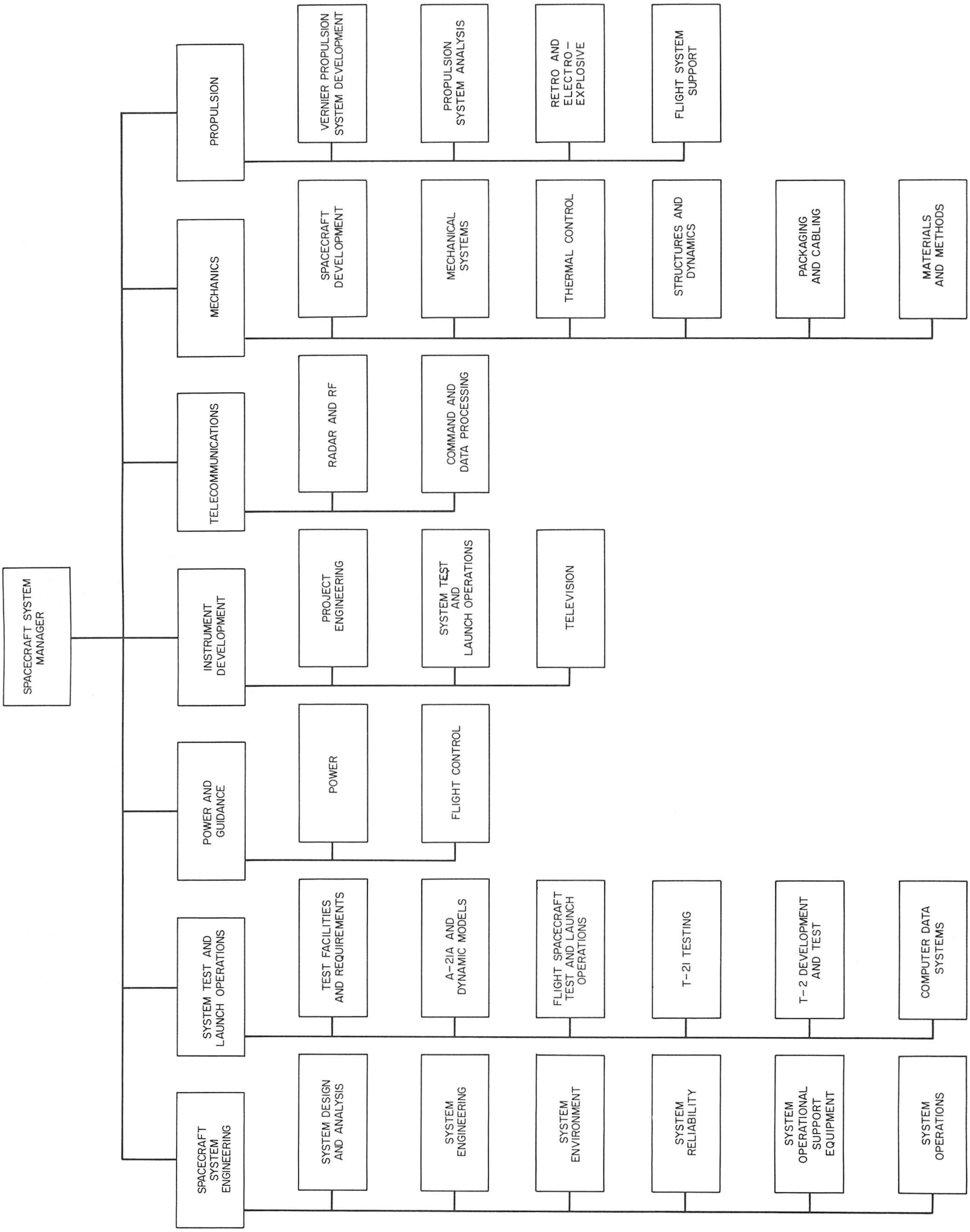
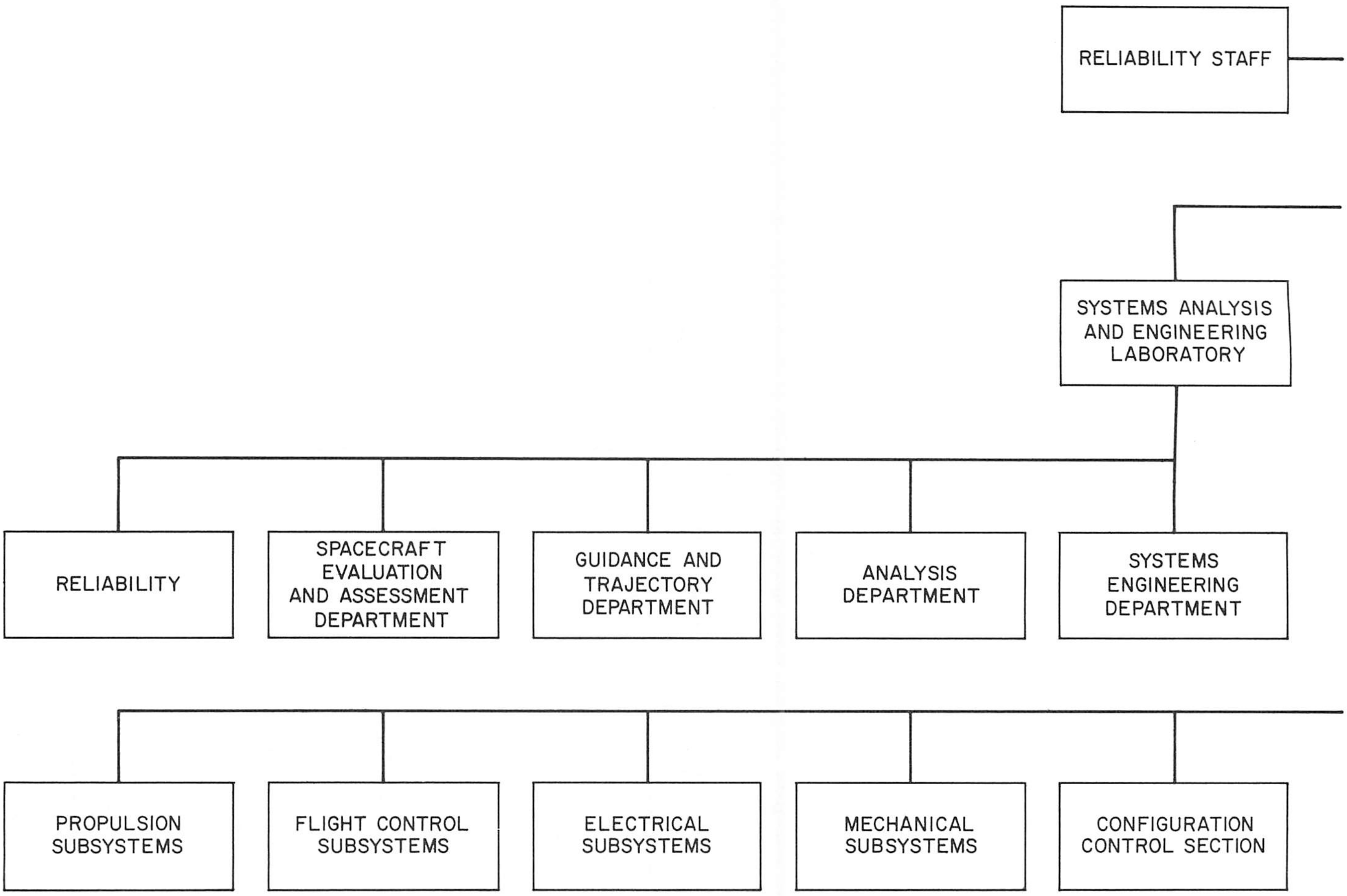


Fig. V-5. Spacecraft System organization


RELIABILITY STAFF
SYSTEMS ANALYSIS AND ENGINEERING LABORATORY
RELIABILITY
SPACECRAFT EVALUATION AND ASSESSMENT DEPARTMENT
GUIDANCE AND TRAJECTORY DEPARTMENT
ANALYSIS DEPARTMENT
SYSTEMS ENGINEERING DEPARTMENT
PROPULSION SUBSYSTEMS
FLIGHT CONTROL SUBSYSTEMS
ELECTRICAL SUBSYSTEMS
MECHANICAL SUBSYSTEMS
CONFIGURATION CONTROL SECTION

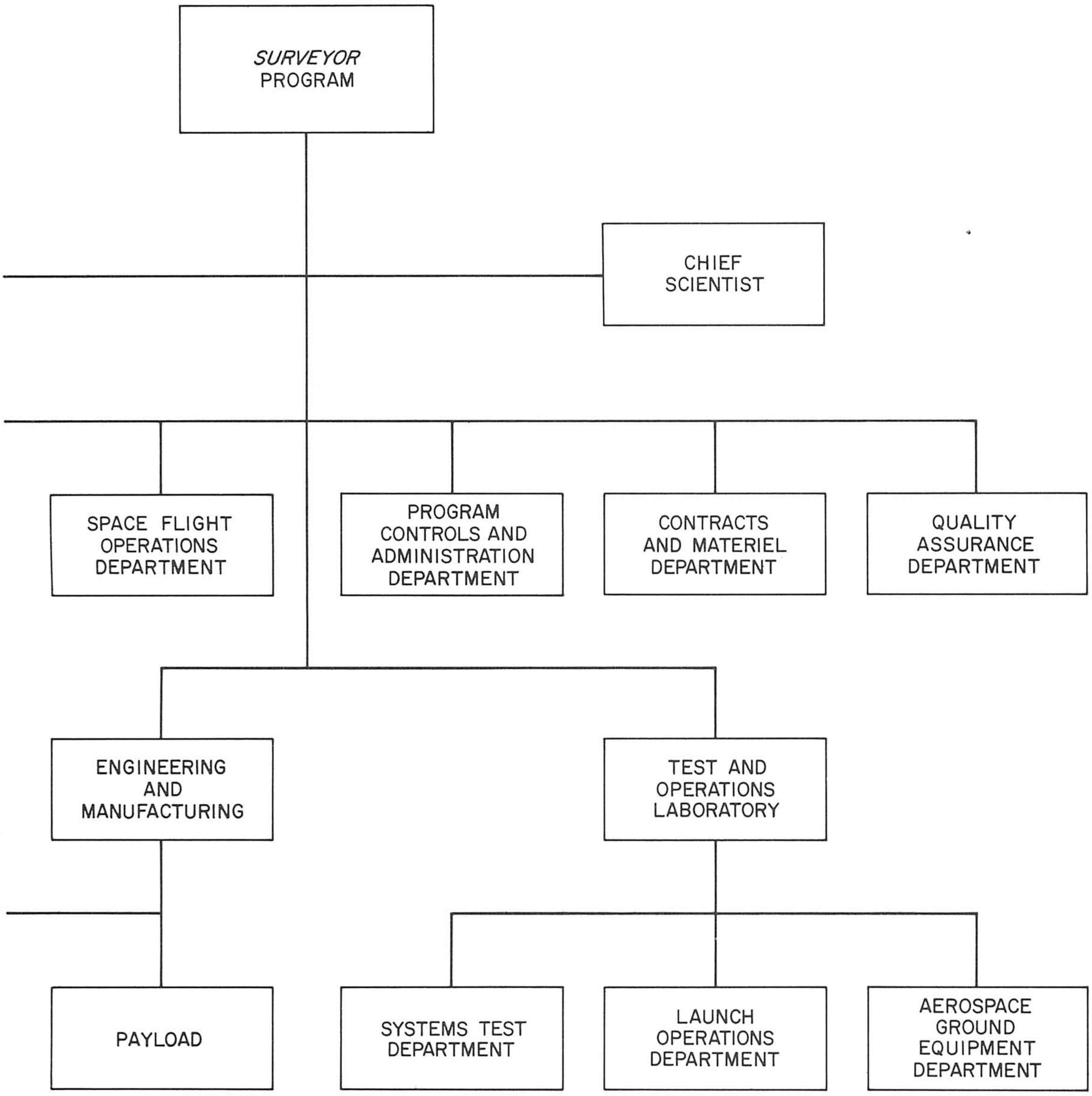


Fig. V-6. HAC program organization

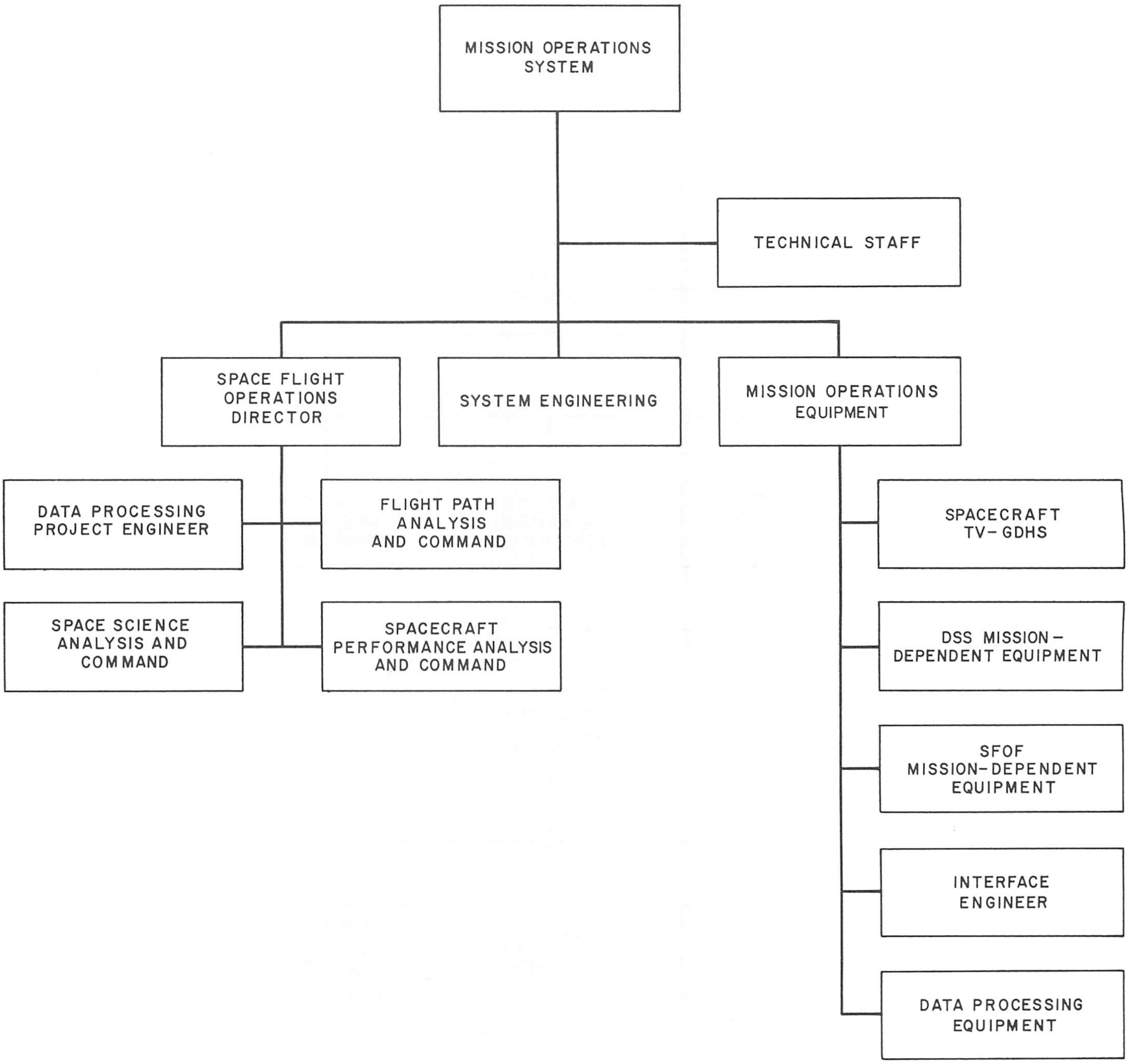


Fig. V-7. MOS organization

conducted under the *Surveyor* Program Space Flight Operations Department. This was primarily a team organized and trained to operate the spacecraft throughout its space flight period and during the lunar operations.

The JPL MOS was organized to prepare the operation system configuration for each *Surveyor* mission, which included the determination of mission-peculiar equipment, the DSN support requirements, the data processing, and other activities related to flight operations. As noted in Fig. V-7, considerable support was provided for the development, procurement, and validation of mission-peculiar equipment.

The JPL T&DS (Fig. V-8) was organized to support the *Surveyor* Project. This included the preparation and coordination of the DSN, AFETR, and other NASA facilities. The formation of the SFOF was also performed; this activity drew heavily from the JPL/DSN organization.

B. Project Controls

1. Contractual Aspects

Aside from the HAC *Surveyor* systems contract, all of the major JPL contracts on the *Surveyor* Project incorporated provisions and controls, which were basically structured for government and JPL contracting practices. However, several contractual aspects of this contract related to the implementation and application of important project controls into a major systems contract.

Because of the complex and changing nature of the Project, the HAC *Surveyor* contract had many unique contractual aspects. Since the Project environment was characterized by changes in such things as mission objectives, spacecraft design, and schedules, it is not surprising that numerous significant contractual changes were necessary through the life of the contract. The type of contract with HAC was cost-plus-fixed-fee (CPFF) from March 1961 until June 1966. Early in 1966, it was apparent that some contract type other than CPFF might provide the incentive and contractual relationship necessary to better accommodate this unusual environment. Thus, in June 1966, the contract was converted to a combination cost-plus-incentive-fee and award fee (CPIF/AF).

The basic concept of the CPIF portion of the contract was a relationship between the postlaunch performance of each *Surveyor* spacecraft and final costs incurred under the contract. A base fee was established to which the fee for postlaunch spacecraft performance was added. Points valued at a dollar amount set forth in the contract were achievable during each mission for such events as landing accuracy (midcourse maneuvers), touchdown dynamics (strain-gage data at touchdown), and operation of the television camera, surface sampler, and alpha scattering instrument. Table V-1 shows the points assigned to the incentive criteria for each mission. At the completion of the contract, 5% of contract costs incurred in excess of the target cost set forth in the contract was subtracted from the fee earned for spacecraft performance. Of the total of 710 points achievable to the

Table V-1. JPL/HAC *Surveyor* contract performance incentive criteria

Event	*Surveyor I*	*Surveyor II*	*Surveyor III*	*Surveyor IV*	*Surveyor V*	*Surveyor VI*	*Surveyor VII*
Midcourse (landing accuracy)	25	20	10	10	10	10	25
Terminal phase	25						10
Touchdown dynamics	25	30	8	8	7	7	5
Soft landing			20	20	20	20	15
Footpad picture	25	50	10	10	10	10	5
Panscan television			40	40	53	53	40
Surface sampler			[10][a]	[10][a]			10
Alpha scattering					[13][a]	[13][a]	[13][a]
Other postlanding operations			12	12			
Total points	100	100	100	100	100	100	110
Points earned	100	100	100	10	100	100	110

[a][ ] indicates points could be earned in lieu of panscan television.

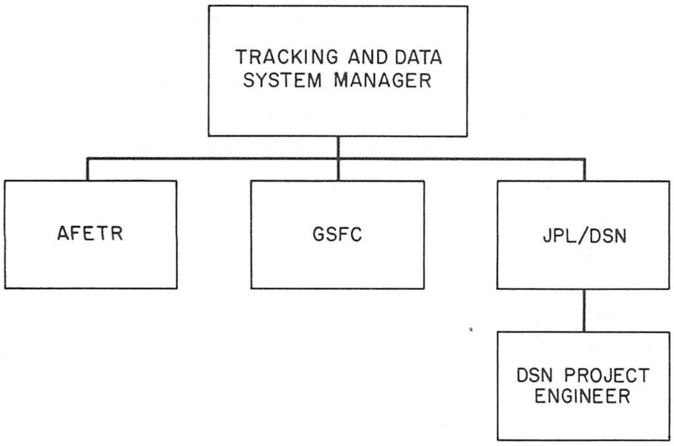


Fig. V-8. T&DS organization

contractor, 520 were earned by the contractor for performance during the seven missions.

The award fee portion of the contract called for $3 million to be achievable by the contractor. Of this amount, $2 million was allocated to, and divided among, six 3-mo periods beginning in July 1966. The balance of $1 million was set aside for the contractor's overall performance during the period from July 1966 through contract completion. Award criteria were established and submitted to the contractor before the start of each period. The dollar budget amounts for the individual criteria, however were withheld from the contractor. At the end of each period, the contractor's performance was assessed by the Laboratory, and this assessment plus all or a part of the achievable fee was submitted in writing to the contractor.

The criteria established before each award period were divided into five main categories: (1) management, (2) engineering, (3) manufacturing, (4) test and operations, and (5) support equipment. Although the individual items were varied somewhat, depending on which areas of performance were particularly applicable during any individual period, the following list of criteria represents the criteria established for the contractor's overall performance and is generally typical of the criteria for each of the six 3-mo individual award periods:

Management.

(1) Continued and consistent interest, awareness, objectivity, and dynamic response to technical and administrative problems at corporate and Project management levels.

(2) Overall skill and integrity in applying manpower in consonance with technical and administrative requirements, problems, and resources.

(3) Effectiveness in distributing and manipulating manpower skills specifically related to transferring to and using Project experience in current and future operations.

(4) Overall consistency, objectivity, thoroughness and sufficiency of the consent-to-ship and consent-to-launch efforts.

(5) Consistency, objectivity, thoroughness and dynamics of Trouble/Failure Report (T/FR) closure effort.

(6) Development and sponsorship of reliable and articulate cost and performance data, with timely and appropriate management reaction to such data.

(7) Effectiveness in handling and disposition of project material and property.

Engineering.

(1) Completeness and integrity of a configuration management system specifically related to the ability to ascertain a total and reliable description of spacecraft design, manufacture, test, and operation.

(2) Completeness, objectivity, ingenuity, and integrity of engineering efforts including problem solving related to manufacturing, test, and operation.

Manufacturing.

Effectiveness of overall manufacturing performance including the establishment, maintenance, and performance of uniform manufacturing, handling, and rework procedures.

Test and operations.

(1) Recognition of, respect for, and appropriate response to the interface relationships between the various organizations participating in the Project.

(2) Concern and effectiveness in establishing, maintaining, and performing prelaunch test and operating procedures.

(3) Sponsorship of and cooperation in furnishing data and tests to overcome or ameliorate known or suspected violations of configuration and testing integrity.

(4) Preparedness for, and proficiency in, performing mission operations, including effectivity of handling nonstandard mission requirements.

Support equipment.

Timely maintenance of support equipment compatible with spacecraft testing and mission operation requirements, which provides a consistent and appropriate level of quality and capability.

The fee available for each individual period was divided into discrete fixed allocations for each specific criterion established. The allocations were established before the start of each period, were approved by the Director or Deputy Director of the Laboratory, and were disclosed to only four or five members of JPL management. At the end of each period, when the contractor's performance had been assessed in each category of criteria, the allocations for each criterion were disclosed to the contractor together with the fee he had been awarded in each. Concerted effort was made by JPL to prevent performance in any one category of criteria from influencing the award in other categories of award criteria. Therefore the contractor's performance in each criterion was, in effect, held independent of his performance in the other areas of criteria.

A review of the contractor's performance in the award area was conducted by two review boards. A Performance Evaluation Board was selected from the senior technical and administrative personnel who were to be directly involved in the *Surveyor* Project activities. The members of this board were required to submit written reports describing their personal assessment of the contractor's performance compared with the award criteria set forth.

A Final Review Board was composed of the *Surveyor* Project Manager, a Deputy Project Manager, and the *Surveyor* Contracts Manager. This board:

(1) Generated the criteria and the dollar allocations for the criteria in each period.

(2) Obtained the approval of the criteria from the JPL management.

(3) Presented the criteria in writing to the contractor's management prior to the beginning of each period.

(4) Designated and instructed the members of the Performance Evaluation Board prior to each award period.

(5) Solicited, evaluated, and reviewed the written assessment reports from the members of the Performance Evaluation Board.

(6) Developed a final, integrated, and balanced assessment of the contractor's performance in each category of award criteria. This assessment was translated into a letter which was submitted to the contractor after approval of that assessment and fee awarded as granted by JPL management.

The incentive provisions of the HAC contract described appeared to provide an important aspect of management control on the Project. Two other aspects, the JPL right of technical direction and method of implementing control documentation, were also included in the HAC *Surveyor* contract, and are described in the following paragraphs.

2. Technical Direction

Because of the need to transfuse JPL knowledge and experience into the detailed design, manufacture, test, and operation of the spacecraft, the contract provided JPL with the right of technical surveillance and direction over the contractor. JPL gave such technical direction through the use of an action directive system. The purpose of this system was to establish a controlled administrative means by which JPL, within its contractual rights of technical direction under the terms of the contract, would unilaterally cause HAC to take a specific action as part of its performance under the contract. Each directive specifically described the actions required of the contractor by JPL, the reasons why the actions were necessary, and why they were considered already covered by contract requirements. The reasons stated later served as the basis for negotiations with the contractor in the event he felt the actions were outside the contract coverage. These action directives also served as the formal JPL documentary record of technical direction to HAC.

Any individual at JPL was empowered to initiate a request for action, but such action was normally initiated by a cognizant *Surveyor* Project engineer. The originator prepared a request-for-action form; upon proper completion and approval by Project management, it was converted to an action directive. Approval by Project management personnel included an assessment of the effect of the directive on other technical areas, and whether or not a formal contract modification should be issued in lieu of an action directive. The initiator delivered the fully executed action directive to the *Surveyor* Contract Office, and it was sent to HAC.

In addition to the regular action directive procedures, emergency action directives were issued to expedite action on problems that occurred during extremely important on-going operations such as system level testing. These directives were issued primarily to limit the necessity of repeating tests if results were marginal or suspect, to change the order of planned tests to accommodate schedules, or to change test requirements, setups, or methods provided that there was no equipment or major schedule change.

When the contractor received an action directive, he would sign it, begin work immediately or notify JPL that the work required was, in his opinion, outside the coverage of the contract and that a contract change would be required before the work would be started. JPL would then automatically issue the requested contractual change and, in the transmittal of such change, would clearly state that the change was issued in lieu of an action directive and that any adjustment to the fee or target cost was subject to later negotiation. The contractor was required to include justification for such adjustment in his proposal submitted in response to the change.

The method of directing the contractor in the manner described above proved to be workable and effective, and overcame the singular problem always inherent in directing a contractor in a contractual environment, that of the delaying of work being directed until a negotiated settlement is effected with relation to whether or not the cost of the change is fee bearing. The method used by the *Surveyor* Project enabled the work to be continued and placed the negotiations of the parties at an appropriate and convenient time.

3. Control Documentation

To establish a basic agreement between the contractor and JPL as to minimum requirements for such key areas as spacecraft configuration, performance requirements, the content of spacecraft test plans, mission operations requirements, and equipment interface requirements, certain key specifications were designated in the incentive contract as controlling documents. These particular documents were elevated to control status such that they could be revised only by contract modification. JPL agreed that any requirements over and above those specified in the controlling documents would be added by formal contract modification. It should be noted, however, that since the contractor was responsible for all performance under the contract, these documents represented only the minimum JPL requirements and any efforts necessary, above and beyond these controlling documents, to meet performance requirements would be at the discretion of the contractor. Such additional effort was carefully monitored by, and required the approval of, JPL technical personnel.

The actual method used for making changes in the controlling documents was for such changes to be handled by the Change Control Board, which is described in Subsection B-4 of this report.

4. Configuration Management

The term "configuration management" refers to identification, accounting, and control of spacecraft, aerospace ground equipment (AGE), operation support equipment (OSE), and functional model hardware and their spare parts, plus the software that supports them. Another major element of configuration management is the development and retention of test procedures, test configuration, and test results. As the basic tool by which configuration of any given piece of hardware is known at any given time, configuration management is concerned with change control and those documents that reflect the as-designed vs the as-built hardware configuration. It permits clear visibility of individual equipment and spacecraft on a day-to-day basis and at key points for interim design reviews, consent-to-ship and consent-to-launch meetings. It provides for disclosure and more timely resolution of discrepancies that could constitute liens against the support equipment, spacecraft or software, and thus endanger flight schedules or performance.

a. Configuration change control. Change control served a dual functional role in *Surveyor* Project Management. Although it provided for control of specifications and contractually controlling documents, an additional function was to provide a control system for procedures, lower tier specifications, and engineer drawings.

During the critical period before the launching of *Surveyor I*, senior management focused attention on the various changes proposed for the spacecraft. The joint JPL/HAC Change Control Board met biweekly and was chaired jointly by the JPL Spacecraft System Manager and the HAC Program Manager, who, in turn, were supported by their technical managers, engineers, and contract people. All changes were reviewed for technical justification and for impact on the Project. When the spacecraft were successfully flown and spacecraft design and manufacture proved, the change process became more routine and the Change Control Board was chaired by the manager of the HAC System Analysis and Engineering Laboratory and the JPL Spacecraft System Manager, with participation by senior management, as required.

An internal HAC Change Control Board was also in effect. The *Surveyor* Change Control Board Manager, as the HAC director of engineering change operation, was responsible for the day-to-day function of review and evaluation of change proposals. He was the final authority in recommending changes to JPL via the JPL Change Control Board representative, and the joint JPL/HAC Change Control Board. Because of the criticality of change control within configuration management, the Change Control Board Manager was supported by a Change Control Board consisting of high level personnel. Among its members were the Spacecraft Subsystem Managers, the Spacecraft Managers, the managers of affected system engineering functions, the chairmen of engineering change centers, representatives of Quality Control, Reliability, Material, Contracts, and Space Flight Operations. These same members sat on joint JPL/HAC Change Control Boards.

b. Control documentation. Certain critical documents were designated as contractually controlling documents (CCDs) and placed under change control of JPL System Engineering. By mandate, changes to these documents then required the approval of JPL.

Among the CCDs chosen were the Spacecraft Development Plan, Test and Operations Plan, Quality Assurance Plan, and the major interface specifications for propulsion, power, and overall system test requirements. Additionally, the Space Flight Operations and *Centaur* Vehicle interface control documents were included.

Technical review of contractor-proposed changes was established for changes to CCDs, and for changes to spacecraft hardware and software. Changes that affected contract schedules or costs were excluded from this view, and required submission of a formal Engineering Change Proposal to JPL by the cognizant contractor.

A representative from JPL System Engineering was a permanent member of the contractor's Change Control Board. His responsibilities included obtaining advance information on proposed changes, coordinating the review of those changes at JPL, and determining the criteria against which review judgment was rendered. Further, as a representative of System Engineering, it was the responsibility of the Change Control Board representative to ensure that all interests of the Spacecraft and Mission Operations Systems, in both hardware and software areas, were incorporated into the reviewing process and the resulting decision for or against the change. Such decisions, properly documented and formally approved or denied, were returned through the Change Control Board. A contract modification was prepared and issued to implement the individually approved change.

A parallel and equally binding procedure was established for ground support and aerospace ground equipment. It included those controlling documents that technically and operationally reflected the supporting hardware and software critical to the success of *Surveyor*. Although these boards functioned separately, close coordination was exercised between them. Thus, Spacecraft System management was assured that spacecraft hardware changes were carried through into the operational equipment and personnel training procedures for spacecraft control during missions by the SFOF and DSN. Project MOS, SFOF, and DSN representatives sat on the Change Control Boards to ensure early exposure to, and proper implementation of, approved hardware and software changes into operational control consoles and ancillary equipments.

As a direct adjunct to configuration management, control of these key documents provided an effective management tool for day-to-day control of the as-built configuration of the individual spacecraft and its auxiliary ground equipment, subsystem by subsystem. It permitted insight and analysis for total integration of the *Surveyor* Spacecraft System, which included over 50 hardware subcontractors and operational units throughout the world.

5. Cost Control

a. Task/work item concept. At the outset of the *Surveyor* Project, the work that the contractor was to perform was segregated into discrete work packages called tasks and subtasks. These tasks represented the functional areas of performance such as electronics, propulsion, structure, etc. For the first 5 yr, the Project was monitored and controlled on the basis of these tasks and subtasks.

It was concluded, however, that effective control and monitoring of the contractor's performance would be facilitated by establishing work packages aligned to the individual performing groups in the contractor's organization. The functional tasks such as electronics efforts spread across too many organizational elements, and it was difficult to assess the contractor's performance in any individual organizational group. After careful consideration, it was decided that a revised work breakdown structure would better serve the need for more meaningful cost information.

In September 1966, a new work package structure was established on the HAC contract. Basically, it consisted of six major categories of work. These categories were management, engineering, manufacturing, test and operations, support equipment, and completed effort. These six categories were further segregated into 32 work items, which became the basic elements of effort used for monitoring and controlling the contractor's performance on other elements of the program such as the program plan and cost accounts; the financial reporting and management assessments were aligned and referenced to the revised work breakdown structure.

Figure V-9 shows the detail of the revised work breakdown structure. The monthly Financial Management Reports (FMR) submitted by the contractor are referenced. All of the cognizant engineer and contract administration assignments at JPL and HAC were aligned to the work items shown.

b. Monthly FMR. Hughes was required to submit a monthly FMR which was restructured to the new work item breakdown. The report was submitted in six parts as:

Report	Description
1	Project level report
2	Project/work items report
3	Work item report
4	Other direct cost (ODC) report
5	Cost and manpower curves
6	Narrative

The six parts of the report are described in detail in the following paragraphs:

Report 1: Project level report. This report was intended to meet all of the requirements of the traditional Project-level NASA type 533 report. Direct labor was displayed by categories of work, labor burden was a line item, and ODC was presented in the same elements of cost as given in contract negotiations. The report was on a cost-incurred basis. The outstanding commitments were shown for easy conversion to a committee cost basis.

Report 2: Project/work item report. This report provided the work item "total cost" from report 3; it summarized through the manufacturing cost level, and presented committed costs. At the end of the report, the calculations were made to bring the report to the total program committed cost and incurred cost level for easy balancing to reports 1 and 3. Equivalent manpower was also shown. Total commitments were shown separately to reconcile these costs with report 1.

Report 3: Work item report. This report displayed the same elements of cost used in the HAC internal cost control systems. These elements were labor costs and other direct costs: equivalent manpower. The manpower was obtained by dividing the total direct labor hours worked by the number of hours in the HAC accounting calendar for the report month. Internal HAC direct labor expended on fabricating special tools and test equipment was reported as "other direct cost," and was not included in the labor line item. All costs were at the manufacturing cost level and were committed costs. Labor costs included direct labor overhead, overtime premium and overease allowance. ODC included special tooling and test equipment, labor and material burden.

Report 4: ODC distribution report. The ODC report displayed the monthly committed cost of nonlabor items, and showed the elements of cost by work item. A subtotal was provided to balance the report to the ODC column on reports 2 and 3. A grand total was shown to balance the report to report 1.

Report 5: Manpower and cost charts. This report consisted of a series of financial status charts: total labor dollars, total ODC dollars, total dollars, and equivalent manpower. These charts were produced for the total program, each of the five categories, and for each of the work items. The data on the charts depicted graphically the committed cost at the manufacturing cost level.

Report 6: Narrative. A narrative was prepared for the total program and for each of the work items. It was a serious management assessment of the status of the work being reported. The narrative contained a paragraph on each of the following topics at the total Project and work item level:

(1) *Percentage of completion.* The percentage was the work item managers' assessment of the completion status of this work item. The percentage was not to be determined by dividing the dollars spent by the dollars authorized.

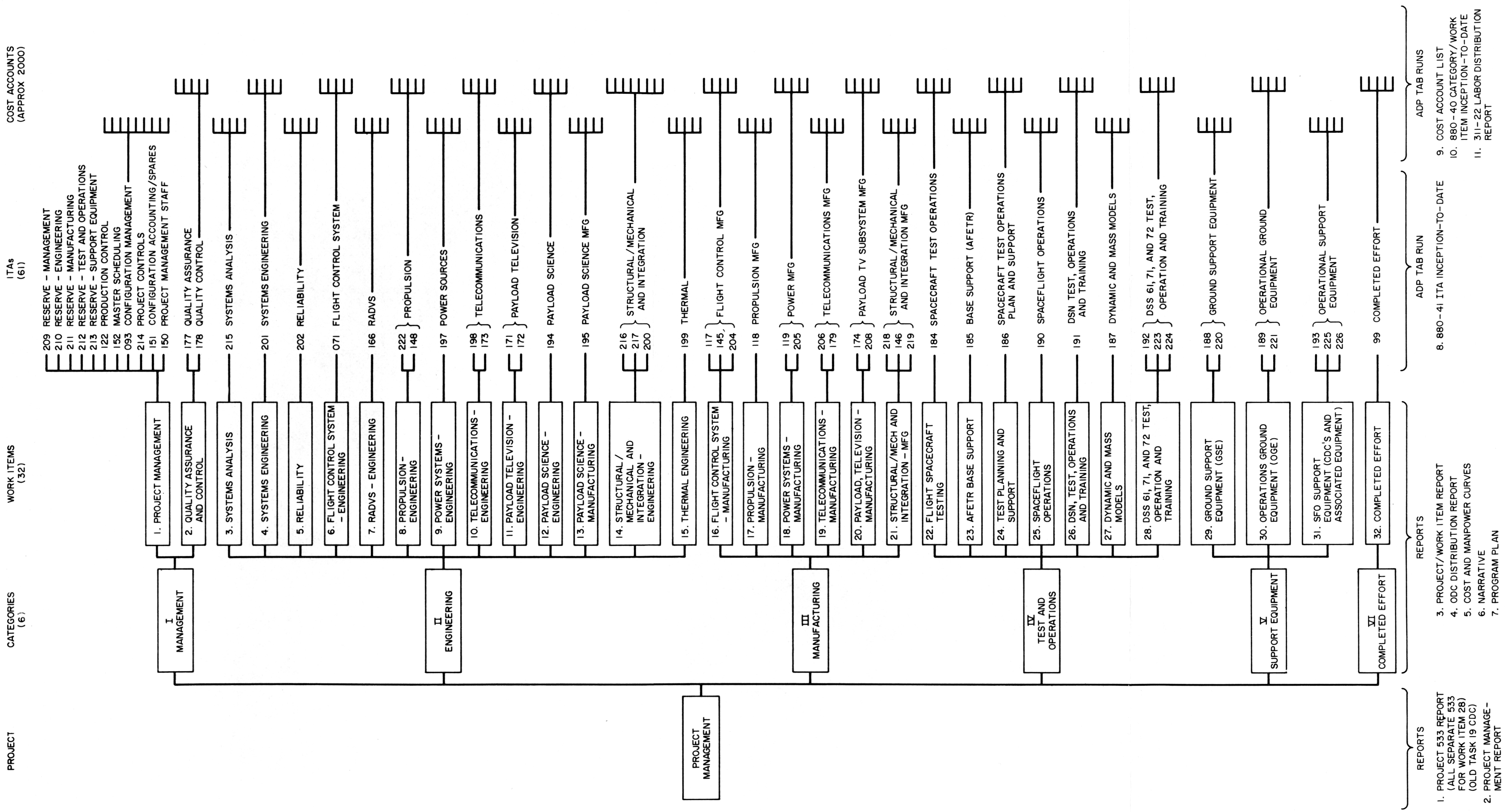


Fig. V-9. Surveyor cost and manpower reporting levels (HAC contract)

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(2) *Significant problems and their effects.* The more important items in each work item were discussed and their effect on the financial status of the work was assessed.

(3) *Management action and expected results.* The action HAC was taking, or planned to take, in solving the problems mentioned were discussed here.

(4) *Schedule status.* The schedule status, as it relates to cost, was discussed.

(5) *Changes from prior estimates.* If the estimated cost changed during the report period, the change was discussed.

(6) *Cost overrun/underrun.* As a result of the above analysis, the overrun/underrun status was discussed.

(7) *Significant journal entries.* A summation of the report period journal entries and their impact on each work item was tabulated in the narrative summary. Each work item manager reported on the entries to his work item.

(8) *Detailed explanation of contents of reports 1 through 7.* The change orders and modifications and the dollar amounts included or excluded were discussed.

(9) *Backup reports.* JPL received copies of the following tab runs on a monthly basis 4 days after the date of the reports: cost accounts list, category/work item inception-to-date tab run (880-40), and internal task authorization inception-to-date tab run (880-41). In addition, JPL received the labor distribution report tab run weekly, 4 days after the date of the report.

(10) *Supplemental reports.* The following supplemental reports were provided along with the regular 533 report: controller's report, report of undefinitized contract changed by work item, management reserve report.

c. Cost negotiations. Cost negotiations were used on the *Surveyor* Project for two reasons: (1) to accommodate the contractual changes made during the history of the Project, and (2) to reach a clear and detailed agreement between JPL and HAC on the complete meaning of each change.

Each time a contractual change was made to the contract, a formal detailed cost proposal was required from HAC. These proposals included a reiteration of the statement of work for the change; a detailed breakdown of the estmated cost of the change; supporting cost data and its rationale; the impact on schedule, if any; and a justification and rationale for any fee claimed.

The manner in which *Surveyor* negotiations were handled was somewhat dependent on the size and magnitude of the change; normally, however, negotiations took place between two teams representing JPL and HAC. Each team was chaired by the cognizant contract administrator from the *Surveyor* Contract Office. Team members generally included the cognizant engineers, the cost analyst, and any other parties interested or involved in the change. The major changes and the proposals and negotiations were resolved into smaller groups that often involved all cognizant engineers and administrators on the Project.

Technical personnel were required to review in depth the proposed direct labor and other direct costs. The cost analyst reviewed the direct labor rates, direct labor overhead rates, general and administrative expense rates, and other direct costs. The JPL Negotiating Team usually visited HAC and conducted fact-finding meetings before each negotiation. After all proposal and fact-finding information had been evaluated, the Negotiation Team would prepare a prenegotiation plan; upon approval of this plan, formal negotiations took place.

In addition to agreeing upon the equitable adjustments in the contract necessitated by the change, the negotiations also served a second purpose. The contractor's proposal represented his understanding of the change and the depth of review by JPL; the subsequent detailed communications and negotiations necessary to settle the change represented a valuable means of reaching an agreement with the contractor on the work to be performed and baseline from which the contractor's performance could be judged and controlled. Since the final negotiation represented a commitment and formal position on the part of the contractor, those agreements were carefully monitored and any deviation from those agreements, related to costs which were subsequently incurred, or to estimated manpower or manpower which had been agreed to be required, was carefully assessed. As a device for controlling their performance, HAC was often asked to explain differences between a negotiated position and their actual performance.

d. Management reporting. There were three basic ways to report *Surveyor* Project cost information to Laboratory management and to NASA Headquarters. (1) Financial management reports (FMRs) of the major

Surveyor contractors, including HAC, RMD, General Precision Instruments, EOS, and their ancillary reports. (2) Accounting reports generated by the JPL Financial Management Division, which contained performance information on Project obligations, commitments, costs, expenditures, and manpower utilization. (3) A semiannual budget submission (the Program Obligation Plan) generated by the *Surveyor* Project and, in the last years of the Project, a supplemental monthly Program Management Report, which presented actual performance information and estimates for cost and manpower for the completion of the Project.

The major contract administered by the *Surveyor* Project was with HAC for the design, fabrication, test, and operations of the *Surveyor* spacecraft. For the management of this contract, the *Surveyor* Spacecraft Development Plan provided the framework upon which the Management Control System section of the HAC contract was based. In addition to outlining hardware delivery requirements and contractual milestones, this document defined the work structure of the Project. The cost collection for the program was accomplished through the use of the HAC work authorization and accounting system and was based on the work structure for the program under contract at the time. Cost reports were submitted to JPL monthly from the inception of the Project. However, since the life span of the *Surveyor* Project was more than 7 yr, the method of financial management and the format of the cost reports were modified periodically in order to remain compatible with changes in HAC's corporate cost accounting procedures, *Surveyor* Project configuration, and government reporting requirements.

The initial work structure was patterned after the technical disciplines of HAC and JPL. The original contract was resolved into 14 major tasks (or work packages), each task composed of a number of subtasks. As the Project matured, additional work was authorized, resulting in new tasks and subtasks; and by mid-1966, the contract consisted of 20 tasks and nearly 100 subtasks.

In the early stages of the Project, monthly financial reports from HAC were prepared on an expenditure plus outstanding commitment basis at a total task level. Cost analysis by JPL was primarily limited to that performed by the JPL *Surveyor* Contract Administration Office.

Along with the expansion of the Project and extension of its schedule and cost came the need for greater visibility into the contractor's financial performance and involvement of the technical organizations and the *Surveyor* Resources Management Office at JPL in financial data analysis. The FMR format was converted to the NASA type 533 report and expanded to include cost collection and projections at the subtask level.

In late 1966, JPL and HAC agreed that the work breakdown structure had become unwieldy in size and content and the FMR equally unwieldy in detail. Agreement was reached to redefine the work breakdown structure so that it was more compatible with the existing JPL and HAC internal organizations, resulting in a structure consisting of 5 major active categories and, within them, 31 work items.

At the same time, a comprehensive cost and manpower reporting system was developed and served as the contractor FMR. This new report system satisfied a variety of reporting requirements and included a top level cost incurred 533-type report for reporting to Headquarters, a commitment and equivalent man report in the work item level, which allowed easy comparison between last month's estimates and those of the current month, and a comprehensive narrative report on each work item, emphasizing the problem areas and the corrective actions taken. After review and analysis by the JPL technical representatives responsible for each work item, major items concerning financial and manpower performance were reviewed by JPL and HAC at each of the monthly executive management meetings. The elements of the HAC FMR and the manner in which this review was accomplished are outlined in Fig. V-10.

The cost and manpower performance of the other major contracts let by the *Surveyor* Project was monitored through their monthly FMR in a similar, but far less rigorous, manner. These reports were in different formats, depending on the size of the contract, its scope of work, and the duration of its performance. Finally, cost reports were submitted to JPL by HAC for several of HAC's major subcontractors, notably Electric Storage Battery, Ryan, Thiokol Elkton, and National Waterlift.

Starting in 1962, JPL submitted, on a quarterly basis, a Program Obligation Plan (POP) describing the resource requirements for each of its NASA-funded projects. Originally, the POP presented cumulative commitments to date and projected resource requirements for the current fiscal year at gross cost elements. Over the years, the POP has expanded in detail and content to include each project by system and within each system major cost categories, major contract detail, calendarized costs to complete, and ancillary narrative and manpower

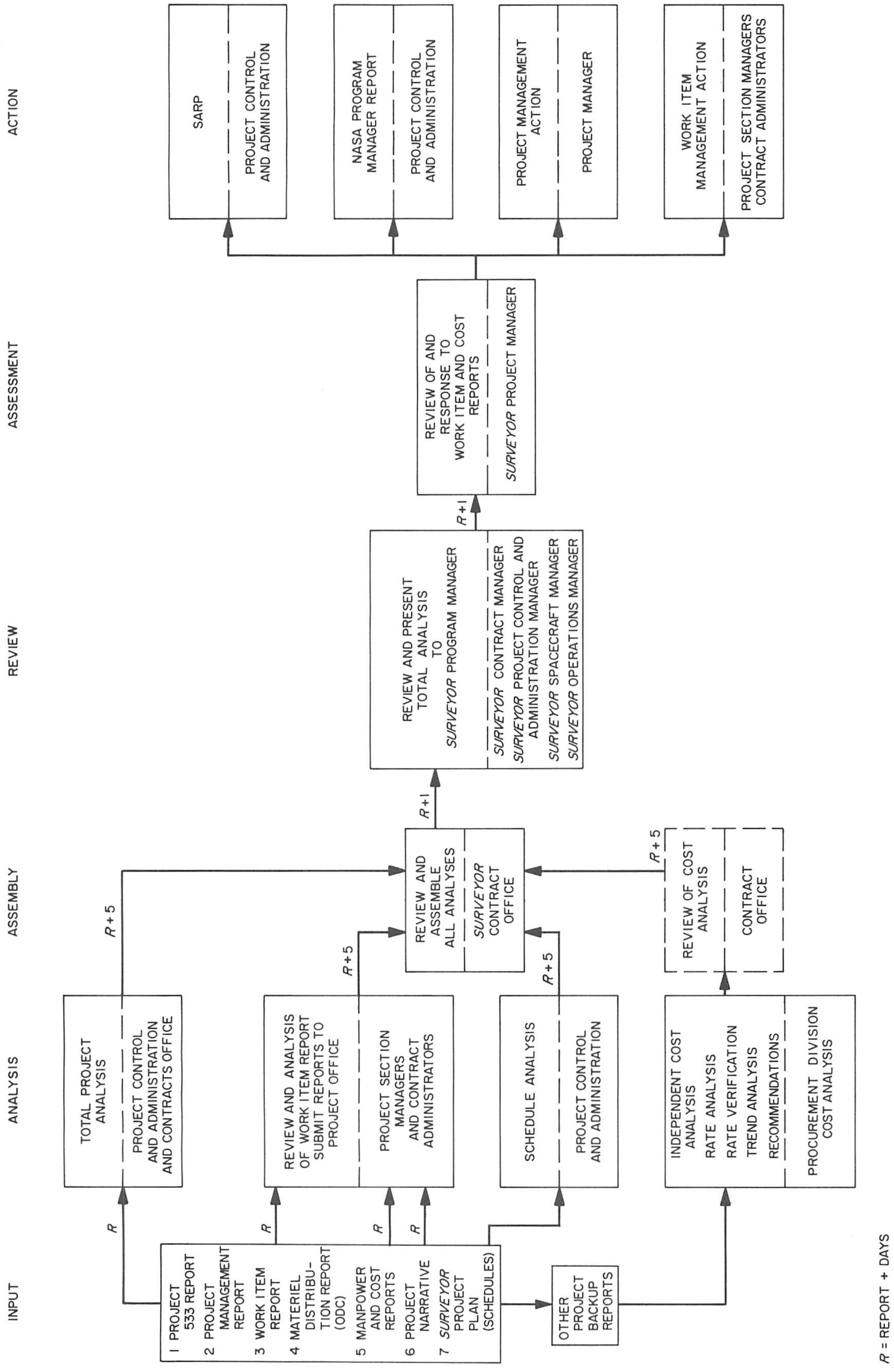


Fig. V-10. *Surveyor* cost, manpower, and schedule control system (HAC contract)

detail. In 1966, the *Surveyor* Project began complementing its POP submission with a formal presentation to representatives of NASA Headquarters; and in 1967, the POP was changed from a quarterly submission to semiannual.

In the early stages of the Project, the POP was supplemented with a biweekly status report, which contained manpower and dollar performance and projections, as well as schedule and technical information. In 1966, the biweekly status report was superseded by a monthly formal submission to Headquarters, originally known as SARP, later as MICS and finally as the Project Management Report (PMR). This report contained a brief narrative of recent events and problems, schedule and resource information, thus providing the Project Office with a mechanism for updating its POP projections more frequently.

The third means for reporting cost information to Laboratory management and to NASA Headquarters consisted of the detailed accounting reports generated by the Financial Management Division. These reports are, for the most part, on a monthly basis and include the Financial Activity Report, Expenditure Report, Cost Incurred Report and various funding and manpower reports. These reports allow one to monitor the fiscal performance of the Project against its POP projections, its internal budget, and its latest manpower plans.

Internal to the Project, several mechanisms were used to keep Project management informed as to financial and manpower status and requirements. In addition to the periodical preparation of the budget submission to Headquarters and the monthly Project reports, a formal review procedure was implemented within the Project for the orderly review of the FMR from HAC. This review consisted of analyses at the work item level by each cognizant engineer; the analyses were combined with an overall contract analysis performed jointly by the Project Control and Administration and the Contract Administration sections. These analyses were summarized and presented to the Project Office monthly. In addition, the Project Control and Administration Section prepared a monthly comptroller's report to the Project Office, outlining financial and manpower status and projected requirements and identifying major financial problem areas and recommended actions. Finally, the Project received a monthly Laboratory Budget Performance Report from the Financial Management Division; this report presented the current-year manpower and obligational status of each job with respect to its currently approved budget. This report was summarized and charted by the Project Control and Administration Section for distribution and review by elements of Project management.

The overall JPL and HAC manpower profiles for the life of the *Surveyor* Project are presented in Figs. V-11 and V-12, respectively.

6. Schedule Monitoring and Control

HAC was obligated to provide:

(1) Basic planning and programming support at the spacecraft system level.

(2) Master program plans, Project task plans, and detailed development plans.

(3) Schedule control of all hardware items.

(4) A Program Evaluation and Review Technique (PERT) and control system.

These items were the prime responsibility of the HAC Program Control and Administration Office.

Basic planning and the development of master program plans, detailed development plans, etc., were a continuing effort until the launch of *Surveyor VII*. Biweekly dissemination of this information was accomplished primarily by publication of the Management Control Report which evolved, in September 1965, into the *Surveyor* Program Plan. These reports included master program plans, individual spacecraft assembly and test plans, delivery schedules for major items of hardware, engineering schedules, etc. Issuance continued on a biweekly basis through September 1967 when it became a monthly publication.

PERT was developed and maintained by HAC to cover the Project from the spacecraft subsystem level through assembly, test, and launch of each spacecraft and included a master program network. Copies of the updated PERT networks, with computer printouts and critical path analyses, were delivered to JPL on a monthly basis. With the SC-7 (*Surveyor VII*) spacecraft delivery to system test, it was agreed that PERT would be of little, if any, further value, and the decision was made to terminate the effort. The last reports delivered to JPL reflected status as of February 24, 1967.

Schedule control of hardware items was the prime responsibility of the Master Scheduling and Program Control Section of HAC. Configuration Status Reports were prepared weekly for each spacecraft and set of spares. These reports identified each item of hardware, per spacecraft, by part number and serial number and

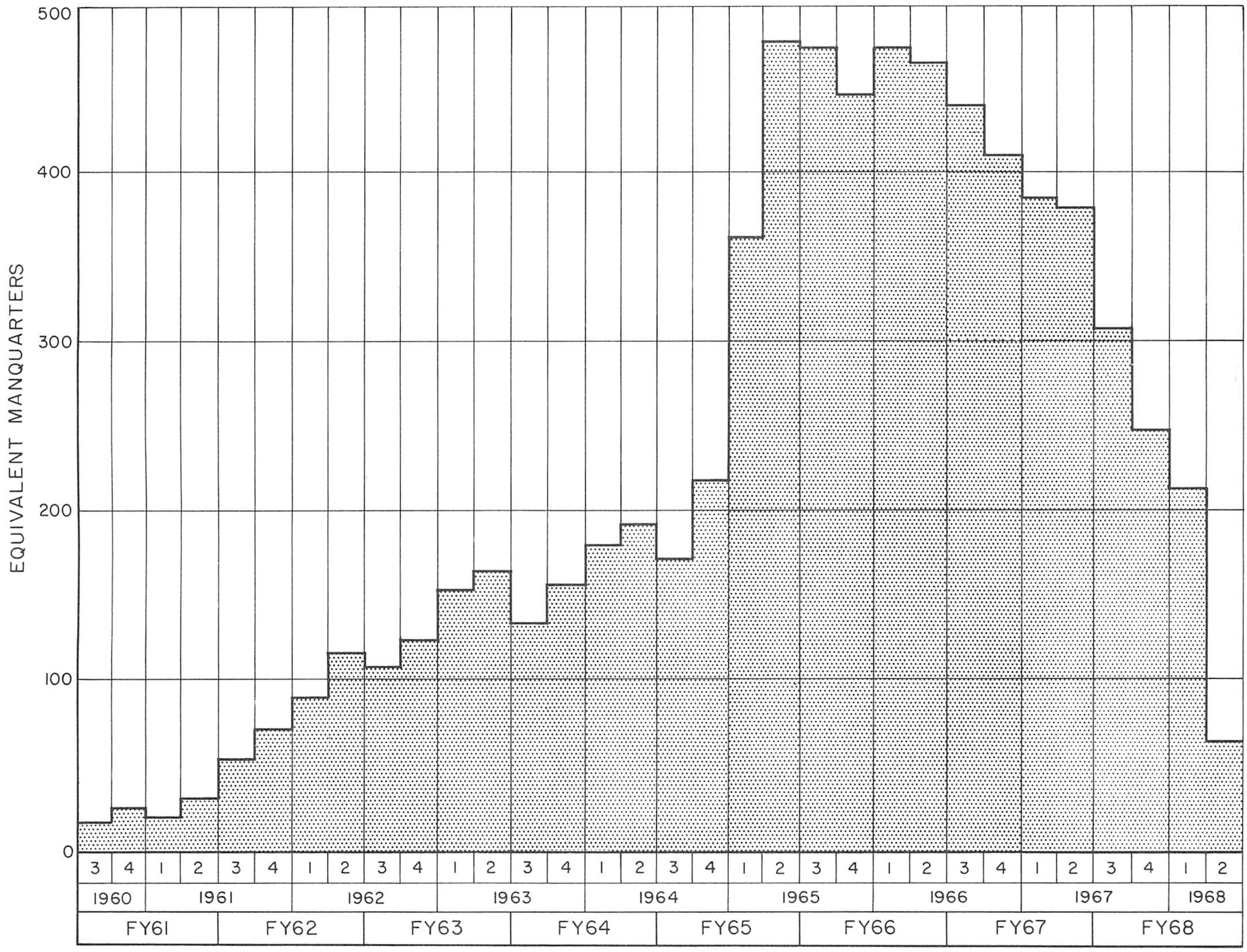


Fig. V-11. Manpower levels at JPL during *Surveyor* Project

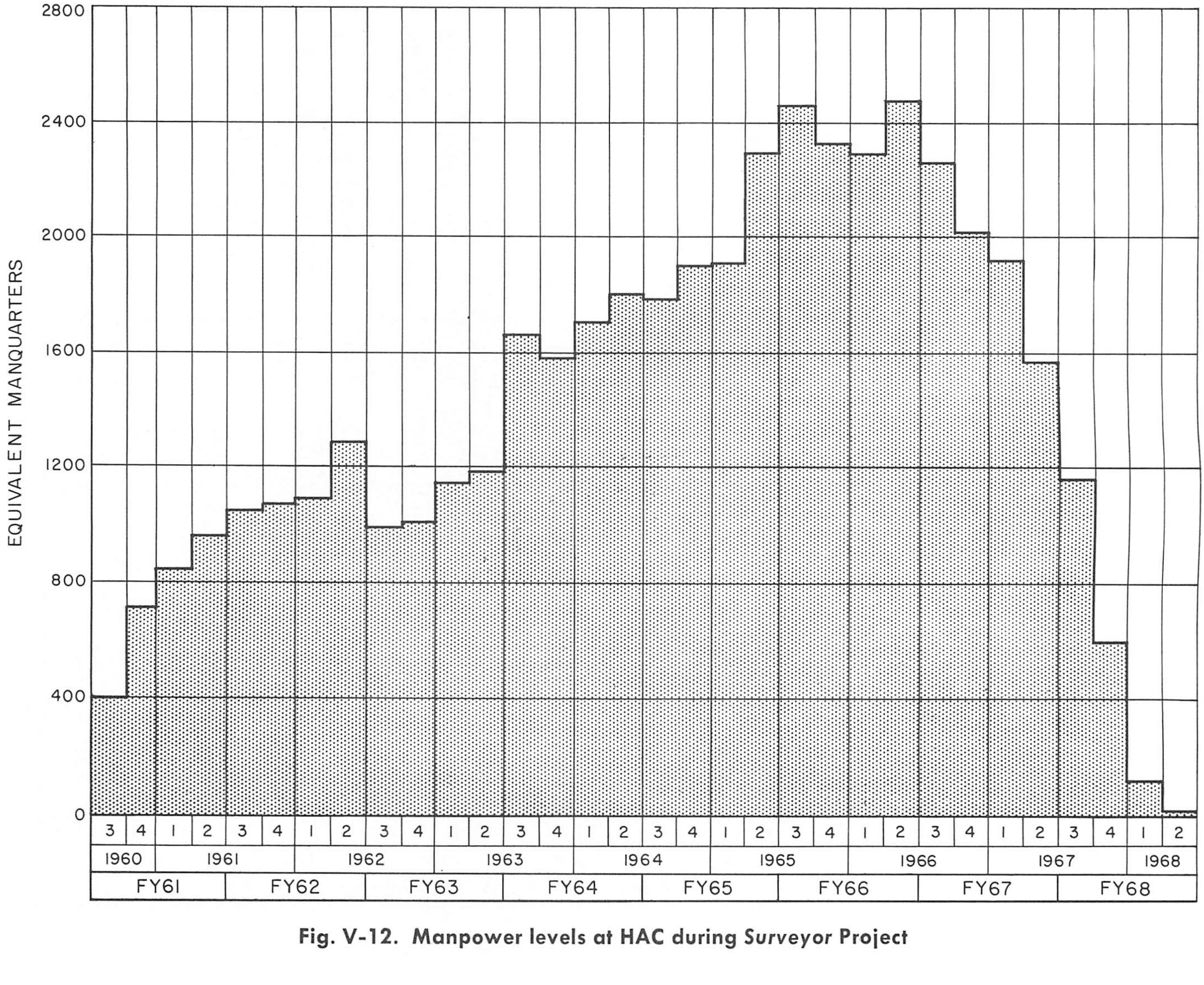


Fig. V-12. Manpower levels at HAC during *Surveyor* Project

reported status on a continuing basis through final acceptance of each spacecraft. Changeouts, reworks, etc., were shown if they covered more than a full reporting period.

The effort of the JPL *Surveyor* Project Control and Administration Section in schedule monitoring and control was divided into four major functions:

(1) Personal contact and observations at the HAC El Segundo and Culver City sites.

(2) Review of the HAC published schedules and related information.

(3) JPL in-house discussion and evaluation of HAC published information and the personal observations of JPL personnel.

(4) Periodic reports to NASA Headquarters.

7. Quality Assurance and Reliability

The *Surveyor* Quality and Reliability Assurance (Q&RA) Office, established within the JPL Project, reported functionally to the Project Manager's Office and administratively to the JPL Q&RA Office. The *Surveyor* Q&RA Office assisted in the conduct of specific tasks, recommended measures to be taken to improve mission success, and monitored mission operation readiness tests, spacecraft system and subsystem tests. As a part of the Project management team, their role was to bring a constant reminder to all personnel of a continuous need for quality, reliability, and minimum risk considerations in all decisions. The latest organizational interface elements of JPL and HAC Q&RA activities are shown in Fig. V-13.

a. Reliability.

1. Responsibilities. Responsibilities of the JPL Q&RA Office included establishing a reliability program for the JPL *Surveyor* Project Office, auditing all reliability tasks and performing specific tasks that other elements of the Project Office did not perform. Responsibilities of the Spacecraft System included management control and engineering support to reliability system design analyses and spacecraft subsystem interface problems. The spacecraft subsystem sections were responsible for subsystem reliability activities.

Reliability tasks were identifiable in all organizational systems of the Project. Reliability mission success modeling was performed at the mission level. Failure reporting and corrective action programs were established and conducted on each major system element. For the spacecraft contract, an extensive and rigorous reliability program was establishesd. Thus, the reliability programs on mission and system elements were different, but were developed to provide a minimum, but adequate and effective, response to NASA contractual requirements for a reliability program.

2. Reliability Program at HAC. Specific identifiable tasks were conducted by HAC Reliability Department in consonance with the requirements of NASA Publication NPC 250-1. Among these tasks, and perhaps most rewarding to the *Surveyor* Project in terms of both its significance and its contribution to the flight mission was the T/FR task effort. The details of this task, which impacted all levels of Project management, senior Project engineers, and cognizant design personnel, are described elsewhere in this report.

The HAC Reliability Department provided the mechanics of the basic T/FR system. A HAC management team sat as peers in "open" T/FR reviews and critiqued the status and significance of each open T/FR to the Project. The JPL Project Office conducted a "second look" closed-loop T/FR review to provide assurance of adequate analysis and effective closeout action. T/FR material was, in fact, a significant portion of the documentation which was offered as visibility for spacecraft shipping or launch-readiness reviews. The JPL Q&RA Office provided control loop for JPL cognizant approval as well as responsibility for the HAC control T/FR loop.

Other key tasks performed within the HAC Reliability departmental cognizance were: parts and materials program, subcontractor program, and spacecraft reliability assessment. The total reliability program is described in Section XII-B of this report.

b. Quality assurance. The basic responsibilities of the JPL *Surveyor* Q&RA Office were to:

(1) Prepare and implement requirements in consonance with applicable tasks of NASA Publications NPC 200-1A and NPC 200-2.

(2) Approve, audit and monitor all JPL *Surveyor* contractor quality systems.

(3) Participate in accepting, and certifying flight hardware, flight spacecraft, and support equipment for the Project.

(4) Represent JPL in contractor Material Review Board activities.

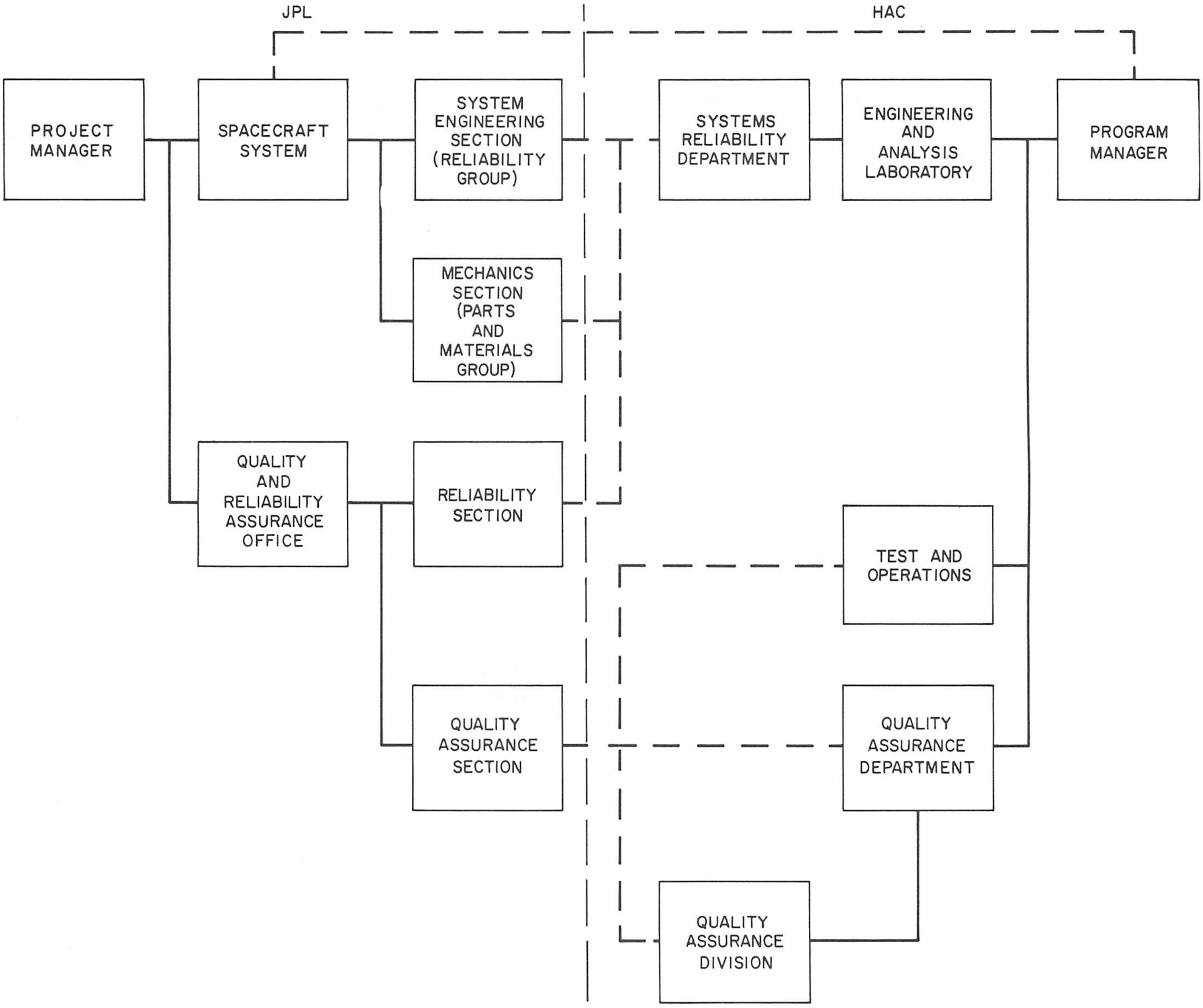


Fig. V-13. JPL/HAC interface diagram

(5) Provide assistance to the Project in all pertinent aspects.

(6) Establish and perform necessary inspection of hardware at key/critical points.

(7) Report quality status and recommend to the Project measures necessary to enhance product quality.

Each JPL contractor was responsible for establishing a JPL-approved Quality Assurance Program in consonance with JPL requirements. Basic responsibilities were to:

(1) Evaluate subcontractors and perform source inspection.

(2) Establish and perform necessary inspections and controls to ensure hardware integrity, conformance, and function.

(3) Identify nonconforming material and ensure initiation of corrective action to prevent recurrences.

(4) Participate in determining acceptability of hardware for flight.

The total quality assurance program is described in detail in Section XII of this report.

8. Management Participation

a. Executive management meetings. Meetings were held between the executive management personnel from HAC, JPL, and NASA Headquarters, normally on a monthly basis, for the purpose of reviewing the status of key aspects of the Project. In the early part of the program, these meetings were held primarily for the purpose of informing the management attendees on the important details of the Project. During the last 3 yr of the Project, these meetings took on the aspect of a forum for discussing critical problems, reaching decisions, and making management commitments. The agenda for each meeting was normally coordinated between the HAC Program Manager, the JPL Project Manager, and the NASA Program Manager. The meeting usually took a full day, and a typical agenda is listed below:

(1) Review of the minutes of the previous meeting: NASA/JPL/HAC.

(2) Cost and manpower status and projections: HAC. Key HAC management personnel would distribute the most current data available showing actuals and trends in the areas of cost and manpower and would be asked to explain the variation from previous predictions.

(3) Contract status: JPL. The JPL Contract Manager would report on pertinent aspects of contract activity including such things as the status of changes made to the contract, and the schedule for definitizing and incorporating the changes into the contract.

(4) Spacecraft status: HAC. HAC would present summary results of any mission accomplishments since the previous meeting, and a review of key aspects of each spacecraft being readied for future missions. The contractor was always required to review in depth the major problems that currently existed on each spacecraft.

(5) Control item hardware summary: JPL/HAC. Reports relating to the control item hardware deliveries that HAC was either manufacturing or obtaining from subcontractors. JPL would report on control items furnished by JPL for use by HAC.

(6) Milestone status: HAC. HAC was required to provide a graphical analysis of the number of milestones achieved during the reporting period vs the number of milestones that HAC predicted would be achieved.

(7) Other topics: NASA/JPL/HAC. This item was placed on the agenda for the purpose of offering attendees an opportunity to bring up subjects which were not specifically listed on the formal agenda.

(8) Review action items: NASA/JPL/HAC. Action items generated from previous management meetings were reviewed with relation to compliance with the directions from those meetings.

(9) Summary of action items: NASA/JPL/HAC. Action items for the meeting were summarized and agreement was reached as to who was responsible for taking the required action.

The executive management meetings contributed a great deal to the success and continuity of the *Surveyor* Project. Since the participants represented the top level of management of each of the agencies involved, the meetings were an effective means for complementing communications and action.

b. Consent-to-ship meetings. Consent-to-ship meetings were instituted to: (1) permit senior management review and acknowledgment of the demonstrated readiness of the spacecraft for shipment; (2) ensure a complete, formal overview of past decisions, problem dispositions and performance of the spacecraft as it was tested through the system test cycle; (3) define future action required for completion before launch; and (4) obtain from NASA and JPL Project Management formal concurrence for shipment of the spacecraft from HAC to either Combined Systems Test (CST) at San Diego or to AFETR for launch preparation.

The meetings were held a few days before the scheduled date of shipment. A data package was transmitted by HAC to JPL and NASA a few days before the meetings to allow adequate time for final review of the material. The data package consisted of:

(1) T/FR status report for spacecraft.

(2) T/FR status report for AGE/CDC.

(3) Spacecraft configuration index.

(4) A volume containing: schedules, test results summary, hardware changeout, hardware repairs, hardware modifications, open vehicle discrepancies, open test requirements, operational constraints/waivers, reliability assessment, spares status, AGE status, procedure status, personnel deployment schedule, and subsystem review.

It is to be noted that the continuous participation by JPL cognizant personnel provided insight into these areas so the data package represented a documented collection of the information.

The chairman of this meeting, which was held at HAC, was the JPL Spacecraft System Manager. The meeting was attended by a Consent-to-Ship Board consisting of representatives from NASA Headquarters, JPL, and HAC. The Consent-to-Ship Board typically consisted of the NASA *Surveyor* Program Manager, NASA Program Engineer, JPL *Surveyor* Project Manager, JPL *Surveyor* Spacecraft System Manager, JPL Assistant Project Manager for Operations, HAC Associate Manager of Space Systems Division, and HAC *Surveyor* Program Manager. Cognizant members of the JPL and HAC *Surveyor* Project also participated.

The agenda covered:

(1) Previous spacecraft significant problem status.

(2) Spacecraft history and status: history and schedules, design aspects, test program summary, and assessment of test requirements.

(3) System review.

(4) Subsystem reviews.

(5) Reliability assessment.

(6) AFETR readiness.

(7) Management review.

As the various areas were discussed, the Board dispositioned the various unresolved problems as:

(1) Category A: unrelated or insignificant.

(2) Category B: problems that required additional work to classify or close out before launch.

(3) Category C: problems that could not be solved before launch and as such were recognized and accepted liens against potential spacecraft mission performance.

At the end of the presentation, the Consent-to-Ship Board convened to assess the various problems and liens in force and to decide whether the spacecraft and its supporting elements were in an adequate state to warrant shipping.

c. Consent-to-launch meetings. The consent-to-launch meetings were similar to the consent-to-ship meetings in that the material presented and discussed and personnel participation were similar. The consideration at this time was the spacecraft readiness and its condition for launching and performing its intended mission. At this time, all problems were resolved or the risk was considered acceptable against spacecraft mission performance.

d. Management interface relationship. The organization of the Project by major systems resulted in the establishment of several important interfaces, all of which required exacting definition and comprehensive verification. Management control of these and lesser interfaces was essential since the systems or subsystems (spacecraft telecommunications) involved were dispersed geographically and managed by several different agencies or companies. The main interfaces considered were the Launch Vehicle System Spacecraft System, T&DS, and MOS.

In organizing for control over these interfaces, it was essential that well delineated responsibility and authority be established, that individuals and/or groups be identified for formal communications and that good documentation be developed and maintained. At the same time, it was important that the system of control allow easy interchange of information at the working level. It also was considered essential that each interface undergo testing for adequate verification. This entailed testing each side of an interface separately, followed by combined testing during joint operations.

The overt and concerted efforts made to maintain constant and appropriate management communications contributed to the success of the *Surveyor* Project. This communication was necessary within each of the participating elements of the Project; that is, at NASA, JPL, HAC and other major contractors, as well as between those participating elements.

Because of the magnitude, importance, and urgency of the *Surveyor* Project, it was necessary that an effective day-to-day interface exist between the key management personnel of NASA/JPL/HAC. Although the guidelines for the relationship and roles of these personnel were informal and never put in writing, several key factors in the relationship were important.

(1) HAC management personnel were always accessible and responsible to JPL management personnel.

(2) NASA management restricted relationship with HAC management personnel to the lowest practical minimum. The exceptions were those instances in which all three parties met, JPL, HAC, and NASA, such as at executive management meetings, and consent-to-ship and consent-to-launch meetings. Information was passed from HAC to NASA via JPL. NASA directions were passed to HAC via JPL.

(3) An environment was created where problems were surfaced and exposed at the earliest appropriate time and handled at the most appropriate level of management.

(4) An effort was exerted to maintain the closest possible relationship between like levels of JPL and HAC. Since JPL and HAC had intentionally formed similar Project management organizations, managers of specific disciplines were expected to maintain a close and continuous interface to deal with progress and problems in that discipline top level management also maintained the same day-to-day relationship.

Another significant interface, which is briefly mentioned here as an example, was that between the Spacecraft System and the Launch Vehicle System. This was an important interface because of the functional and physical complexity between the Spacecraft and Launch Vehicle Systems and because of the organizational complexity involving several agencies and companies (Fig. VIII-4). In addition to the formal relationship between the *Surveyor* Project Manager (at JPL) and the *Centaur* Project Manager (at LeRC), informal information flow channels permitted technical information to be exchanged at the working level between each of the agencies involved.

Verification of the Spacecraft System/Launch Vehicle System interface was carried out by CST at a specially constructed CST stand at San Diego, as well as at Cape Kennedy. These tests provided for direct hookup of all interface elements between the Spacecraft Launch Vehicle Systems and the exercising of the interface by a series of tests that provided functional simulation of the countdown and launch phase of flight. The test was designed to check the proper operation of each function, as well as to demonstrate the absence of any degrading interaction.

During early phases of Spacecraft System/Launch Vehicle System and spacecraft interface development, joint meetings were held in which representatives of all elements of the interface met to define, review, and evaluate all facets of the interface. As the program progressed, such action was reduced since the test and flight results continued to demonstrate a well qualified and trouble-free interface.

A similar pattern of design, review, test and control was conducted between the various other system interfaces, i.e., Spacecraft System/MOS, Spacecraft System/T&DS and MOS/T&DS. These interfaces are described in greater detail in the related sections of this report.

e. Special management action. During the Project history, events occurred that occasionally posed a serious threat or concern to the Project. Such situations varied from a relatively simple technical problem of a repetitive or chronic nature, a failure causing interruption of a key test, or the extreme of an inflight, abortive failure, i.e., SC-2, SC-4.

In such cases, it became imperative that top management, from within the Project or, in the more serious cases, when special outside consultants were designated, participate in the review action. If the situation warranted or prevailed, key members of management were convened as a review team to assess the situation and recommend or initiate corrective action. In simple situations, 1 day might be adequate for review while a more serious situation might entail several months of intensive investigation and review.

Several instances of abortive failure, such as the early crash of the T-2 radar-controlled landing test vehicle, the SC-2 flight failure, or the SC-4 flight failure, were of such importance that various members of upper management from HAC, JPL, and NASA were convened in Failure or Technical Review Boards. The main review board usually asked for special technical working groups to be convened for in-depth studies of a particular technology under scrutiny. Such groups would then report their findings and recommendation to the review board. Thus, independent, experienced individuals were provided any reasonable resource they might request to explore the conditions leading up to the problem, to assess the problem, and to recommend corrective action. In addition, followup reviews were undertaken to assess the adequacy of the results achieved.

The primary function of such special groups was to cause or direct comprehensive studies and analyses of the problems for consideration. This would then permit certain actions to be decided or directed from the review body.

Where smaller problems were involved, outside management or consultants were not used, but a similar procedure was followed. Usually, technical specialists and members of management were brought together in a group to conduct the review. Such groups were usually kept small; but if it became too large and unwieldy, splinter groups were organized to provide inputs to the main reviewing group.

Frequently, in either the case of major review boards or the lesser review committees, many organizational and technical interfaces were crossed. In such cases, clear and concise delineation of what was desired, who was to be involved, responsibility, and schedule of events was mandatory. Thus, a chairman for such action was required who would be firm, yet equitable, to provide comprehensive coverage of the problem under consideration. This was particularly true where large-scale activities were involved requiring significant commitment of resources from parties that might not have been responsible for the situation under review. However, the cooperation of such parties was essential to the integrity and thoroughness of the final outcome. In the instances cited, as well as others not discussed, the outcome demonstrated the success of such reviewing action conducted by *Surveyor* Project management.

VI. Mission Analysis and Engineering

The Mission Analysis and Engineering function was composed principally of Mission Design and Analysis, Telecommunication System Analysis, and Launch Vehicle Integration. Mission Design and Analysis included trajectory design, navigation and guidance, launch constraints definition, and related functions. These activities, which are described in Subsections A–I, were supported through the coordinated efforts of JPL, HAC, LeRC, GD/C, TRW Systems, and the AFETR. The Telecommunication System Analysis activities are described in Subsection J. Launch Vehicle Integration was particularly concerned with assuring spacecraft/launch vehicle hardware compatibility, but also monitored all other spacecraft/launch vehicle interfaces. This section presents a discussion of these activities.

A. *Surveyor* Trajectory Design

Surveyor trajectories consisted of three distinct phases: (1) The boost phase, during which the *Atlas/Centaur* launch vehicle lifted the spacecraft from Launch Complex 36 at Kennedy Space Center (KSC) to the required translunar injection conditions. (2) The translunar phase, during which the spacecraft separated from the *Centaur* and began its ballistic flight to the moon. Translunar injection occurred between about 11 and 34 min after launch, corresponding to a location anywhere from the North Atlantic ocean to South Africa, as shown in Fig. VI-1. The time and location of injection were determined by the earth–moon geometry at lunar encounter, the ascent mode, and launch azimuth. Duration of the ballistic, or cruise, phase of flight ranged from 62 to 67 hr, depending on lunar declination at encounter. (3) The descent phase, during which the spacecraft made a powered descent to the lunar surface. This phase began with the ignition of the spacecraft's vernier engines immediately before main retroignition, and usually lasted 2 to 3 min.

The boost phase of the trajectory was designed by LeRC and GD/C with regard for the constraints set forth by JPL. Determination of the programmed steering rates used during programmed *Atlas* booster phase and the guidance logic used during *Atlas* sustainer and *Centaur* phases of flight were involved, as well as many other complex functions. Because the boost phase and the translunar phase were strongly coupled, the boost phase had to be simulated in the design of the translunar phase. However, the translunar phase and the powered-descent phase were coupled through only two ballistic (i.e., unbraked) lunar-impact parameters: speed and incidence angle. Consequently, the translunar phase was

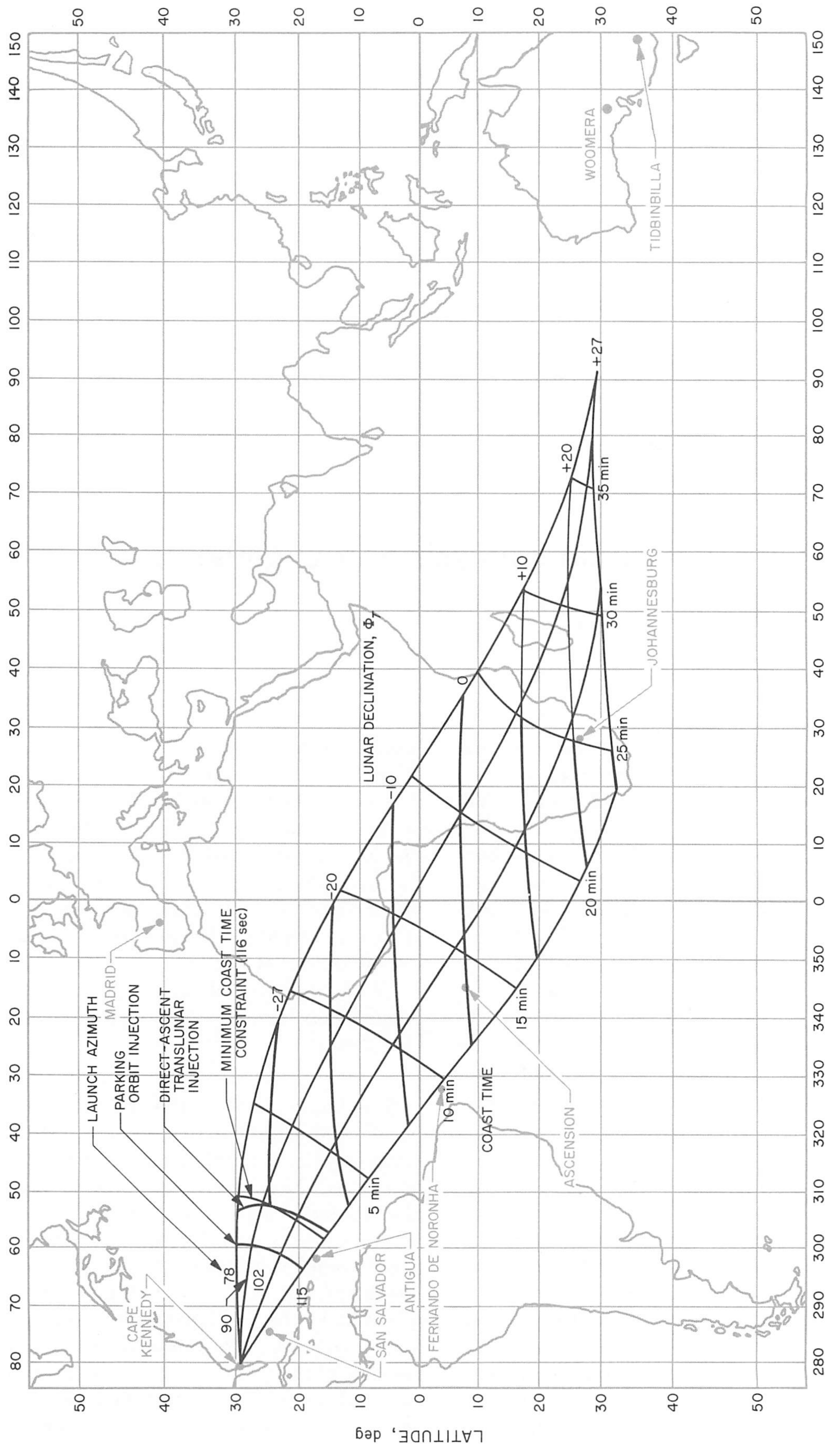


Fig. VI-1. Parking orbit and translunar injection loci designed to unbraked lunar-impact conditions, which eliminated any need to simulate the powered-descent phase in series with the translunar phase. The translunar trajectory design was, therefore, reduced to the process of selecting, from the continuum of ballistic translunar trajectories, those trajectories that could achieve the mission objectives while satisfying the mission constraints and ground rules. To examine the great number of trajectories of interest in the selection process, highly efficient, approximate, trajectory generation techniques were developed. These techniques are discussed in this section of this report; the ascent modes used for *Surveyor* also are discussed and compared. The 90-hr trajectories, which were considered and rejected, are compared with the 66-hr trajectories used by *Surveyor*. The final trajectory design process is discussed in some detail (also see Section XIII for a discussion of terminal-descent-phase trajectory design).

1. Computational Techniques

a. Conic approximations. For the trajectory design (and all other preflight analysis), the *Surveyor* translunar trajectories were adequately approximated by modified conic techniques. That is, the basic two-body equations of motion were used in conjunction with several empirical correction devices. These equations dictate that the translunar trajectory is a portion of an ellipse with one focus at the earth's center as shown approximately to scale in Fig. VI-2. The plane of the ellipse (i.e., the trajectory plane) must contain the earth's center, the injection point, and the moon[1] at the time of lunar arrival. Although the boost trajectory is not exactly planar, it can be assumed to be for the following discussion. For a planar boost trajectory the launch site also lies in the translunar trajectory plane. Figure VI-3 presents this three-dimensional geometry. Knowing that the moon's position at arrival changes very slightly during a launch window,[2] it can be seen from Fig. VI-3 that the launch azimuth Σ_L and in-plane central angle ϕ_{LT} required to intercept the moon vary with launch time because of the earth's rotation. The ranges of the values to which these two parameters were restricted determine the initial definition of the launch window.

The size and shape of the translunar ellipse (Fig. VI-2) are determined by the perigee radius r_p and the geocentric energy C_{3E} as given by:

[1]To compensate for the bending of the trajectory caused by the increasing influence of lunar gravity as the spacecraft approaches the moon, the conic approximations were "aimed" for an empirically determined point slightly ahead of the moon rather than the moon.

[2]A launch window is the interval of time on a given day when a launch may be attempted.

Eccentricity

$$e = 1 + \frac{r_p C_{3E}}{\mu_E}$$

Semimajor axis

$$a = -\frac{\mu_E}{C_{3E}}$$

where μ_E is the earth's gravitational constant.

The perigee altitude was fixed at 90 nmi based on consideration of injected spacecraft weight requirements (i.e., *Atlas/Centaur* payload performance capability) and aerodynamic heating constraints. Geocentric energy C_{3E}

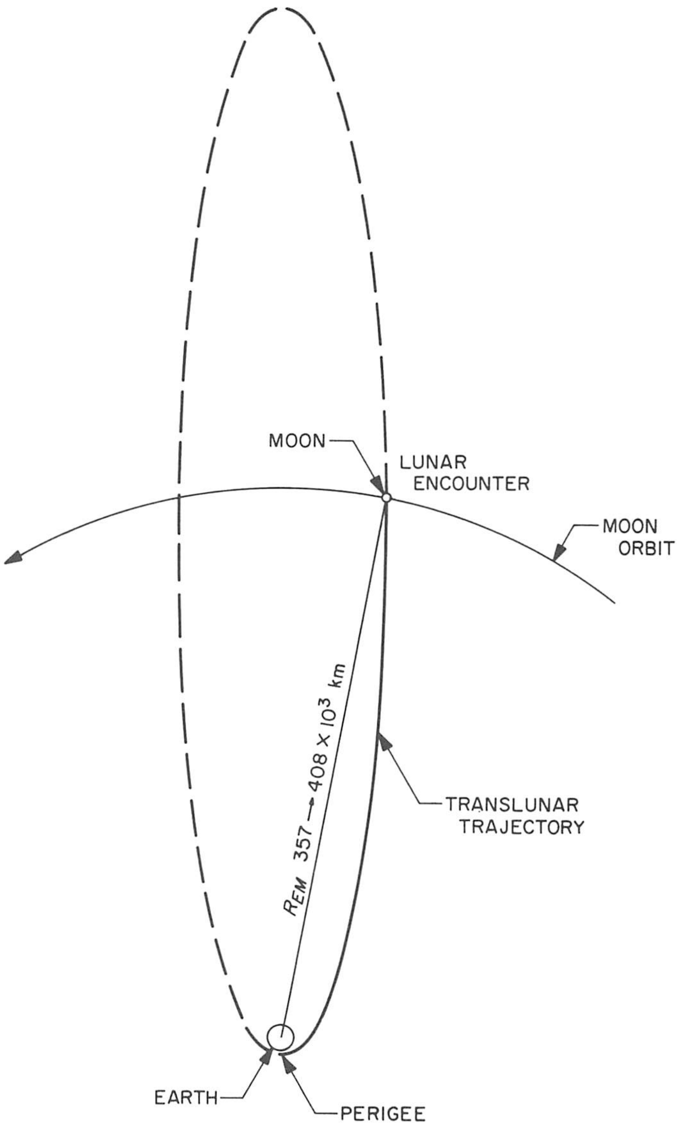
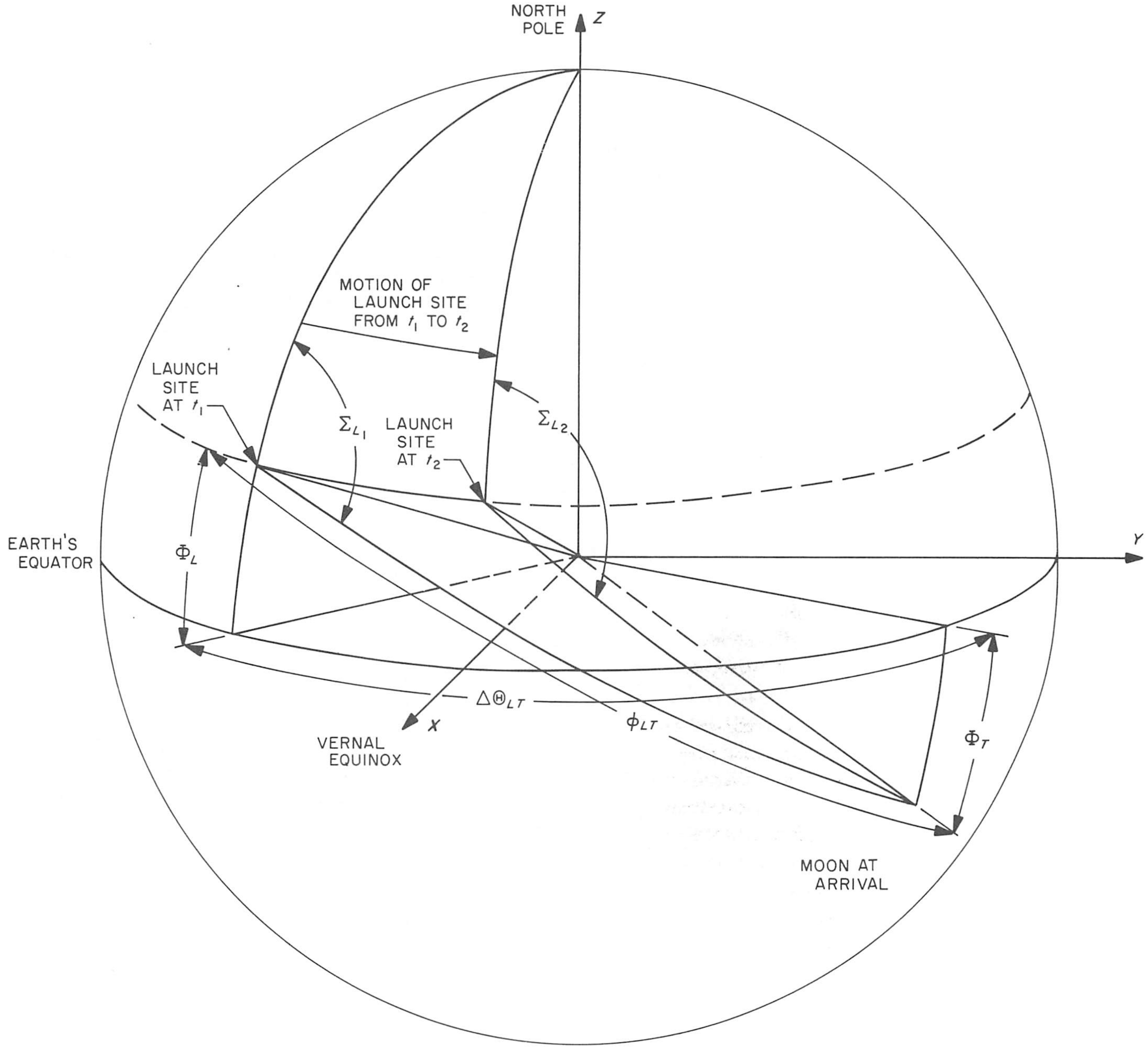


Fig. VI-2. In-plane translunar geometry (approximately to scale)



Φ_L = LAUNCH SITE DECLINATION

Φ_T = LUNAR DECLINATION AT TIME OF ARRIVAL

$\Delta\Theta_{LT}$ = DIFFERENCE IN RIGHT ASCENSION FROM LAUNCH TO LUNAR ENCOUNTER

ϕ_{LT} = IN-PLANE ANGLE FROM LAUNCH TO LUNAR ENCOUNTER

Σ_L = LAUNCH AZIMUTH

Fig. VI-3. Lunar launch geometry

was then the key parameter in defining the translunar trajectory. The required geocentric energy was determined from a polynomial in required translunar flight time and earth–moon distance at lunar arrival:

$$C_{3E} = C_{3E}\,(T_f, r_m)$$

where T_f is translunar flight time and r_m is earth–moon distance at arrival. The coefficients of this polynomial were determined by curve-fitting precision, numerically integrated, n-body trajectory data. This polynomial was one of the most significant empirical devices used to achieve the required accuracy in the design process.

To determine the unbraked lunar-impact conditions, it was necessary to "patch" the translunar geocentric conic to a moon-centered (selenocentric) conic. This is known as the patched conic technique, illustrated in Fig. VI-4. In the vicinity of the moon, the trajectory is essentially a hyperbola with the principal focus at the moon's center. The hyperbolic excess velocity[3] $V_{PM\infty}$ is obtained by vectorially subtracting the geocentric (i.e., earth-relative) velocity of the moon from the geocentric velocity of the spacecraft. The corresponding unbraked impact speed V_{IM} is then:

$$V_{IM} = \left[\,|\,V_{PM\infty}\,|^2 + \frac{2\mu_M}{R_M}\right]^{1/2}$$

where μ_M is the moon's gravitational constant and R_M is the moon's radius (1738 km). It was also necessary to determine the selenographic latitude and longitude of the unbraked vertical impact point. This is the unique point at which the spacecraft would impact if it approached the moon along a straight path (see Fig. VI-4). This point is found analytically by determining the point at which a line parallel to the hyperbolic excess velocity vector and through the moon's center pierces the surface. The incidence angle Γ_{IM} at any point on the lunar surface was determined from the basic conic equations to be given by:

$$\Gamma_{IM} \cong 0.7\theta$$

where θ is the angle measured at the moon's center between the vertical impact point and the point in question, as illustrated in Fig. VI-4. However, in generating the conic design trajectories, location of the landing site was of little concern. In fact, the design frequently was initiated before the landing site was selected.

*b. **Boost models.*** As previously noted, the boost phase had to be simulated in the design of the translunar trajectories. Only the end points (i.e., launch and injection) had to be modeled; the powered flight path was not required. The modeling differed for the direct- and parking-orbit ascent modes; therefore, they are treated separately below.

Direct ascent. The direct-ascent boost phase was characterized by nearly continuous thrusting from launch to translunar injection (i.e., there was no parking-orbit coast phase). To compensate for the fact the boost phase was nonplanar because of earth rotation and some yaw steering, the trajectory was constructed in the plane containing the launch site and injection point, but, consequently, not the actual velocity vector. The orientation of this plane was specified by empirically adjusting the launch azimuth. The adjustment, determined from precision AC-5 and AC-6 powered flight simulations generated by GD/C, proved to be adequate for all direct-ascent flights.

Figure VI-5 shows the adjusted, in-plane, direct-ascent geometry. To intercept the moon, the spacecraft must sweep out the central angle ϕ_{LT}, which is determined by launch azimuth and the moon's declination at arrival. This dictates the following geometrical requirement:

$$\phi_B - \nu_I = \phi_{LT} - \nu_T$$

where ϕ_B is the burn arc, ν_I is injection true anomaly, and ν_T is target (moon) true anomaly. The target true anomaly, ν_T, was relatively fixed at about 170 deg, but exactly determined by the earth–moon distance at arrival and the geocentric energy, C_{3E}. The *Atlas/Centaur* burn arc ϕ_B was directly related to the injection true anomaly ν_I, as shown in Fig. VI-6. Note that the burn arc is relatively fixed at 28 deg, which would be expected for a single-burn boost phase. Consequently, ϕ_B was curve-fit to the quantity $\phi_B - \nu_I$ using the AC-5 and AC-6 data mentioned. Once the quantity $\phi_{LT} - \nu_T$ was determined, ϕ_B and ν_I could be readily obtained and the injection point located.

The amount of weight the *Centaur* could inject into the translunar trajectory was a strong function of the injection flight-path angle[4] Γ_I and a weaker function of

[3]Hyperbolic excess velocity is the theoretical velocity the spacecraft would have possessed at an infinite distance from the moon if it arrived at the moon from this distance in the absence of all forces except lunar gravity.

[4]Γ_I is readily determined from ν_I, C_{3E}, and e ($\nu_I \cong 2\Gamma_I$).

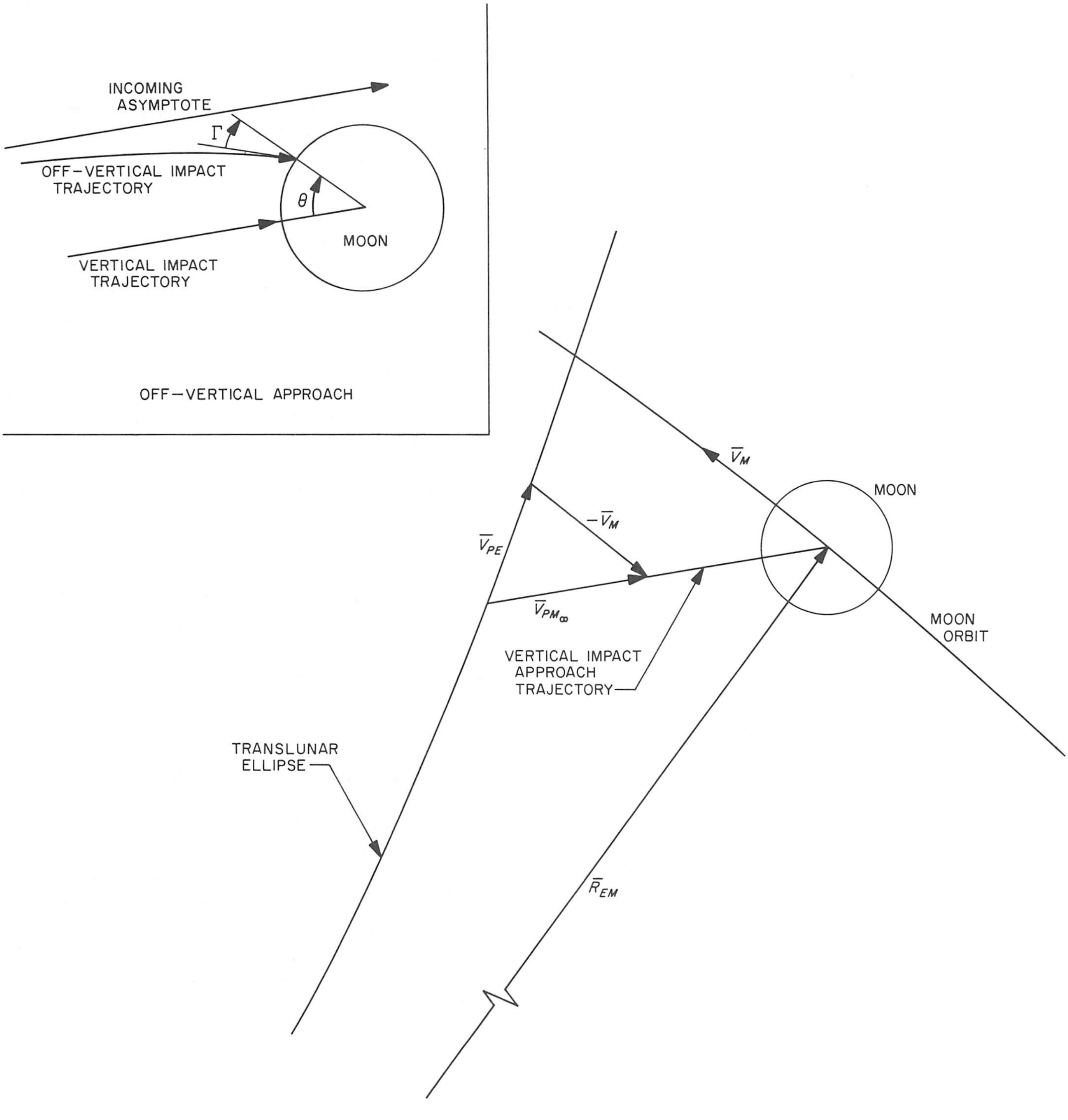


Fig. VI-4. Lunar encounter geometry

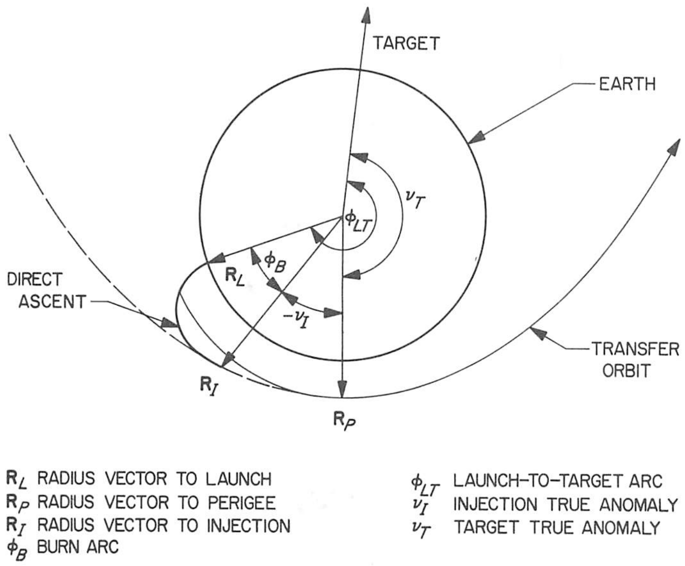


Fig. VI-5. In-plane direct-ascent geometry

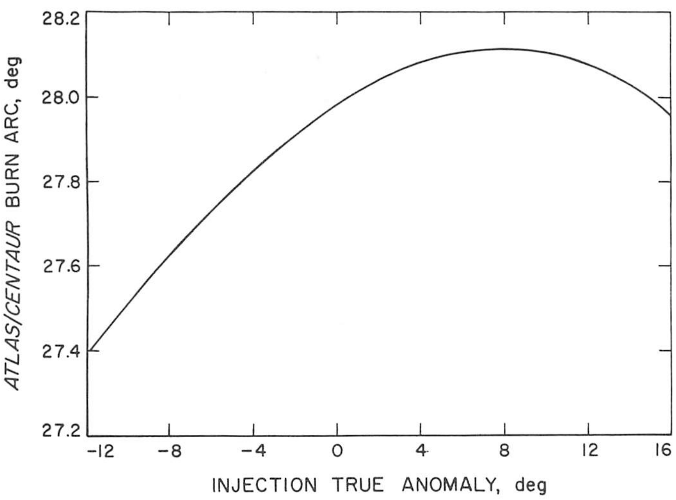


Fig. VI-6. Direct-ascent burn arc

launch azimuth Σ_L, and geocentric energy C_{3E}. *Centaur* payload capability (i.e., allowable spacecraft weight) was reported by GD/C at the performance reporting point given by $\Gamma_I = -2.5$ deg, $\Sigma_L = 108$ deg, and $C_{3E} = -0.85$ km²/sec². The variation in payload capability about this point, called the payload capability margin, was curve-fit in terms of Γ_I, Σ_L, and C_{3E} using the AC-5 and AC-6 data. The resulting curves are shown in Fig. VI-7. If, for example, GD/C reported that the "net margin" was 150 lb, this meant that *Centaur* could inject *Surveyor* at a performance point 150 lb more stringent than the reporting point. Thus, the capability margin given by Fig. VI-7 could be as low as -150 lb; or the total margin (equivalent to excess propellant) was the sum of the GD/C reported net margin and the computed payload capability margin. *Atlas/Centaur* could inject *Surveyor* at any point where the total margin was positive. In summary, the direct-ascent boost model was used to determine the burn arc ϕ_B, injection true anomaly ν_I, and *Centaur* excess propellant in addition to adjusting the trajectory-plane orientation. The mission design implications of direct ascent are discussed in Subsection A-3. Note that the propellant availability manifests itself principally in limitations in injection true anomaly ($-6° < \nu_I < 14°$) that, in turn, limits ϕ_{LT} ($184 < \phi_{LT} < 204$), thereby restricting direct-ascent missions to lunar encounter declinations below approximately -16 deg, a very severe limitation.

Parking-orbit ascent. The parking-orbit ascent mode consisted of three phases: (1) the first burn from launch to parking-orbit injection, (2) the coasting period in a circular parking orbit to satisfy the in-plane geometrical requirement, and (3) the second burn from parking orbit to the translunar trajectory. The state vector (i.e., position and velocity) at parking-orbit injection was a function of launch azimuth only. Moreover, since the geocentric radius was fixed at 6545 km (based on an equatorial altitude of 90 nmi), the path angle necessarily zero, and the speed fixed by the radius, only three parameters remained to be specified as a function of launch azimuth: geocentric latitude, longitude, and the azimuth of the inertial velocity vector at parking-orbit injection. These parameters were tabulated against launch azimuth at intervals small enough to allow linear interpolation. For a given launch azimuth, the parking-orbit injection point was immediately defined and then treated as the launch site for a boost trajectory with parking-orbit coast and second-burn phases. This method accounted for all nonplanar effects of the first-burn phase. Furthermore, as the coast and second-burn phases were truly coplanar with the translunar trajectory, a high degree of accuracy was achieved with the conic approximation.

Figure VI-8 shows the in-plane parking-orbit geometry. The required launch[5]-to-target central angle, ϕ_{LT}, had to be achieved by adjusting the coast arc, ϕ_c, as specified by:

$$\phi_c = \phi_{LT} - \phi_{B2} + \nu_I - \nu_T$$

Since the parking orbit was circular, the required coast time t_c was a linear function of coast arc ϕ_c given by:

$$t_c = 14.638 \frac{\text{sec}}{\text{deg}} \phi_c$$

[5]Recall launch was assumed to be parking-orbit injection.

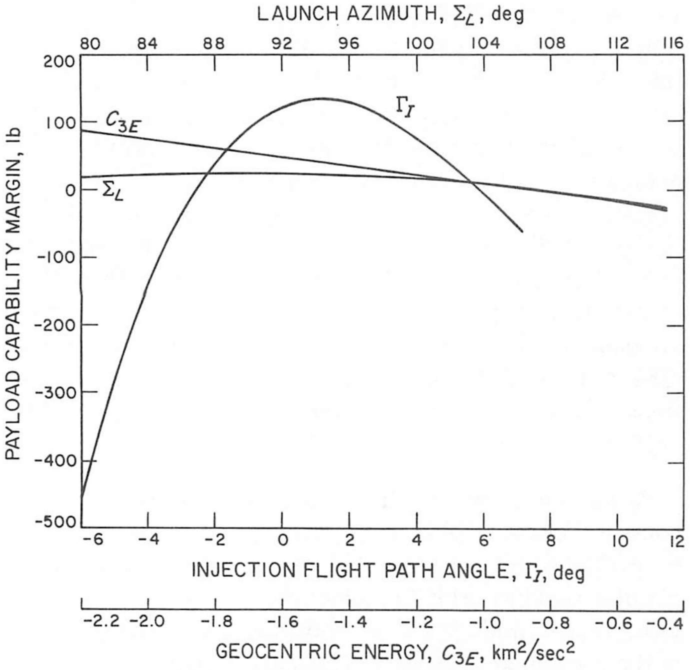


Fig. VI-7. Payload capability margin

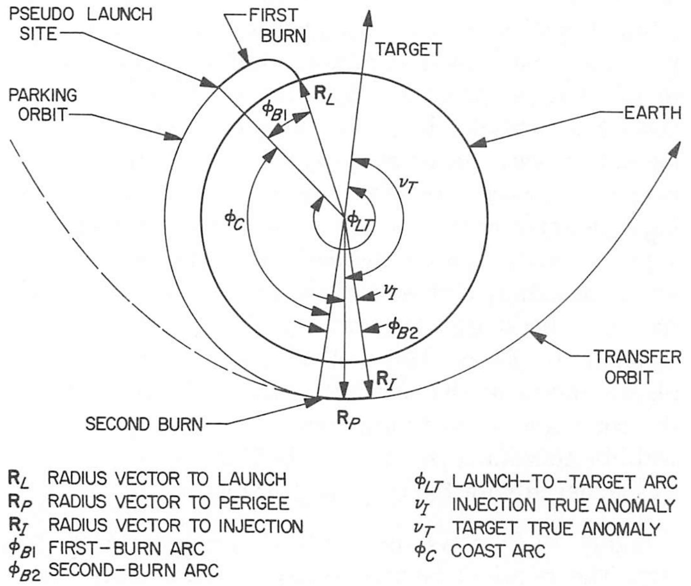


Fig. VI-8. In-plane parking-orbit ascent geometry

The second-burn arc, ϕ_{B2}, and injection true anomaly, ν_I, were essentially fixed at 9.3 and 4.9 deg, respectively, based on optimization of the second burn. To summarize, the parking-orbit boost model was used to determine the parking-orbit injection point (which was then used as a pseudo launch site) and the required coast time. Mission-design aspects of the parking-orbit ascent mode are discussed in Subsection A-3.

c. Powered-descent interface. The nominal powered-descent trajectory, which was a gravity turn, was essentially coplanar with the unbraked approach trajectory. These trajectories are shown in Fig. VI-9. The unbraked approach trajectory was completely defined (in-plane) by its impact speed and incidence angle. Consequently, unbraked impact speed and incidence angle were the only parameters required from the translunar trajectory in the design of the powered-descent trajectory. Once the ignition altitude was selected,[6] basic two-body equations were used to determine the speed and flight-path angle at ignition. The altitude, speed, and flight-path angle were then used as initial conditions for a three-degree-of-freedom (two translational, one rotational), numerically integrating, powered-flight simulation. Once the impact speed and maximum incidence angle were determined, the spacecraft ballast loading could be determined. The allowable range of ballast loadings dictated the range of allowable impact speeds, which was a constraint on the translunar trajectory design.

d. Conclusions. The conic approximation techniques and boost models described in this section were incorporated in the JPL near-earth conic computer programs, which were based largely on techniques developed during the *Ranger* Project. These computer programs proved to be highly efficient tools for generating the masses of required trajectory data with adequate accuracy. This demonstrated the wisdom of devoting the necessary resources to the development of appropriate computer tools rather than using existing overaccurate and inefficient ones. It is emphasized that the computational techniques described were not used in the final targeting. All real-time trajectory computations were done with the JPL precision, numerically integrating, *n*-body space trajectory program.

2. Ascent Mode

Surveyors I, II, and *IV* were injected into translunar trajectories via the direct-ascent mode; *Surveyors III, V, VI,* and *VII* used the parking-orbit ascent mode. The parking-orbit ascent mode was clearly superior from a mission design standpoint, since, using a parking-orbit ascent, it was geometrically possible to launch on any day of the month. The execution of a parking-orbit coast sequence introduced considerable complexity into the launch vehicle; the direct-ascent mode permitted some simplification in the launch vehicle, but severely restricted launch opportunity.

[6]Ignition altitude was determined by iteration to achieve the desired main retro burnout conditions.

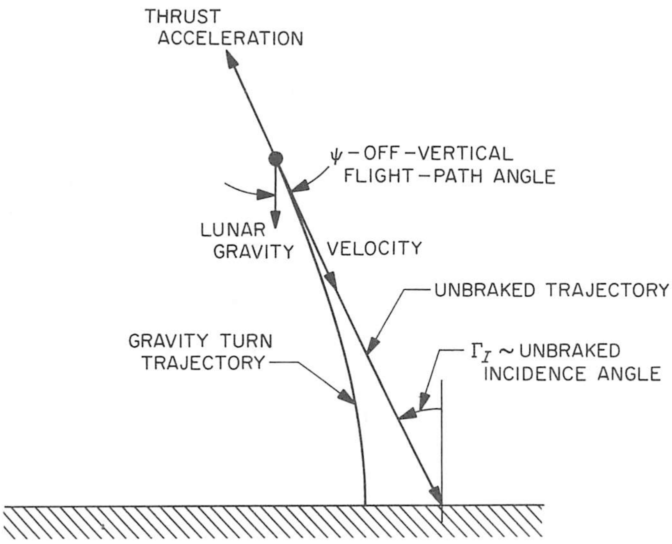


Fig. VI-9. Gravity turn descent geometry

a. Characteristics. As discussed previously, the spacecraft had to sweep out the launch-to-target central angle, ϕ_{LT}, to intercept the moon. The ϕ_{LT} is determined by launch azimuth and target declination (i.e., declination of the moon at encounter), as shown in Fig. VI-3. Once ϕ_{LT} is determined, the trajectory problem becomes essentially planar. In-plane geometry for both the direct- and parking-orbit ascent modes is presented in Fig. VI-10. Since the true anomaly of the target, ν_I, is nearly constant at 170 deg, the launch-to-perigee angle ϕ_{LP}, must be adjusted to achieve the required value of ϕ_{LT}. The value of ϕ_{LP} equals the sum of the burn arcs[7] and coast arc minus the injection true anomaly ν_I.

Atlas/Centaur payload capability was highly sensitive to the injection anomaly ν_I. For parking-orbit ascent, ν_I was fixed at the optimum value of 4.9 deg; the coast arc varied to achieve the required ϕ_{LP}. Because of the fixed duration of the propellant settling sequences, the minimum allowable coast time was 116 sec. The maximum coast time limit of 25 min was adopted as a compromise between launch opportunity and required *Centaur* operating time. These limits restricted ϕ_{LT} to the range from about 202 to 296 deg. For direct ascent, the only means of adjusting ϕ_{LP} was to vary the injection time anomaly at the expense of payload capability. As the payload (i.e., *Surveyor* spacecraft) weight was fixed during a launch period, variations in payload capability were manifested as variations in excess *Centaur* propellant. Consequently, injection true anomaly could be varied only within the range corresponding to greater than zero excess propellant. This range was from about -6 to $+14$ deg, restricting ϕ_{LT} to the range from about 184 to 204 deg.

Launch window duration was closely related to the ϕ_{LT} limits because ϕ_{LT} decreased monotonically with launch time because of the earth's rotation (see Fig. VI-3). Figure VI-11 presents the launch windows available for both direct- and parking-orbit ascent missions as a function of lunar declination at arrival. Parking-orbit launches were possible for all lunar declinations and the associated launch windows always exceeded 1 hr. Direct-ascent launch windows were available only for lunar declinations less than -16 deg.

Lunar declination varies sinusoidally, having a period of 27.3 days as shown in Fig. VI-12. During the 1966–1968 time period, the moon was achieving its near maximum excursion in declination of ± 28 deg yielding 7–9 days each cycle with declinations below -16 deg. Only during the summer and fall months did any of these days also correspond to acceptable lighting conditions at the planned landing sites.

Predicted payload capability of each *Atlas/Centaur* launch vehicle was reported by GD/C monthly, based on these mutually established ground rules:

Parameter	Direct ascent	Parking orbit
Launch azimuth, Σ_L deg	108	114
Geocentric energy C_3, km²/sec²	-0.85	-0.85
Injection true anomaly ν_I, deg	-5.0	~ 4.5
Coast time t_c, min	—	20

The set of direct-ascent parameter values was adopted as a reasonable performance goal at the time direct ascent was established as the primary ascent mode. A reported direct-ascent net margin[8] of zero implied a 7- to 8-day launch period with launch windows of 5–20 min on the first and last days and about 80 min on the other days. It is emphasized that increases in direct-ascent net margin implied increases in launch opportunity. In contrast, parking-orbit ascent launch opportunity was

[7]The burn arc for direct ascent and the sum of the two burn arcs for parking-orbit ascent were nearly constant at 28 deg.

[8]Net margin = *Atlas/Centaur* payload capability − *Surveyor* spacecraft weight.

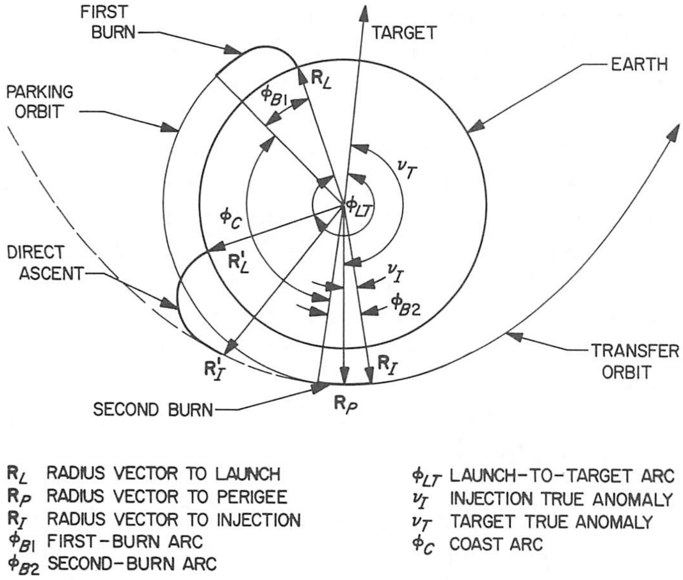


Fig. VI-10. Parking-orbit vs direct-ascent geometry

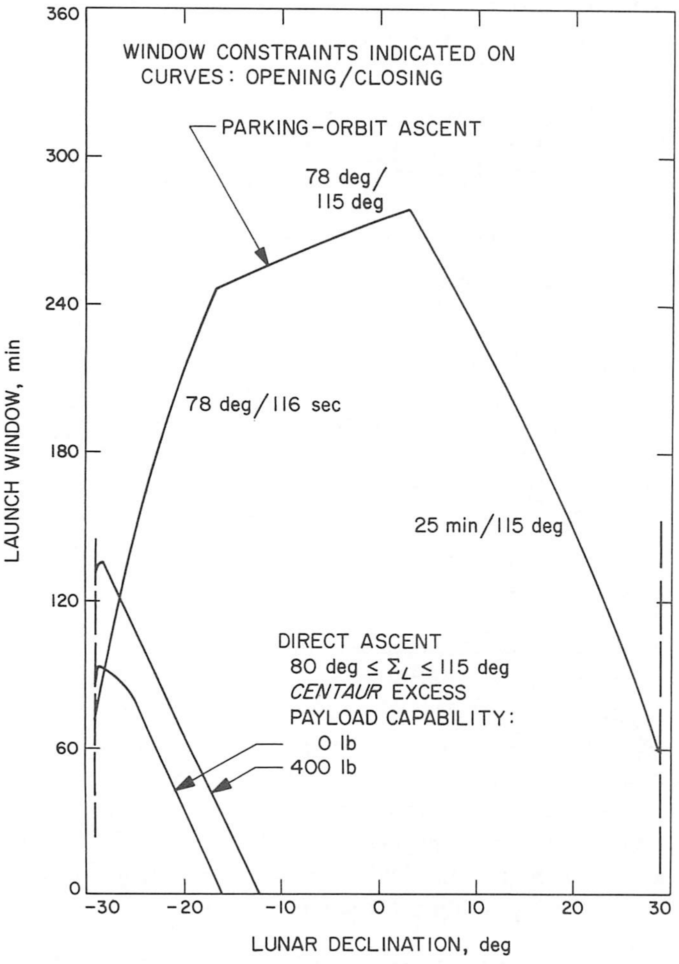


Fig. VI-11. Launch window vs lunar declination

determined by the launch azimuth and coast-time constraints. Changes in positive net margin had no effect on launch opportunity. Negative net margin implied the vehicle could not launch the *Surveyor.*

For a given *Atlas/Centaur* vehicle, the parking-orbit net margin was typically about 150 lb less than the direct-ascent net margin. This difference was primarily due to the additional *Centaur* hardware required for the parking-orbit mode. For example, if a vehicle had a parking-orbit net margin of −150 lb, parking-orbit ascent would be precluded. The same basic vehicle, however, configured for direct ascent could provide the limited direct-ascent launch opportunity.

b. Historical note. During early stages of the *Surveyor* Project, all missions were planned for the parking-orbit ascent mode. However, difficulties experienced in *Centaur* development indicated that *Centaur* would not be capable of launching a 2500- or even a 2100-lb *Surveyor* in the parking-orbit ascent mode. This dictated drastic reduction in spacecraft weight to 2100 lb and, in 1962, adoption of the direct-ascent mode for the *Surveyor I* through *IV* missions. This performance goal appeared feasible; and elimination of the parking-orbit ascent had the additional benefit of substantially reducing *Centaur* development time. Development of the parking-orbit capability, however, continued at a low level of effort in the hope that it might be available for the *Surveyor V* mission, which was to use a 2450-lb spacecraft.

In 1965, the parking-orbit ascent mode was reinstated only for the *Surveyor IV* mission with the AC-12 launch vehicle. Subsequently, parking-orbit ascent was reinstated as the prime ascent mode for the *Surveyor V, VI,* and *VII* missions. By mid-1966, schedule delays made advantageous the reassignment of the AC-12 parking-orbit vehicle to the *Surveyor III* mission and the AC-11 direct-ascent vehicle to *Surveyor IV,* which was rescheduled for a summer launch. This was the final reassignment of ascent modes.

3. Flight Time Considerations

The requirement that the spacecraft be visible from Goldstone (DSS 11) during the lunar encounter phase constrained allowable translunar flight time within certain intervals. The difference in right ascension between launch and arrival at the moon, $\Delta\Theta_{LM}$, is a function of launch azimuth and lunar declination, as shown in Fig. VI-3. The launch position must be located at the angle $\Delta\Theta_{LM}$ behind the moon's position at arrival, as indicated on Fig. VI-13. Goldstone Tracking Station is 36.31 deg in right ascension behind the launch site at

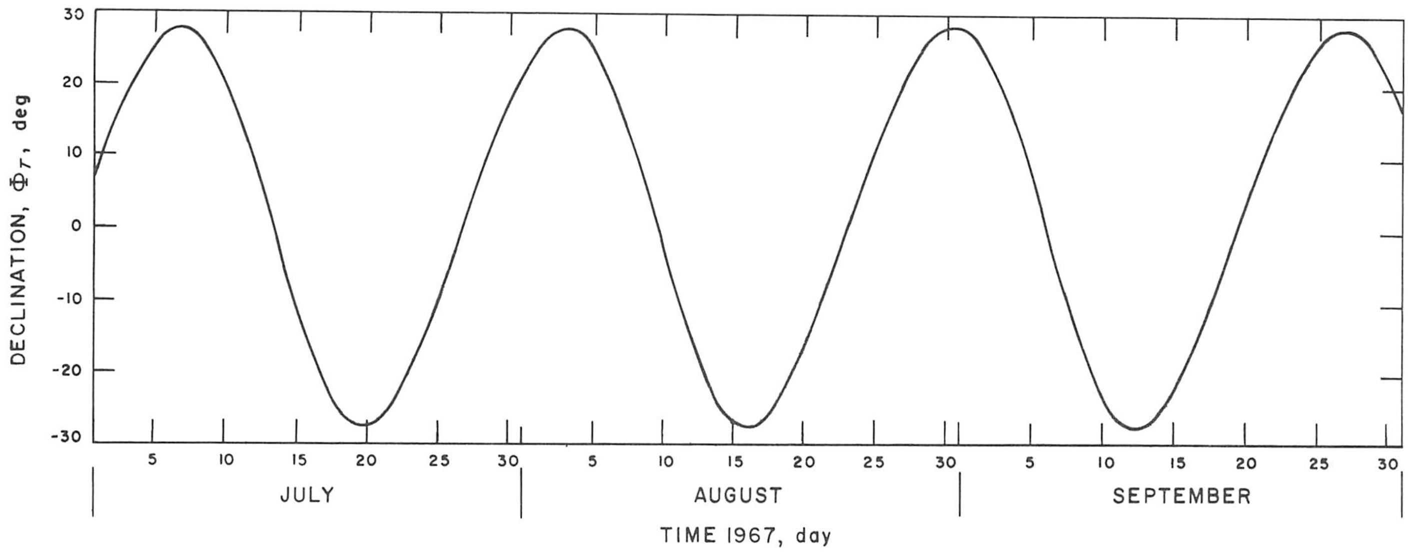


Fig. VI-12. Lunar declination vs time

the time of launch; therefore, the earth must rotate through the angle $\Delta\Theta_{LM} + 36.31 + 360N$ deg (where N is an integer) during the flight so that Goldstone will be under the moon at arrival. Since the earth's rotation rate is uniform, there is a unique correspondence between this required rotation angle and the flight time. An average value of $\Delta\Theta_{LM}$ yields flight times of 66 and 90 hr (for $N = 2$ and 3, respectively), which were of interest for *Surveyor* missions. Flight time could vary as much as 4–6 hr from these values because of the variation of $\Delta\Theta_{LM}$ and because the moon did not have to be precisely at the Goldstone meridian at arrival. Selection of the 66-hr rather than the 90-hr flight time class for *Surveyor* missions was based on the following factors:

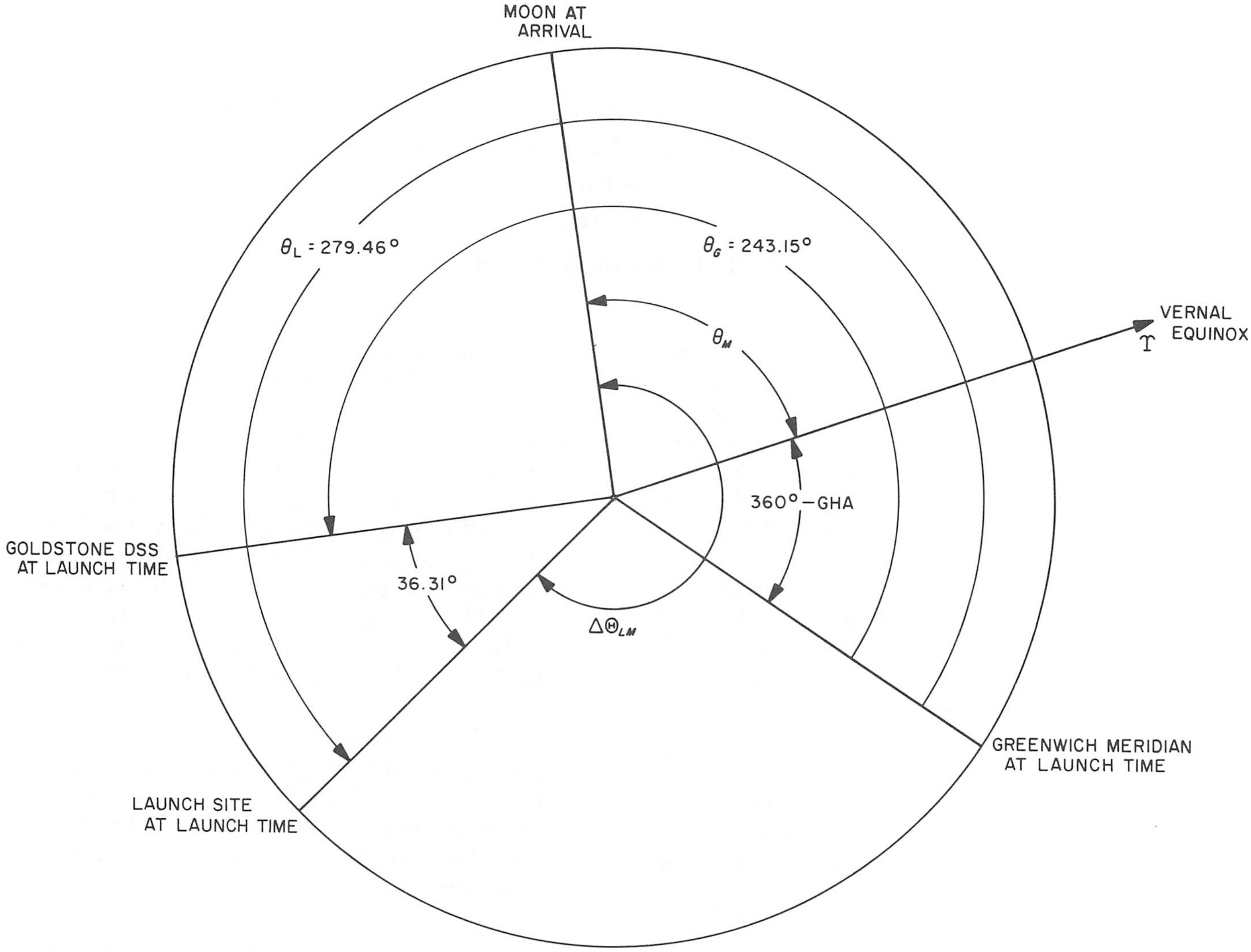
(1) *Payload.* In general, the shorter the flight time, the higher the energy requirement at each end of the trajectory. Thus, for a given *Atlas/Centaur* performance capability, the maximum weight injected into translunar orbit decreases as the flight time is shortened. Also, the fraction of this weight that can be soft-landed on the moon is lower because of the higher de-boost velocity requirement at the moon.

(2) *Accuracy.* Shorter flight time higher-energy trajectories are less sensitive to injection guidance errors (in terms of lunar miss for a given injection error) than trajectories with longer flight times. Although this effect is partially offset by a similar difference in trajectory sensitivity to midcourse corrections, the midcourse correction requirements for 90-hr trajectories were expected to be somewhat larger than for 66-hr trajectories.

(3) *Landing site accessibility.* Usually, if the flight time was increased, the vertical impact point moved westward across the lunar surface. Since the spacecraft off-vertical incidence capability was fixed at 45 deg, the eastern boundary of the accessible landing area also moved westward, diminishing available landing area. For example, use of the 90-hr flight time class would have precluded the *Surveyor V* landing in Mare Tranquillitatis.

(4) *Spacecraft performance.* Various aspects of spacecraft design and operation were directly affected by flight time class; thermal, power, reliability, and flight control performances would have been degraded with the longer flight times requiring more stringent design requirements.

At the outset of the *Surveyor* Project, the 66-hr flight time class was selected. In 1964, concern over the ability of the *Atlas/Centaur* to inject the weight of *Surveyor* into translunar orbit reopened the flight time question. Extensive recomparison of the 66- and 90-hr trajectories was performed. It was found that *Atlas/Centaur* direct-ascent payload capability could be increased by about 200 lb (maintaining the same launch opportunity) by changing to 90-hr trajectories, which, however, would have necessitated significant spacecraft changes. It was decided not to change to 90-hr trajectories because of renewed confidence in *Atlas/Centaur* performance.



$180° \leq \Delta\Theta_{LM} \leq 360°$

Fig. VI-13. Required excursion in right ascension ($\Delta\Theta_{LM}$)

4. Final Design

The trajectory selection was governed by the following ground rule:

> Trajectories shall be designed and targeted to impact the lunar surface with a constant unbraked impact speed. This speed, known as the *design impact speed,* shall remain fixed over the launch period. However, the actual impact speeds may vary over the daily launch windows when it is impossible to design trajectories to the *design impact speed* without violating the arrival time constraints. In this situation, the trajectories that would violate the time constraint shall be designed to arrive at that constraint.

a. Constraints. The constraints pertinent to the trajectory selection are presented in Table VI-1. Final launch windows were invariably less than those dictated by these constraints because of the development of launch constraints as discussed in Subsection I.

b. Design charts. To facilitate the trajectory selection, design charts were developed which displayed all the pertinent trajectory data relative to the constraints. Examples of these charts for direct-ascent and parking-orbit ascent missions are presented in Figs. VI-14 and VI-15, respectively. The charts present the unbraked impact speeds required to arrive at the early arrival constraint (i.e., 2 hr after DSS 11 rise) and the late constraint (i.e., 3 hr before DSS 11 set) as a function of launch time. Data were presented only for launch times within the launch azimuth and coast time (excess propellant for direct ascent) constraints. Since direct-ascent launches were geometrically possible only 8 or 9 days each month, data for all possible launch days were presented on the direct-ascent charts. Parking-orbit launches were geometrically possible each day. The launch dates presented on the parking-orbit charts were selected on the basis of corresponding lunar lighting conditions at all landing sites under consideration. (See Subsection A-4-c for a discussion of lighting criteria and launch periods.) Since the partial derivative of arrival time with respect to impact speed was nearly constant at about -7 min/m/sec, the pre and postlanding visibility for a given launch time and impact speed could be readily scaled from the charts. For example, Fig. VI-15 shows that an impact speed of 2630 m/sec is required to arrive at the late constraint for a launch of 08:30 GMT on launch day 3. Therefore, an impact speed of 2640 m/sec would yield about 4.2 hr of postlanding visibility (3 hr + 7 min $\times$ 10 m/sec = 4.2 hr).

Table VI-1. Trajectory design constraints

Parameter	Allowable range or value	Ascent mode: Direct	Ascent mode: Parking orbit	Reason
Launch azimuth, Σ_L, deg	78 to 115		X	Range safety
	80 to 115	X		
Parking-orbit coast time, sec	116 to 1500		X	*Centaur* propellant management; reliability
Minimum flight performance reserve (FPR), lb	235[a]	X		Ensure with high probability sufficient *Centaur* propellant
Minimum pre-landing Goldstone (DSS 11) visibility, hr	2	X	X	Allow adequate time for acquisition and commanding of terminal maneuver over DSS 11 with allowance for flight time dispersion caused by injection errors
Minimum post-landing Goldstone (DSS 11) visibility, hr	3	X	X	Allow adequate time for post-landing engineering assessment and television over DSS 11
Design impact speed range	[b]	X	X	Spacecraft propulsion and ballast capability
Allowable variation in nominal impact speed, m/sec	12		X	Allocation of midcourse maneuver capability

[a]175 lb for *Surveyor I* (AC-10).
[b]Varied with spacecraft.

The charts also present the launch azimuth and parking-orbit coast time or payload capability margin as a function of launch time. This facilitated translation of design impact speed selections into usable launch azimuth sectors and coast time intervals or payload margins. The use of these charts is discussed further in Subsection A-4-d.

c. Launch period selection. Launch periods for the parking-orbit missions were constrained primarily by the lunar lighting conditions at the landing sites under

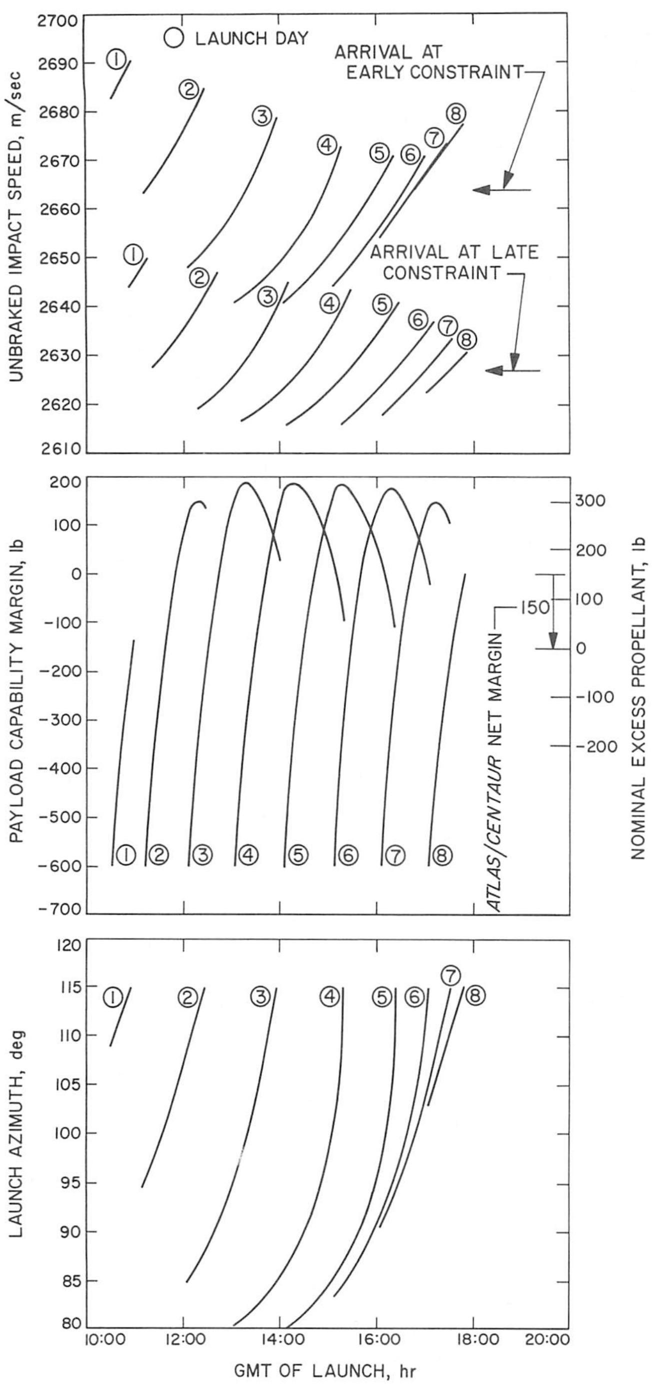


Fig. VI-14. Design chart for typical direct-ascent launch period

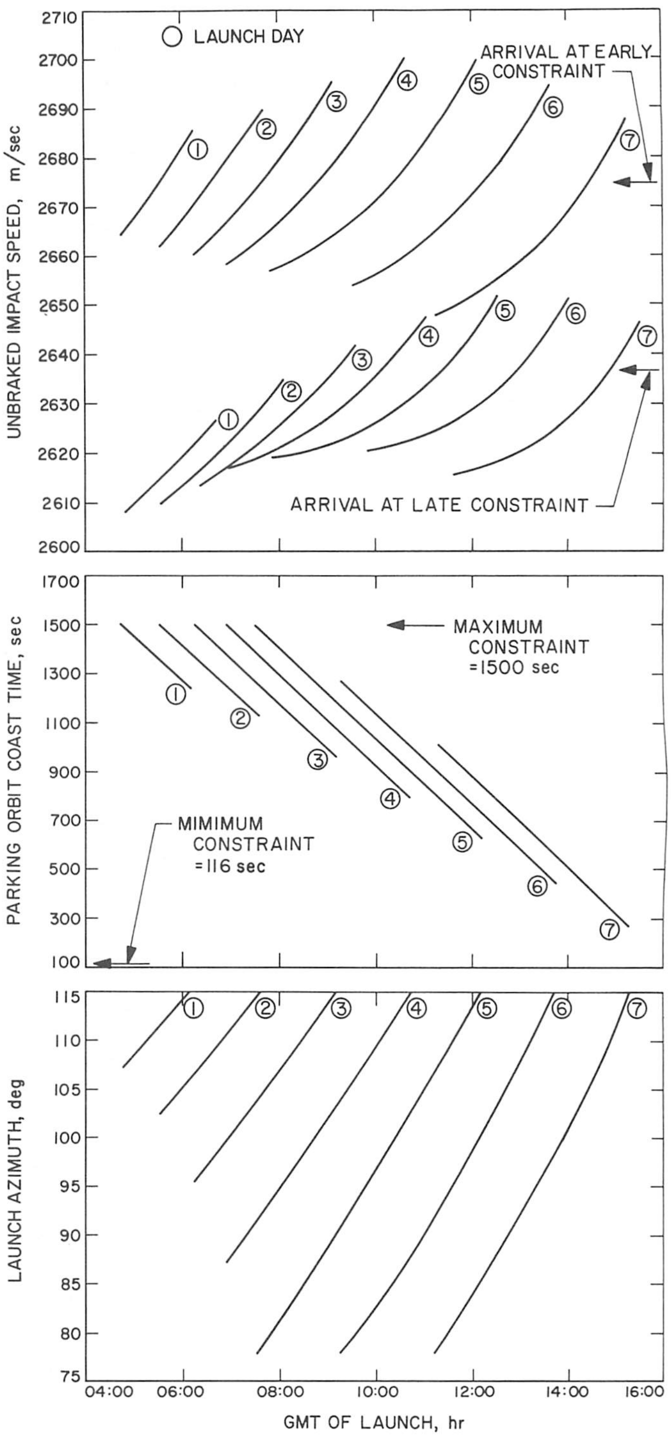


Fig. VI-15. Design chart for typical parking-orbit ascent launch period

consideration, while the direct-ascent launch periods were dictated by both lunar lighting and lunar declination. It was highly desirable to land the spacecraft as early in the lunar day as possible to maximize the lunar operating time and to obtain television pictures at valuable low-sun elevation angles.[9] It was, in fact, desirable to land immediately before dawn to study the sunrise phenomena. However, the *Surveyor* spacecraft were limited in survival capability following a pre-dawn landing, and the time of sunrise could not be predicted accurately because of uncertainty in the local slope at the planned landing site. (An adverse slope of 15 deg could delay sunrise for 30 hr.) Consequently, a mission design constraint dictated that nominal (i.e., zero slope) landings were to occur in sunlight. Compatible with the above objectives and this constraint, the first day of a launch period was the first launch date when landing at the most easterly landing site under consideration would nominally occur in sunlight.

For direct-ascent missions, the launch period included all subsequent days that were geometrically available, which yielded an 8-day period at most. However, there was no hard constraint on the length of the launch periods for the parking-orbit missions. Also, there was not sufficient launch experience to quantitatively determine the launch period length required to achieve a specified probability of launching. Project Science requested that landings be restricted to lunar morning at sun elevations below 70 deg; this would yield 6-day launch periods, which were acceptable from the software generation standpoint and appeared reasonable, based on limited *Atlas/Centaur* launch-on-time experience (see Subsection I). The 70-deg maximum sun elevation ground rule was thus adopted.

d. Design impact speed selection. Once the launch period was selected, the trajectory selection was reduced to selecting the optimum design impact speed for the launch period. At this point, however, the launch period often consisted of the superposition of launch periods for two or more landing sites, resulting in a period slightly longer than the final launch period. In some instances, this resulted in a design impact speed that was not optimum for the final launch period.

The primary consideration in the selection of this design impact speed was that it be within the spacecraft capability, which was defined by the range of design speeds that could be adequately ballasted for, based on the spacecraft weight and main retro motor capability. Selection of the design impact speed within this available range was based primarily on pre and postlanding Goldstone (DSS 11) visibility. The prelanding DSS 11 visibility was constrained to be at least 2 hr. Although the postlanding visibility was constrained to be at least 3 hr, it was desirable to maximize the postlanding DSS 11 visibility. For nominal landings, postlanding DSS 11 time was valuable in maximizing low-sun elevation television data return after the standard engineering assessment.[10] For nonstandard landings such as the *Surveyor III* mission, postlanding DSS 11 time was extremely valuable for extended engineering and television investigations of the spacecraft condition. Since postlanding DSS 11 visibility increased with increasing impact speed, the highest available impact speed was selected subject to launch window considerations.

As noted in Table VI-1, trajectories could be designed to impact at speeds as much as 12 m/sec less than the design impact speed. Therefore, when a trajectory violated the early arrival constraint at the design impact speed, it was retargeted to arrive at the arrival constraint. This procedure, known as corner targeting, is illustrated in Fig. VI-16. Any part of the launch window that required an impact speed more than 12 m/sec less than the design speed was eliminated. Consequently, selection of the design impact speed involved a tradeoff between

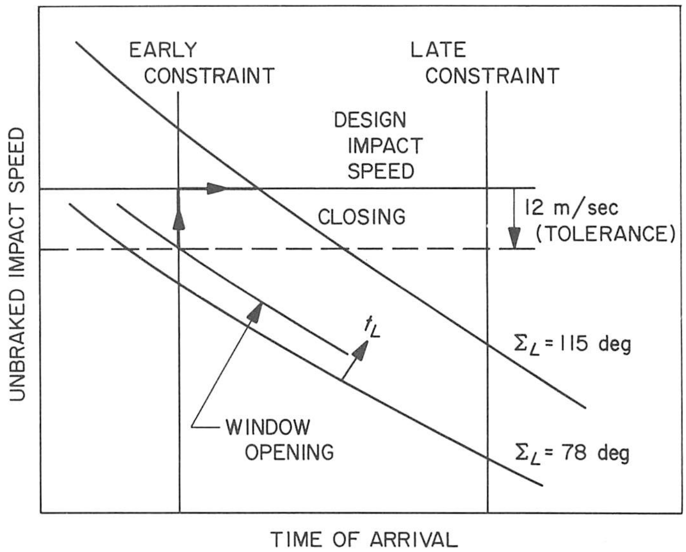


Fig. VI-16. Corner targeting

[9]The lunar day and lunar night are both approximately equal to 354 hr (except very near the lunar poles) corresponding to the east-to-west terminate movement of 0.508 deg/hr.

[10]It was not possible to obtain real-time television at the SFOF via the overseas Deep Space Stations.

postlanding visibility and launch opportunity. The criteria used in this tradeoff were: (1) reducing launch windows to less than 2 hr should be avoided, (2) postlanding visibility should be at least 5 hr. The earlier launch period days were more influential in the selection because of the high probability of launching early in the period. Preference was sometimes given to certain launch azimuth sectors because of anticipated launch constraints.

For a given launch period, several candidate design impact speeds were selected by inspection of the design charts (Figs. VI-14 and VI-15) considering all of the above factors. For each candidate speed, minimum postlanding visibility, available launch window, and usable launch azimuth sector, were tabulated for each launch date. These parameters were obtained directly from the design charts. The optimum design impact speed was determined from the tabulations and from additional inspection of the design charts.

The design impact speeds for *Surveyors I* and *II* were the lowest speeds that would satisfy the 3-hr postlanding DSS 11 visibility constraint. The importance of postlanding visibility was not fully recognized at that time. In January 1967, preliminary design impact speeds were specified for each launch period from April 1967 through February 1968. These speeds were not optimum in all cases because the numerous landing sites still under consideration necessitated consideration of launch dates subsequently eliminated. This preliminary selection was used to determine which spacecraft would require the larger A-22 main retro motor. Immediately before issuance of the target criteria for a mission, the final design impact speed was selected following the procedure given and weighing all refinements in landing site selection and trajectory constraints that occurred subsequent to the preliminary selection. The design impact speed selection is summarized below:

Spacecraft	Design impact speed, m/sec
Surveyor I	2662
Surveyor II	2663
Surveyor III	2670
Surveyor IV	2662
Surveyor V	2642
Surveyor VI	2635
Surveyor VII	2660

e. Target criteria and trajectory characteristics. Target criteria for each mission consisted primarily of the specification of the launch dates to be targeted and the design impact speed, landing site coordinates,[11] and arrival time constraints to be used. Issuance of the target criteria specification for a given mission concluded the trajectory design for that mission. Concurrent with issuance of the target specification, a memorandum was issued summarizing preliminary characteristics of the selected trajectories, based on the conic results. The following data were presented for each launch date: time of launch window opening and closing, and duration; usable azimuth sector; coast-time variation or propellant margin variation; pre and postlanding DSS 11 visibility; and, for each landing site under consideration, the sun elevation at landing and the unbraked impact incidence angle. These data were used in the development of mission sequences and support plans.

Upon receipt of the target specification, GD/C targeted the mission as described in Subsection B. To verify the targeting and provide the required precision trajectory data to JPL, HAC, LeRC, and AFETR, GD/C generated precision, numerically integrated, trajectory simulations from launch to unbraked lunar impact at launch time intervals specified in the target specification. From this point, trajectory activities were centered on the analysis of the data provided by these simulations. (Subsection C).

B. *Atlas/Centaur* Targeting

GD/C targeted the *Atlas/Centaur/Surveyor* trajectories in compliance with the targeting specifications issued by JPL. A precision closed-loop targeting technique was developed by GD/C which proved to be more accurate and less expensive than previous techniques. This technique incorporated a precision simulation of the trajectory from liftoff to unbraked lunar impact in which closed-loop control of the vehicle by the guidance system was fully modeled. This permitted iterating the guidance constants directly until the specified lunar impact conditions and other key parameter values were achieved.

For a given launch date, several discrete trajectories were targeted at nearly equal intervals of launch time throughout the launch window. Guidance constants generated for these trajectories were then curve-fit as functions of launch time. The resultant polynomials were

[11]The desired soft-landing coordinates were also used as the target coordinates for the unbraked trajectory. No attempt was made to account for the known bias between the nominal unbraked and soft-landing points in the targeting specification because this bias was negligible compared with the injection guidance errors.

known as the prelaunch polynomials and their coefficients were known as *J*-constants. The *J*-constants constituted the launch-date-dependent input to the *Centaur* guidance computer. When the "go-to-flight-mode" signal was given 9 sec before liftoff, the prelaunch polynomials were evaluated to generate the actual guidance constants to be used in the flight.

To facilitate verification of the targeting and provide the required precision trajectory data to all involved agencies, GD/C generated precision trajectory simulations from launch to unbraked lunar impact for seven launch times approximately equally spaced in each launch window. The spacecraft translunar injection conditions for these trajectories were then curve-fit as functions of launch time to obtain the injection condition polynomials, which were used extensively to define the nominal translunar trajectory for any given launch time.

The launch-date-dependent *J*-constants and launch-time-dependent launch azimuth settings were documented by GD/C in the firing tables. Variations in *Centaur* excess propellant, coast time, and the launch azimuth were presented graphically and were included in tabulations of the translunar injection conditions at 1-min intervals of launch time. Injection condition polynomials and detailed trajectory listings were also provided.

C. Final Standard Trajectories

The final standard trajectories for each mission were defined by the precision GD/C targeting. That means that the boost phase of these trajectories was generated using the GD/C precision, closed-loop guidance, powered-flight simulation and that the postinjection phase was based on the injection condition polynomials provided with the targeting. Generation and analysis of the final trajectories was necessary to fulfill various requirements. Some of these requirements are discussed in the following paragraphs.

A postinjection standard trajectories document was prepared and published by HAC for each mission. These documents presented the pertinent trajectory data in handbook form. The injection and lunar impact conditions, representative earth tracks, Deep Space Station view periods and elevation angles, and earth cone and clock angles were presented for each launch date.

To familiarize the Deep Space Stations with the trajectories and provide adequate information for initial spacecraft acquisition (assuming a near-nominal injection), a preflight nominal predictions document was generated for each mission. This document consisted of time-tagged tabulations of station observables; i.e., look angles, doppler frequencies, and their respective rates. Observables were usually tabulated for each 3 deg of launch azimuth for each launch date, which resulted in a large volume of data. Considerable efficiency was achieved by using conic trajectories defined by the injection condition polynomials in generating the predictions. These predictions were a backup to the more accurate predictions generated at the SFOF and transmitted to the stations in real-time. Initial acquisition was always accomplished using real-time predictions.

The injection condition polynomials were also used in real-time to generate nominal injection conditions for the estimated launch time immediately before liftoff. These conditions were then used to generate the nominal trajectory and the associated observable predictions; in some cases, they were used as starting conditions for the orbit-determination process.

Precision powered-flight trajectory data (position, velocity, acceleration, attitude, etc.) from liftoff through *Centaur* retro maneuver were supplied to the AFETR by GD/C on magnetic tapes in the standard AFETR 80-4 trajectory data format. Trajectories were provided for each 3 deg of launch azimuth. These tapes were processed by the AFETR to generate the AFETR tracking stations observable predictions and the range safety charts. The range safety charts displayed the flight envelopes for normally functioning vehicles and, thus, were an essential part of the vehicle destruct criteria. Magnetic tapes were also supplied to JPL where they were used to generate station observables and to obtain the data necessary to refine the boost models used in the trajectory design.

D. AFETR Flight Plan

To obtain permission to launch the *Surveyor* series of spacecraft from KSC, formal flight plan approval requests were submitted to the AFETR. Two *Surveyor* flight plan approval requests were submitted: one for the direct-ascent missions and one for the parking-orbit ascent missions. These requests were basically concerned with justifying the use of launch azimuths outside of the 93- to 111-deg launch azimuth sector, which the AFETR had approved for *Surveyor* before submission of any request.

Establishment of an approved launch azimuth sector is based on a tradeoff of range safety hazards and mission objectives. The 93- to 111-deg sector presented relatively low risk to life and property in the event of a vehicle malfunction; it did not, however, provide adequate launch opportunity for *Surveyor* spacecraft. Larger azimuth sectors were requested and approved. Although it was seldom possible to use the full approved sectors because of various launch constraints, availability of the larger sectors made possible the design of adequate launch windows within the constraints.

The 78- to 115-deg azimuth sector was requested and approved for parking-orbit missions. Permission to launch as far north as 78 deg was requested to provide for launch windows greater than 1 hr for the most negative lunar declination of about −28 deg. Southern limit of 115 deg was based on criteria stated for direct ascent.

For the direct-ascent missions, the 80- to 115-deg azimuth sector was requested and approved. This sector was highly desirable because it permitted maximum possible use of the *Atlas/Centaur,* without overflying the Bahama islands. *Atlas/Centaur* performance limitations precluded launching north of 80 deg; launching south of 115 deg could have resulted in direct overflight of the Bahamas, thus presenting an unacceptable range safety hazard.

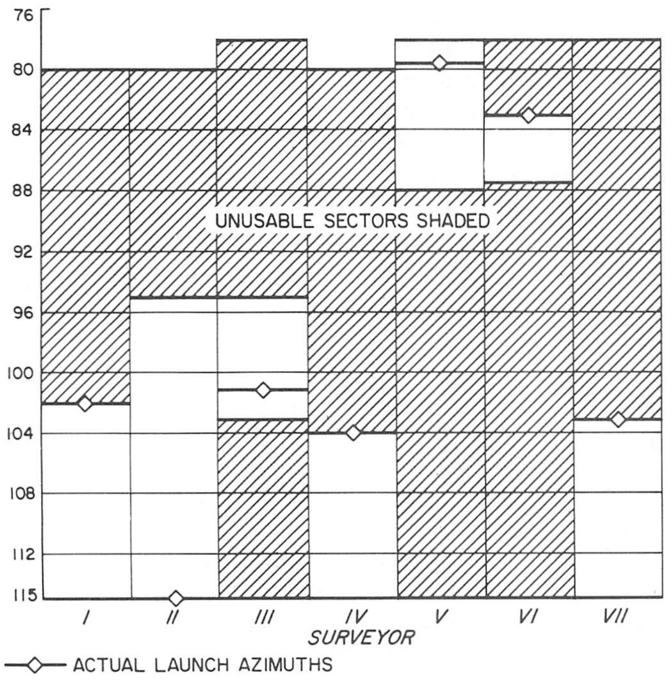


Fig. VI-17. Final launch window designs (including refinements made during countdowns) and actual launch azimuths

Figure VI-17 illustrates for each *Surveyor* launch the final launch window designs available after all constraints were applied, and the actual launch azimuth. The final designs include refinements made during countdowns.

E. Centaur/Surveyor Performance and Trajectory Working Group

The *Centaur/Surveyor* performance and trajectory working group was formed in 1963 and functioned throughout the *Surveyor* Project. Its membership consisted of representatives from JPL, LeRC, GD/C, HAC, TRW Systems, and KSC. Meetings were held approximately every 2 mo. Through this group, the member agencies gained an understanding of the problems each member had relating to vehicle performance, guidance, and trajectories. Deliberation of these problems by the group facilitated solutions that were effective from a total system viewpoint.

To provide additional insight into the group's activities, several of these problems are mentioned here. In order to determine launch vehicle performance (particularly payload capability), ground rules were established. The maximum *Centaur* parking-orbit coast time constraint was defined as a compromise between reliability and launch opportunity. The launch azimuth sector to be requested from AFETR range safety was discussed. Spacecraft heating constraints were examined to determine the proper *Centaur* nose fairing jettison time and parking-orbit altitude. The *Surveyor/Centaur* separation sequence and *Centaur* retro maneuver were established in agreement with the spacecraft Canopus sensor constraint. Various methods were developed and used to determine and monitor *Centaur* injection guidance accuracy. After considering several other methods, it was established that GD/C would target the trajectories to unbraked lunar impact conditions. Formats of the targeting specifications and firing tables were reviewed.

Two other technical working groups were closely related to the group. These were the guidance equation review board (GERB) and the range safety and planning working group (RSPWG). The GERB was responsible for the overall guidance equation development in addition to the detailed verification of the guidance constants for each mission. The RSPWG was responsible for determining and fulfilling the AFETR trajectory data requirements for the support of *Centaur/Surveyor* missions. Activities of these two groups were reported at each meeting.

A key function of the performance and trajectory working group was the establishment of the spacecraft/launch vehicle guidance and trajectory software interface schedule. The group was to ensure that all interface tasks were clearly understood and completed in a timely manner for each mission. The software schedule was compressed several times as the efficiency of performing the requisite tasks improved as the project progressed. The original schedule specified that targeting was to begin 23 wk before launch; the schedule for the last two missions specified the targeting only 15 wk before launch. The schedule for the mission-dependent software items used on the *Surveyor VI* and *VII* missions is shown in Table VI-2. Descriptions of these terms are given in Ref. VI-1. Initially, generation of software was planned for the prime and first and second backup launch periods for each mission. An accelerated schedule was developed for generating software for the third backup launch period, which was to begin immediately upon failure to launch during the prime period. Compression of the schedule achieved after a few flights permitted limiting software preparation to the prime and first backup only, with an emergency schedule to be implemented for subsequent backups, if necessary.

When hardware difficulties caused the rescheduling of launches, software generation was frequently aborted after considerable effort had been expended for a given launch period. These situations made it difficult to adhere to the software schedule. Although software items were frequently completed late, software delivery never presented any threat to the launch schedule. For the later missions, the software schedule was often compromised by late issuance of the targeting specifications, which was a consequence of delays in the landing site selections. (The landing site coordinates were part of the target criteria.) Landing site selection for *Surveyor VII* was delayed so long that the targeting was performed before the site selection. As a result, it was necessary to target to two sites, which provided access to other prime sites via the midcourse maneuver (see Subsection H).

F. Spacecraft Navigation and Guidance

1. Philosophy

To guide *Surveyor* to the desired lunar landing site, the spacecraft was commanded to execute one or more midcourse velocity correction maneuvers. These maneuvers were performed by orienting the spacecraft's thrust axis in the direction of the required velocity increment and commanding the vernier engines to burn at a fixed acceleration level for the time increment required to achieve the velocity change. Midcourse maneuvers were used to correct *Centaur* injection guidance errors and, in some cases, to change the aiming point from that targeted.

Computation of the midcourse maneuver required a highly precise determination of the pre-midcourse orbit. Clearly, errors due to the uncertainty in this orbit would not be corrected. Also, the terminal descent required that the main retro thrust be oriented relative to the approach trajectory with high precision. These requirements dictated stringent orbit determination accuracy

Table VI-2. Mission-dependent guidance and trajectory software schedule

Mission-dependent milestones	Issued by	Item	Due date, weeks before launch
Design specification: *Surveyor*/*Centaur* target criteria *Surveyor* mission X	JPL	Document	15
Rough draft of final trajectories	GD/C	Magnetic tape listings, cards	11
Airborne guidance computer tapes	GD/C	Paper tape	11
Nominal AC–Y Range Safety trajectories tape	GD/C	Magnetic tape	10
Final guidance equations and performance analysis for *Centaur* AC–Y	GD/C	Document	9
Preinjection trajectory characteristics report AC–Y	GD/C	Document	8
Range Safety trajectory data AC–Y	GD/C	Document	8
AC–Y trajectory dispersion tape	GD/C	Magnetic tape	8
Range Safety aerodynamic data for *Atlas*/*Centaur* AC–Y	GD/C	Document	8
Postinjection standard trajectories *Surveyor* mission X	HAC	Document	5
Approval of milestone 13	LeRC	TWX	5
Centaur propellant margin recommendation	LeRC	TWX	4
AC–Y firing tables	GD/C	Document	4
Approval of milestone 13	JPL	TWX	3
Launch constraints addendum	JPL	Document	3

requirements. *Surveyor* orbit determination was achieved by performing a modified least-squares fit to the DSN two-way doppler tracking data obtained throughout the flight. Orbit determination techniques were essentially those developed for the *Ranger* and *Mariner* missions.

The nominal sequence of events called for the performance of a single midcourse maneuver to be executed during the first Goldstone (DSS 11) view period, about 15 to 20 hr after launch. Following acquisition of the spacecraft signal by the Deep Space Station within 1 hr after launch, tracking data were continuously received at the SFOF and processed in the orbit determination program (**ODP**) at frequent intervals to update the estimate of the orbit. During this same time, midcourse maneuver alternatives were analyzed. About 2½ hr before maneuver execution time, the final pre-midcourse orbit estimate would be completed and used to refine the command magnitudes for the selected alternative.

After the midcourse maneuver, the orbit was redetermined to compute the spacecraft rotations for the terminal maneuver. These rotations and the ignition delay time were computed based on the orbit estimate available about 4 hr before encounter. Special procedures then were used to determine the unbraked impact time with the highest attainable accuracy to effectively back up the altitude marking radar (AMR). Orbit determination and maneuver analyses were the responsibility of the *Surveyor* flight-path analysis and command team, jointly staffed by JPL and Hughes.

2. Orbit Determination

a. Orbit determination program. The JPL single-precision orbit determination program (**SPODP**) (Ref. VI-2), the principal analysis tool used for *Surveyor* orbit determination, utilized an iterative, modified[12] least-squares technique to find, at a given spot, that set of initial conditions that causes the weighted sum of squares of the tracking data residuals[13] to be minimized. The single-precision Cowell trajectory program (**SPACE**) (Ref. VI-1) and the double-precision JPL Development Ephemeris 19 were used in conjunction with the **SPODP**.[14]

The weighted-least-squares technique used for the parameter estimates had the refinement that *a priori* information on the parameters and their statistics influenced the estimate. Basic equations are:

$$\Delta q_i = [A^T W A + \Gamma^{-1}]^{-1} [A^T W (O - C) + \Gamma^{-1} \Delta q_i]$$

and

$$q_{i+1} = q_i + \Delta q_i$$

where

q_i = the estimate of the solution parameter vector $(m \times 1)$ on the ith iteration.

A = the matrix of first order partial derivatives on each observable with respect to each solution parameter $(m \times n)$.

W = the diagonal weighting matrix formed by taking the reciprocal of the *a priori* estimated effective variance on each observable $(n \times n)$.

Γ = the *a priori* covariance matrix on the solution parameters $(m \times m)$.

$O - C$ = the vector of differences between the observed data and the calculated data $(n \times 1)$.

Δq_i = the difference between the *a priori* solution estimate and the ith iteration estimate $(m \times 1)$.

The statistics associated with the parameter estimates are given in the covariance matrix $[A^T W A + \Gamma^{-1}]^{-1}$. This expression shows that the statistics are a direct reflection of the data weights.

b. Deep Space Stations and data types. The Deep Space Stations that tracked *Surveyor* are listed in Table VI-3; those that supported the various missions are indicated. The geocentric station location varied very slightly

Table VI-3. Deep Space Station locations[a]

DSS[a]	Surveyors tracked							Geocentric radius, km	Geocentric latitude, deg	Longitude, deg
	I	*II*	*III*	*IV*	*V*	*VI*	*VII*			
11	X	X	X	X	X	X	X	6372.020	35.20822	243.15070
42	X	X	X	X	X	X	X	6371.691	−35.21942	148.98140
51	X	X	X	X	X	X	X	6375.506	−25.73926	27.68568
61			X	X	X	X	X	6370.012	40.23882	355.75110
72		X		X	X			6378.239	−7.8999	345.6736

[a]Locations given are specifically for *Surveyor III*.

[12]"Modified" indicates that the weighting of individual data types is accomplished in a different manner than the usual least-squares method.

[13]Defined as "observed minus computed $(O - C)$ values.

[14]Before *Surveyor IV*, JPL Ephemeris EPHEM-1 was used.

(less than 10 m) from mission to mission because of polar motion. The locations given are those that were used for the *Surveyor III* mission.

The Deep Space Stations obtain both angle and doppler data. Angle data are simply the antenna pointing angles (hour angle and declination) for all stations except DSS 72 at Ascension island. DSS 72 has an azimuth/elevation type mount. There were three types of doppler data obtained from *Surveyor*: (1) one-way doppler, where the spacecraft receives no signal, but transmits its signal to ground stations operating in receive mode only, (2) two-way doppler, where a ground station transmits a reference signal, which is received and retransmitted to the same ground station by the spacecraft, and (3) three-way doppler, where a ground station receives the signal the spacecraft has received from another ground station and retransmitted. Because all other data types are much less accurate, all orbit estimates, upon which commands were based, were obtained using two-way doppler data only.

Figure VI-18 shows the tracking coverage for *Surveyor II* and intervals during which the various data types were obtained. In this respect, *Surveyor III* is representative of all *Surveyors*.

c. Data weighting, error sources, and sample rates. The philosophy used for weighting data in the **SPODP** was to calculate a weight value based on the effective (or expected) variance of a given data type. The effective variance for a given data type was determined by summing up the variances caused by all known error sources.

For two-way doppler data, error sources were divided into two general classes: (1) hardware or station equipment errors, and (2) software; i.e., computing and model errors. For the first class of errors, such items were considered as transmitter reference oscillator stability, doppler counter roundoff error or quantization, and doppler counter error caused by dropped or added cycles in the presence of a low signal-to-noise ratio. Of these items, the major contributor is counter quantization error, which was estimated to be 0.0011 m/sec for a data sample rate of 60 sec. For the second class of errors, certain model errors exist that are not adequately accounted for in the **SPODP** and are not sufficiently known that they may be reflected in the effective variance. Among these are planetary and earth–moon ephemerides errors. Planetary ephemerides errors are negligible for a lunar trajectory, but earth–moon ephemerides errors affect such quantities as predicted unbraked impact time. This was evidenced by the fact that the predicted impact time varied as more near-moon tracking data were included in the orbit solution. The error in the refraction correction model used to correct low-elevation data contributed a maximum of 1.07×10^{-4} m/sec for a 60-sec sample rate. In the **ODP**, statistics are based upon single-phase data weights modified by an empirical refraction

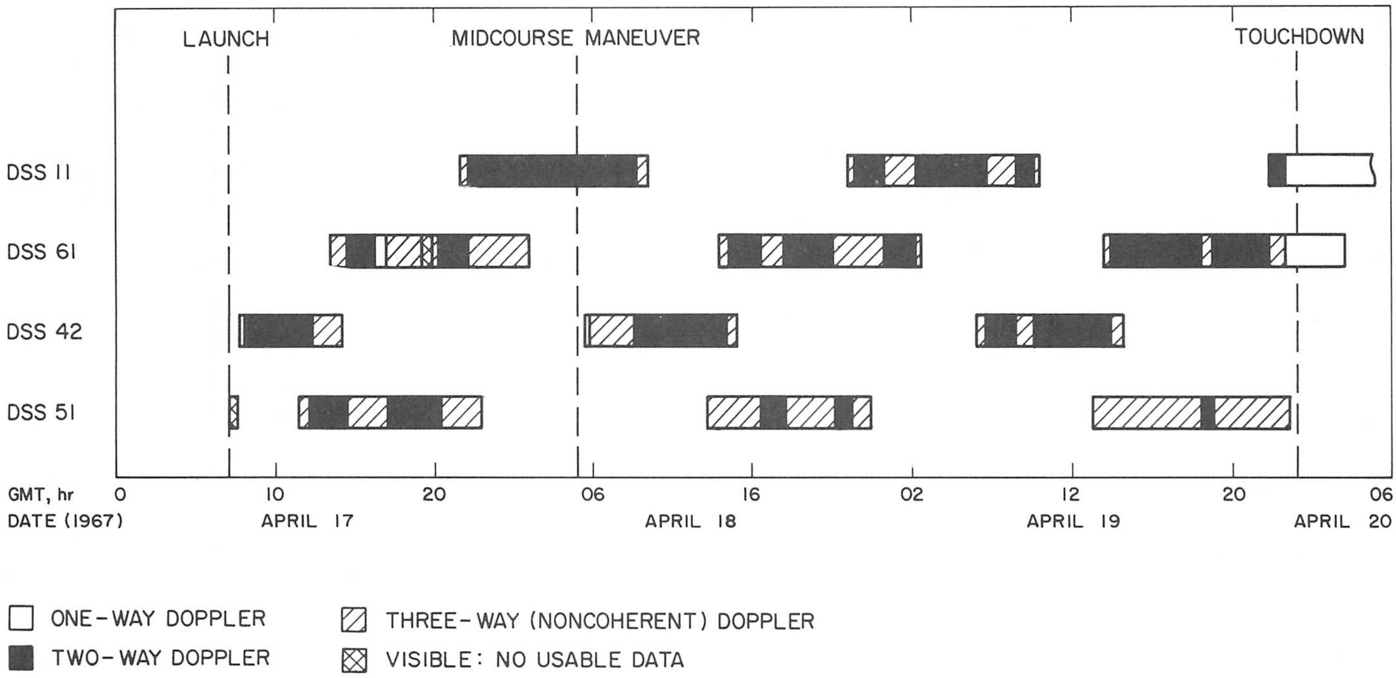


Fig. VI-18. Tracking coverage for *Surveyor III*

formula to account for varying elevation angles. Computing errors incurred within the program were the major contributors to the two-way doppler data weight. These errors (about 0.012 m/sec for a 60-sec sample rate) occurred because most of the computations were made in single precision, which results in interpolation errors and a buildup of roundoff errors. Based on the above error sources mentioned, effective two-way doppler data weight is 0.013 m/sec, which corresponds to 0.2 Hz for S-band stations.

Error sources associated with angle data are:

(1) Angle jitter or variation about the aiming point caused by antenna drive servomechanisms.

(2) Angle correction errors caused by differences between the empirical correction model, which is based on the antenna optical axis and the RF pointing axis.

(3) Angular encoder readout errors caused by inaccuracies in the compensation cams. Resolution of the encoder is plus or minus one count which corresponds to 0.002 deg.

(4) Refraction correction errors caused by the difference between the atmospheric model used in the **SPODP** and the actual atmosphere at a given time.

Of these, dominant error sources are angle correction errors that contribute an estimated variance of 0.033 deg^2 for a sample rate of 60 sec. Based on this, an effective data weight of 0.18 deg was used for angle data. Biases usually remain after corrections are applied to angle data. Since the applied corrections are based on optical star tracking, the biases are probably caused by differences between the antenna optical and RF axes. Efforts are now underway to use RF sources such as postlanding *Surveyor* tracking to improve the corrections. Even though the applied corrections did not completely remove the systematic pointing errors, corrected angle data were extremely valuable in converging to an orbit solution during the early part of the missions. However, angle data were omitted from the orbit solutions as soon as enough two-way doppler data were available to obtain a good solution.

The data sample spacing used at the tracking stations was determined by the tradeoff between doppler counter rounding error and truncation errors occurring in the doppler frequency computations. Sample spacing had to be reduced during two phases of flight: (1) near-earth and (2) during the midcourse maneuver. For these phases, a sample spacing of 10 sec was used. A spacing of 60 sec was used at all other times.

d. Data editing. The JPL tracking data processor (**TDP**) and orbit data generator (**ODG**) programs[15] were used to edit all incoming tracking data and to prepare data files for input to the **SPODP**. Data points are first read into the **TDP**, which checks each data sample for acceptable format; i.e., it checks to determine whether the sample is one of 30 acceptable message formats, whether each item is in the proper field, and whether any item contains a missing or illegal character. During flight operations, time does not permit reconstruction of data points rejected for bad format. The next item checked by the **TDP** is the data-condition code. A data point is given a bad-data-condition code when automatic detectors at the station sense that the data would be unusable. These detectors have manual overrides that are used whenever an equipment malfunction is suspected and during periods when the transmitter is being returned before transferring transmitting assignment to another station. A course in range value check is made by the **TDP** to determine whether each data type is within an acceptable limit; i.e., 360 deg for angles and 10^4 cycles for doppler. All data that passed these checks and were not rejected by a user option were time-sorted and written on disk and magnetic tape for access by the **ODG**. The **ODG** reads the data file and, if it includes angular data from DSS 42 or DSS 51, the values are corrected to remove systematic antenna-pointing errors. The doppler data were then checked for monotonicity, valid tracking mode, valid sample rate, and converted from cycles to cycles/sec by differencing adjacent samples and dividing by the sample time. Pertinent transmitter and receiver frequencies are entered on the file with each doppler sample. (These frequencies were read in by the user or, in some formats, were included with the data sample.) The data were then written on disk and magnetic tape for access by the **SPODP**.

Blunder points are the data points rejected by the **TDP** and **ODG** during validity checks, or by applying the user rejection limits during the orbit computation. These limits were based on experience gained in previous missions, and on the philosophy that it is better to immediately reject questionable points, which could create difficulties in converging to an orbit, than to attempt to salvage every point. This is particularly true when few data are available during the early phase of the mission.

[15] R. E. Holzman, *User's Guide to the Tracking Data Processor and Orbit Data Generator Programs*, Reorder 62-605, Pasadena, Calif., May 27, 1965.

e. In-flight computational sequence and solution types. During each flight, the orbit solution was periodically updated as new tracking data became available. Since the computers were heavily loaded throughout most of the mission, type of orbit solution was held to a minimum. That is, the number of parameters estimated in a solution had to be restricted to the minimum set that would still allow the orbit determination accuracy goals to be met. In general, estimating only the position and velocity ("the standard six") of the spacecraft at a given epoch was the best compromise between accuracy and computer time for in-flight *Surveyor* orbit determination. Improved physical constants and station location parameter solutions obtained from the *Ranger* block III and *Mariner II* and *IV* tracking data were used as nominal values.

In the pre-midcourse maneuver phase, all orbit solutions were obtained by estimating the standard six parameters only. After midcourse maneuver execution, all pre-midcourse tracking data, from initial acquisition until start of maneuver roll turn, were used to obtain a best estimate pre-midcourse 6×6 orbit solution. The state vector (probe position and velocity) at injection epoch was integrated forward to the end of midcourse motor burn and incremented by the commanded midcourse velocity change. The resulting vector was then used as the initial estimate of the spacecraft postmidcourse orbit.

During the postmidcourse maneuver phase from end of midcourse motor burn until approximately lunar encounter minus 4 hr, the orbit solutions were based on estimating only the standard six parameters. Spacecraft terminal attitude maneuvers were based upon 6×6 orbit solutions that included data from end of motor burn to approximately 4 hr before lunar encounter. To provide an effective backup for the *Surveyor* AMR the orbit solution type had to be changed during the last few hours of each mission. The type of orbit solution used and the computation of the AMR backup command are discussed in Subsection F-6.

f. Orbit determination accuracy. Orbit determination errors and midcourse maneuver execution errors are principal contributors to landing site error. Preflight and inflight orbit determination accuracy could only be treated statistically; consequently, it was expressed in terms of the standard deviations (i.e., 1σ errors) of the key target parameters. Preflight accuracy estimates were required to achieve a compatible spacecraft system design and to establish reasonably achievable mission goals. Preflight estimates were made of both the "guaranteed" and the "expected" accuracies and documented. Table VI-4 summarized these accuracies for the midcourse maneuver situations of primary interest. The "guaranteed" accuracies, used for designing spacecraft capability and simulating worst-case conditions, were based on assumed data biases and some loss of crucial early data. The expected accuracies, used to estimate nominal total system performance, incorporated no data losses or biases. Orbit determination accuracy is decided principally by the data weights and the amount of data used in the solution.

In-flight orbit determination activities focused on obtaining the most accurate orbit estimates possible in the available time. Postflight analysis allows more extensive processing of tracking data and results in the best possible estimate of the orbit, based on all usable tracking data. Comparison of the inflight and postflight orbit estimates provides one of the best measures of the orbit determination function performance. This comparison is

Table VI-4. Preflight orbit determination accuracy estimates

Midcourse time (approximate time from launch), hr	Direct-ascent missions (lunar declination: < -15 deg)				Parking-orbit ascent missions (lunar declination: > -15 deg)			
	1σ uncertainty ellipse in B-space at midcourse, km $\times$ km		1σ impact time uncertainty 40 min before encounter, sec		$1\ \sigma$ impact time uncertainty B-space at midcourse, km $\times$ km		$1\ \sigma$ impact time uncertainty 40 min before encounter, sec	
	Expected	Guaranteed	Expected	Guaranteed	Expected	Guaranteed	Expected	Guaranteed
15	5×15	20×50	0.5	0.7	2×9	3×14	0.6	0.7
40	2×3	2×4	0.5	0.7	2×4	2×5	0.6	0.8
15/40[a]	3×4	4×7	0.5	0.7	2×6	4×14	0.6	0.8

[a]Midcourses at 20 and 45 hr; uncertainties quoted for the 40-hr midcourse.

presented in Table VI-5. For the postmidcourse estimate, additional comparisons are provided by postlanding tracking solutions and correlation of *Lunar Orbiter* and *Surveyor* pictures (see Subsection F-5).

References VI-3, VI-4, and VI-5 contain detailed discussions of the *Surveyor* orbit determination analyses and results.

3. Midcourse Maneuver

a. Introduction. To compensate for *Centaur* injection guidance errors, the *Surveyor* spacecraft was designed with a midcourse correction capability. Spatial miss at the target can be reduced to two components: (1) the component of miss in the plane tangent to the lunar surface at the target, referred to as "miss only"; and (2) the component normal to this plane, referred to as "time-of-flight" error. Original estimates of *Centaur* guidance accuracy indicated that a 50-m/sec correction capability would be required to have a 3σ miss correction probability at the nominal midcourse time (15–20 hr after launch). Since previous experience indicated that miss only and time-of-flight errors were highly correlated, a 50-m/sec midcourse capability was established as a design requirement for incidence angles not exceeding 25 deg. Later information indicated that correcting miss plus time of flight could require significantly more capability than correcting miss only; however, confidence in the *Centaur* guidance accuracy had increased sufficiently

Table VI-5. Comparison of inflight and postflight orbit determination results

Mission	Computuation	Premidcourse			Postmidcourse[a]		
		B_{TT}, km	B_{RT}, km	t_{IM}, hr:min:sec, msec	B_{TT}, km	B_{RT}, km	t_{IM}, hr:min:sec, msec
Surveyor I	Inflight solution	−671.1	73.6	05:29:10.724	−111.9	−392.6	06:15:14.519
	Postflight solution	−657.2	68.0	05:28:59.537	−113.0	−395.6	06:15:15.066
	Inflight −postflight	−13.9	5.6	11.187	1.1	3.0	0.547
	Inflight 1σ uncertainty[b]	2.2 × 10.4 km, 40.0 deg		7.628	3.3 × 0.8 km, 98.8 deg		0.44
Surveyor II	Inflight solution	1331.5	6.8	03:19:55.424			
	Postflight solution	1333.8	5.2	03:19:54.223	*Surveyor II* failed at midcourse		
	Inflight −postflight	−2.3	1.6	1.201			
	Inflight 1σ uncertainty	2.2 × 25.1 km, 57.5 deg		7.214			
Surveyor III	Inflight solution	815.6	−97.8	23:58:16.856	1469.4	−390.5	00:01:47.984
	Postflight solution	817.0	−97.4	23:58:16.297	1469.5	−388.7	00:01:48.158
	Inflight −postflight	−1.4	−0.4	0.559	−0.1	−1.8	−0.174
	Inflight 1σ uncertainty	1.8 × 6.6 km, 71.4 deg		1.767	0.6 × 2.6 km, 82.9 deg		0.55
Surveyor IV	Inflight solution	1768.0	−238.7	02:11:42.145	1952.4	−341.9	02:02:30.397
	Postflight solution	1762.4	−236.1	02:11:44.824	1954.3	−340.5	02:02:31.171
	Inflight −postflight	5.6	−2.6		−1.9	−1.4	
	Inflight 1σ uncertainty	1.2 × 2.1 km, 35.0 deg		0.822	2.3 × 11.3 km, 87.2 deg		0.47
Surveyor V[c]	Inflight solution	2894.9	−219.7	23:25:13.907	2992.4	−177.5	00:45:15.162
	Postflight solution	2893.4	−217.8	23:25:14.318	2991.5	−177.8	00:45:15.318
	Inflight −postflight	1.5	−1.9	−0.411	0.9	0.3	−0.156
	Inflight 1σ uncertainty	3.7 × 8.8 km, 78.0 deg		1.710	2.2 × 7.8 km, 79.5 deg		0.559
Surveyor VI	Inflight solution	1753.3	439.0	00:35:42.987	1692.2	257.6	00:58:32.972
	Postflight solution	1752.4	440.9	00:35:43.638	1691.3	258.1	00:58:32:855
	Inflight −postflight	0.9	−1.9	−0.651	0.9	−0.5	0.117
	Inflight 1σ uncertainty	4.1 × 11.1 km, 95.3 deg		1.840	8.2 × 14.6 km, 116.5 deg		0.503
Surveyor VII	Inflight solution	2044.2	359.0	01:02:53.534	1036.7	2012.2	01:02:47.700
	Postflight solution	2044.8	362.7	01:02:52.983	1034.7	2014.0	01:02:47.914
	Inflight −postflight	−0.6	−3.7	0.551	2.0	−1.8	−0.214
	Inflight 1σ uncertainty	16.7 × 40.0 km, 112.7 deg		5.630	5.5 × 11.7 km, 68.2 deg		0.700

[a] Inflight *B*-space estimate from orbit used for terminal maneuver computation; inflight t_{IM} estimate from orbit used to compute altitude marking radar backup time.

[b] Uncertainty given in terms of 1σ dispersion ellipse in *B*-space. Orientation angle is measured from *T*-axis counterclockwise to semimajor axis of ellipse.

[c] Pre-midcourse: prefirst midcourse, post-midcourse: postfinal midcourse.

to indicate that 50 m/sec would be more than adequate to correct both *Centaur* miss and time-of-flight errors. Miss plus time of flight represented an upper bound on midcourse requirement since it was planned to correct the time-of-flight error only to the extent that optimized the terminal parameters.

Several of the later spacecraft had midcourse capabilities significantly less than 50 m/sec because of weight growth and increase of the incidence angle requirement to 45 deg. These losses in capability were inconsequential because of the *Centaur* guidance accuracy improvements. Before the *Surveyor VI* flight, LeRC guaranteed the 3σ *Centaur* guidance accuracy to be 30 m/sec for miss plus time of flight. This was a major factor in allowing the preflight allocation of 13 m/sec midcourse capability to the large landing site change (about 1200 km on the lunar surface) made in flight with *Surveyor VII*. As expected, the correction required to achieve the landing site change was an order of magnitude greater than would have been required to correct the *Centaur* injection error.

The midcourse maneuver and terminal descent were intimately coupled. Midcourse maneuver analysis consisted largely of the optimization of critical terminal-descent parameters.

Before each mission, the more probable maneuver situations were analyzed parametrically for each planned launch date; the results were documented. Also documented are system parameter values used in midcourse and terminal guidance programs during the mission.

b. Computational techniques. The miss parameter **B** was used to measure miss distances at the moon because it is almost a linear function of velocity at any point along the translunar trajectory, thus permitting extensive use of linear mapping techniques. The osculating conic either at impact or at closest approach to the moon is used in defining **B**. **B** is the vector from the moon's center to the incoming asymptote of the hyperbolic approach trajectory (Fig. VI-19). **S** is a unit vector in the direction of the incoming asymptote. The plane normal to **S** is referred to as the "*B*-plane." The orientation of **B** in the *B*-plane (or "*B*-space" as it is frequently called) is described in terms of the unit vectors **R** and **T**, normal to **S**. **T** is taken parallel to a fixed reference plane (either the moon's equator or the earth's equator) and **R** completes the right-handed orthogonal system, *R–S–T*. **B** was usually expressed in terms of its components, $\mathbf{B}\cdot\mathbf{T}$ and $\mathbf{B}\cdot\mathbf{R}$. The *R–S–T* system is used for all JPL space projects.

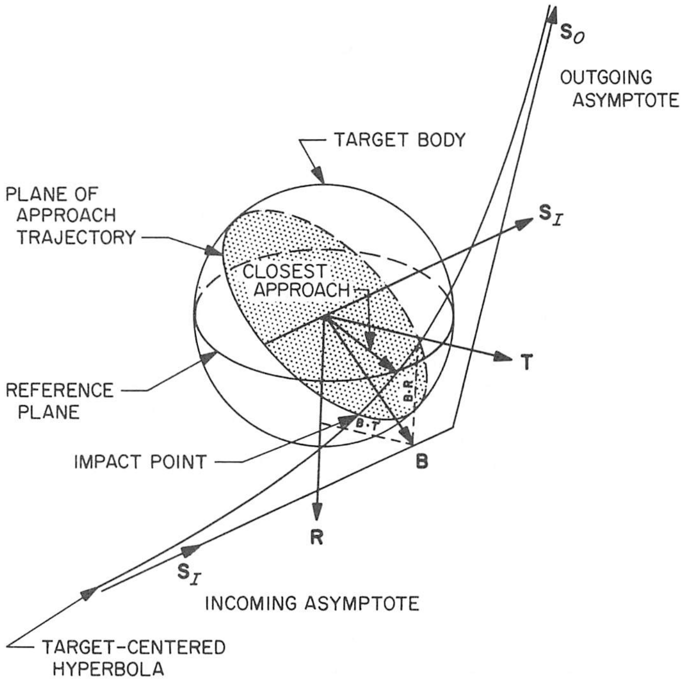


Fig. VI-19. Definition of $\mathbf{B}\cdot\mathbf{T}$, $\mathbf{B}\cdot\mathbf{R}$ system

Since **S** is essentially invariant with midcourse velocity correction, values of $\mathbf{B}\cdot\mathbf{T}$ and $\mathbf{B}\cdot\mathbf{R}$ that correspond to given achievable values of selenographic latitude and longitude are uniquely determined once the impact speed is specified. A typical mapping from selenographic coordinates to *B*-space is given in Fig. VI-20. The nonlinearity between selenographic coordinates and *B*-space, illustrated in this figure, precluded targeting directly to selenographic coordinates.

For calculation of velocity correction, linear perturbation theory was used to relate changes in target parameters to midcourse velocity increment. For example,

$$\Delta\mathbf{B}\cdot\mathbf{T} = \frac{\partial\mathbf{B}\cdot\mathbf{T}}{\partial\mathbf{x}}\,\Delta\mathbf{x} + \frac{\partial\mathbf{B}\cdot\mathbf{T}}{\partial\mathbf{y}}\,\Delta\mathbf{y} + \frac{\partial\mathbf{B}\cdot\mathbf{T}}{\partial\mathbf{z}}\,\Delta\mathbf{z}$$

or introducing the concept of the gradient vector in velocity space

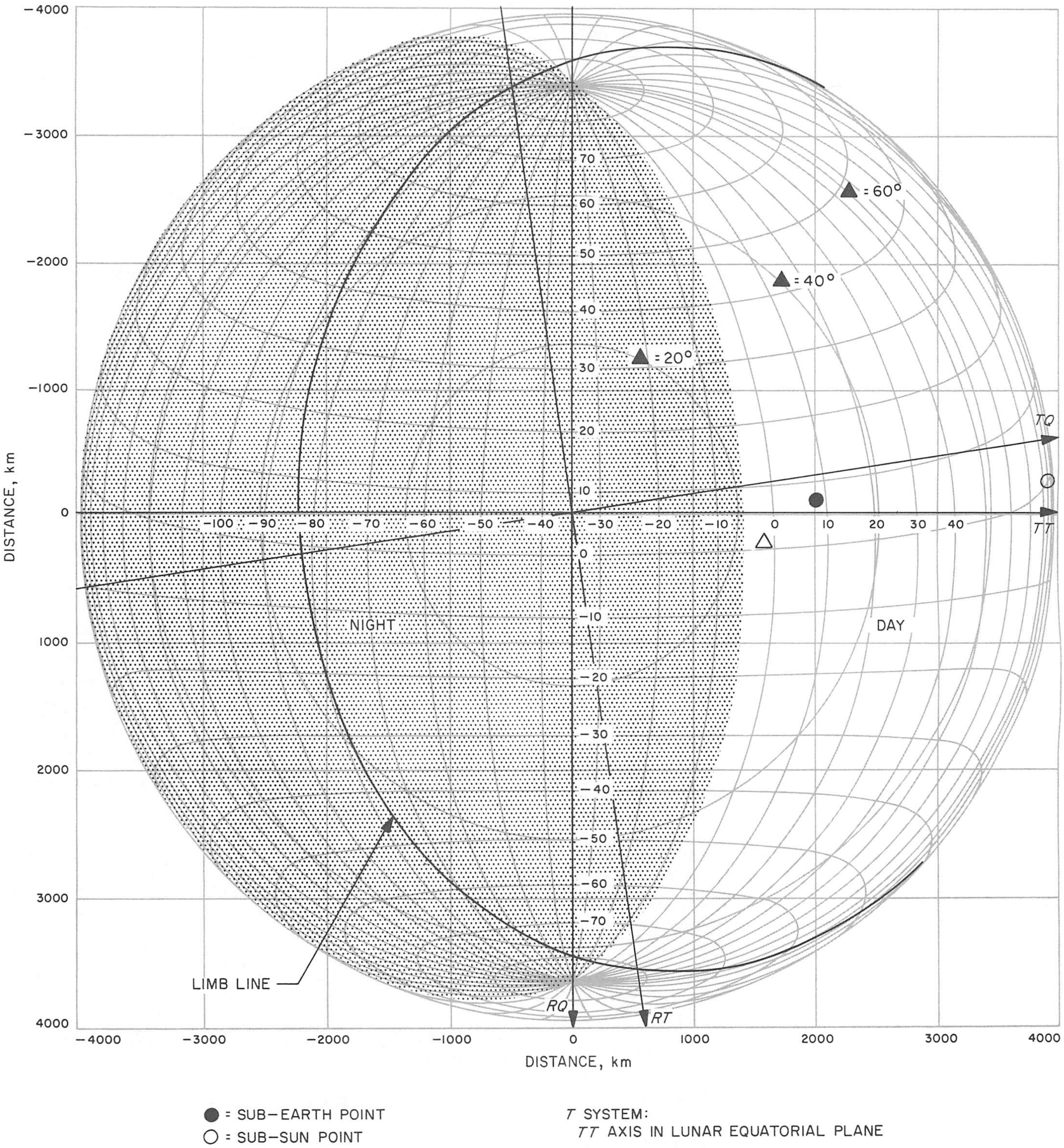
$$\Delta\mathbf{B}\cdot\mathbf{T} = \nabla\mathbf{B}\cdot\mathbf{T}\cdot\Delta V$$

and similarly

$$\Delta\mathbf{B}\cdot\mathbf{R} = \nabla\mathbf{B}\cdot\mathbf{R}\cdot\Delta V$$

$$\Delta V_{IM} = \nabla V_{IM}\cdot\Delta V$$

$$\Delta TF = \nabla TF\cdot\Delta V$$



● = SUB–EARTH POINT
○ = SUB–SUN POINT
▲ = UNBRAKED IMPACT ANGLE, Γ
△ = AIMING POINT

T SYSTEM:
TT AXIS IN LUNAR EQUATORIAL PLANE

Q SYSTEM:
TQ AXIS IN EARTH EQUATORIAL PLANE

ANGLE BETWEEN *T* AND *Q* SYSTEMS: −8.27 deg

IMPACT VELOCITY = 2.634 km/sec

Fig. VI-20. Moon in *B*-space for typical trajectory

where

$$V_{IM} = \text{impact speed}$$

$$TF = \text{time of flight}$$

The change in $\mathbf{B} \cdot \mathbf{T}$ is maximum if ΔV and $\nabla \mathbf{B} \cdot \mathbf{T}$ are collinear and zero if they are perpendicular. The most efficient means of correcting miss only (i.e., $\mathbf{B} \cdot \mathbf{T}$ and $\mathbf{B} \cdot \mathbf{R}$) is to confine ΔV to the plane of $\nabla \mathbf{B} \cdot \mathbf{T}$ and $\nabla \mathbf{B} \cdot \mathbf{R}$. This plane, known as the critical plane, is defined by its normal vector, the noncritical direction:

$$\dot{\mathbf{U}}_3 = \frac{\nabla \mathbf{B} \cdot \mathbf{T} \times \nabla \mathbf{B} \cdot \mathbf{R}}{|\nabla \mathbf{B} \cdot \mathbf{T} \times \nabla \mathbf{B} \cdot \mathbf{R}|}$$

The sense of $\dot{\mathbf{U}}_3$ is established so that $\nabla TF \cdot \dot{\mathbf{U}}_3 > 0$. The component of the midcourse velocity increment along $\dot{\mathbf{U}}_3$ is called $\dot{\mathbf{U}}_3$ and produces no first order change in the miss parameter, $\mathbf{B}$. The component in the critical plane (ΔV_{CR}) required to correct the miss was uniquely defined by the following:

$$\Delta V_{CR} = -\mathbf{K}^{-1} \Delta \mathbf{B}$$

where

$$\mathbf{K} = \begin{bmatrix} \dfrac{\partial \mathbf{B} \cdot \mathbf{T}}{\partial x} & \dfrac{\partial \mathbf{B} \cdot \mathbf{T}}{\partial y} & \dfrac{\partial \mathbf{B} \cdot \mathbf{T}}{\partial z} \\ \dfrac{\partial \mathbf{B} \cdot \mathbf{R}}{\partial x} & \dfrac{\partial \mathbf{B} \cdot \mathbf{R}}{\partial y} & \dfrac{\partial \mathbf{B} \cdot \mathbf{R}}{\partial z} \\ \dot{\mathbf{U}}_{3_x} & \dot{\mathbf{U}}_{3_y} & \dot{\mathbf{U}}_{3_z} \end{bmatrix}$$

$$\Delta \mathbf{B} = \begin{bmatrix} \Delta \mathbf{B} \cdot \mathbf{T} \\ \Delta \mathbf{B} \cdot \mathbf{R} \\ [0] \end{bmatrix}$$

(i.e., the miss to be corrected)

The $\mathbf{K}$ matrix was determined by numerical differencing. An iterative procedure was used to determine ΔV_{CR} in order to overcome errors inherent in the linear analysis.

The value of the noncritical component, $\dot{\mathbf{U}}_3$, was selected to optimize the following terminal parameters, maximizing the probability of a successful landing: main retro burnout velocity, vernier propellant margin, flight time, and landing accuracy. To select the optimum $\dot{\mathbf{U}}_3$, these four parameters were automatically plotted as a function of $\dot{\mathbf{U}}_3$ over the $\dot{\mathbf{U}}_3$ range of interest. Figure VI-21 is typical of these plots, known as $\dot{\mathbf{U}}_3$ scans. Note that landing accuracy was expressed in terms of 99% circular miss dispersion (CMD)[16] on the lunar surface. A $\dot{\mathbf{U}}_3$ scan was generated by evaluating the four terminal parameters at many discrete values of $\dot{\mathbf{U}}_3$.

Flight time and impact speed for a given $\dot{\mathbf{U}}_3$ were determined by linear mapping. Impact speed, incidence angle,[17] and the midcourse propellant consumption were then used in an analytic terminal-descent model to compute main retro burnout velocity and vernier propellant margin. Vernier propellant margin was defined as the amount of usable propellant that would be onboard the spacecraft after a nominal descent; consequently, dispersions were not considered in its determination. Dispersion allowances were provided for by the required propellant reserve, which was the amount of propellant required to have 99% probability of sufficient propellant considering all dispersions minus the amount required for 50% (i.e., the nominal consumption). The required propellant reserve was determined by extensive mission-independent Monte Carlo analyses and found to be primarily a function of burnout velocity and incidence angle as shown in Fig. VI-22. When the propellant margin equalled the required propellant reserve, the probability of having enough propellant was 99%. Detailed descriptions of the terminal guidance and propellant analysis techniques are contained in Ref. VI-6.

The 3σ midcourse execution errors were expressed in a covariance matrix determined by the error sources and the total velocity correction corresponding to the given $\dot{\mathbf{U}}_3$. This covariance matrix was mapped linearly to B-space and then added to the 3σ orbit determination B-space error covariance matrix. The 3σ total error ellipse (midcourse execution plus orbit determination errors) in B-space was determined from the resulting matrix and then mapped to the lunar surface. The equivalent 99% CMD was obtained by a table lookup that assumed surface errors to be normally distributed.

The rapid approximation techniques (i.e., linear mapping and analytic terminal-descent model) used in generating the $\dot{\mathbf{U}}_3$ scans were invaluable to the *Surveyor* guidance analysis. Once the value of $\dot{\mathbf{U}}_3$ was selected, the trajectory perturbed by the total midcourse velocity

[16]99% CMD was the radius of the circle within which 99% of all cases would lie.

[17]Incidence angle was obtained from the final trajectory run in the determination of ΔV_{CR}.

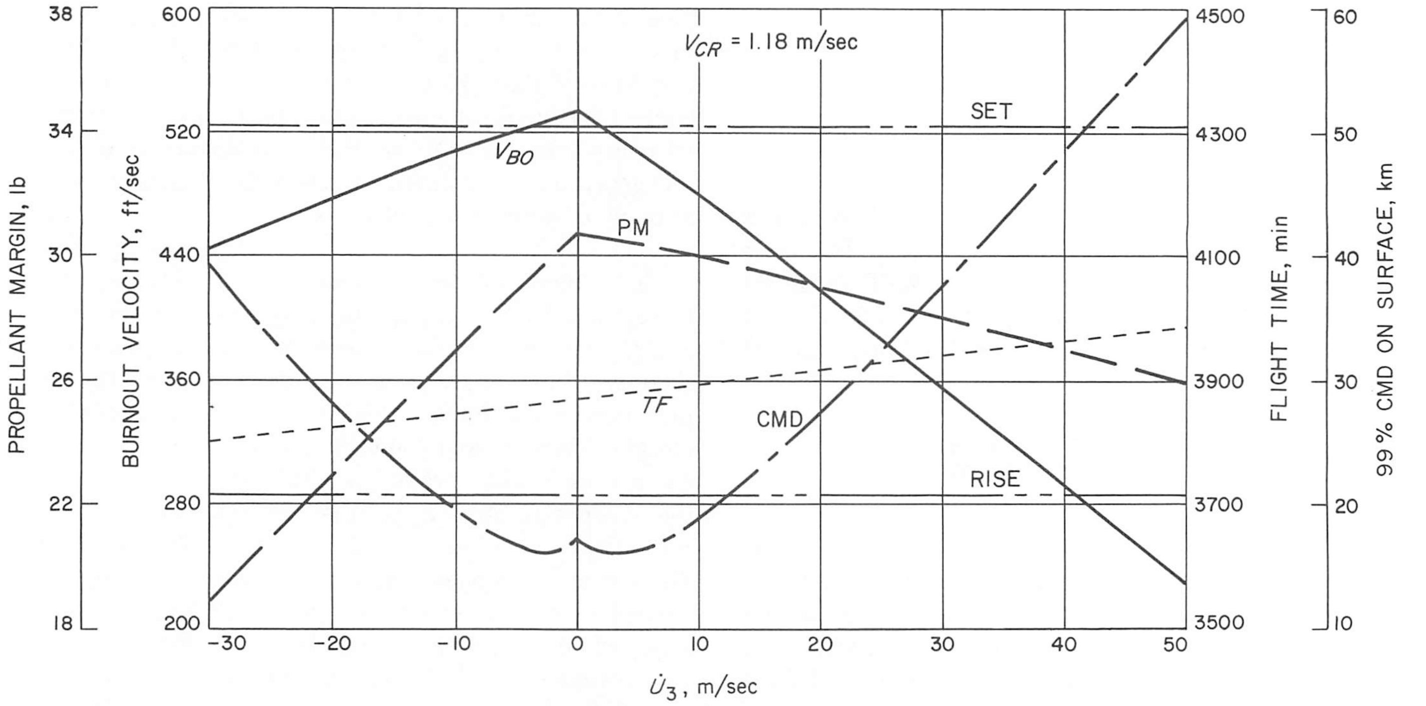


Fig. VI-21. Typical $\dot{\mathbf{U}}_3$ scan

correction was numerically integrated to unbraked lunar impact. The resulting impact speed and incidence angle were then used in a planar, three-degree-of-freedom, numerically integrating, powered-flight program to obtain a precision simulation of the nominal terminal descent. This precision simulation of the flight from midcourse to soft-landing provided an overall check on the selected midcourse velocity correction.

c. Capability. As previously stated, a 50-m/sec maximum midcourse capability was a design requirement for the *Surveyor* spacecraft. The predicted midcourse capability of each spacecraft was monitored closely and reported monthly by HAC. Actual inflight midcourse capability often differed from the best preflight prediction because of factors such as nonnominal impact speed resulting from injection dispersions. Inflight capability was determined from the $\dot{\mathbf{U}}_3$ scans.

Figure VI-23 presents the terminal-descent range velocity phase plane for an unbraked impact incidence angle of 25 deg. The nominal burnout locus, command-descent segments, RADVS constraints, and 3σ burnout dispersion contours are shown. The RADVS constraints restricted nominal burnout velocity to the 250 (280)[18] to 525-ft/sec range. Effect of the midcourse was to reduce spacecraft weight, resulting in an increase in the retro motor velocity increment and, consequently, a decrease in burnout velocity. Thirty-six feet per second of the allowable burnout velocity range were allocated to variations in the nominal targeted unbraked impact speed. The remaining 239 (209) ft/sec could be translated into midcourse capability subject to vernier propellant sufficiency. All spacecraft were sufficiently heavy so that this translation provided more than a 50-m/sec capability; however, vernier propellant availability limited the midcourse capability to less than 50 m/sec for *Surveyors III, IV, V,* and *VII*.

A zero minimum midcourse requirement was specified for *Surveyors I* through *IV* to provide capability to perform a mission without a midcourse maneuver. No such requirement existed for *Surveyors V* through *VII*. Center of gravity offset and weight increases on *Surveyors V* through *VII* due to payload changes resulted in predictions of possible loss of attitude control at descent segment intercept from vernier engine thrust saturation. A nonzero minimum midcourse was thus specified for these spacecraft to reduce the weight enough at midcourse to preclude thrust saturation during terminal descent. Before the *Surveyor IV* mission, however, the tag values of the thrust acceleration limiters and vernier engines were determined to preclude thrust saturation, even if no midcourse were executed, and the zero minimum midcourse capability was reinstated for *Surveyors*

[18]Numbers in parenthesis are for $\Gamma_{IM} = 45$ deg.

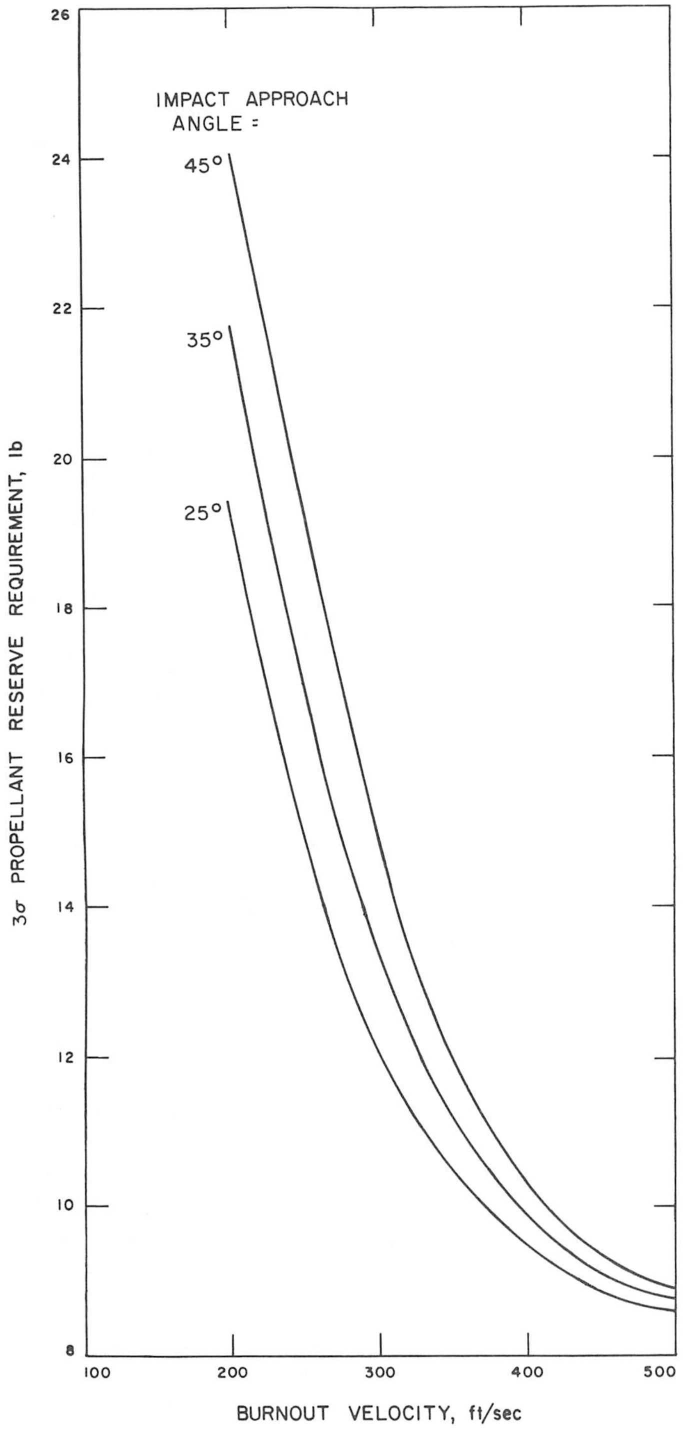


Fig. VI-22. Vernier propellant reserve

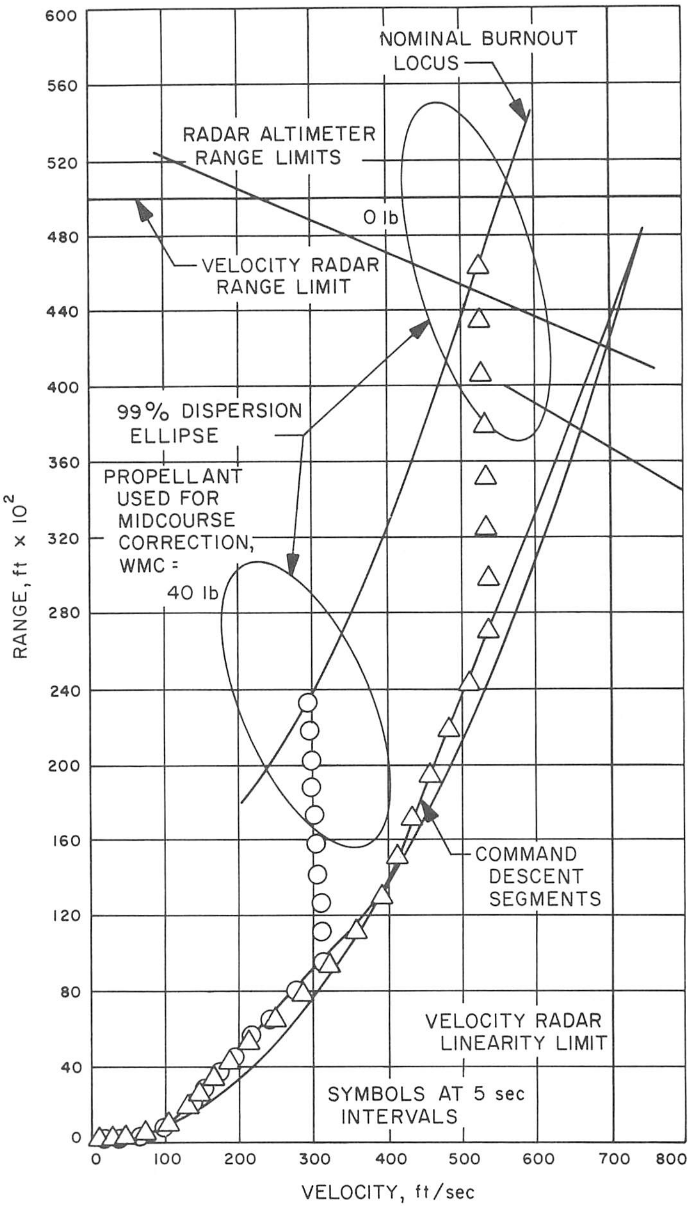


Fig. VI-23. Vernier descent phase (unbraked impact angle: 24 deg)

VI and *VII*. To provide more flexibility in landed roll orientation, however, it was specified that if a midcourse were performed, its magnitude must be sufficient to reduce nominal burnout velocity to 500 ft/sec and thus, avoid instability associated with high burnout velocities and helium saturation of the vernier propellant. Table VI-6 summarizes the midcourse capability of each spacecraft as known immediately before flight.

d. Accuracy. Midcourse maneuver execution error consisted of magnitude and pointing errors, as shown in Fig. VI-24. The magnitude error included both bias and proportional components. The contributions of the various error sources to the overall errors shown in Fig. VI-24 are given in Ref. VI-6 and are also discussed in Section XIII. The magnitude and pointing errors produced an ellipsoidal error distribution. The resulting two-dimensional error distribution in the critical plane was of primary concern, since it contributed to landing site error.

Table VI-6. Midcourse capability[a]

Parameter	Mission						
	Surveyor I	Surveyor II	Surveyor III	Surveyor IV	Surveyor V	Surveyor VI	Surveyor VII
Maximum incidence angle,[b] deg	7	25	26	36	49	25	36
Midcourse capability, m/sec							
Maximum	50	50	50	46	46	47	40
Minimum	0	0	—[c]	0	14	0	0

[a]As known immediately prior to launch and based on maximum incidence angle shown.
[b]Maximum incidence angle required to land at desired landing site during the prime launch period. Not to be confused with the spacecraft design requirement.
[c]Not available.

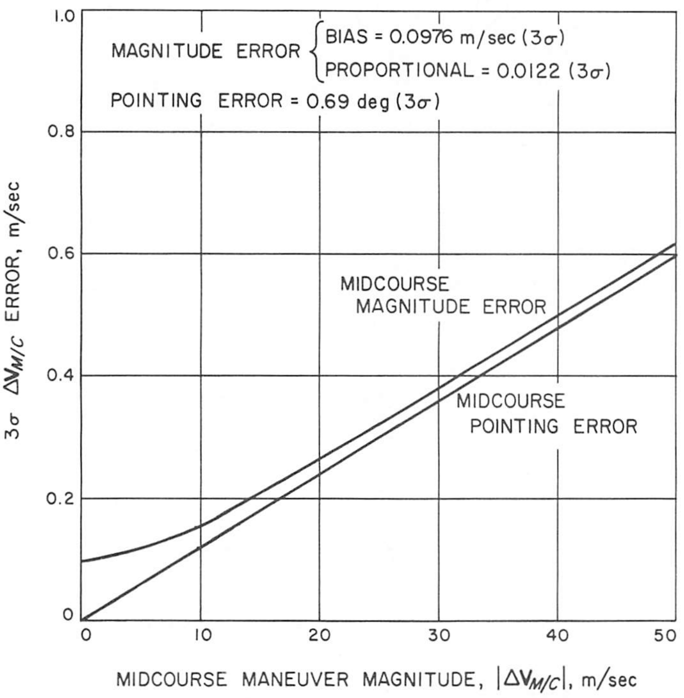


Fig. VI-24. Midcourse execution error

At first, preflight error analysis was simplified by assuming a spherical error distribution with a radius equal to the magnitude error, which was conservative, but quite realistic for midcourse magnitudes greater than about 10 m/sec. The spherical distribution mapped to a circular distribution in the critical plane. As additional emphasis was placed on landing accuracy and the expected critical plane velocity correction diminished to about 5 m/sec, the ellipsoidal error distribution was used in preflight analyses. A significant reduction in critical plane error was achieved by introducing a $\dot{U}_3$ component to the correction when the critical plane correction was small. This is illustrated in Fig. VI-21. Addition of $\dot{U}_3$ rotates the total velocity correction and the relatively large bias error out of the critical plane, thereby reducing critical plane error until $\dot{U}_3$ becomes large enough to cause the pointing error to exceed the magnitude error. As discussed previously, inflight error analysis utilized the actual execution error covariance matrix.

Several error compensation techniques were introduced as the project progressed. Measured misalignment between the vernier thrust, the spacecraft Z-axis, and inflight measured gyro drift rates was compensated for by adjusting the maneuver rotation magnitudes. The selected value of $\dot{U}_3$ was also adjusted slightly to eliminate timer quantization error.

e. Selection of maneuver. Since the *Surveyor* trajectories were designed for lunar encounter to occur during the third Goldstone view period following launch, two opportunities occurred to command a midcourse maneuver from Goldstone. Executing the maneuver during the first Goldstone view period resulted in maneuver times of 15–23 hr after launch; execution during the second view period resulted in maneuvers 38–45 hr after launch. These maneuver alternatives were simply referred to as 15- and 40-hr maneuvers, respectively. Since the orbit determination uncertainty decreased with time, greater landing accuracy could be achieved with a 40-hr maneuver. The 15-hr maneuver was favored, because it allowed additional time to overcome anomalies that might develop as a result of the maneuver. During a mission, the 15- and 40-hr maneuver characteristics were each presented to the Mission Director who, after considering all facets of the mission, ultimately decided on either the 15- or 40-hr maneuver.

For a given maneuver time, the critical plane velocity correction required to correct the miss was uniquely determined. The noncritical component, $\dot{U}_3$, could be adjusted to optimize the terminal parameters. Although, initially, an effort was made to mathematically model this optimization, the variety of factors affecting each mission made this impractical. In practice the $\dot{U}_3$ was selected on the basis of engineering judgment. This was done with the aid of $\dot{U}_3$ scans of the key terminal parameters shown in Fig. VI-21. The critical plane correction was usually small (< 5 m/sec), so sufficient $\dot{U}_3$ was introduced to reduce burnout velocity below 500 ft/sec, providing additional flight control and RADVS margins. Generally, negative values of $\dot{U}_3$ were selected to provide additional postlanding Goldstone visibility except when the negative value would violate the prelanding visibility constraint. However, absolute value of $\dot{U}_3$, large enough to jeopardize landing accuracy goals, was never selected. Only on *Surveyor V* was propellant margin influential in selecting $\dot{U}_3$.

Once the desired velocity correction was determined, the various two-rotation attitude maneuvers that would orient the thrust axis in the direction of the velocity correction were computed. The omnidirectional gain histories for each maneuver pair were then computed and analyzed. Selection of the optimum maneuver pair was based primarily on the following considerations:

(1) A roll-yaw maneuver pair was preferred because it yielded the best accuracy and thermal environment.

(2) It was desirable to remain on one omniantenna during the maneuver; however, any required antenna switching was to be performed between, rather than during, the rotations.

*f. **Results.*** Table VI-7 gives the commanded midcourse velocity correction for each mission and an estimate of the achieved execution accuracy. The accuracy estimates are based on the best postflight determination of the pre- and postmidcourse orbits.

4. Terminal-Descent Control Parameters

The terminal-descent control parameters that could be adjusted until just prior to descent were ignition altitude, thrust bias angle, and vernier thrust level used during main retro burn. Main retro burnout velocity and propellant margin were determined by the midcourse maneuver selection assuming the standard 200-lb vernier thrust level during main retro burn. Subsequent to midcourse, these parameters could be changed by commanding an alternate 150-lb thrust level.[19] With burnout velocity resolved, the ignition altitude required for nominal burnout to occur on the burnout locus (Fig. VI-23) was uniquely determined. Since propellant margin was usually more than adequate, ignition altitude was increased slightly ($\sim$1000 ft) to provide additional margin against falling below a command descent segment.

[19]Only *Surveyor V* utilized the 150-lb thrust level.

Table VI-7. Commanded midcourse velocity increments and estimated errors

Mission	Commanded midcourse velocity increment, m/sec			Estimated error[a]	
	V_{CR}	U_3	ΔV	Magnitude, m/sec	Pointing, deg
Surveyor I	3.74	20.00	20.35	−0.11	0.29
Surveyor II	1.18	9.50	9.59	—[b]	—[b]
Surveyor III	4.19	0.00	4.19	−0.04	0.10
Surveyor IV	2.47	−10.00	10.27	−0.05	0.31
Surveyor V[c]	3.87	3.60	5.32	0.04	2.42
Surveyor VI	1.18	10.00	10.06	0.06	0.26
Surveyor VII	11.05	0.80	11.08	0.02	0.04

[a]Based on difference of best pre-midcourse and post-midcourse orbit estimates. The 1σ uncertainties associated with these determinations of midcourse velocity errors are of the same order as the errors themselves. However, these determinations have particular merit because of their independence of the spacecraft system.

[b]*Surveyor II* failed at midcourse.

[c]Final maneuver.

The thrust bias angle was the nominal angle between the retro thrust axis and the velocity vector at ignition. The value of this angle ($\sim$1 deg max) was selected to minimize the effect of flight-path angle dispersions at main retro burnout.

5. Landing Accuracy

Landing accuracy was resolved primarily by premidcourse orbit determination errors and midcourse maneuver execution errors, which combined as independent error sources. It was expressed in terms of probability ellipses both in *B*-space and on the lunar surface. Frequently, these ellipses were replaced by equivalent error circles to facilitate analysis. The accuracy achievable in *B*-space was independent of landing site location. Since the focusing from *B*-space to the lunar surface reduced with increasing incidence angle (Fig. VI-20), landing accuracy depended upon incidence angle and on landing site location. For example, if landing accuracy was 30 km for 0 deg off vertical incidence, it would degrade to 45 km at 45 deg off vertical.

During the site selection for *Surveyor I*, it was determined (based on the orbit determination and midcourse execution accuracies discussed previously, and on the expected *Centaur* injection accuracy) that 50-km radius landing sites were compatible with *Surveyor* capabilities. Several 50-km sites were identified; eventually, three of these were assigned to *Surveyors I, II,* and *III*. After *Surveyor II*, increasing pressures for flexibility in landing

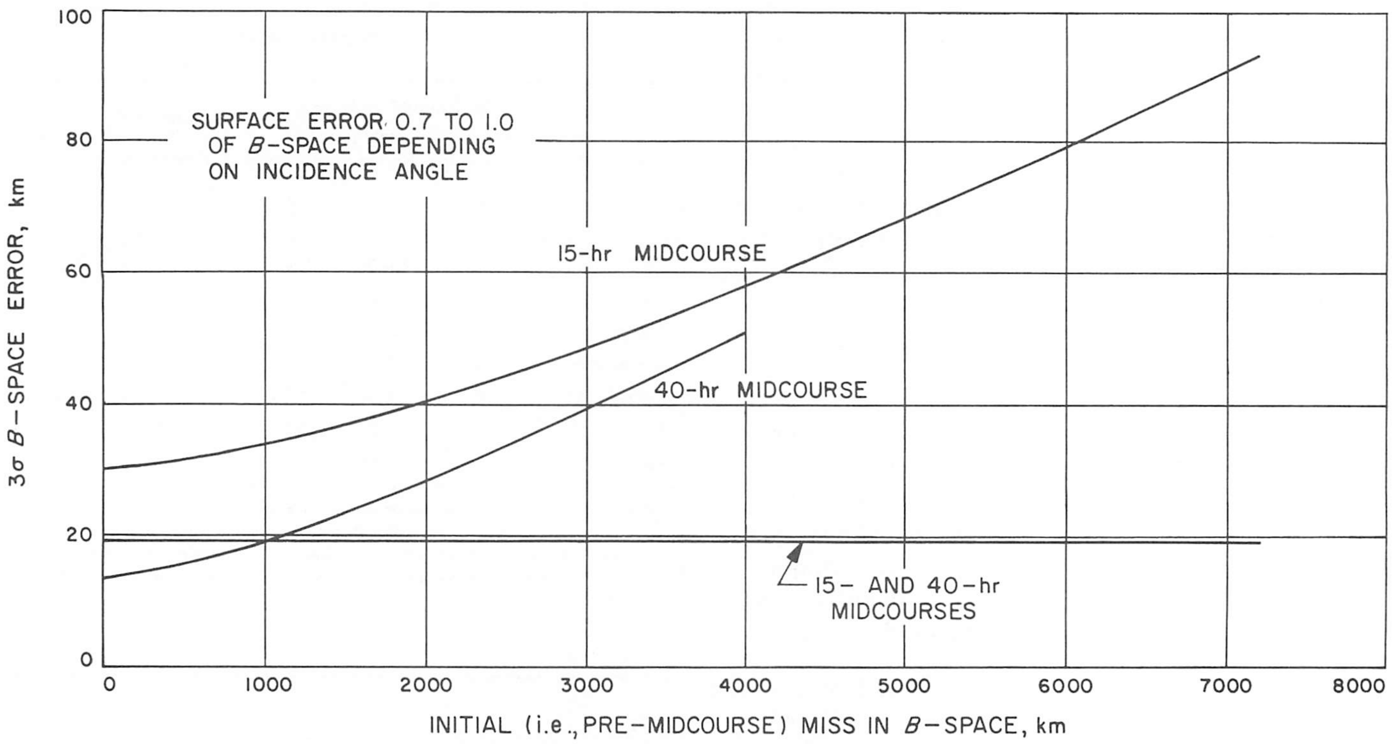


Fig. VI-25. Guidance accuracy

site selection resulted in a re-evaluation of landing accuracy capability. At that time, the demonstrated accuracy of the *Centaur* guidance system largely influenced the specification of landing accuracy at 30 km. *Centaur* injection accuracy was influential in specifying landing accuracy because it dictated the expected midcourse maneuver magnitude and, consequently, the expected execution errors. (If two midcourse maneuvers, rather than one, had been the standard mission mode, *Centaur* accuracy would have been of much less consequence.) A large group of 30-km radius sites was identified, and the landing sites for *Surveyors IV*, *V*, and *VI* were selected from this group.

Before *Surveyor III*, *Lunar Orbiter* high-resolution photographs of selected *Surveyor* landing sites became available. These photographs revealed terrain hazards that could be avoided by improved landing accuracy. Additional emphasis was placed on landing accuracy, and midcourse maneuver strategy (including the possibility of two midcourses) was re-examined. Figure VI-25 was used to develop probable maneuver strategies (i.e., a 15- or 40-hr midcourse, or both) required to achieve given accuracy goals as a function of *Centaur* injection error. From the overall mission viewpoint, it was evident that the preferred maneuver alternatives in order of decreasing preference were: (1) a 15-hr midcourse, (2) a 40-hr midcourse, and (3) both a 15- and a 40-hr midcourse. Figure VI-25 shows that, for *Centaur* injection errors up to 1000 km in *B*-space, greater accuracy could be achieved with a single 40-hr midcourse than with two midcourses. This was most significant; in fact, actual *Centaur* injection errors never exceeded 800 km.[20] Only *Surveyor IV* executed a single 40-hr midcourse. It was expected that two midcourses would be required for *Surveyor VII*; however, the orbit determination accuracy and execution accuracy of the first midcourse were phenomenally good and a second midcourse was not required. Table VI-8 summarizes the landing accuracy predicted at midcourse and the accuracy achieved for each mission. It is seen that, except for *Surveyor V*, the landing accuracy goals were more than met.

6. AMR Backup Command

The AMR onboard the spacecraft initiated the terminal retroignition sequence. The AMR generated a signal when it sensed the slant range of 60 statute mi to the lunar surface. This signal started a timer countdown that terminated with ignition of the vernier engines followed by main retroignition 1.1 sec later. A procedure to back

[20]*Surveyor VII* required about a 2000 km *B*-space correction because of an inflight landing site change. The actual *Centaur* injection error was only 99 km in *B*-space.

up the AMR by ground command was developed and implemented. Command transmission time was intentionally biased late (i.e., delayed) so that the AMR had ample to time function, but in time to save a significant percentage of missions if the AMR had not functioned. Optimum delay time was specified in terms of vernier propellant availability and the uncertainty in the delay time (based on unbraked impact time uncertainty and execution time uncertainty) assuming the AMR reliability to be 0.999 as shown in Fig. VI-26. The propellant availability was expressed in terms of the altitude parameter N (the number of standard deviations in burnout altitude that could be tolerated without propellant depletion occurring). Known delays in command receipt at the spacecraft, such as command console operator delay and transmission time, were compensated for.

To effectively back up the AMR, special procedures were developed to predict the unbraked impact time to within a 1σ uncertainty of 0.5 sec. Error sources other than tracking data errors, which significantly affected the prediction accuracy were: (1) assumed lunar radius at the impact point, (2) error in earth–moon ephemerides, and (3) timing errors. The best estimate of lunar radius at the impact point was obtained through consultation with the ACIC and NASA Langley Research Center. An *a priori* 1σ uncertainty of 1 km (roughly equivalent to 0.4 sec) was assigned to the radius. The last two error sources could be adequately reduced by relying heavily on the near-moon tracking data and processing data in the following manner:

(1) Process all available two-way doppler data from midcourse to about encounter minus 5 hr, 40 min, and map the resulting solution, plus covariance matrix, to the time of the last data point. No significance was attached to encounter minus 5 hr, 40 min epoch other than its consistency with nominal sequence of events times. Degrade the diagonal elements of the mapped covariance matrix by 0.5 km on position components and 0.01 m/sec on velocity components.

(2) Expand the estimate list to include geocentric radius (corresponding to distance of tracking station off-earth spin axis) and longitude of the two observing stations; that is, the type solution was expanded to a 10×10. *A priori* uncertainties of 12 m in spin axis distance, 40 m in station longitude, and 25 m in longitude difference between the two stations were added to the mapped covariance matrix.

(3) Reduce effective data weight to 0.003 m/sec to obtain realistic statistics on predicted unbraked impact time. This reduction was valid, since computational errors were no longer major error sources; i.e., the trajectory is only being integrated over a 6-hr period. The model errors were considered by degrading the covariance matrix and by adding the station parameters to the estimate list.

Table VI-9 compares selected nominal AMR backup delay times with the actual measured delays. In all instances, the AMR functioned properly. In each case, the backup command would have saved the mission if the AMR had failed.

G. Landed Roll Orientation

Orientation of the spacecraft Z axis was determined by the required main retro thrust direction (approximately antiparallel to the velocity vector at ignition). However, there was some flexibility in the specification

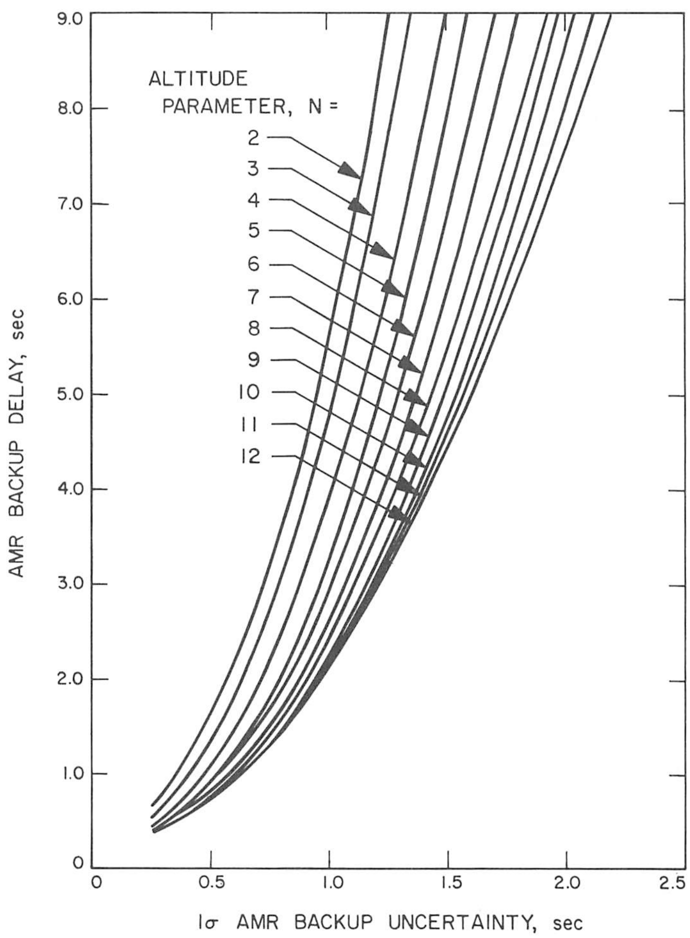


Fig. VI-26. AMR backup delay time (radar reliability: 0.999)

Table VI-8. Landing accuracy

Parameter	Mission						
	Surveyor I	*Surveyor II*	*Surveyor III*	*Surveyor IV*	*Surveyor V*	*Surveyor VI*	*Surveyor VII*
3 σ landing site uncertainty at midcourse (OD + execution errors)							
Semimajor axis, km	38.7	53.9	15.1	10.8	~100[a]	18.5	75
Semiminor axis, km	28.7	17.7	10.6	7.2	~100[a]	12.6	25
Orientation,[b] deg	33	57	58	7	—	−42	−57
Estimated actual landing site error							
Magnitude, km	19.1	—[c]	2.8	6.8	33.1	8.4	2.6
Orientation,[b] deg	−11.0	—[c]	−167.5	−167.2	152.7	165.5	−110.7

[a]Large uncertainty caused by short data span between maneuvers.
[b]Angle, measured north of east, to major axis/miss vector.
[c]*Surveyor II* failed at midcourse.

Table VI-9. AMR backup delay times

Delay times	Mission						
	Surveyor I	*Surveyor II*	*Surveyor III*	*Surveyor IV*	*Surveyor V*	*Surveyor VI*	*Surveyor VII*
Specified	1.16	—[a]	1.73	1.27	—	1.28	3.09
Observed	1.05	—[a]	0.61	0.30	7.20	0.44	3.92

[a]Not applicable.

of the spacecraft roll orientation. The spacecraft was roll stabilized throughout the terminal descent; therefore, the roll orientation achieved at the end of the terminal attitude maneuver also determined the nominal landed roll orientation.

Originally, it was planned to establish the roll orientation to point the planar array (high-gain) antenna at the earth to transmit approach television pictures. Approach television was abandoned in favor of engineering data transmission before the *Surveyor I* flight. For *Surveyor I*, no roll constraints or landed roll orientation preferences were specified, and roll orientation was selected to optimize communication over the deployed omnidirectional antenna.[21]

Between the flights of *Surveyors I* and *II*, RADVS antenna pattern measurements were obtained which indicated the possibility of signal acquisition on the sidelobe of one of the three beams from transmission via the mainlobe of another beam. This cross-coupling phenomenon would have resulted in incorrect attitude steering and possibly catastrophic failure. Probability of cross-coupled side lobe (CCSL) acquisition was a strong function of spacecraft attitude relative to the lunar surface and to the velocity vector. RADVS logic changes, which would preclude CCSL acquisition, were designed; however, schedule requirements did not permit incorporation of this logic in the RADVS units on *Surveyors II* and *III*. Additional incidence angle (18 deg $\leq \Gamma_{IM}$) and roll attitude constraints were imposed on *Surveyors II* and *III* to minimize possibility of CCSL acquisition. Project science specified landed roll orientation preferences for *Surveyors II* and *III* which were incompatible with the RADVS constraints. These preferences were based on a desire to optimize lighting conditions for television operations. Because proper RADVS operation was critical, science preferences had to be compromised. Figure VI-27 shows RADVS constraints, science preferences, and the selected roll orientation for *Surveyors II* and *III*.

The CCSL logic was incorporated in the RADVS units for *Surveyors IV* through *VII*. Although this logic offered complete protection against CCSL acquisition, it was discovered that it could cause a main-lobe to be rejected as a CCSL for certain geometry. Analysis further revealed that a beam approaching the vertical scintillation could occur that would cause main-lobe rejection as a CCSL. Roll attitude constraints were imposed on *Surveyors IV* through *VII* to preclude such main-lobe rejections. Roll attitude constraint regions were defined by extensive Monte Carlo analyses, which used actual RADVS test measurements on each unit in modeling the RADVS performance under all influential dispersions. After the constraint regions were defined for a given spacecraft, project science selected the optimum landed orientation that satisfied the RADVS constraints with some margin.

[21]Omniantenna A; omniantenna B did not deploy on *Surveyor I*.

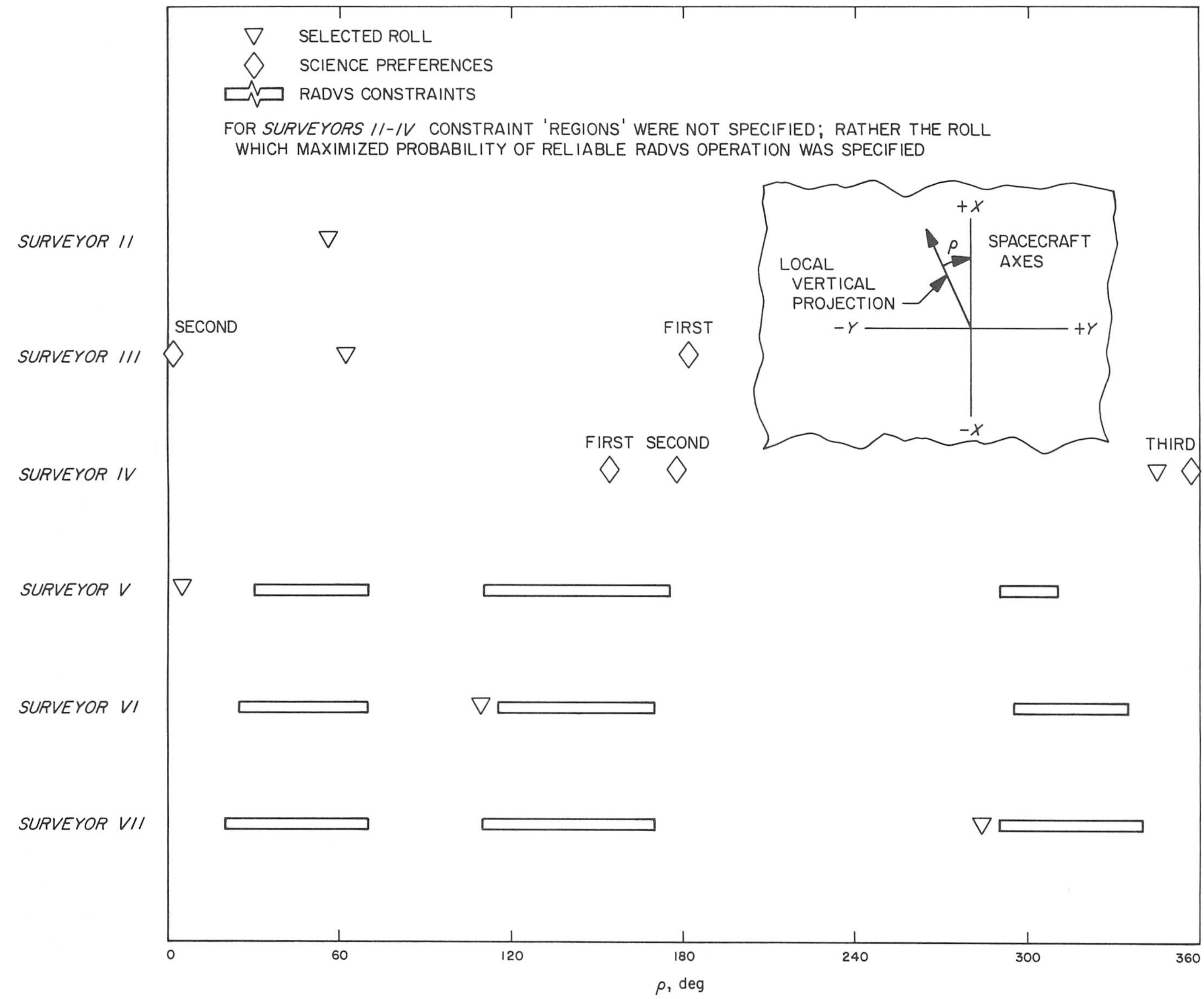


Fig. VI-27. Roll attitude constraints and selections

Main retro burnout velocities above 500 ft/sec would have required additional roll constraints to preclude attitude control instability from helium saturation of the vernier propellant. These constraints were avoided by specifying that the midcourse maneuver be sufficiently large to preclude burnout velocities above 500 ft/sec.

H. Landing Site Selection

The *Surveyor* spacecraft were designed for unbraked impact (off-vertical) incidence angles from 0 to 45 deg.[22] This provided accessibility to landing sites up to about 63 deg of surface angle from the vertical impact point. Since the location of the vertical impact point was launch-date-dependent (see Subsection A-4), site accessibility also depended on launch date. Figure VI-28 presents the envelope of vertical impact points for the time period of interest to *Surveyor*, (1966–1968). Corresponding accessibility regions are also shown. Sites within the shaded region were available only part of the time. The accessibility region was bounded on the west by the limb restriction that prohibited landing at any site that could come within 19 deg of the lunar limb considering lunar libration. This was a communication constraint. Since a principal objective of the *Surveyor* Program was to gain knowledge of the moon directly applicable to the *Apollo* Program, all except *Surveyor VII* were further constrained in landing within the *Apollo* zone of interest.

The *Surveyor* spacecraft could not be expected to survive the lunar night. The survival capability, following a predawn landing, was quite limited. A guideline specifying that landings should occur during the lunar morning at sun elevation angles less than 25 deg was established to optimize television viewing and to provide near-maximum on-surface operating time. To facilitate site selection, the sunrise terminator, subsolar point, and vertical impact point associated with each possible launch date in a given launch period were displayed on a grid of selenographic latitude and longitude. These displays were used to readily determine the accessibility and launch period for each site under consideration for a given mission.

Landing accuracy was a key factor in selection of the landing sites. Estimated 3σ landing accuracy for *Surveyors I* and *II* was a 50-km radius circle on the lunar surface. Consequently, a large group of 50-km sites within the *Apollo* zone of interest was identified. Landing sites for *Surveyors I* and *II* were selected from this group. Immediately after the *Surveyor II* mission, increasing emphasis on greater site selection flexibility and the high injection accuracy demonstrated by the *Atlas/Centaur* launch vehicle led to specification of the landing accuracy of the remaining *Surveyors* as a 30-km radius circle. An extensive group of 30-km sites was then defined from which all except the *Surveyor VII* site were selected. *Surveyor VII* was assigned to a 10-km radius site on the Tycho blanket. The risk associated with this very small site was considered to be warranted by the tremendous scientific potential offered by the site.

Since both the *Surveyor* Project and the *Lunar Orbiter* Program were to obtain lunar data in support of *Apollo*, NASA formed the *Surveyor/Orbiter* Utilization Committee to oversee site selections for each project. With the exception of *Surveyor I*, all *Surveyors* were assigned to landing sites that had previously been photographed in high resolution by *Lunar Orbiter*.[23] This dual coverage was planned to promote generalization of *Surveyor* findings to other surface areas. However, just before *Surveyor III*, analysis of high-resolution photographs obtained by *Lunar Orbiter* indicated unanticipated landing hazards. Consequently, a computer program was developed to estimate landing success probability by using a lunar surface model based on the crater size density distribution obtained from crater counts on the high-resolution *Lunar Orbiter* photographs. Rocks greater than 12 cm in diameter were particularly hazardous. The surface model assumed rock densities to be correlated with crater size and distance, as observed by *Surveyor I*. Landing success probability estimates obtained from this model were considered in all subsequent *Surveyor* site selections.

The locations of the *Surveyor* landing sites are indicated in Fig. VI-28. Coordinates of these sites are given in Table VI-10. Selection of sites within the *Apollo* zone of interest at widely spaced intervals was not only consistent with *Apollo* needs, but enhanced the overall scientific return of *Surveyor*. *Surveyor VI* concluded the required support to *Apollo*. The prime goal of *Surveyor VII* was to obtain data at a site offering the greatest chemical diversity from the mare sites of the previous landings. This dictated selection of the site on the northern Tycho blanket.

[22]*Surveyors I* and *II* were constrained to a maximum incidence of 25 deg.

[23]*Surveyor I* landed before the *Lunar Orbiter I* flight. *Lunar Orbiter III* subsequently photographed the *Surveyor I* site with enough resolution to identify *Surveyor I* on the photograph.

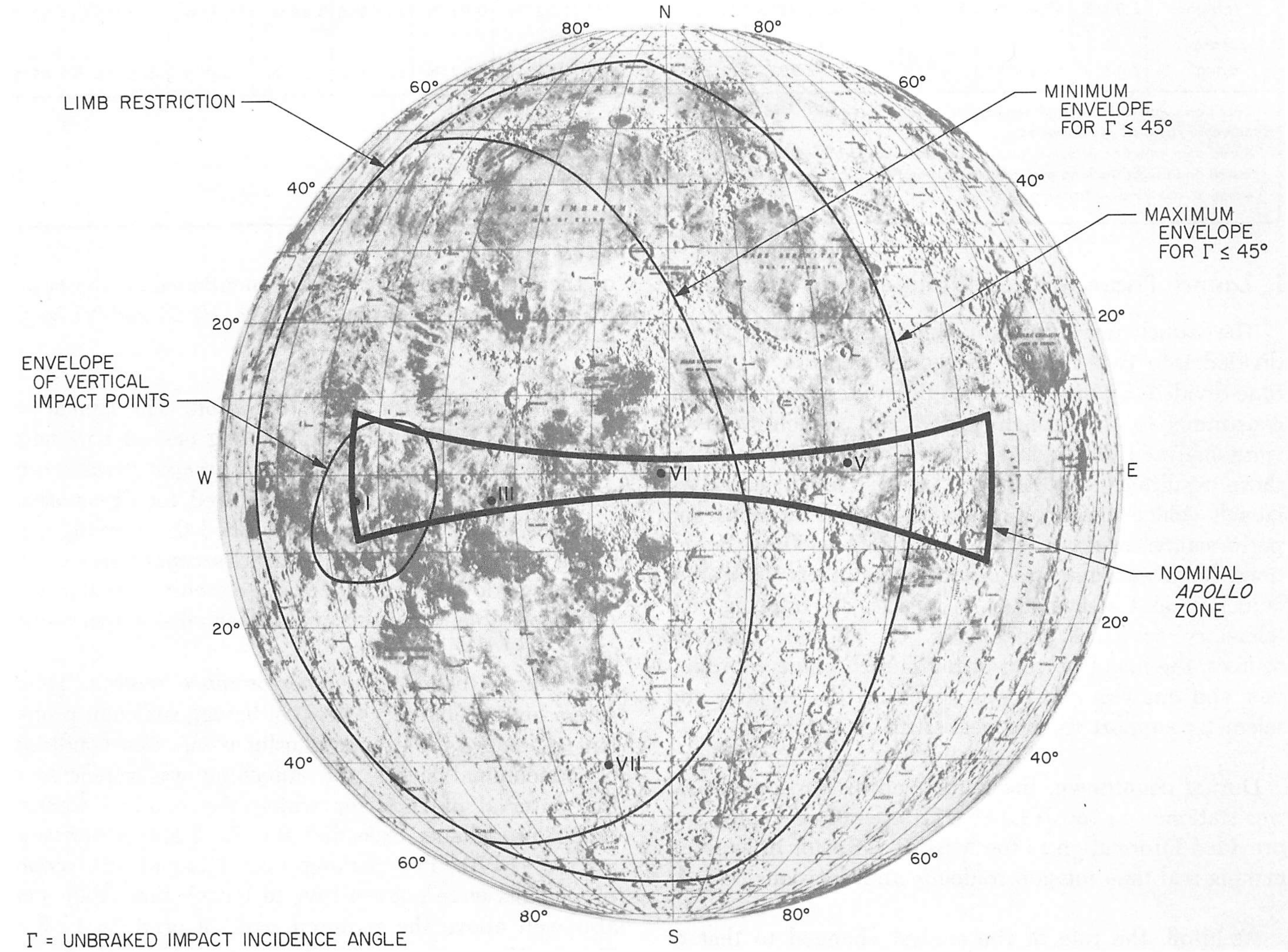


Fig. VI-28. Lunar accessibility regions and landing sites

Table VI-10. Landing site selenographic coordinates

Coordinates	Mission													
	Surveyor I		*Surveyor II*		*Surveyor III*		*Surveyor IV*		*Surveyor V*		*Surveyor VI*		*Surveyor VII*	
	Lat	Long	Lat	Long	Lat	Long	Lat	Long	Lat	Long	Lat	Long	Lat	Long
Selected preflight	3.25°S	43.83°W	0.00°N	0.67°W	3.33°S	23.17°W	0.58°N	0.83°W	0.00°N	24.00°E	0.42°N	1.13°W	40.87°S	11.37°W
Selected inflight	2.33°S	43.83°W	0.55°N	0.83°W	2.92°S	23.25°W	0.42°N	1.33°W	0.92°N	24.08°E	0.42°N	1.13°W	40.87°S	11.37°W
Estimated actual	2.40°S[a]	43.23°W[a]	—[b]	—[b]	2.99°S[c]	23.34°W[c]	0.37°N[d]	1.55°W[d]	1.42°N[e]	23.11°E	0.51°N[c]	1.39°W[c]	40.88°S[c]	11.45°W[c]

[a]Based on *Lunar Orbiter III* photograph of landed *Surveyor I*.
[b]*Surveyor II* failed at midcourse.
[c]Based on correlation of *Surveyor* pictures and *Lunar Orbiter* photographs.
[d]Based on inflight tracking and unlikely assumption that *Surveyor IV* executed a nominal terminal descent.
[e]Based on postlanding tracking.

I. Launch Phase Mission Planning and Analysis

The launch phase mission planning and analysis was divided into two general categories of tasks that were time-divided by liftoff event. Before each launch, all constraints to the launch period and launch windows were analyzed and identified. Such constraints included those resulting from Range Safety launch corridors, launch vehicle parking-orbit coast time limitations and performance reserves, spacecraft shadow time limitations, lunar encounter requirements, conflicts with other launches, and deficiencies in near-earth tracking and telemetry coverage capabilities. Of all the constraint sources, the major portion of the launch phase coordination and analysis effort was spent in the tracking and telemetry support for the near-earth phase.

During countdown, the launch phase mission analyst was stationed in the JPL/AFETR operations center and provided information to the Mission Director to assist in making real-time mission tradeoffs and decisions.

At liftoff, the role of the analyst changed to that of continually evaluating the state of the mission during the early phases of flight; he was especially concerned with the time history of the launch vehicle performance and associated flight operations to determine their possible influence on initial acquisition by the DSN and subsequent space flight operations.

1. Launch Constraints Determination

Delineated below are those launch constraints that influenced the determination of launch periods and launch windows for the various *Surveyor* missions. The behavior of the constraining parameters mentioned in the subsequent paragraphs is shown in Figs. VI-29 and VI-30 for typical launch windows.

a. Launch azimuth sector limitation. The launch azimuth sector between 78 and 115 deg east of true north was approved by the AFETR for *Surveyor* parking-orbit missions; 80 to 115 deg was approved for direct-ascent missions. These limitations established the absolute maximum windows considered for any *Surveyor* mission. The launch window design utilized these sectors to the greatest extent possible, consistent with other mission constraints.

b. Launch vehicle flight performance reserve. To ensure a high probability (99%) of having sufficient propellant to achieve the proper translunar injection conditions on nonnominal flights, the launching was restricted to that interval of time for which the nominal *Centaur* excess propellants exceeded the 3σ flight performance reserve (FPR). For parking-orbit missions, the excess propellants were not sensitive to launch time; they were also well above the required parking orbit 3σ FPR of 255 lb. However, for direct-ascent missions the excess propellants were highly sensitive to launch time (discussed in Section VI-A-3 and shown in Fig. VI-30). The direct ascent 3σ FPR requirement of 235[24] lb generally dictated the openings of the final launch window designs.

c. Launch vehicle heating constraint. To preclude excessive heating on the aft end of the *Atlas* sustainer during direct ascents, the launching was restricted to the launch time interval for which the flight path angle at

[24]175 lb applied to *Surveyor I*.

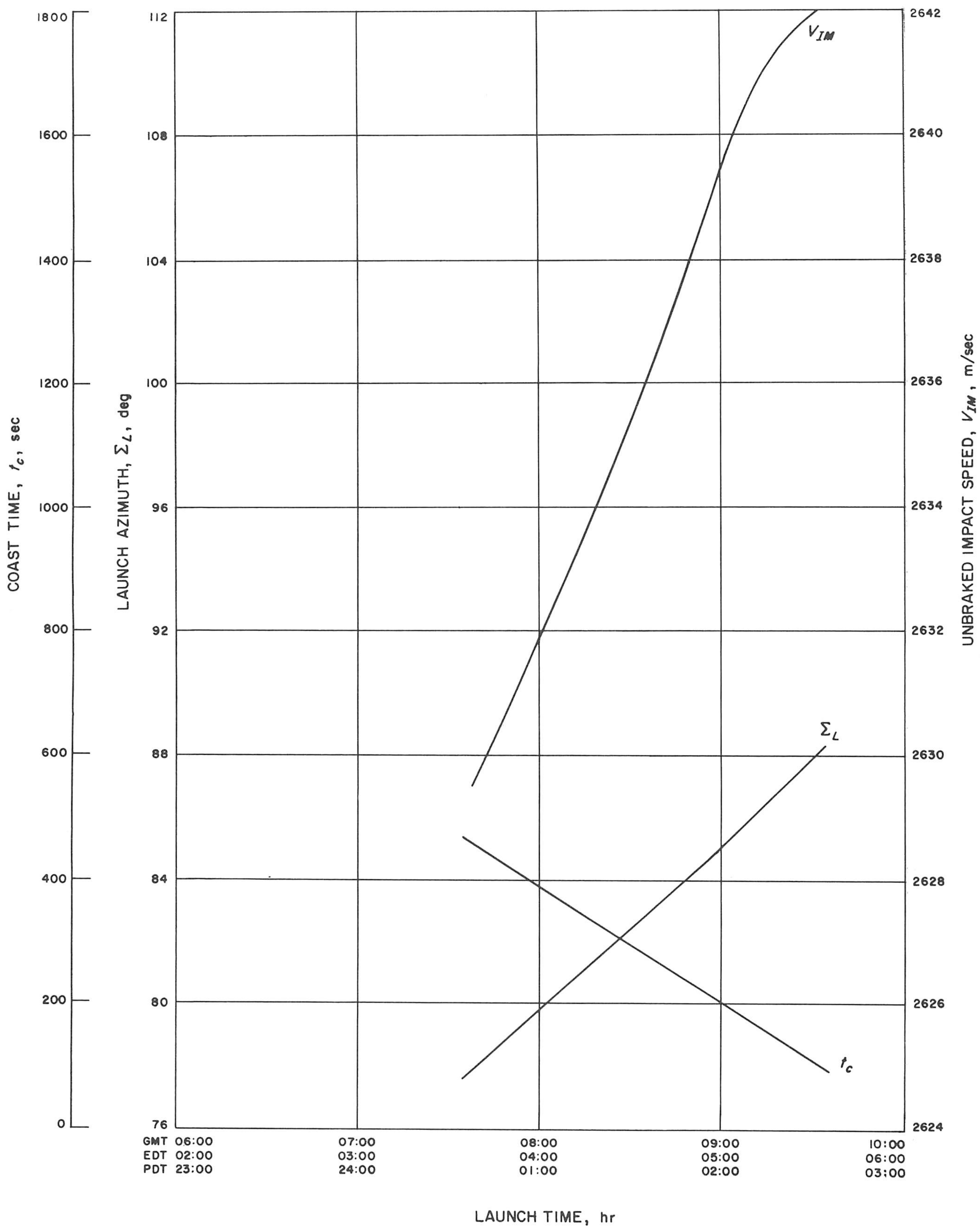


Fig. VI-29. Typical graph of unbraked impact speed, launch azimuth and parking orbit coast time vs launch time

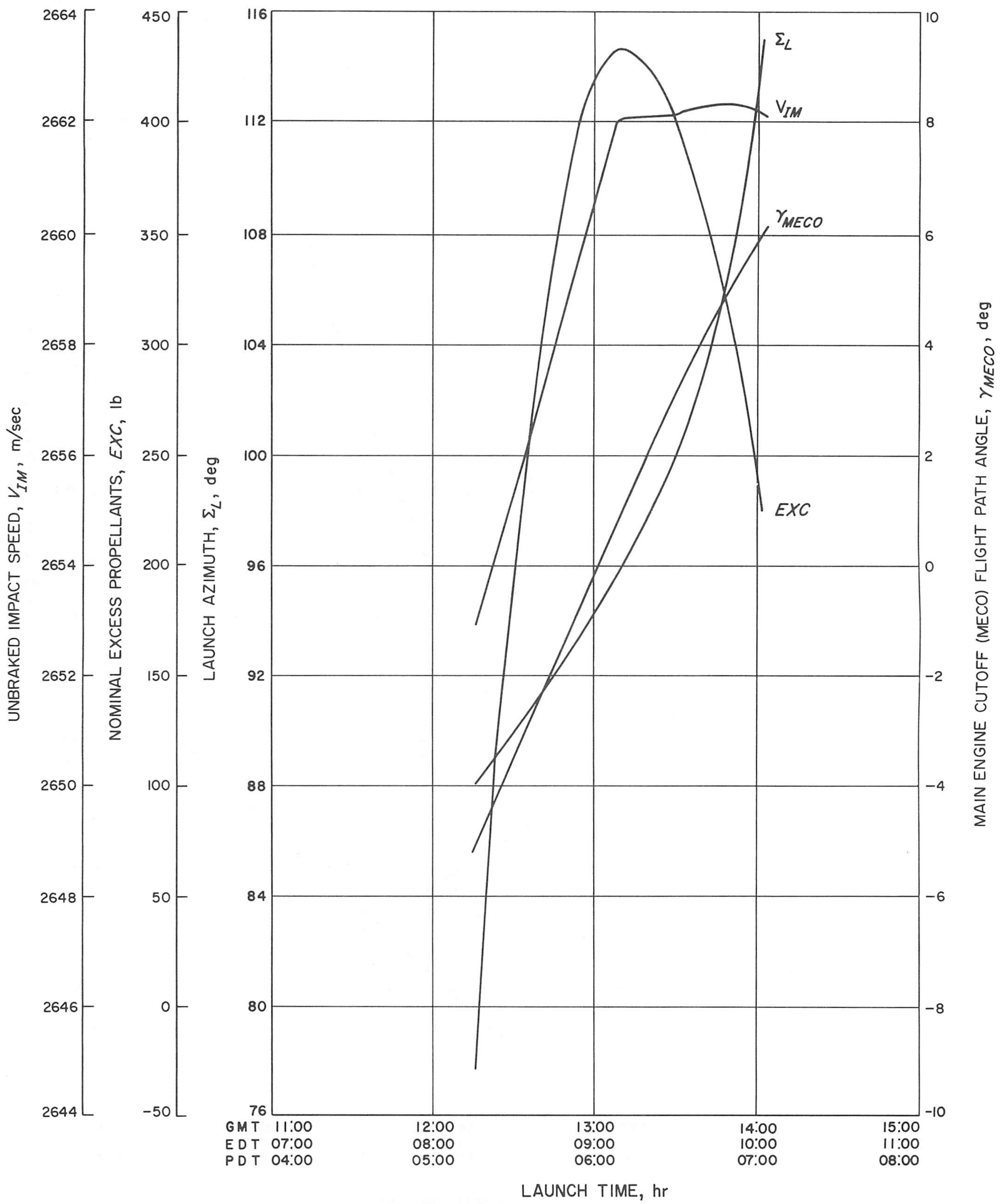


Fig. VI-30. Typical graph of unbraked impact speed, nominal excess propellants, launch azimuth, and MECO flight path angle vs launch time

Centaur main engine cutoff would be greater than −2 deg. This constraint was not identified until just before the *Surveyor IV* launch.

d. Launch vehicle parking-orbit coast time limitations. To complete propellant settling sequences between the first and second burns of the *Centaur* during parking-orbit missions, a minimum parking-orbit coast time of 116 sec was required. The *Centaur* was also limited to a maximum parking-orbit duration of 25 min because of a design constraint.

e. Spacecraft unbraked lunar impact speed limit. As discussed previously, the nominal unbraked lunar impact speed was fixed at the design impact speed across a launch window except when it had to be varied to satisfy arrival time constraints. Launching was prohibited when the nominal impact speed would have been more than 12 m/sec less than the design impact speed.

f. Spacecraft high-power transmitter operating limit. The *Surveyor* was launched with its transmitter operating in the low-power mode of 100 milliwatts. However, 11 sec before spacecraft separation from *Centaur*, the transmitter was commanded to operate in the high-power mode at 10 W. This power increase ensured adequate signal strength margins for the DSN to achieve initial acquisition of the spacecraft. To avoid overheating, the spacecraft transmitters operated in the high-power mode for no more than 60 min. Consequently, since the Deep Space Stations were the only facilities with command capability, the initial acquisition station was locked up in two-way with the spacecraft by 60 min following the high-power-on event. Thus, those portions of the launch windows, where the DSN could not meet this requirement, were eliminated.

g. Earth shadow limitation. The spacecraft was limited to a maximum time of 42 min in earth's shadow immediately after launch. Because shadow time varied over the launch window for those days when shadow entry was possible, this limitation could have impacted certain windows.

h. Inadequacies in Tracking and Data System (T&DS) support of the near-earth class I tracking and telemetry coverage requirements. Inasmuch as a considerable amount of effort was expended in analyzing the T&DS support of the near-earth tracking and telemetry coverage requirements and the resultant impact on launch windows, the following separate section is devoted to the discussion of the analytical procedures involved in the area of this support.

2. Near-Earth T&DS Coverage Analysis

Delineated in this section are the detailed procedures used for establishing launch constraints due to the limited capability of the T&DS to meet the requirements for tracking and telemetry coverage during the near-earth phase of the missions. These requirements were levied, primarily, to (1) perform a near-real-time mission evaluation, and (2) receive and record data for postflight analysis. Since these items were considered prime objectives, those requirements specifically needed to meet them were defined as class I. They are shown schematically for direct ascent and for parking-orbit missions in Figs. VI-31 and VI-32, respectively. Wherever reference is made to the coverage requirements in the following discussion, class I requirements only are implied.

a. Tracking and Data System. The T&DS is composed of facilities of the AFETR, the Manned Space Flight Network (MSFN), and the DSN. A list of the facilities available to support the *Surveyor* missions and their capabilities is provided in Table VI-11. Land stations and ships are included in the AFETR facilities. Generally, at least two of the three ships listed were available for support. AFETR aircraft were used for special support to cover critical launch vehicle telemetry intervals.

b. Near-earth support plan of T&DS. During the planning stage of the near-earth T&DS support, the launch phase mission analyst coordinated the project's inputs with the T&DS analyst at JPL/AFETR; this analyst, in turn, interfaced with the AFETR and MSFN planners. The mission analyst reviewed the support plans as they were formulated and provided recommendations to the T&DS analyst.

The ability of the T&DS to satisfy the tracking and telemetry coverage requirements was heavily dependent upon (1) the characteristics of the trajectories, (2) the availability of the T&DS resources, and (3) the capability of these resources. The prime objective during the planning stage was to use these three inputs to devise an optimum support plan. To achieve this objective, the most important consideration to an optimum plan is the planning of ship support, which is the foremost significant variable in the hands of the planners. Ship support planning is also time-critical because of the relatively long lead time required to schedule the ships, analyze ship locations, and for the ships to travel across oceans (in some cases) to be at their designated locations. Therefore, the number of ships and their support positions must be optimized early in the planning phase.

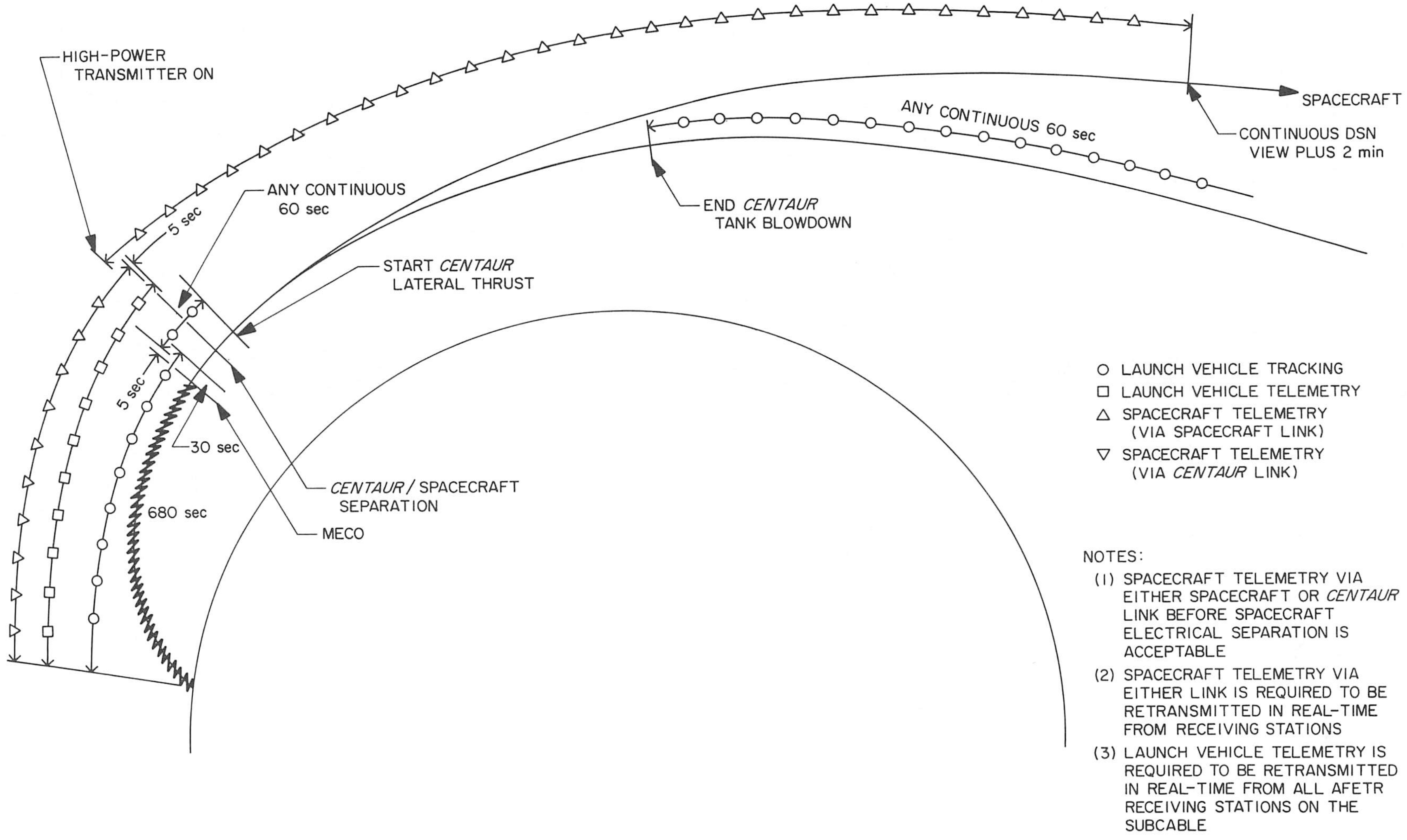


Fig. VI-31. Near-earth class I T&DS requirements for direct-ascent launches

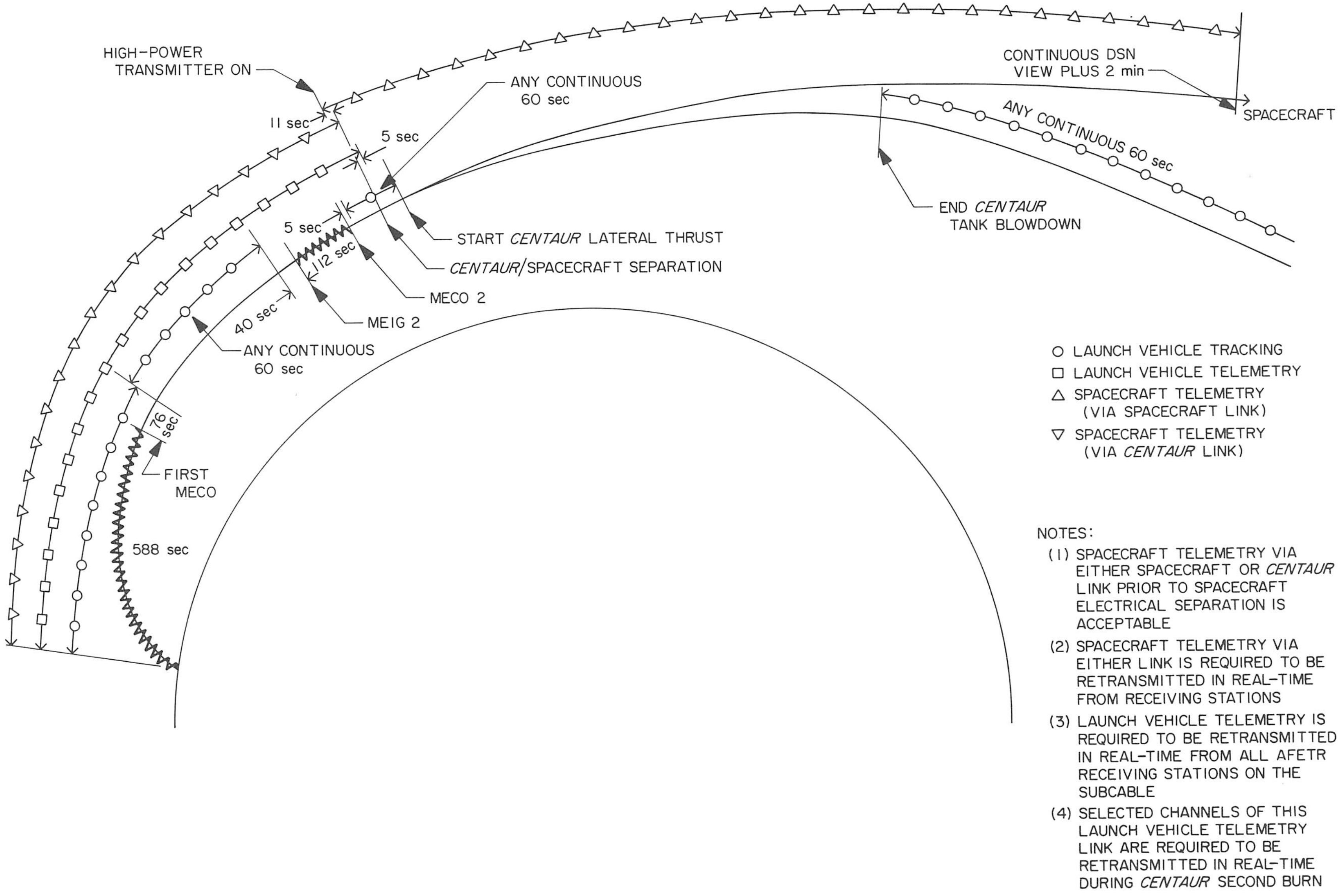


Fig. VI-32. Near-earth class I T&DS reauirements for parking orbit launches

Table VI-11. Tracking and data system configuration

AFETR		C-band tracking		Telemetry	
Station	Station location	Capability	Antenna type	Capability	Antenna type
0	Patrick Air Force Base	Beacon/skin	FPQ-6		
1	Cape Kennedy	Beacon/skin	FPS-16	VHF S-band	TLM-18 3-ft-parabolic
19	Kennedy Space Center, Merritt Island (Tel 4)	Beacon/skin	TPQ-18	VHF S-band	TAA-2A TAA-3A
3	Grand Bahama	Beacon Beacon/skin	FPS-16 TPQ-18	VHF S-band	LH tri-helix TAA-2
7	Grand Turk	Beacon/skin	TPQ-18		
91	Antigua	Beacon/skin	FPQ-6	VHF S-band	TLM-18 TAA-3A
40	Trinidad	Skin	FPS-43		
12	Ascension	Beacon Beacon	FPS-16 TPQ-18	VHF S-band	TLM-18 TAA-3A
13	Pretoria	Beacon	MPS-25	VHF S-band	TLM-18 3-ft-parabolic
UNI	RIS *Twin Falls*	Beacon	FPS-16	VHF S-band	16 crossed dipole TAA5-12
WH1	RIS *Coastal Crusader*			VHF S-band	TAA-1 TAA5-24
YAN	RIS *Sword Knot*			VHF	TAA-1 TAA5-24
BDA	Bermuda	Beacon/skin	FPS-16	VHF	Quad helix
CRO	Carnarvon	Beacon/skin	FPQ-6	VHF S-band	Quad helix 30-ft-parabolic
TAN	Tananarive	Beacon	FPS-16	VHF	Quad helix
CYI	Grand Canary	Beacon	MPS-26	VHF	Quad helix
Deep Space Stations		Tracking capability		Telemetry	
DSS 51	Johannesburg	One-, two-, and three-way doppler and antenna angles		S-band	85-ft-parabolic
DSS 42	Tidbinbilla			S-band	85-ft-parabolic
DSS 72	Ascension			S-band	30-ft-parabolic
DSS 61	Robledo			S-band	85-ft-parabolic
DSS 11	Goldstone			S-band	85-ft-parabolic
DSS 71	Cape Kennedy			S-band	4-ft-parabolic

An optimum support plan was considered to be one in which the T&DS was able to satisfy the tracking and telemetry coverage requirements over the greatest possible launch window within those windows dictated by all other constraints. For the T&DS planners to arrive at an optimum plan, some guidelines, which indicated how the other constraints affected the windows, had to be supplied for them. These guidelines were prepared by the mission analyst, after review of all project constraints.

For the mission analyst to evaluate planned AFETR ship positions, both land and ship station views were generated using the time data sequential processor program. The required trajectory parameters and the station view constraints were inserted into this program and the resultant station views plotted on earth maps and in view period graphs similar to those presented in Figs. VI-33 and VI-34, respectively. Plotted with the station views were the key launch vehicle and spacecraft events. These view period graphs are very useful tools in establishing deficiencies in the T&DS coverage and the resultant T&DS launch window constraints. The earth maps are particularly useful in selecting initial ship locations and, if necessary, in improving the locations over the previous selection.

Preflight T&DS working group meetings were held at AFETR with representatives from AFETR, MSFN, and DSN in attendance. The mission and T&DS system analysts were also present. These meetings provided discussions of the guidelines, planned ship locations, potential conflicts with other launches, interface problems, etc. The meetings were held early enough so that the final support plan reflected the project's guidelines.

3. Final Launch Window Specification

In establishing final launch windows, several steps were involved. First, a final review of all project launch window constraints was performed so that the windows would be based upon the most recent constraints. The most restrictive constraints were then determined essentially as shown in Fig. VI-35. Plots of the constraining parameters vs launch time are compared to establish the constraints that dictated the earliest and latest permissible launch times.

4. Real-Time Mission Analysis

It was important for two reasons to continually evaluate the state of the mission during the early phases of the flights. First, a requirement existed for the DSN to initially acquire the spacecraft within a relatively short time (about 30 min) of its scheduled time. Thus, the analyst had to be aware of any mission anomalies that would affect this acquisition. Second, early identification of anomalies was imperative to ensure maximum opportunity for corrective action.

The near-earth mission analysis is based on space vehicle flight events reported in near-real-time, the reported acquisition characteristics of the T&DS stations, the reported space vehicle performance evaluations based on real-time telemetry data, the monitored powered flight trajectory characteristics, and the orbit determination calculations of the real-time computer system (RTCS) at KSC.

Actual launch vehicle and spacecraft flight event times (known as *Mark* times) were reported in near-real-time from the supporting AFETR and MSFN stations viewing the vehicle at the time of the particular event. Their occurrences were reported and then, after further examination of the telemetry data by station personnel, the actual times were reported. A comparison between these actual in-flight event times and the nominally expected times provided the analyst with one of the data sources by which to establish mission status.

In addition to reporting *Mark* events, the station announced acquisition of signal and loss of signal. By noting the times of these occurrences and comparing them with the expected rise and set times, a further indication of mission status was available.

During the powered flight from launch to main engine cutoff, a real-time evaluation of the launch vehicle performance was received from the launch vehicle telemetry laboratory at AFETR via the AFETR missile operations paging system (MOPS). This evaluation was based upon launch vehicle telemetry transmitted to the laboratory in real-time from the uprange stations. A real-time commentary was monitored over the MOPS from AFETR central control giving the trajectory status, as automatically plotted on the range safety instantaneous impact point and present position charts. The voice reports from each of these sources combined provided an excellent account of the launch vehicle performance. When available during a parking orbit, *Centaur* second-burn telemetry data were retransmitted in real-time to the telemetry laboratory for analyses of this portion of the flight.

Spacecraft telemetry data were available from the stations in near-real-time from launch to DSN initial acquisition, via the *Centaur* link or via the spacecraft link,

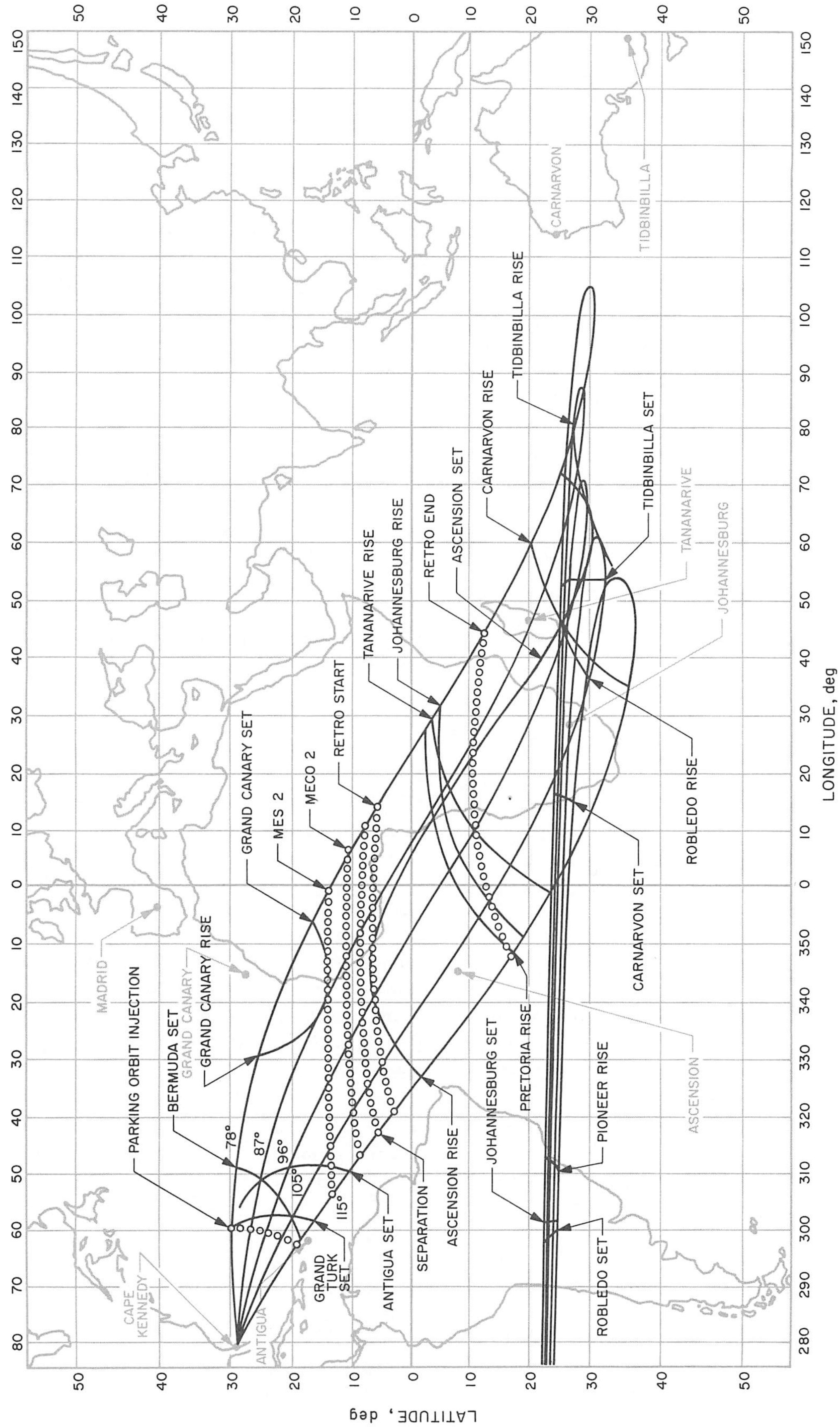


Fig. VI-33. Typical earth map of T&DS station coverage

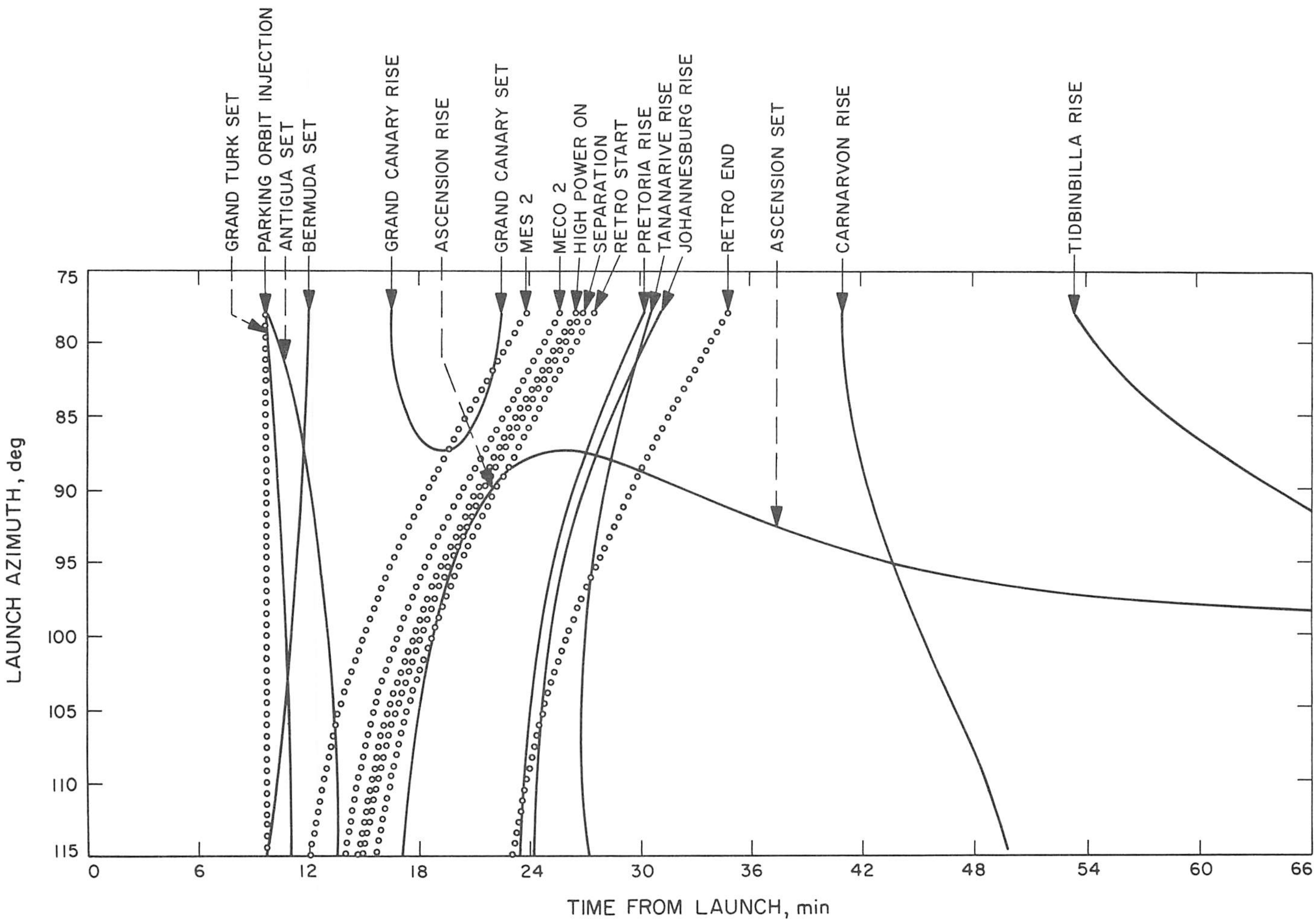


Fig. VI-34. Typical view period graph for first hour

except, during portions of the parking orbit, for that launch mode. Because of limited T&DS facilities, complete coverage was not provided. These data were transmitted to both JPL/AFETR and the SFOF for analysis.

Parking-orbit launch vehicle C-band tracking data received by AFETR and/or MSFN were used by the RTCS to first compute the actual parking orbit. The resultant orbital parameters were compared with the expected values to determine normalcy. A theoretical transfer orbit was subsequently calculated, based upon the actual parking orbit and nominal *Centaur* second burn. The first set of DSN acquisition information prepared by the real-time computer system was based upon this theoretical orbit in the case of parking-orbit missions.

Transfer orbit C-band tracking data, taken immediately following translunar injection, were processed to determine the actual spacecraft translunar trajectory. This trajectory was mathematically mapped to the moon by the RTCS to evaluate the unbraked lunar impact conditions.

Some early DSN two-way spacecraft tracking data were back-fed from the SFOF to the RTCS for their use in additional transfer orbit computations. This provided a dual source of orbit calculations for comparison, based upon the same tracking data source.

A detailed report of the launch phase mission analysis activities was presented. The launch operations plan and the T&DS support during the near-earth phase were described.

5. Launch-On-Time Experience

A summary of the launch-on-time history for the seven *Surveyor* missions is presented in Table VI-12 (also see Section VIII). As seen in Table VI-12, all *Surveyor* flights were launched on the first attempt except *Surveyor IV*, which was scrubbed before the countdown began, but was launched on the following day.

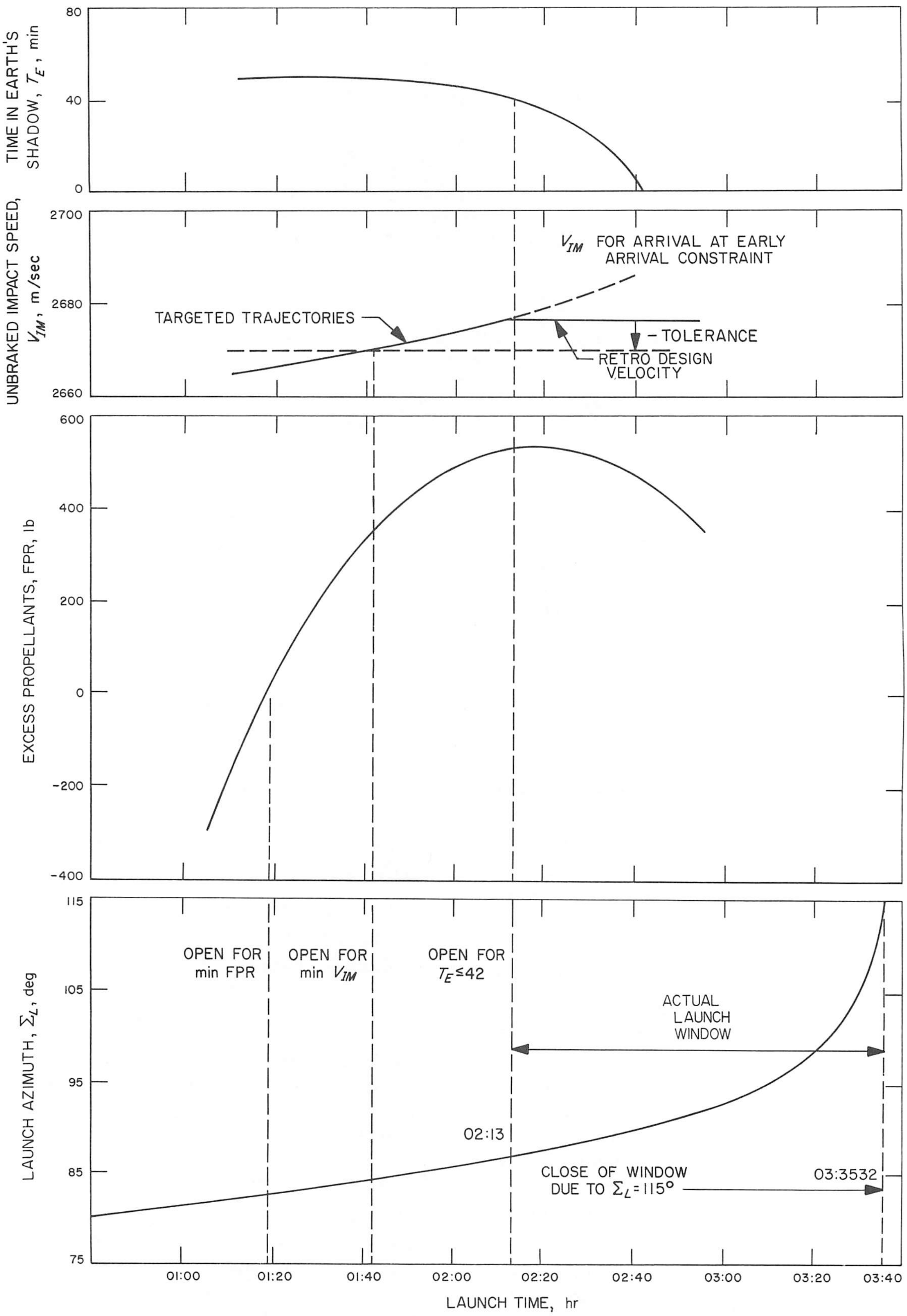


Fig. VI-35. Sample launch window refinement

Table VI-12. *Atlas/Centaur/Surveyor* launch-on-time history[a]

Mission	Launch attempts	Scheduled hold at $T-$ 90-min, min	Extension of $T-$ 90-min hold, min	Scheduled hold at $T-$ 5-min, min	Extension of $T-$ 5-min hold, min	$T-$ 5-min recycled hold, min
Surveyor I	1	60	0	21	0	0
Surveyor II	1	70	0	15	7	29
Surveyor III	1	60	0	10	51[b]	0
Surveyor IV	2	60	0	15	0	0
Surveyor V	1	60	0	10	18	0
Surveyor VI	1	60	0	10	17[c]	0
Surveyor VII	1	60	35[c]	10	0	0

[a]Also see Section VIII.
[b]Only hold because of spacecraft anomaly.
[c]Preplanned possible holds.

Although there were extended holds beyond those planned for *Surveyor VI* and *VII* countdowns, they were precountdown-planned to improve the T&DS coverage capabilities.

J. Telecommunications System Analysis

1. Mission Design and Summary of Performance

The *Surveyor* overall telecommunications system consisted of transmitters, receivers, transponders, antennas, signal conditioning and processing units, and command decoders. It was the function of this system to: (1) provide digital engineering data transmission from the spacecraft at data rates compatible with specific mission phase requirements; (2) provide analog data, such as that from television, science experiments, and touchdown strain gages, at signal levels sufficient for proper discrimination; (3) provide phase-coherent, two-way doppler data for tracking and orbit determination; and (4) provide a highly reliable spacecraft command reception capability throughout the mission to allow for complete control of the spacecraft from the ground.

The *Surveyor* spacecraft was designed to be almost entirely controllable by ground command. This technique imposed several constraints on the design of the telecommunications system. Specifically, the telecommunications system was required to: (1) provide high data rates such that a nearly complete analysis of spacecraft performance could be accomplished in real-time for control and decision making purposes, (2) provide a command subsystem with performance and reliability compatible with the ground control philosophy, and (3) provide reliable telemetry and command capability for all possible spacecraft attitudes. Additionally, the system design was constrained by the requirements to (4) be compatible with the NASA DSN ground tracking facilities, both during normal tracking and initial acquisition phases, and (5) to provide wide-band lunar data for scientific purposes.

To meet the constraints imposed on the telecommunications system, a highly redundant and flexible system was developed.

In addition to the redundancy of the *Surveyor* telecommunications system, the spacecraft subsystem could operate in a large number of different communication modes. Not only could the redundant components be switched into many different combinations, but each of the spacecraft receivers could be operated in either an automatic frequency-controlled (AFC) mode or an automatic phase-controlled (APC) mode. Furthermore, there was a large number of telemetry mode operations for the downlink *Surveyor* telemetry. All these factors tended to make the mission design and analysis with respect to the telecommunications system a complex task.

To simplify the mission design and analysis of the *Surveyor* telecommunications system, each mission was divided into phases in which each phase represented a different mode of attitude control of the spacecraft. Different modes of attitude control represented different types of problems and, therefore, this was a convenient way to separate the types of analyses involved.

One of the primary analyses which was performed for each mission was the generation of predictions of the telecommunications system performance. These predictions were on the performance during the coast phases of the missions when the three-axis attitude of the spacecraft was celestially referenced. Preflight predictions were generated for each mission using nominal trajectories for the available launch period. The preflight predictions were used to plan the mission with respect to the optimum choice of omniantenna, the optimum choice of transmitter/transponder, and the telecommunication bit rate capability. Later, during the mission, inflight predictions were generated using the actual trajectory. The in-flight predictions were then used to access the performance of the telecommunications system.

Both the preflight and in-flight predictions were made using telecommunications prediction computer programs. The preflight predictions were made using a Hughes version of the program and the in-flight predictions were made using JPL version of the program. The two programs performed identical calculations and both utilized magnetic tape for inputs. One of the magnetic tapes contained trajectory information applicable to the specific trajectory and another of the magnetic tapes contained digitized antenna patterns. The outputs of the programs provided signal level-vs-time information based on the slant range from the appropriate ground station and the omniantenna gain for the particular spacecraft attitude when locked to the sun and Canopus. Figures VI-36 and VI-37 show some sample time variable predictions of the uplink and downlink performance for *Surveyor IV*.

a. Prelaunch phase. The primary consideration during the prelaunch phase was to assess the subsystem performance and confirm its readiness for launch. In addition, important spacecraft transmitter and receiver frequency data were measured at this time. These frequency data were used to predict the frequencies at initial acquisition, thus improving the process of acquisition by the Deep Space Stations.

Since the spacecraft transmitter was commanded to high-power operation shortly before spacecraft separation from the *Centaur*, the compartment temperatures began to increase. This temperature rise caused the receiver best-lock frequency and transmitter frequency to change from prelaunch values. Therefore, based on the predicted temperature at DSS initial acquisition and the measured frequencies before launch, frequency vs temperature data for both spacecraft transmitter and receiver were used to predict frequencies at acquisition.

No telecommunications system anomalies or performance degradations were observed during any of the prelaunch countdown phases, with the exception of *Surveyor II*. On *Surveyor II*, receiver B registered abnormally low received signal levels and appeared to have degraded performance. At the time, however, the reduced signal levels were attributed to the multipath conditions commonly associated with the RF link at AFETR. It was later determined in flight that the receiver was indeed not performing normally.

b. Initial DSS acquisition phase. The initial acquisition phase denoted the period of the mission when RF contact, both uplink and downlink, was established for the first time by a Deep Space Station. In order for the acquisition to be accomplished normally, four requirements had to be satisfied:

(1) Doppler rates not in excess of receiver tracking rate capabilities.

(2) Maximum antenna tracking rate not exceeded.

(3) Spacecraft visible for a sufficient period of time.

(4) Adequate signal power received at both the spacecraft receivers and the DSS receivers.

Items (1) through (3) were determined from the trajectory data for each mission; however, item (4) was determined from a statistical analysis of the omniantenna patterns. During this period the spacecraft was sun-oriented. The tipoff rate and roll position constraints imposed on the *Centaur* limited the uncertainty in spacecraft roll attitude to a ± 60-deg region. This same region of the antenna patterns was integrated to determine the percentage of the region which contained gains above the required antenna gain. This percentage of coverage was then a measure of the probability of achieving the required signal power on the uplink and downlink. For all missions, it was determined that the percentage of coverage was at least 99%. This was confirmed by mission data, which showed that the received signal power was never a limiting factor during the acquisition.

In addition to the desirability of assessing the spacecraft performance as soon as possible after launch, there was also a thermal constraint which required an immediate initial acquisition. This was due to the high compartment temperatures which occurred during high-power operation of the spacecraft transmitter. It was required that the spacecraft transmitter be switched from high-power to low-power operation within 1 hr maximum after injection (spacecraft separation).

Before initial acquisition by the Deep Space Station, the spacecraft was transmitting in high power at 550 bits/sec with a special low-modulation index. Since the spacecraft was not roll-attitude controlled, it was not known

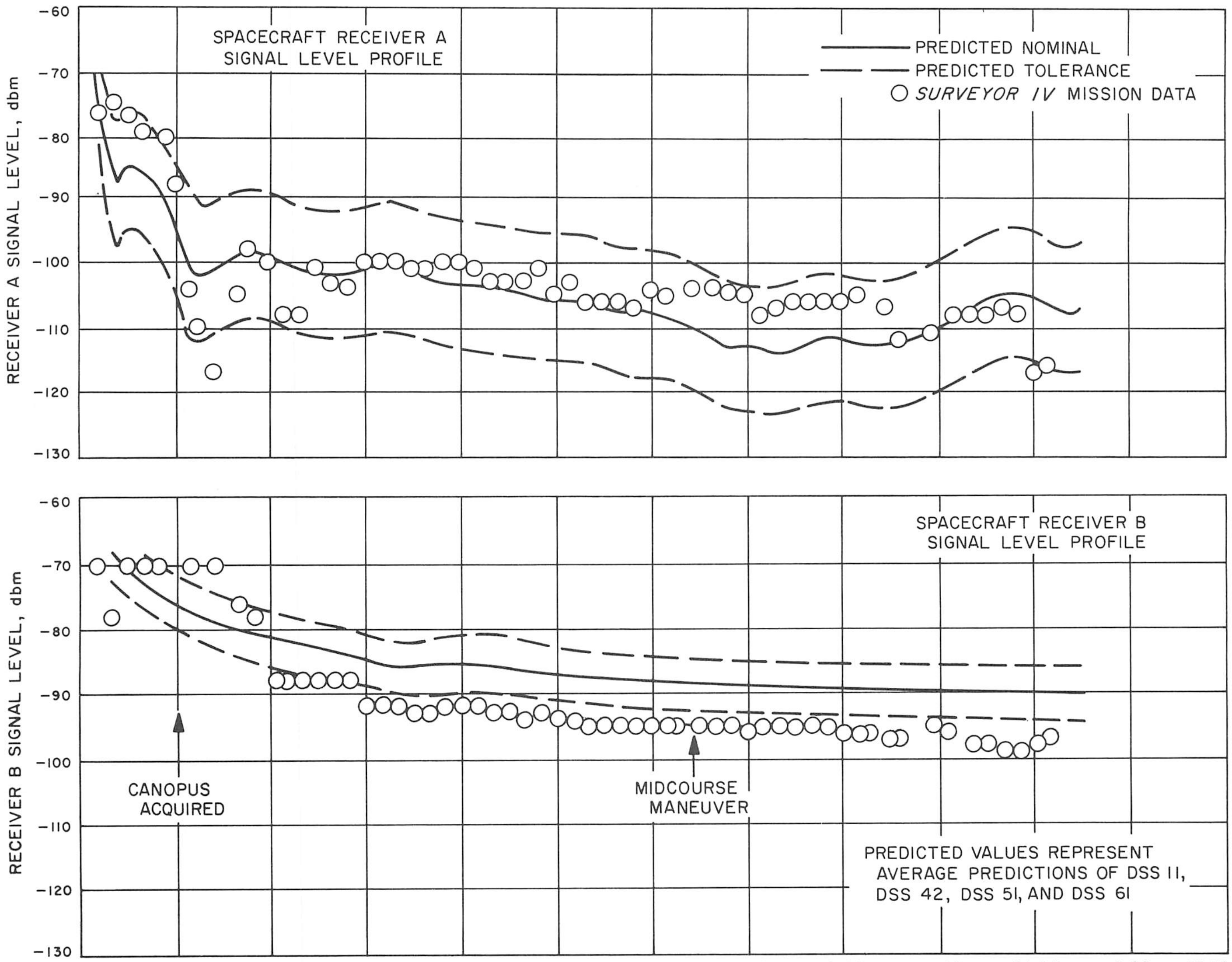


Fig. VI-36. *Surveyor IV* spacecraft receiver signal level profile

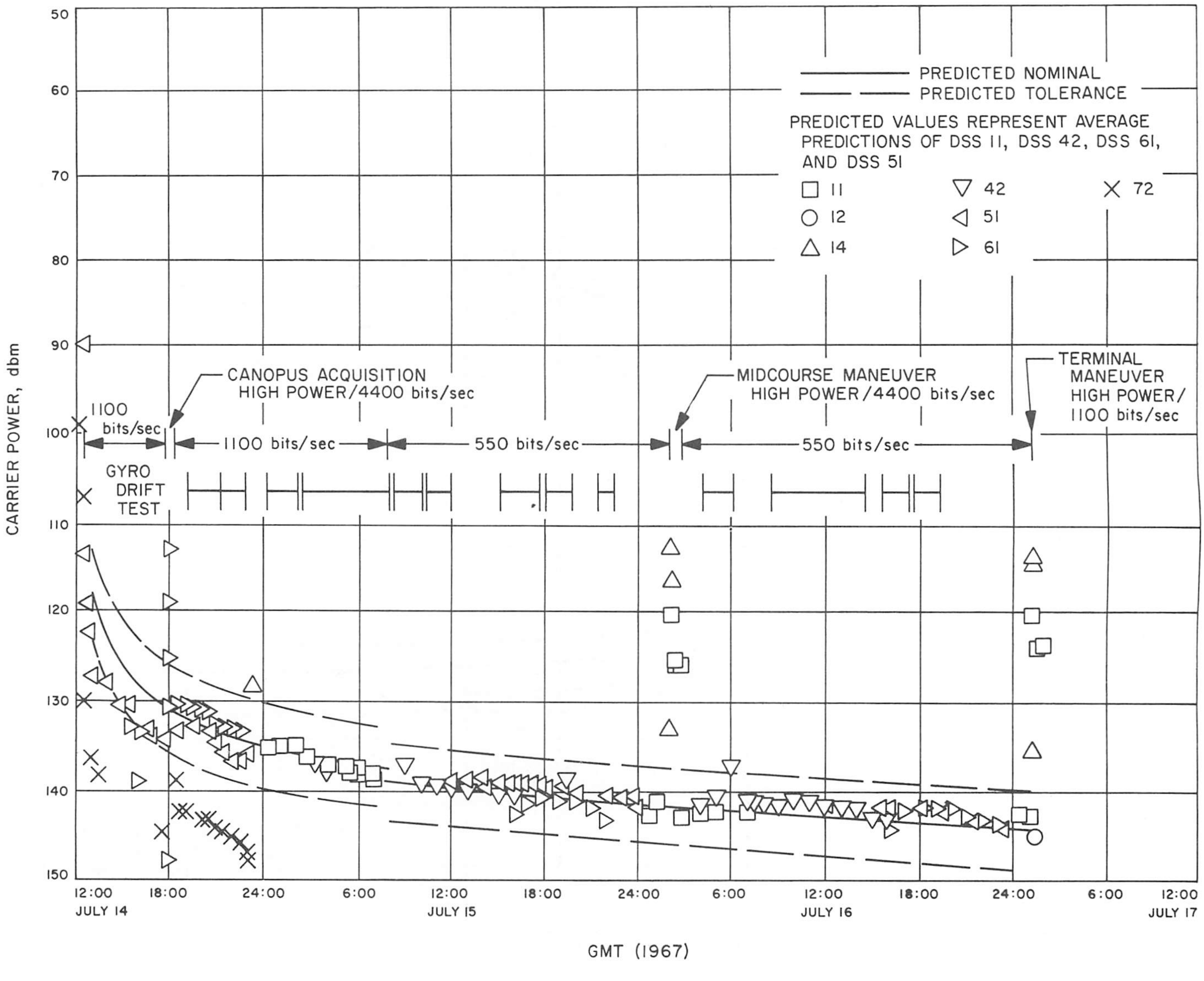


Fig. VI-37. Predicted signal levels for *Surveyor IV*: DSS received carrier power

in advance which antenna would be best. Therefore, omniantenna B was used. In addition, the spacecraft was launched with receiver B transponder on and receiver A AFC mode on. The frequency predictions given to the ground station were for acquisition of receiver B.

The initial acquisitions of each of the *Surveyor* spacecraft were accomplished in a nominal manner. In all cases, the spacecraft transmitter was transferred from high-power to low-power operation within the required 1-hr period. Table VI-13 summarizes the performance for each of the missions.

After the initial acquisition had occurred, the spacecraft was switched to 1100 bits/sec. At this time, the performance of the spacecraft receivers was assessed. After spacecraft receiver B was adjusted for zero static phase error (SPE), the deviation from center frequency for spacecraft receiver A was checked. If the deviation, or AFC error, was greater than ± 25 kHz, however, which could occur for some receivers, both receivers were phase-locked and pulled in frequency an equal distance away from their respective zero SPE positions. This procedure ensured a command capability in either of the two receivers.

The only telecommunication problem encountered during the acquisition phase was the failure of omniantenna A to deploy for *Surveyor I*. Since omniantenna B was the prime antenna during the transit phase, the stowed antenna did not adversely affect the mission. Omniantenna A later extended to its normal operating position at touchdown.

c. Transit phase I (sun acquired only). This phase covers the period from initial spacecraft acquisition until Canopus acquisition. During this time, the spacecraft attitude was uncertain in roll, and the roll axis was oriented toward the sun. Because of the uncertainty in roll attitude, the omniantenna gains could vary appreciably during this period. Predictions of performance were not attempted at this time; however, the relative performance of spacecraft receivers A and B was an indication of the attitude of the spacecraft as a basis for assessment of telecommunication performance. The normal mode of operation at this time was at 1100 bits/sec over omniantenna B. The antennas were not switched unless there were excessive bit error rates at 1100 bits/sec, although a greater gain of omniantenna A could occur. *Surveyor V* was the only flight in which a switch to omniantenna A was necessary. There were no telecommunication problems encountered during this mission phase for any of the spacecraft.

d. Canopus acquisition. The spacecraft having acquired the sun shortly after separation from the *Centaur*, remained sun-oriented, but the roll attitude was not celestially referenced until Canopus acquisition occurred. At approximately 6–10 hr after launch, the spacecraft was rolled in order to acquire Canopus. As explained here, the spacecraft attitude in the X–Y (roll) plane was uncertain to ± 60 deg before Canopus acquisition. Since the earth–probe–sun (EPS) angle was known, the estimated attitude for the start of the roll and the EPS angle could be used to analyze the antenna patterns for a 360-deg roll. The considerations for execution of the Canopus acquisition were:

(1) One-way or two-way lock was recommended based on the probability of maintaining phase lock on the uplink. If the spacecraft transponder dropped phase lock, the spacecraft transmitter reverted to the narrow-band voltage-controlled crystal oscillator (NBVCXO) for its frequency control; the shift in frequency that commonly occurred could also cause the ground receiver to phase-lock. Thus, there was a possible reacquisition problem under these circumstances.

(2) The desired spacecraft transmission mode during Canopus acquisition was in high power at 4400 bits/sec. *Surveyors I* and *II* had erroneous temperature measurements at 4400 bits/sec; thus, 1100 bits/sec was used instead. This problem was corrected for subsequent spacecraft.

(3) The number of rolls and the choice of antenna were based on the null regions of the spacecraft antenna. In order to map all the stars, the selected antenna must not have caused a data outage in the vicinity of any one star. If there were stars in the null regions of both of the antennas, one roll on each antenna was necessary for a complete map.

The recommendations for execution of the Canopus acquisition and the subsequent performance of each of the *Surveyor* spacecraft are shown in Table VI-14. This table shows the missions which were executed in two-way and in one-way, as well as the missions when antenna nulls caused losses of data.

All Canopus acquisitions were nominal with the exception of *Surveyor VI*. For *Surveyor VI*, the uplink omniantenna B null region had lower gains than anticipated, and two-way lock was lost for a period of time. It was necessary to complete the roll in one-way lock. This caused a loss of some data and a slight delay of final Canopus acquisition.

Table VI-13. *Surveyor initial acquisition performance*

Mission	Station and time of initial acquisition	Initiation of high-power operation	Transfer to low-power operation	Elapsed time in high power	Comments
Surveyor I	DSS 51 $L + 23$ min, 20 sec	$L + 13$ min, 32 sec	$L + 39$ min, 42 sec	26 min, 10 sec	
Surveyor II	DSS 51 $L + 23$ min, 07 sec	$L + 12$ min, 21 sec	$L + 44$ min, 33 sec	32 min, 12 sec	High tracking rates caused a slight delay in acquisition of two-way lock and subsequent command transmissions
Surveyor III	DSS 42 $L + 50$ min, 09 sec	$L + 34$ min, 53 sec	$L + 64$ min, 56 sec	30 min, 35 sec	
Surveyor IV	DSS 72 $L + 16$ min, 27 sec	$L + 12$ min, 26 sec	$L + 40$ min, 39 sec	28 min, 13 sec	A transient in the DSS 72 tracking servo system caused the antenna to slew 15 deg off the spacecraft. This resulted in a delay of two-way lock
Surveyor V	DSS 51 $L + 27$ min, 24 sec	$L + 19$ min, 15 sec	$L + 39$ min, 59 sec	20 min, 44 sec	High tracking rates were responsible for two-way lock onto a sidelobe of the ground station antenna. This delayed the confirmation of a normal spacecraft transmitter power transfer
Surveyor VI	DSS 51 $L + 29$ min, 50 sec	$L + 25$ min, 19 sec	$L + 40$ min, 32 sec	15 min, 13 sec	
Surveyor VII	DSS 42 $L + 50$ min, 24 sec	$L + 34$ min, 59 sec	$L + 61$ min, 46 sec	26 min, 47 sec	

Table VI-14. *Surveyor star acquisition performance*

Mission	Recommendations for maneuver	Performance	Comments
Surveyor I	(1) Maneuver can be executed in two-way (2) Spacecraft configuration should be high power, 1100 bits/sec, and mode 5 data (3) Execute one roll on omniantenna B	(1) Maneuver was executed in two-way using Transponder B. Phase lock was maintained throughout the maneuver and there was no loss of downlink carrier at any time (2) Spacecraft configuration was at high-power, 1100 bits/sec, and mode 5 data (3) One roll was executed on omniantenna B (4) There were no data outages	
Surveyor II	(1) Maneuver should be executed in one-way (2) Spacecraft configuration should be high-power, 1100 bits/sec, and mode 5 data (3) Execute one roll on omniantenna B. (If lost stars are acquired, there is a possibility of an additional roll needed on omniantenna A.) (4) There is a possibility of the loss of data and downlink carrier at some time during a 360-deg roll on either omniantenna A or B	(1) Maneuver was executed in one-way. There was no loss of downlink carrier at any time (2) Spacecraft configuration was of high-power, 1100 bits/sec, and mode 5 data (3) One roll on each omniantennas B and A was necessary for a complete star map (4) Data was noisy within the predicted regions; however, no significant data outages occurred	
Surveyor III	(1) Maneuver can be executed in two-way, if required. Loss of the downlink may occur when passing through the null; however, the uplink should maintain lock throughout the complete roll	(1) Maneuver was executed in one-way. There was no loss of downlink carrier at any time (2) Spacecraft configuration was at high-power, 4400 bits/sec, and mode 5 data (3) One roll was executed on omniantenna B	One-way operation was selected in preference to two-way because of an FPAC input that two-way tracking was not required

Table VI-14 (contd)

Mission	Recommendations for maneuver	Performance	Comments
Surveyor III (contd)	(2) Spacecraft configuration should be high-power, 4400 bits/sec, and mode 5 data (3) Execute one roll on omniantenna B (4) There is a possibility of the loss of data between 220 and 250 deg in a positive roll sense from Canopus	(4) Some parity errors were noted in the data: minor difficulties were experienced in maintaining decommutator lock. No significant data outages occurred	
Surveyor IV	(1) Maneuver can be executed in two-way. Loss of the downlink may occur when passing through the null; however, the uplink should maintain lock throughout the complete roll (2) Spacecraft configuration should be high-power, 4400 bits/sec, and mode 5 data. (3) Execute one roll on omniantenna B. An additional roll on omniantenna A may be necessary if Eta Ursa Majoris does not appear on first roll (4) There is a possibility of the loss of data between 217 and 237 deg in a positive roll sense from Canopus	(1) Maneuver was executed in two-way using transponder B. Phase lock was maintained throughout the maneuver and there was no loss of downlink carrier at any time (2) Spacecraft configuration was of high-power, 4400 bits/sec, and mode 5 data. (3) Only one roll on omniantenna B was required (4) Minor difficulties were experienced in maintaining decommutator lock; however, no significant data outages occurred	
Surveyor V	(1) Maneuver can be executed in two-way. Loss of the downlink may occur when passing through the null; however, the uplink should maintain lock throughout the complete roll (2) Spacecraft configuration should be high-power, 4400 bits/sec, and mode 1 data (3) Execute one roll on omniantenna B (4) There is a possibility of the loss of data between 160 and 240 deg in a positive roll sense from Canopus. No stars will be missed because of data outages on omniantenna B	(1) Maneuver was executed in two-way using transponder B. Phase lock was maintained throughout the maneuver and there was no loss of downlink carrier at any time (2) Spacecraft configuration was at high-power, 4400 bits/sec, and mode 1 data (3) One roll was executed on omniantenna B (4) Data outages occurred within the predicted region	
Surveyor VI	(1) Maneuver can be executed in two-way. Loss of the downlink may occur when passing through the null on omniantenna B during the second roll; however, the uplink should maintain lock throughout both rolls (2) Spacecraft configuration should be high power, 4400 bits/sec, and mode 1 data. (3) Start the roll on omniantenna A and remain until either the moon is seen or for 180 deg, whichever occurs first. Switch to omniantenna B and complete map. Remain on omniantenna B and roll to acquire Canopus (4) There is a possibility of the loss of data between 187 and 237 deg in a positive roll sense from Canopus during the second roll on omniantenna B	(1) Maneuver was executed in two-way until phase lock was lost. The subsequent part of the maneuver was executed in one-way. The downlink carrier was lost during the same period that uplink phase lock was lost (2) Spacecraft configuration was at high-power, 4400 bits/sec, and mode 1 data (3) The initial roll was executed on omniantenna A. The switch to omniantenna B took place in accordance with the recommendations. After the period of loss of lock, one complete roll was executed on omniantenna B (4) Data outage occurred during the loss of downlink carrier period. Intermittent decommutator lock occurred within the predicted null region	An unexpected deep null in the uplink omniantenna B pattern caused a loss of phase lock during the initial roll. The loss of uplink lock also caused a loss of downlink lock. The loss of lock condition existed for about 9 min during which the spacecraft had been allowed to roll 100 deg. The maneuver was resumed in one-way lock and all subsequent events were normal

Table VI-14 (contd)

Mission	Recommendations for maneuver	Performance	Comments
Surveyor VII	(1) Maneuver can be executed in two-way. Loss of the downlink may occur when passing through the null; however, the uplink should maintain lock throughout the complete roll (2) The spacecraft configuration should be high power, 4400 bits/sec, and mode 1 data. (3) Execute one roll on omniantenna B (4) There is a possibility of the loss of data between 204 and 234 deg in a positive roll sense from Canopus	(1) The maneuver was executed in one-way. There was no loss of downlink carrier at any time (2) Spacecraft configuration was at high-power, 4400 bits/sec, and mode 1 data (3) One roll was executed on omniantenna B (4) Intermittent decommutator lock occurred within the predicted null region	One-way operation was selected in preference to two-way because of an FPAC input that two-way tracking was not required

e. Transit phases II and III. Transit phase II covers the period from Canopus acquisition until the midcourse maneuver, and transit phase III covers the period from the midcourse maneuver until the terminal maneuver. Since the three-axis attitude of the spacecraft was celestially referenced during these phases, preflight and inflight predictions of telecommunication performance could be generated.

Some sample time-variable predictions of the uplink and downlink performance for *Surveyor IV* are shown in Figs. VI-36 and VI-37. The actual measured values during the mission are superimposed on the plots. Throughout the coast phases for each mission, gyro drift checks were performed periodically. These gyro drift checks produced spacecraft attitude variations which were not considered when the predictions were generated. Therefore, many of the deviations in actual signal levels from the nominal predicted values can be attributed to this cause; however, the tolerances on the predictions bound these variations in most cases.

The normal data transmission mode for transit phase II was at 1100 bits/sec for all missions with the exception of *Surveyor IV*. For *Surveyor IV*, the midcourse maneuver was delayed until the second pass over the Goldstone Tracking Station and it was necessary to switch to 550 bits/sec before the maneuver. There were no telecommunication problems on any of the missions during this mission phase; however, the degraded performance of receiver B on *Surveyor II* was investigated during this period. It was determined that the −16-db bias in the receiver operation would still provide an adequate command margin at lunar distances.

The minimum data transmission mode for transit phase III was at 550 bits/sec for all missions with the exception of *Surveyor III*. For *Surveyor III*, it was necessary to reduce the data rate to 137.5 bits/sec in order to obtain good data quality. There were no telecommunication problems which occurred during this phase for any of the missions; however, there were some operational difficulties associated with the tumbling spacecraft of *Surveyor II*. The spacecraft telecommunication system performed as expected under the conditions.

f. Midcourse maneuver. As soon as enough tracking data were received during a mission in order to define the trajectory, the necessary midcourse maneuver possibilities to correct for injection errors were investigated (see Section XI). In general, for each type of maneuver, there were eight attitude maneuver pairs considered. The eight pairs consisted of two sets of roll–yaw, roll–pitch, pitch–yaw, and yaw–pitch combinations. One set was identified as the standard maneuvers and corresponded to maneuvers which resulted in minimum time durations of rotations. The other set was identified as the optional maneuvers and were related to the standard maneuvers, but were not for minimum time durations. Since initially both 15- and 40-hr midcourses were considered, the maneuvers for both had to be analyzed.

To analyze all the maneuver possibilities, contour plots of the antenna patterns were used to great advantage to assess the performance margins for uplink and downlink communications during the maneuvers. The midcourse and terminal guidance system (MTGS) computer program was also used for the analysis. The MTGS computer program had as one of its outputs a tabulation of look angles, antenna gains, command margins, telemetry margins, and a summary of the maximum and minimum margins to be expected for each maneuver. The program incorporated digital antenna data, fixed system parameter values, and spacecraft attitude information in order to compute the desired information. The primary considerations for analysis of the maneuvers from the telecommunications standpoint were:

(1) The normal spacecraft transmission mode was in high power at 4400 bits/sec.

(2) It was desired that there be adequate minimum telemetry performance margins during the turns and good margin at the end of the turns.

(3) It was desired to operate in two-way lock during the turns and during the thrusting.

(4) It was desired to maintain command capability throughout the maneuver period with *both* spacecraft receivers, if possible. The most important consideration, however, was that there be command capability with both receivers during the thrusting period.

(5) It was operationally undesirable to switch antennas between turns or before thrusting.

Figure VI-38 is a sample antenna contour plot showing the maneuver performed for *Surveyor II*. No telecommunication subsystem problems were encountered during the midcourse maneuver period for any of the missions. Two-way tracking was maintained throughout the maneuver and thrusting periods. With the exception of *Surveyor V*, the maneuvers were performed with data transmission via omniantenna B in high power at 4400 bits/sec. During the *Surveyor V* mission when six maneuvers were performed, the fourth was executed in low power at 1100 bits/sec. Table VI-15 summarizes the performance of *Surveyor V* for each of six maneuvers.

g. Terminal phase. The terminal phase covers the time from the start of the terminal maneuver until the time of touchdown. For the purpose of analysis of the telecommunications system, the terminal phase is divided into the following three sections:

(1) Terminal maneuver.

(2) Retroignition through vernier descent.

(3) Touchdown.

1. Terminal maneuver. The terminal maneuver was performed in order to orient the spacecraft for the predetermined landing sequence and final touchdown orientation.

The preliminary maneuvers for the terminal sequence were selected before the actual mission. This allowed the telecommunication performance to be assessed prior to flight in the same manner that the midcourse maneuvers were analyzed in flight. The terminal maneuver possibilities consisted of the same sets of maneuvers as described for the midcourse maneuvers, with the exception that each maneuver set could be followed by a roll maneuver to optimize the spacecraft orientation for scientific and/or engineering considerations. Because of the availability of the 210-ft antenna at DSS 14, there was virtually complete flexibility in the choice of the roll attitude. As in the case of the midcourse maneuvers, the telecommunication subsystem capability for the terminal maneuvers could be readily assessed with the use of the MTGS computer program information.

No telecommunications problems were encountered during the terminal maneuver phase of any mission.

The maneuvers for all the missions were performed with data transmission via omniantenna B in high power at 1100 bits/sec. The one-way tracking mode was used even though adequate margins were available for two-way operation. The one-way configuration was desired for the terminal-descent sequence and, operationally, it was considered safer not to make a change following the terminal maneuver.

2. Retroignition through vernier descent. The spacecraft attitude was held inertial following completion of the terminal maneuvers and throughout the retro engine burn phase. The expected telecommunication performance during this period could, therefore, be determined by the existing performance margins at the last rotation of the terminal maneuver.

Following the retro engine burnout, the spacecraft attitude was controlled by doppler steering. The burnout velocity dispersions, however, could result in different trajectories during the vernier descent. In order to analyze the telecommunication performance during this period, a Monte Carlo simulation by computer program was used.

For each run, the simulation was made for 100 cycles where each cycle corresponded to a possible trajectory. The program subsequently generated histograms of the lowest omniantenna gains which occurred during RADVS acquisition and vernier descent for each cycle. Thus, histograms indicated the percentage of 100 possible trajectories for which the minimum available antenna gains were greater than, or equal to, the required antenna gain. This percentage was used with plots of confidence intervals to express the link performance during this phase. The actual expected link performance was expressed in terms of the minimum probability of occurrence at the 95% confidence level for 100 samples.

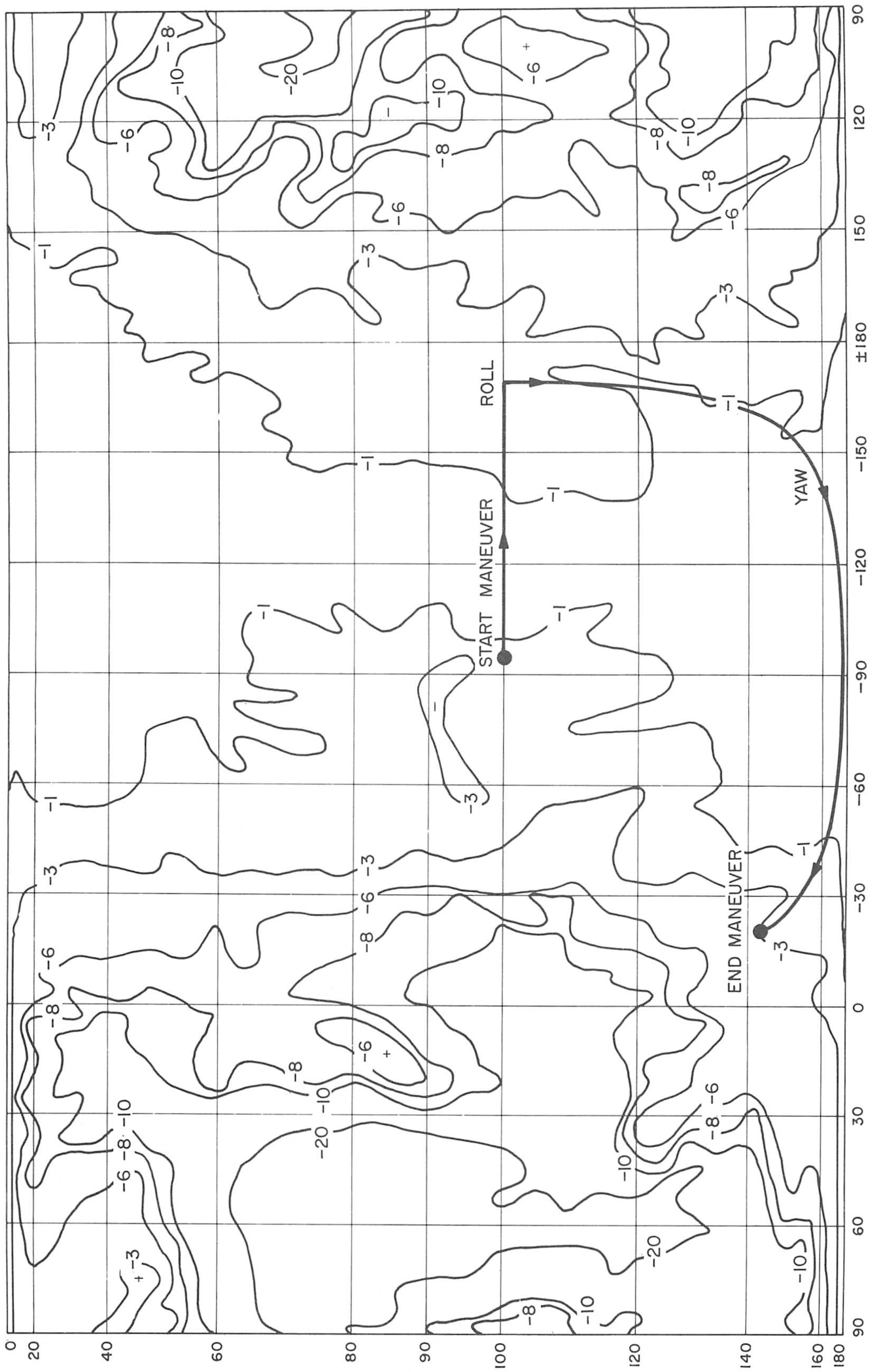


Fig. VI-38. *Surveyor II* midcourse maneuver trace: omniantenna B downlink contour map

Table VI-15. *Surveyor V* midcourse maneuvers

Maneuver	Start time (GMT) on Sep 9, 1967 hr:min:sec	Attitude rotation and thrusting	Signal level variation, db						Spacecraft to earth PCM data rate, bits/sec	Minimum margin, db						Comment
			Spacecraft to earth (omniantenna B)		Earth to spacecraft					Spacecraft to earth (PCM data)		Earth to spacecraft (command)				
					Receiver A		Receiver B					Receiver A		Receiver B		
			Predicted	Observed	Predicted	Observed	Predicted	Observed		Predicted	Observed	Predicted	Observed	Predicted	Observed	
Midcourse I (nominal midcourse)	01:32:57	Pre-thrust roll: +71.8 deg	2.8	3.3	18.2	17.5	5.4	7.1	4400	15.2	17.0	8.4	4.7	23.1	21.8	High power on at 01:15:20
	01:38:06	Pre-thrust yaw −35.6 deg	1.7	2.4	3.9	3.5	2.4	3.2	4400	15.8	17.4	18.8	21.9	25.9	25.7	High power not commanded off until after midcourse 3
	01:45:02	Thrust: 14.25 sec	—	0.2	—	0.3	—	0.0	4400	—	19.3	—	25.1	—	25.7	
	01:53:51	Post-thrust yaw: +35.6 deg	1.7	2.3	3.9	2.7	2.4	3.8	4400	15.8	17.0	18.8	22.7	25.9	24.7	
Midcourse 2 (sunline maneuver)	02:12:02	Thrust: 10.05 sec	—	0.4	—	1.2	—	0.4	4400	—	17.2	—	20.2	—	28.5	High power on before midcourse 1; not turned off until after midcourse 3
Midcourse 3 (anti-sunline maneuver)	02:24:33	Pre-thrust yaw: −180 deg	10.0	10.9	13.0	11.8	8.0	9.3	4400	5.8	9.4	11.0	12.7	20.0	19.2	High power off at 02:55:41
	02:39:51	Thrust: 23.05 sec	—	0.7	—	0.2	—	0.0	4400	—	8.7	—	19.2	—	22.9	High power operation for 1 hr, 40 min, 21 sec
	02:45:22	Post-thrust yaw: −180 deg	11.0	11.5	7.0	10.2	17.0	9.7	4400	4.8	8.8	17.0	15.8	11.2	19.2	
	04:00:47	Post-thrust roll: −71.8 deg	2.8	—	18.2	13.2	5.4	7.0	1100	18.3	—	8.4	7.5	22.9	21.2	Roll executed in low power
Midcourse 4	04:05:55	Pre-thrust roll: +68.5 deg	2.8	—	18.2	14.0	5.4	7.0	1100	17.9	—	8.4	7.9	22.5	21.5	Maneuver executed in low power
	04:09:33	Pre-thrust yaw: +106.8 deg	1.0	—	6.0	6.4	8.0	7.4	1100	17.7	—	17.6	13.7	19.6	21.5	
	04:18:48	Thrust: 13.0 sec	—	—	—	0.7	—	0.4	1100	—	—	—	18.4	—	22.4	Post-thrust maneuvers executed over DSS 42
	04:58:13	Post-thrust yaw: −106.8 deg	1.0	—	6.0	8.1	8.0	7.0	1100	17.7	—	17.6	14.6	19.6	17.7	
	05:04:11	Post-thrust roll: −68.5 deg	2.8	—	18.2	10.9	5.4	6.7	1100	17.9	—	8.4	12.0	22.5	17.7	
Midcourse 5	07:54:30	Pre-thrust roll: +64.6 deg	2.8	—	18.2	9.8	5.2	5.9	4400	13.8	—	7.0	11.8	21.7	21.5	High power on at 07:49:35 and off at 08:46:51
	07:58:31	Pre-thrust yaw: +143.2 deg	4.3	—	6.7	9.9	9.3	10.1	4400	12.6	—	17.0	14.2	13.2	19.2	
	08:24:04	Thrust: 33.0 sec	—	—	—	0.0	—	0.0	4400	—	—	—	24.1	—	20.6	High-power operation for this maneuver lasted 57 min, 16 sec
	08:33:51	Post-thrust yaw: −143.2 deg	4.3	—	6.7	9.3	9.3	8.8	4400	12.6	—	17.0	14.8	13.2	19.7	
	08:40:35	Post-thrust roll: −64.6 deg	2.8	—	18.2	11.6	5.2	5.3	4400	13.8	—	7.0	8.5	21.7	21.8	
Midcourse 6	23:13:51	Pre-thrust roll: −76.0 deg	8.0	7.0	15.0	19.3	3.0	7.5	4400	5.4	6.5	−3.1	−2.4	17.9	17.5	High power on at 23:06:30 and off at 23:51:35
	23:19:36	Pre-thrust yaw: −100.6 deg	10.0	10.0	10.0	10.9	13.0	5.9	4400	3.7	5.9	9.9	10.0	6.9	12.8	
	23:30:58	Thrust: 5.45 sec	—	1.4	—	0.0	—	0.3	4400	—	15.9	—	11.4	—	15.0	High-power operation for this maneuver lasted 45 min, 5 sec
	23:36:07	Post-thrust yaw: +100.6 deg	10.0	10.8	10.0	11.2	13.0	7.9	4400	3.7	5.8	9.9	9.5	6.9	11.8	
	23:44:05	Post-thrust roll: +76.0 deg	8.0	6.9	15.0	11.0	3.0	5.1	4400	5.4	6.1	−3.1	5.7	17.9	18.9	

No telecommunication problems were encountered during the retro engine firing and vernier descent during any mission except *Surveyor IV* when the spacecraft signal was abruptly lost just before the end of retro burn. Although there are failure modes that could have caused the observed signature, this anomaly has not been attributed to the telecommunication system.

3. *Touchdown.* The touchdown phase required an analysis that took into account the statistical nature of the earth vector position at touchdown. There was a region of uncertainty in which the spacecraft-to-DSS vector could lie, owing to possible changes in the spacecraft roll orientation during vernier descent and as a result of touchdown. Using the look angle regions determined by the possible earth vector position as a result of spacecraft motion, an antenna gain coverage was computed by integrating the antenna gain values over the regions to determine the gain distribution within the regions. The link performance was assessed relative to the percent of the omniantenna gains available which were equal to or greater than the required gain.

The desired operating mode at touchdown was 1100 bits/sec and strain gages; the alternate modes were 1100 bits/sec only, 550 bits/sec plus strain gages, and 550 bits/sec only. Since the 1100 bits/sec strain-gage data mode required relatively high performance margins, the recommendation to use this mode was made only after careful consideration of many factors. The performance history of the telecommunication subsystem for the mission, the Monte Carlo analyses, the integration of antenna gains over the expected touchdown region, and the expected ground station performance were all determining factors.

The desired 1100 bits/sec plus strain-gage data mode was achieved for all applicable missions. This data mode was operated just before touchdown for *Surveyors I* and *III*. Because of the type of failure of *Surveyor IV*, however, subsequent missions used the 1100 bits/sec plus strain-gage data mode throughout retro burn and descent. Good pulse code modulation (PCM) and strain-gage data were obtained for all applicable missions with the exception of *Surveyor III*. At *Surveyor III* touchdown, the received signal quality was good; however, the PCM data became erroneous. Further investigation indicated that one or more commutator switches in the ESP and AESP had shorted. Further analysis permitted corrections to selected data channels at low bit rates so that lunar operations were not impaired; all science objectives of the mission were conducted.

h. Anomalies and conclusions. Table VI-16 lists only five mission anomalies associated with or affecting the telecommunication system. There was a sixth occurrence, which was originally thought to be an anomaly but later identified as a characteristic of the omniantenna data. It was determined from mission experience that the uplink omniantenna B patterns had a +2-db bias (see Subsection J-2). With the exception of the loss of signal from *Surveyor IV*, no associated telecommunication subsystem anomaly affected the successful completion of the science objectives of any mission. Although the telecommunication subsystem has failure modes that produce the loss of signal observed for *Surveyor IV*, the anomaly has not been attributed to the subsystem.

Table VI-17 shows the signal level performance for each mission. It should be noted that the tolerances on predicted values are not included. Thus, in some cases, the summary of performances does not fully reflect nominal expected performance. In general, with the proper adjustment for uplink omniantenna B patterns, it was found that, in most cases, near nominal performances were experienced in both the uplinks and downlinks. Furthermore, with the exception of receiver B performance on *Surveyor II*, and the unexpected deep null during Canopus acquisition on *Surveyor VI*, the telecommunication subsystem performed as expected.

2. Antenna Measurement Program

Almost $1\frac{1}{2}$ yr before the first *Surveyor* launch, it was determined that the large gain uncertainties of *Surveyor* omniantenna pattern measurements, which had been made by that time, could possibly endanger, if not compromise, the *Surveyor* missions. A study was made that showed the uncertainties in the antenna patterns were ± 9.5 db and ± 14.0 db at antenna gain levels of 0 and -10 db, respectively. For these magnitudes of uncertainties, the following effects on the *Surveyor* missions were concluded:

(1) Command link could not be guaranteed at near lunar distances.

(2) Most of the downlink communication modes could be seriously degraded.

(3) Data link during midcourse maneuvers could not be guaranteed at maximum maneuver angles.

(4) Continuous telemetry could not be guaranteed to lunar ranges.

(5) Inflight assessment of the telecommunication performance would be handicapped.

(6) Probability of initial acquisition of the spacecraft might be reduced.

Table VI-16. Telecommunication subsystem anomalies

Spacecraft	Anomaly	Cause	Effect on mission	General comments
Surveyor I	Nonextension of omniantenna A	Omniantenna A was slightly misaligned with clamp that held it in stowed position. Resulting friction force developed by misalignment prevented antenna from freeing itself from clamp	No adverse effect on mission as a whole. Receiver A experienced large signal variations during maneuvers. With sun/Canopus locks, antenna gain was higher than predicted for extended omniantenna A. Omniantenna B was prime antenna even before antenna A problem	Relating omniantenna gain pattern data for stowed and extended case with observed levels made it possible to determine whether antenna A was stowed or extended verification switch was bad
Surveyor II	Degraded performance of receiver B	Threshold degradation could be duplicated by simulating a loss in gain in either the ×6 multiplier into first mixer or output of oscillator which drives this multiplier. Failure apparently took place before launch, since similar problems were noted prior to countdown	Receiver B was degraded 16 db, but at lunar distances command margin would still be a positive 1 to 2 db	Planned postmidcourse test to determine whether receiver B was degraded was eliminated after mission became nonstandard
Surveyor III	Degradation of telemetry signal processing	One or more commutator switches in both ESP and AESP had shorted at touchdown	Analog data were erroneous. Digital and television telemetry data were not affected	Further analysis permitted corrections to selected data channels at low bit rate so that lunar operations were not impaired
Surveyor IV	Loss of signal during retro firing	No definite cause has been found Two most likely transmitter associated failures that would reproduce failure signature are: (1) opening or shorting of 29-V low ripple line to transmitter, or (2) opening of coax from transfer switch to the SPDT switch	Termination of mission	This anomaly has not been attributed to RF subsystem
Surveyor VI	Loss of signal during Canopus acquisition	An unexpected deep null in the uplink omniantenna B pattern caused the spacecraft receiver to drop phase lock. The subsequent shift in frequency as the spacecraft transmitter reverted to the NBVCXO for its frequency control caused the ground receiver to also lose phase lock	There was a loss of some data and a slight delay of final Canopus acquisition	The unexpected deep null is attributed to the fact that the spacecraft configurations subsequent to *Surveyor III* did not represent the configuration for the measured antenna pattern data

Table VI-17. RF performance: signal level summary of maneuver phases

Mission phase	Receiver A (requirement of signal level: > -114 dbm)		Receiver B (requirement of signal level: > -114 dbm)		Deep Space Station		
	Predicted value, db	Actual performance	Predicted value, db	Actual performance	Predicted value, db	Requirement	Actual performance
Star Maneuver	Time variable predictions. Predicts are some nominal value $\pm$		Time variable predictions. Predicts are some nominal value $\pm$		Time variable predictions. Predicts are some nominal value $\pm$		
Surveyor I	8.0	Level between 4.0 and -20.0 db about nominal and > -104 dbm	7.0	Level between 6.0 and -7.0 db about nominal and > -100 dbm	7.0	> -135.5 dbm (1100 bits/sec/ high power)	Level between 7.0 and -3.0 db about nominal and > -129 dbm
Surveyor II	10.0	Level between 17.0 and -13.0 db about nominal and > -116 dbm	10.0	Level between 6.0 and -7.0 db about nominal and > -112 dbm	10.0	None	Level between 4.0 and -13.0 db about nominal and > -150.0 dbm
Surveyor III	10.0	Level between 5.0 and -6.0 db about nominal and ≥ -100 dbm	10.0	Level between 2.0 and -3.0 db about nominal and > -97 dbm	10.0	None	Level between 6.0 and -5.0 db about nominal and ≥ -146.0 dbm
Surveyor IV	10.0	Level between 8.0 and -7.0 db about nominal and ≥ -105 dbm	8.0	Level between 0 and -9.0 db about nominal and > -104 dbm	10.0	None	Level between 5.0 and -4.0 db about nominal and > -147.9 dbm
Surveyor V	10.0	Level between 8.0 and -6.0 db about nominal and ≥ -108.5 dbm	8.0	Level between 6.0 and -6.0 db about nominal and > -114 dbm	10.0	None	Level between 16.0 and -5.0 db about nominal and ≥ -143.7dbm
Surveyor VI	10.0	Level between 8.0 and -4.0 db about nominal and ≥ -102.5 dbm	8.0	Level between 5.0 and -20.0 db about nominal and < -130 dbm in null region	10.0	None	Level between 2.0 and -7.0 db about nominal and ≥ -153.4 dbm
Surveyor VII	10.0	Level between 5.0 and -10.5 db about nominal and ≥ -111.0 dbm	8.0	Level between 5.0 and -8.0 db about nominal and ≥ -106.7 dbm	10.0	None	Level between 6.0 and -6.0 db about nominal and ≥ -144.4 dbm
Coast phase							
Surveyor I	8.0	Level between 12.0 and 0 db about nominal and ≥ -95 dbm	6.0	Level between 4.0 and -4.0 db about nominal and > -100 dbm	8.0	> -159.3 dbm (carrier power) (17.2 bits/sec threshold)	Level between 3.0 and -3.0 db about nominal and > -146.0 dbm
Surveyor II	12.0	Level between 2.0 and 4.0 db about nominal and ≥ -95 dbm	7.0	Level between $+1.0$ and $+3.6$ db about nominal and ≥ -107 dbm	8.0	> -136.7 dbm (carrier power) (17.2 bits/sec threshold)	Level between $+0.5$ and -2.5 db about nominal and > -139 dbm at 1100 bits/sec
Surveyor III	10.0	Level between 2.0 and -12.0 db about nominal and ≥ -113 dbm	4.0	Level between 0 and -4.0 db about nominal and ≥ -94.0 dbm	6.0	> -157.7 dbm (carrier power) (17.2 bits/sec threshold)	Level between 2.0 and -4.0 db about nominal and > -148.0 dbm
Surveyor IV	10.0	Level between 8.0 and -12.0 db about nominal and ≥ -120 dbm	5.0	Level between -2.0 and -5.0 db about nominal and ≥ -99.0 dbm	5.0	> -157.4 dbm (carrier power) (17.2 bits/sec threshold)	Level between 2.0 and -4.0 db about nominal and > -145 dbm at 550 bits/sec
Surveyor V	10.0	Level between 10.0 and -4.0 db about nominal and ≥ -108 dbm	5.0	Level between 0.5 and -4.0 db about nominal and ≥ -97.0 dbm	5.0	> -157.4 dbm (carrier power) (17.2 threshold)	Level between 3.0 and -1.0 db about nominal and > -145 dbm at 550 bits/sec
Surveyor VI	10.0	Level between 7.0 and 1.0 db about nominal and ≥ -103 dbm	5.0	Level between 2.0 and -4.0 db about nominal and ≥ -99.0 dbm	5.0	≥ -157.4 dbm (carrier power) (17.2 threshold)	Level between 3.5 and -1.5 db about nominal and > -141 dbm at 550 bits/sec
Surveyor VII	10.0	Level between 8.0 and -20.0 db about nominal and as low as at least -130 dbm	5.0	Level between 1.0 and -4.0 db about nominal and ≥ -98.0 dbm	5.0	> -157.4 dbm (carrier power) (17.2 bits/sec threshold)	Level between 3.0 and 0.0 db about nominal and > -141 dbm at 550 bits/sec
Midcourse							
Surveyor I	Same as star maneuver		Same as star maneuver		Same as star maneuver		
Surveyor II	Spacecraft tumbling		No performance evaluation		3.0	> -135.4 dbm (carrier power at 4400 bits/sec high power)	Level between 1.0 and -3.0 db about nominal and > -124 dbm

Mission phase	Receiver A (requirement of signal level: > −114 dbm)		Receiver B (requirement of signal level: > −114 dbm)		Deep Space Station		
	Predicted value, db	Actual performance	Predicted value, db	Actual performance	Predicted value, db	Requirement	Actual performance
Surveyor III	10.0	Level between 5.0 and −6.0 db about nominal and > −105 dbm	3.0	Level between 1.3 and −3.0 db about nominal and > −92 dbm	3.0	> −136.4 dbm (carrier power at 4400 bits/sec high power)	Level between 2.0 and 0 db about nominal and > −121.5 dbm
Surveyor IV	10.0	Level between 4.0 and −9.0 db about nominal and ≥ −107.4 dbm	3.3	Level between −3.0 and −7.0 db about nominal and ≥ −98.7 dbm	2.8	> −136.0 dbm (carrier power at 4400 bits/sec high power)	Level between 0 and −2.5 db about nominal and > −126.9 dbm
Surveyor V							
Surveyor VI	10.0	Level variations of 17.1 db and ≥ −106.5 dbm	6.6	Level variations of 12.1 db and ≥ −99.5 dbm	2.9	> −136.1 dbm (carrier power at 4400 bits/sec high power)	Level variations of 5.6 db and ≥ −126.9 dbm
Surveyor VII	10.0	Level variations of 30.4 db and ≥ −126.3 dbm	5.7	Level variations of 6.1 db and ≥ −90.3 dbm	2.9	> −136.1 dbm (carrier power at 4400 bits/sec high power)	Level variations of 3.5 db and ≥ −123.8 dbm
Terminal							
Surveyor I	Same as star maneuver		Same as star maneuver		Same as star maneuver		
Surveyor II	No terminal maneuver because of spacecraft tumbling		No terminal maneuver because of spacecraft tumbling		No terminal maneuver because of spacecraft tumbling		
Surveyor III	10.0	Level variations of 26.8 and > −123.7 dbm	3.4	Level variations of 6.2 db and > −99.7 dbm	2.7	> −130.7 dbm (carrier power at 1100 bits/sec high power)	Level between 0 and −2.0 db about nominal and > −125 dbm at 1100 bits/sec
Surveyor IV	10.0	Level variations of 8.0 db and > −116.5 dbm	6.4	Level variations of 11.9 db and > −106 dbm	5.0	> −130.4 dbm (carrier power at 1100 bits/sec high power)	Level between 2.5 and −1.0 db about nominal and > −124.4 dbm at 1100 bits/sec
Surveyor V	10.0	Level variations of 22.9 db and > −119.7 dbm	4.8	Level variations of 10.8 db and > −92.6 dbm	3.0	> −130.4 dbm (carrier power at 1100 bits/sec high power)	Level variations of 3.8 db and ≥ −125.1 dbm at 1100 bits/sec
Surveyor VI	10.0	Level variations of 13.8 db and > −108 dbm	5.7	Level variations of 11.6 db and > −106.1 dbm	2.9	> −130.4 dbm (carrier power at 1100 bits/sec high power)	Level variations of 5.9 db and ≥ −127.2 dbm at 1100 bits/sec
Surveyor VII	10.0	Level variations of ≥30.0 db and as low as at least −130 dbm	6.0	Level variations of 10.7 db and > −107 dbm	2.9	> −130.4 dbm (carrier power at 1100 bits/sec high power)	Level variations of 7.0 db and ≥129.7 dbm at 1100 bits/sec

The omniantenna pattern measurements had been made on the HAC 3000-ft antenna range using a *Surveyor* MA-2 full-scale spacecraft model. The cause of the large gain uncertainties were due principally to two factors:

(1) *Surveyor* MA-2 spacecraft model used for the antenna pattern evaluation did not represent in detail the final *Surveyor* SC-1 flight configuration. This was due to lack of definition at the time the model was built.

(2) Patterns measured on the Hughes antenna range contained measurement uncertainties due to instrumentation errors and reflection errors. The reflection errors were caused by undesirable range geometry that could not be eliminated. The HAC antenna range was designed for measurement of directional antennas and was, therefore, not suitable for measurement of omnidirectional antenna patterns. In addition, the measurement uncertainties were accounted for principally by engineering judgments.

In view of the unacceptable pattern inaccuracies, it was required that the omniantenna patterns be remeasured with a substantial increase in accuracy. It was concluded that this could be accomplished by:

(1) Use of a more accurate full-scale antenna model spacecraft.

(2) Use of the JPL 3300-ft antenna range. The JPL antenna range was considered to have the following advantages:

   (a) Extensive component and system calibration procedures developed at JPL.

   (b) Accumulated knowledge of the magnitude of departure from a plane wave of the illuminating field at the receiving site. The JPL range had fewer sources of error than the HAC range because of more desirable range geometry conditions. Knowledge of the range errors was important, since these errors constituted a major contribution to the overall uncertainties assigned to the pattern data.

   (c) A digital antenna pattern recording facility. This facility allowed patterns and associated tolerances to be recorded on magnetic tape to provide increased accuracy in performing *Surveyor* communications analysis.

Therefore, a new spacecraft model was fabricated and a new full-scale measurement program was initiated at JPL.

a. RF-1 spacecraft. A study of the primary patterns of the omniantennas revealed that these antennas heavily illuminated the spacecraft structure. It was, therefore, judged that the secondary antenna patterns of the spacecraft could be perturbed by spacecraft appendage such as cables, substructure, and dielectric surfaces. It was determined that, for accurate antenna patterns, the spacecraft should be designed to represent, as accurately as possible, the nominal SC-1 configuration.

The spacecraft model, designated as the RF-1, was subsequently built to strict guidelines and specifications. The spacecraft frame was constructed of 1020 steel with an empty substructure of lightweight aluminum. All cable harnessing and dielectric surfaces were dummied to approximate the flight specification. Flight-type hardware was used where possible and when available. Major spacecraft components of flight-type quality, actually used on the RF-1 model, consisted of the following items:

Item	Category of hardware
Solar panel	T-21 test model
Planar array	T-21 test model
Omniantennas, booms, and associated RF cabling	T-21 test model
RADVS antenas	Engineering evaluation units
Canopus sensor shield	Engineering evaluation units
Crushable blocks	Class 3 test hardware
Retro nozzle	Scrap hardware

To minimize the measurement errors from the effects of spacecraft appendage deflections in a 1-g field, both analytical and experimental analysis were made of the RF-1 model. It was determined that the omniantenna and the antenna/solar panel positioner (A/SPP) mast displayed the maximum elastic deflections. These deflections were minimized by using RF transparent guy cords and special turnbuckles.

b. Spacecraft support tower. An important part of the *Surveyor* omniantenna measurements was the spacecraft support tower design. In addition to the problem of the heavy and complex *Surveyor* spacecraft configuration, the design of the support tower was affected by:

(1) The objective of obtaining patterns with minimum measurement uncertainties.

(2) The extreme low-gain characteristics of the omniantennas, which caused the patterns to be sensitive to interfering mechanical structures.

(3) The full-sphere pattern coverage requirement for the *Surveyor* spacecraft.

To minimize pattern measurement uncertainties, it was determined that, ideally, the omniantenna under test should not be positioned below the horizontal plane containing the spacecraft Z-axis, nor should the antenna enter the region behind the support tower as viewed from the transmitter. Entering these regions with the antenna would increase errors from range multipath in the case of below the horizontal plane and from support tower diffraction in the case of the region behind the support tower.

It was finally decided that, for a *Surveyor*-type spacecraft, a support tower design for single support of the spacecraft from the retro motor end was the most desirable. This type of design had the following features:

(1) Single retro mounting would reduce the spacecraft handling and realignment. This, in turn, would reduce the possibilities of damage to the spacecraft and reduce the errors in the re-establishment of the coordinate system.

(2) The *Surveyor* retro motor casing could be used to house the roll axis, the drive motor, synchro transmitter, and associated cabling. This would eliminate reflection errors from these items, usually mounted on the support tower.

(3) Full-sphere pattern coverage could be accomplished such that the antenna under test remained in the upper hemisphere at all times. This meant that, for some positions, the support tower would obscure the antenna under test; however, a field probing study was performed to assess the amount of distortion in the *Surveyor* antenna pattern when measured in the region behind the support tower. It was determined that errors on the order of ± 3.0 db at the -10-db level were possible. This amount was considered acceptable since it was a substantial improvement over the tolerances of the original pattern measurements.

c. Antenna pattern tolerances. Extensive procedures were performed to measure or calculate the magnitude of all the contributing sources to measurement inaccuracy. The uncertainties associated with the measured antenna pattern data result from the following sources of error:

1. Range reflection and diffraction. Range reflection and diffraction errors constitute the largest contributor to the total measurement uncertainty. The range errors are due primarily to multipath transmission caused by reflection from the surrounding terrain and by diffraction from the canyon ridge edge in front of the spacecraft. These errors cause a nonuniform illumination at the receiving site, producing uncertainties in the measured pattern data.

To determine the magnitude of measurement uncertainties caused by range reflection and diffraction, it was necessary to probe the field in the vicinity of the receiving aperture. This was accomplished using a *Surveyor* prototype omniantenna. The measurements were made for over 130 omniantenna positions relative to the spacecraft support tower and the reflecting terrain.

2. Instrumentation and calibration tolerances. The remainder of the tolerances attributable to measurement are related to the capabilities of the instrumentation used and the accuracy to which the calibrations are performed. The types of factors contributing to these tolerances are:

(1) *Recording system linearity.* This linearity tolerance is a measure of the departure of the receiving system from a linear response as a function of gain level, and determines the recording system dynamic range.

(2) *Recording system stability.* This stability is a measure of the change in illuminating power level between pattern records taken at different times.

(3) *Transmitting system ellipticity stability.* This stability is a measure of the change in illuminator ellipticity from day to day.

(4) *Gain standard antenna calibration.* This tolerance reflects the accuracy to which the absolute gain of a standard reference antenna can be established.

(5) *Gain comparison.* This tolerance accounts for the error in establishing the standard gain reference level for the spacecraft in the nonuniform illuminated field.

(6) *Cable insertion loss.* This tolerance accounts for the uncertainty of cable and calibrated pad insertion loss caused by measurement inaccuracies.

(7) *Antenna mismatch loss.* This tolerance accounts for the uncertainty in the mismatch loss corrections due to the inaccuracies of making VSWR measurements.

(8) *Polarization loss absolute gain calibration.* This calibration accounts for the uncertainty in establishing the standard gain reference due to the unknown antenna polarization ellipse orientations and the finite ellipticities between illuminator and dual-polarized standard gain antennas.

(9) *Rotary joint wow.* The rotary joint wow is defined as variations in the RF signal caused by rotational asymmetries of the rotary joint.

(10) *Azimuth turntable slip rings.* This tolerance accounts for variations in the amplitude of a 1-kHz audio signal caused by the rotational asymmetries of the azimuth turntable slip rings.

(11) *Spacecraft stability.* This tolerance accounts for pattern variations due to the stability of the spacecraft configuration, as the spacecraft is turned for pattern measurements.

(12) *Spacecraft pattern repeatability.* This is a measure of the ability to record a spacecraft pattern and repeat the pattern at a later date.

(13) *Wind modulation.* Excessive wind conditions, producing perturbations on the antenna gain function due to mechanical vibration of spacecraft and illuminator, are considered as a tolerance contributor.

(14) *Thermal distortions.* The perturbations on the antenna gain function caused by thermal expansion of spacecraft structure and substructure are considered as contributors to the tolerances.

(15) *Test antenna interaction.* This tolerance reflects the uncertainty in establishing a reference gain due to scattering interaction between the illuminator and test antenna.

The magnitudes of all tolerances for the *Surveyor* pattern measurements are shown in Table VI-18. Compared with the original pattern data, the gain uncertainties were reduced on the order of 5.5 and 6.5 db at the 0- and −10-db levels, respectively.

d. Use of data. The measured antenna patterns for *Surveyor* were stored in digital form on magnetic tapes and in analog form on microfilm. In addition to the use of the magnetic tapes with the telecommunications prediction computer program, the pattern data was also computer-processed to generate contour plots. Figures VI-39 to VI-42 show composite contour plots plotted on equal-area grids for each of two omniantennas at 2113.0 and 2295.0 MHz frequencies. These plots were used extensively for analysis of the midcourse and terminal maneuvers. In addition, there were also contour plots plotted on polar grids. These type of contour plots were used for analysis of the touchdown phase of each mission.

During each mission, the Canopus acquisition phase provided an excellent opportunity to assess the accuracy of the antenna patterns. Since the acquisition is performed at a constant cone angle, the ensuing roll provided a 360-deg cut of the patterns. Figure VI-43 shows a sample plot of the predicted vs actual antenna gains for the Canopus acquisition on the *Surveyor III* mission. It can be seen that the predicted and actual values agree relatively well in high-gain regions. It was this type of comparison that enabled telecommunication analysts to detect a 2-db error in the uplink omniantenna B antenna patterns. Although the shape of the actual signal level curves appeared to match the shape of the predicted signal levels for uplink omniantenna B, the pattern data was consistently 2 db higher in magnitude. In addition, the uplink omniantenna B patterns contained higher peak gains than the other patterns. It was concluded that the uplink omniantenna B patterns contained a bias estimated to be approximately 2 db in magnitude. The cause of this bias was not determined; however, the bias was detected and accounted for during all the latter missions.

It can be seen from Fig. VI-43 that the differences between predicted and actual values are greater in the low-gain regions. In most cases, however, it was found that the differences were bounded by the tolerances on the predicted values. In addition, with the proper adjustment for uplink omniantenna B patterns, all the data acquired during coast phases demonstrated an uncertainty less than ±3 db at antenna gain levels greater than −3 db. It was found that the latter missions demonstrated the greatest divergence from the predicted values; however, this was to be expected since the spacecraft configurations subsequent to *Surveyor III* were different than that used for the measurement of the antenna patterns.

Table VI-18. Antenna pattern tolerance values

Tolerance factor	Frequency, MHz	Omniantenna	Derivation method	Constant value, db	Sectionally continuous gain regions,[a] db						
					3 to 0.5	0.5 to −0.5	−0.5 to −2.5	−2.5 to −4.8	−4.8 to −7.5	−7.5 to −17.5	<−17.5
Recording system linearity	2113.3	A	Measured	—	0 + 0.01	0.03 + 0.01	0.04 + 0.01	0.06 + 0.01	0.08 + 0.01	0.11 + 0.01	0.21 + 0.01
		B			0 + 0.01[b]	0.04 + 0.01	0.05 + 0.01	0.07 + 0.01	0.09 + 0.01	0.12 + 0.01	0.22 + 0.01
	2295.0	A–B	Measured	—	0 + 0.01	0.03 + 0.01	0.04 + 0.01	0.06 + 0.01	0.08 + 0.01	0.11 + 0.01	0.21 + 0.01
RF-1 spacecraft stability	Both	A	Measured	—	0.36	0.36	0.36	0.36	0.36	0.36	0.87
		B			0.17[c]	0.17	0.17	0.17	0.17	0.17	0.45
RF-1 pattern repeatability	Both	A	Measured	—	0.76	0.76	0.76	0.76	0.76	1.36	3.43
		B			0.69[c]	0.69	0.69	0.69	0.69	1.47	4.05
Range reflection and diffraction	2113.3	A	Measured	—	1.12 + 0.30	1.87 + 0.60	2.47 + 0.20	2.87 + 0.0	2.87 + 0.48	4.17 + 0.27	6.87 + 0.42
		B			0.83 + 0.30	1.87 + 0.60	2.47 + 0.20	2.87 + 0.0	2.87 + 0.48	4.17 + 0.27	6.87 + 0.42
	2295.0	A	Measured	—	0.65 + 0.30	1.40 + 0.60	2.00 + 0.20	2.40 + 0.0	2.40 + 0.48	3.70 + 0.27	6.40 + 0.42
		B			1.12 + 0.30[b]	1.87 + 0.60	2.47 + 0.20	2.87 + 0.0	2.87 + 0.48	4.17 + 0.27	6.87 + 0.42
Recording system stability	Both	A–B	Measured	±0.25							
Transmitter ellipticity stability	Both	A–B	Measured/ estimated	Negligible							
Gain standard absolute calibration	2113.3 2295.0	A–B	Measured	±0.20 ±0.15							
Gain comparison calibration	Both	A–B	Measured	±0.10							
Omnicable insertion loss	2113.3 2295.0	A–B	Measured/ Calculated	±0.03							
Polarization loss, absolute gain calibration	Both	A–B	Calculated	Negligible							

[a] $X + Y = X$ db $+ Y$ db/db.

[b] Omniantenna B values were recorded at gain region of 4 to 0.5 db.

[c] Omniantenna B values at 2113.3 MHz were recorded at gain region of 4 to 0.5 db.

Table VI-18 (contd)

Tolerance factor	Frequency, MHz	Omniantenna	Derivation method	Constant value, db	Sectionally continuous gain regions,[a] db						
					3 to 0.5	0.5 to −0.5	−0.5 to −2.5	−2.5 to −4.8	−4.8 to −7.5	−7.5 to −17.5	< −17.5
Rotary joint wow	Both	A–B	Measured	±0.03							
Slip ring wow	Both	A–B	Measured	Negligible							
Wind modulation, < 5 mi/hr	Both	A–B	Measured/ estimated	±0.10							
Thermal distortion	2113.3	A–B	Estimated	Negligible							
	2295.0	A	Estimated	±0.2							
		B		Negligible							
Illuminator spacecraft interaction	Both	A–B	Calculated	Negligible							
Totals[d]	2113.3	A	—	—	2.95 + 0.31	3.73 + 0.61	4.34 + 0.21	4.76 + 0.01	4.78 + 0.49	6.71 + 0.28	12.09 + 0.43
		B	—	—	2.40 + 0.31[b]	3.48 + 0.61	4.09 + 0.21	4.51 + 0.01	4.53 + 0.49	6.64 + 0.28	12.30 + 0.43
	2295.0	A	—	—	2.63 + 0.31	3.41 + 0.61	4.02 + 0.21	4.44 + 0.01	4.46 + 0.49	6.39 + 0.28	11.77 + 0.43
		B	—	—	2.64 + 0.31	3.42 + 0.61	4.03 + 0.21	4.45 + 0.01	4.47 + 0.49	6.58 + 0.28	12.24 + 0.43

[a] $X + Y = X$ db $+ Y$ db/db.

[b] Omniantenna B values were recorded at gain region of 4 to 0.5 db.

[d] Totals include constant values and sectionally continuous gain values.

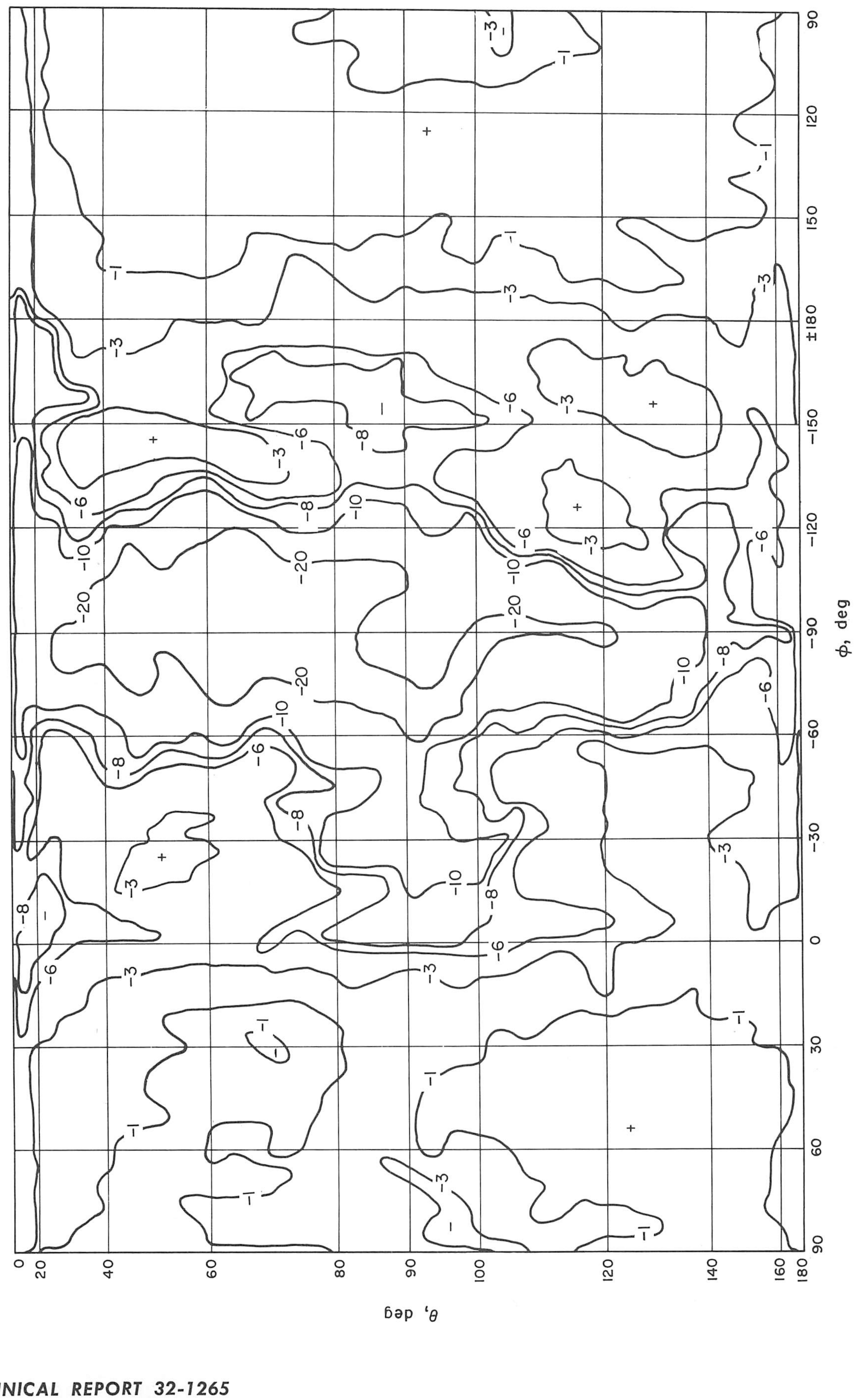


Fig. VI-39. Omniantenna A uplink gain contours: 2113 MHz

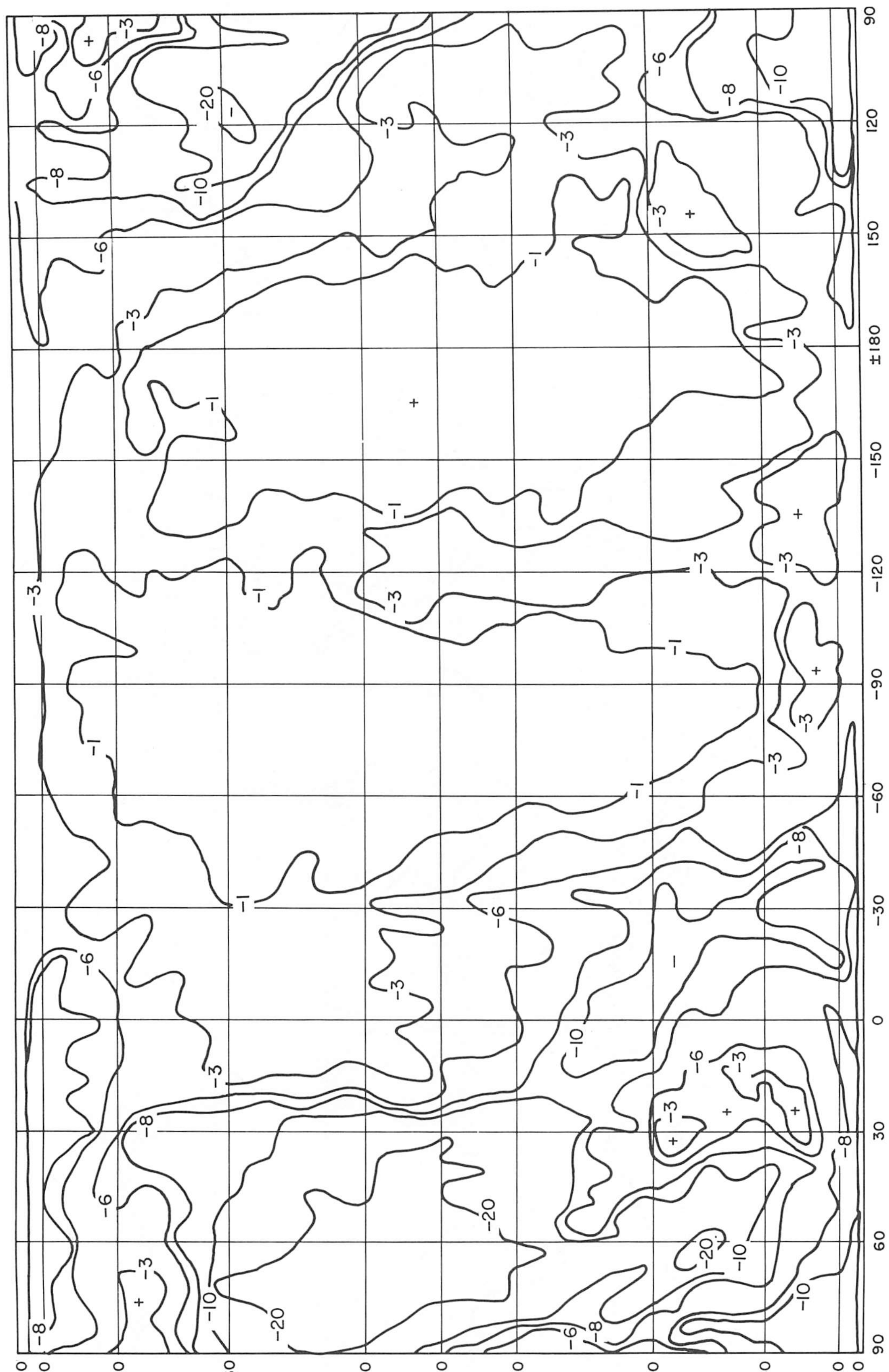


Fig. VI-40 Omniantenna B uplink gain contours: 2113 MHz

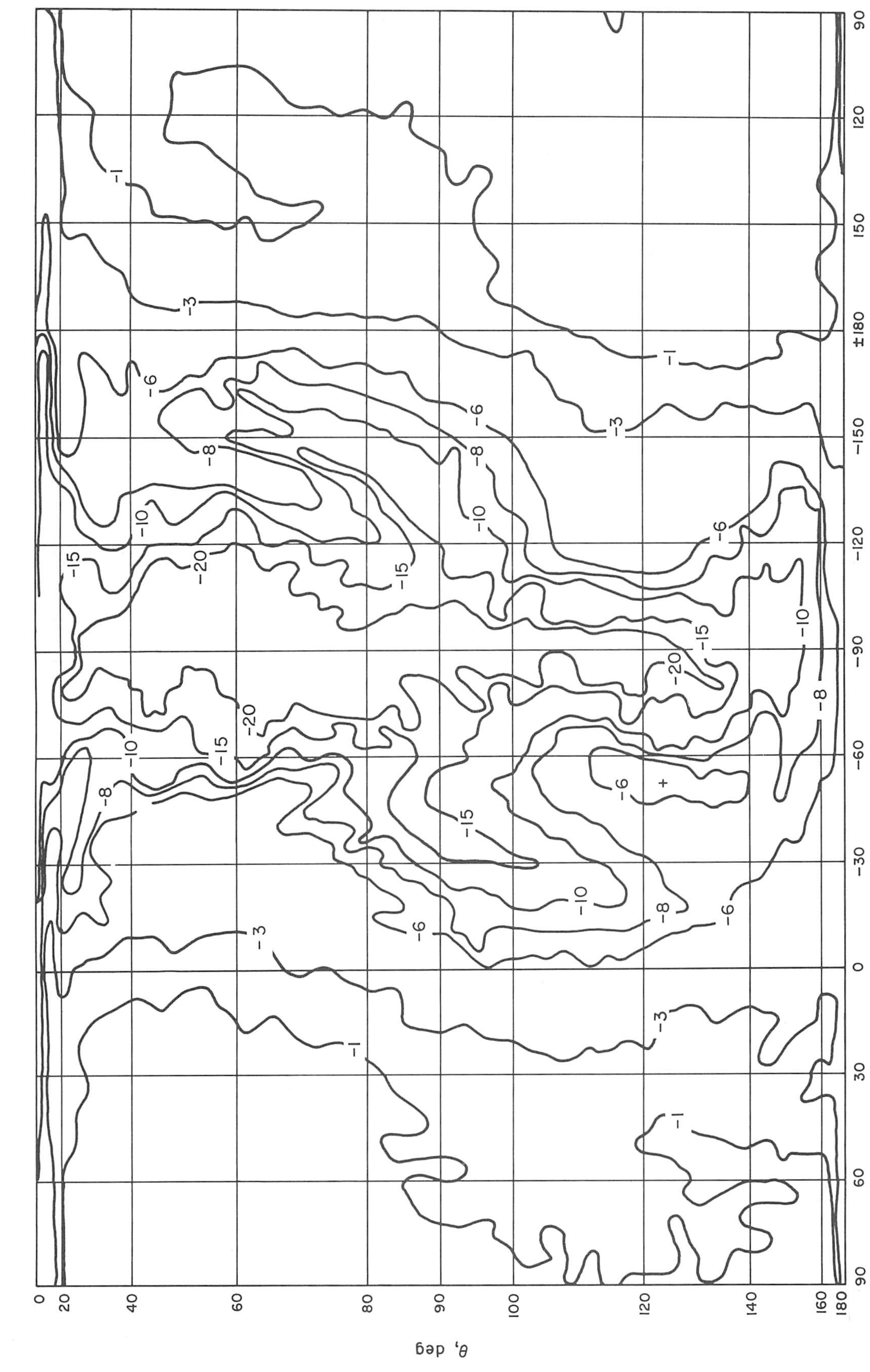


Fig. VI-41. Omniantenna A downlink gain contours: 2295 MHz

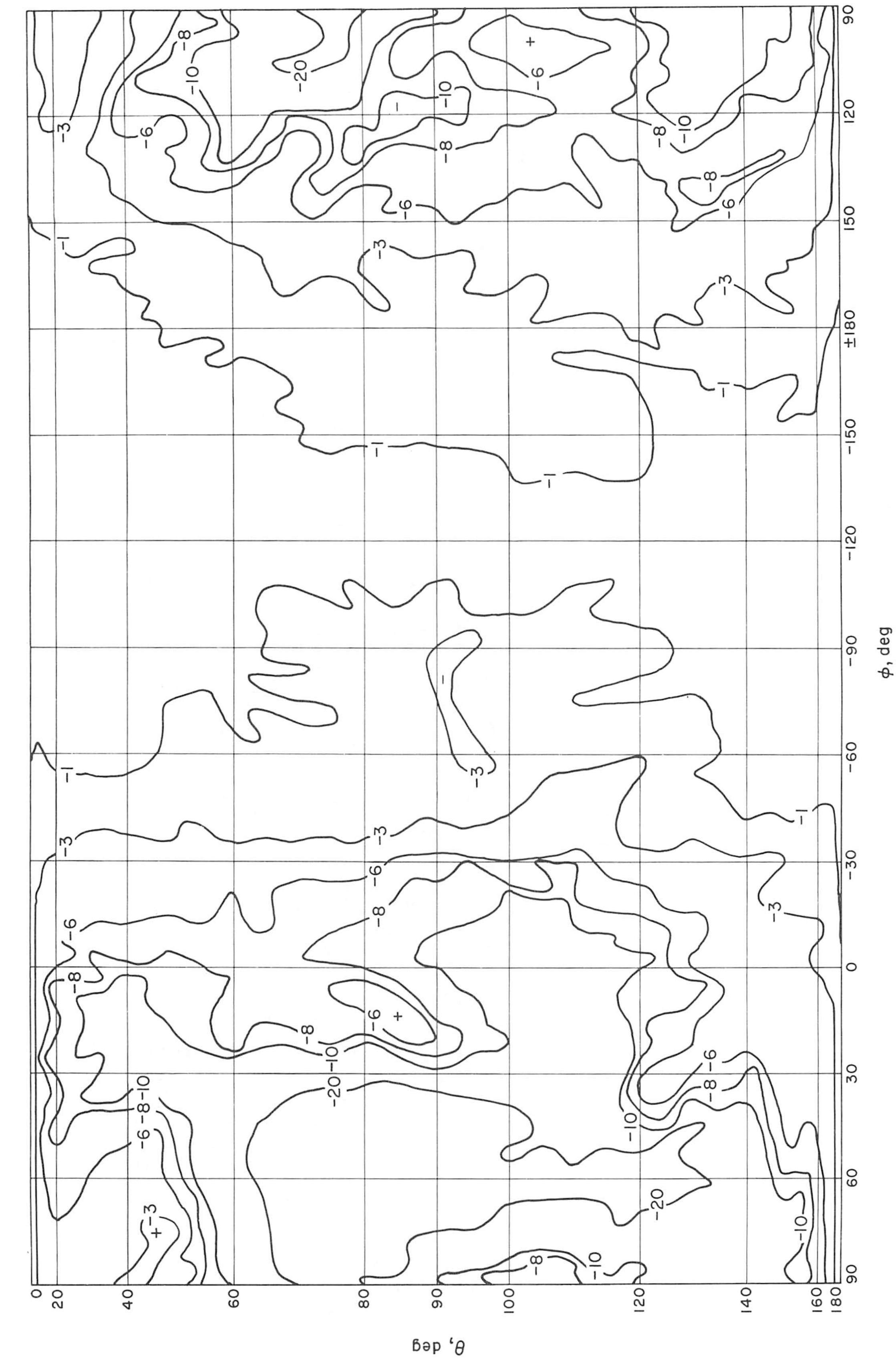


Fig. VI-42. Omniantenna B downlink gain contours: 2295 MHz

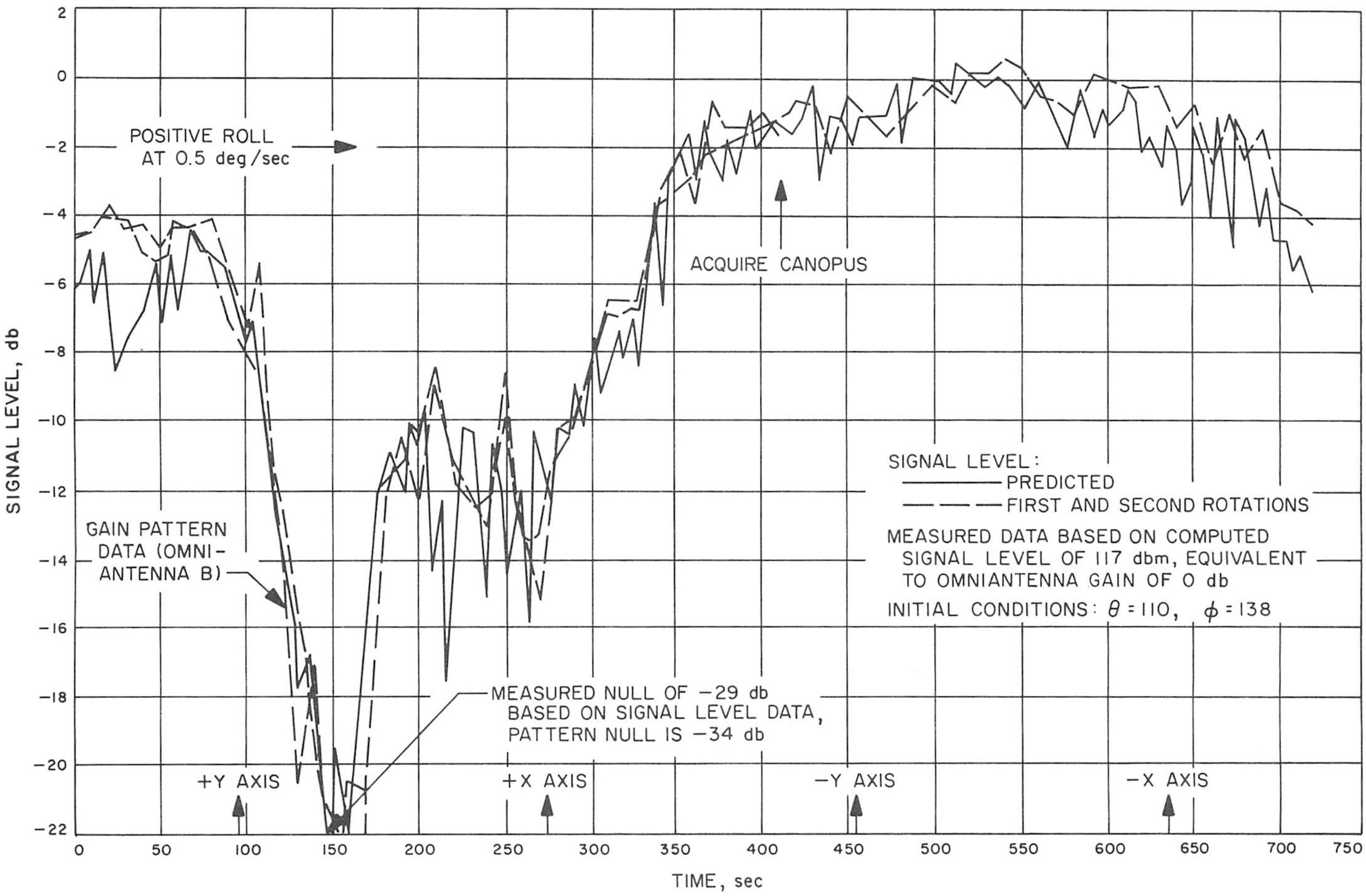


Fig. VI-43. ***Surveyor III* signal level vs predictions during star acquisition for omniantenna B downlink**

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VII. Mission Operations

This section covers the *Surveyor* Mission Operations System (MOS) as it evolved. The MOS configuration was relatively stable during the last several years; it consisted of mission-independent equipment, mission-dependent equipment, procedures, and personnel. The system was used to monitor and control the spacecraft from injection, through lunar transfer orbit, during terminal descent, and through completion of the mission objectives on the lunar surface.

The mission-independent equipment of the MOS included the Tracking and Data System (T&DS) for *Surveyor*, consisting of the Air Force Eastern Test Range (AFETR), Goddard Space Flight Center (GSFC), and the Deep Space Network (DSN). The design of these facilities was general in nature in order to support more than one NASA program.

The mission-dependent equipment of the MOS was unique to *Surveyor*. The command and data handling consoles (CDC) at the *Surveyor* Deep Space Stations, the spacecraft television ground data handling system (TV-GDHS), the Space Flight Operations Director (SFOD), his assistants, and the Space Flight Operation Plan are typical examples of the mission-dependent equipment, personnel, and procedures.

A. Mission Operations System

The MOS encompassed closely knit mission-dependent and -independent equipment that provided mutual support during preparation and accomplishment of a mission.

The overall design of the *Surveyor* Project was unique in several ways. Some concepts had significant effects on MOS design and operation. The concept of controlling a spacecraft by commands from ground stations had many ramifications. This concept argued for a centralized control system, high data rates, real-time processing, display of these data, and highly trained operations personnel. Section IV gives a resume of the MOS evolution.

1. Mission-Dependent Operations

The *Surveyor* space flight operations were complex primarily because of the flexibility built into the spacecraft. This complexity argued for an MOS that was extensive in scope and highly organized. The basic function of the MOS was to control the spacecraft through its defined mission. This involved the following:

(1) Coordination and direction of the Deep Space Instrumentation Facility (DSIF), Space Flight Operations Facility (SFOF), and MOS personnel during

the mission; evaluation of spacecraft progress; and issuance of the requisite commands to the spacecraft.

(2) Evaluation of the spacecraft performance, as well as MOS performance, during flight in real-time to identify and determine the nature of any nonstandard performance that would require a deviation from the nominal mission plan. (Real-time is used to express the idea of timeliness, that is, the need for factual information about an event being made available as close in time to the actual occurrence of the event as possible.)

(3) Determination of corrective actions in real-time that would maximize the mission capabilities of the system under nonstandard performance conditions.

(4) Coordination and direction of corrective actions in real-time.

(5) Recording and dissemination of data for postmission analysis.

a. Organization. The MOS organization used many of the personnel responsible for the MOS design to perform the mission operations. The mission was controlled by *Surveyor* personnel. Other members of the team were DSN and AFETR personnel performing a service for the *Surveyor* Project. The MOS organization for mission accomplishment is shown in Fig. VII-1.

The *Surveyor* Project manager, in the capacity of mission director, was in charge of all mission operations. The assistant mission director aided him in a staff capacity and acted in his behalf on specific request.

The mission operations were under the immediate control of the SFOD. The SFOD made certain that the MOS was ready before the mission began. During space operations, the SFOD maintained control of all operations, participated in making major decisions, and presented alternatives to the mission director. The assistant SFOD aided the SFOD in the conduct and control of the mission and afforded the SFOD backup and relief.

In the event that communication was lost with an overseas tracking station, the mission control authority was delegated to the *Surveyor* Operations Chief (SOC) at the respective overseas station. The tracking station SOC was supplied by the *Surveyor* Project and had intimate knowledge of the *Surveyor* spacecraft. When communications were restored with the SFOF, mission control returned to the SFOD.

During space flight and lunar surface operations, all commands were issued by the SFOD or his specifically delegated authority. Three groups of specialists provided technical support to the SFOD in respect to the flight path, spacecraft performance, and related technologies.

The flight-path analysis and command (FPAC) group handled those space flight functions that related to the location of the spacecraft. The FPAC director maintained control of the group's activities, making recommendations for maneuvers to the SFOD in accordance with the flight plan. The FPAC director was supported by the following five subgroups of specialists within the FPAC group:

(1) The trajectory group determined typical conditions of spacecraft injection and generated lunar encounter conditions based on injection conditions as reported by AFETR and computed from tracking data by the orbit determination group. The actual trajectory determinations were made by computer.

(2) The tracking data analysis (TDA) group made quantitative and descriptive evaluations of the tracking data received from the Deep Space Stations. The incoming tracking data were monitored by the TDA group 24 hr/day. The data processing system (DPS) and computer programs generated for this purpose were used to perform these functions. The TDA group acted as direct liaison between the data users (the orbit determination group) and the DSIF. Predicts were provided to the DSIF.

(3) The orbit determination group determined the actual orbit of the spacecraft by processing tracking data that were received from the tracking stations by the TDA group. Also, statistics on various parameters were generated to evaluate maneuver situations. Tracking predictions were generated for the Deep Space Stations. The orbit of the spacecraft was recomputed by the orbit determination group after maneuvers to determine the success of these maneuvers.

(4) The maneuver analysis group was responsible for developing midcourse and terminal maneuvers in real-time during flight, for both standard and nonstandard missions. When a decision was made regarding the type of maneuver to be performed, the maneuver analysis group generated the proper spacecraft commands to effect these maneuvers. These commands were relayed to the spacecraft performance analysis and command (SPAC) group

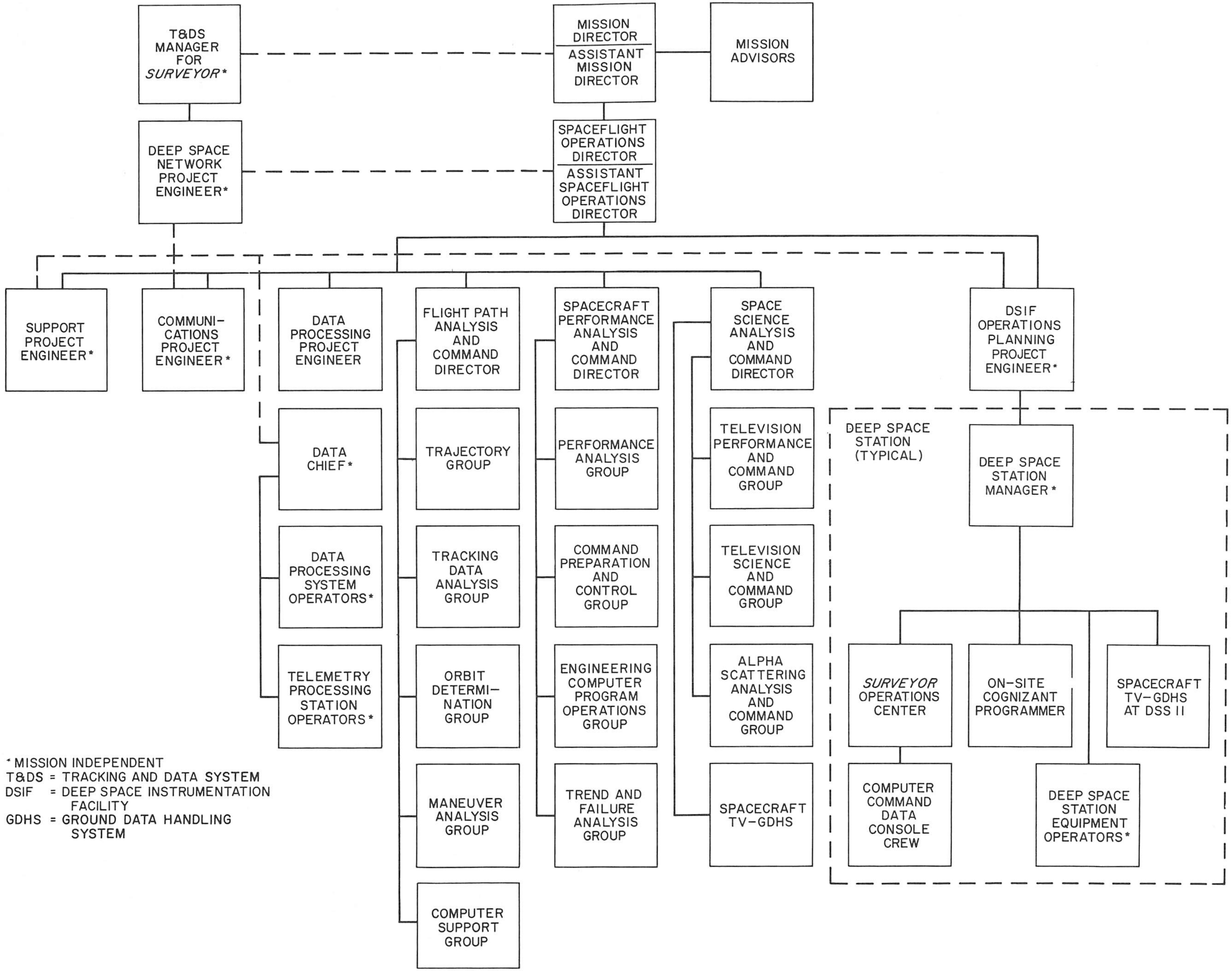


Fig. VII-1. Organization of the MOS for the *Surveyor* missions

to be included with other spacecraft commands. Once the command message had been generated, the maneuver analysis group verified that the calculated commands were correct.

(5) The computer support group acted in a service capacity to the other FPAC subgroups. All computer programs used in space operations were checked out by this group before mission operations began to ensure that optimum use was made of the DPS facilities.

The SPAC group, operating under the SPAC director, was responsible for spacecraft operation. This group was divided into four subgroups:

(1) The performance analysis group monitored the telemetered engineering data from the spacecraft, determined spacecraft status, and maintained spacecraft status displays throughout the mission. The performance analysis group also determined the results of commands sent to the spacecraft. If telemetry data indicated a failure aboard the spacecraft, the performance analysis group made an analysis of the cause and recommended appropriate nonstandard procedures.

(2) The command preparation and control group was responsible primarily for preparing command sequences to be sent to the spacecraft. In so doing they provided inputs for computer programs used in generating the sequences, verified that the commands for the spacecraft had been correctly received at the Deep Space Stations, and then determined that the commands had been correctly transmitted to the spacecraft. If nonstandard operations became necessary, the group also generated the required command sequences. The command performance and control group controlled the actual transmission of commands at the Deep Space Stations by the authority of the SOC.

(3) The engineering computer program operations (ECPO) group included the operators for the DPS input/output console, related card punch, card reader, page printers, and plotters located in the spacecraft performance analysis area. The ECPO group also handled the computing functions for the rest of the SPAC group, and maintained an up-to-date list of parameters for each program.

(4) The trend and failure analysis group consisted of spacecraft design and analysis specialists. They provided in-depth, near-real-time spacecraft performance analysis (in contrast to the performance analysis group's real-time analysis). The group also managed the interface for the spacecraft checkout computer facility (SCCF) at HAC. The SCCF (1219) was used mainly for premission spacecraft ground testing. However, during the mission, two data lines were provided to the SCCF 1219 via the telemetry processing station (TPS). These lines accommodated telemetry rates up to 4400 bits/sec. Eight incoming lines were also provided, terminating at seven teleprinters and at a line printer in the SPAC area. The trend and failure analysis group used the system to process and display engineering data transmitted from the spacecraft. The group also included draftsmen for wall chart plotting and to maintain wall displays of spacecraft condition and performance.

The space science analysis and command (SSAC) group performed those space flight functions related to the operation of the TV camera and science instruments. The SSAC group was divided into the following operating subgroups:

(1) The television performance analysis and command (TPAC) group analyzed the performance of the TV equipment and was responsible for generating standard and nonstandard command sequences for the survey TV camera.

(2) The television science analysis and command (TSAC) group analyzed and interpreted the TV pictures for the purpose of ensuring that mission objectives were being met. The TSAC group was under the direction of the project scientist and performed scientific analyses and evaluations of the TV pictures.

(3) The soil mechanics analysis and command group prepared and recommended the commands to be sent to the soil mechanics/surface sampler instrument (SM/SS), and was also responsible for operating the SM/SS and analyzing its performance.

(4) The alpha scattering analysis and command group prepared and recommended the commands for the alpha scattering instrument. This group also conducted alpha scattering instrument operations during transit and lunar phases and for analyzing its performance.

The portion of the spacecraft TV-GDHS assigned to the SFOF supported the SSAC group with processed electrical video signals and finished photographic prints. The spacecraft TV-GDHS operated as a service organization within the MOS structure. Documentation, system checkout, and quality control within the system were the responsibility of the spacecraft TV-GDHS operations manager. During operations support, the spacecraft TV-GDHS operations manager reported to the SSAC director.

The use of the DPS by *Surveyor* for computer programming was under the direction of the assistant SFOD. The assistant SFOD directed the DPS activities relative to the MOS. The work was directed through the data chief in charge of DPS personnel and equipment. Supporting personnel included the SFOF input/output console operators, as well as equipment operators in the DPS and TPS areas.

The communications project engineer directed the operational communications personnel and equipment within the SFOF, as well as the DSN ground communications system lines to the Deep Space Stations throughout the world.

The support project engineer was responsible for the following:

(1) Ensuring the availability of required SFOF support functions, including air conditioning and electric power.

(2) Monitoring the display of *Surveyor* information on the mission status board and throughout the facility.

(3) Directing the handling, distribution, and storage of data derived from the mission.

(4) Ensuring that only the personnel required for mission operations were permitted to enter the operational areas.

The DSIF operations planning project engineer was responsible for directing the Deep Space Station activities from his post in the SFOF, Pasadena, California. At each station the operations were directed by a station manager. The station manager was in charge of all operational aspects of his assigned station during a mission.

b. Plans and methods. The plans and methods element of the MOS was the means of combining all the other elements (equipment, personnel, and computer programs) into a system that would perform in a consistent and predictable manner. This element was, in essence, the overall plan of the MOS, which included design criteria and mission requirements, development of the mission-dependent equipment, use of the mission-independent equipment, and the procedures needed to operate the system.

The plans and methods were developed in parallel by groups within the MOS and T&DS who were assigned responsibility for specific areas of space flight operation. These groups designed plans and methods for operations based on technical requirements imposed by the MOS plan. These instructions were then submitted to an appropriate authority for approval. When approved, the instructions were published, becoming the official textbooks for the mission. The MOS development is described in several detailed planning documents (see Bibliography).

2. Computer Programs

A comprehensive system of computer programs (and computers) was developed for support of *Surveyor* mission operations. Computer programs selected and combined the extensive data processing capabilities of electronic computers. By electronic data processing, vast quantities of mission-produced data were assembled, identified, categorized, processed, and displayed in various areas of the SFOF where the data were used. The most significant service to the MOS was providing knowledge in real-time relative to the spacecraft status throughout the mission. This service was particularly important to engineers and scientists of the technical support groups. These groups selected, organized, compared, and processed current-status data through use of the computer programs to form time-critical recommendations for the SFOD.

a. Computer system configurations and functions. The Space Flight Operations (SFO) computer programs were real-time and nonreal-time programs operating in various types of interacting computers. These computers, though geographically separated, interrogated each other via hard-line connections or microwave. The function of the SFO programs was to link these computer interfaces and respond to feedback controls generated as a result of data-collection–data-analysis performed by the man-machine interaction.

The SFO computer system for the *Surveyor III–VII* missions consisted of computers listed in Table VII-1.

In addition to the computers given in Table VII-1, a PDP-1 computer was used for telemetry data simulation, and an ASI 6050 computer was used for alpha scattering simulation. Both computers were located in the simulation data control center (SDCC) within the SFOF.

Figure VII-2 shows the total interaction of the SFO computers, the types of data generated, and the data transfer that occurred in the system. Only one computer

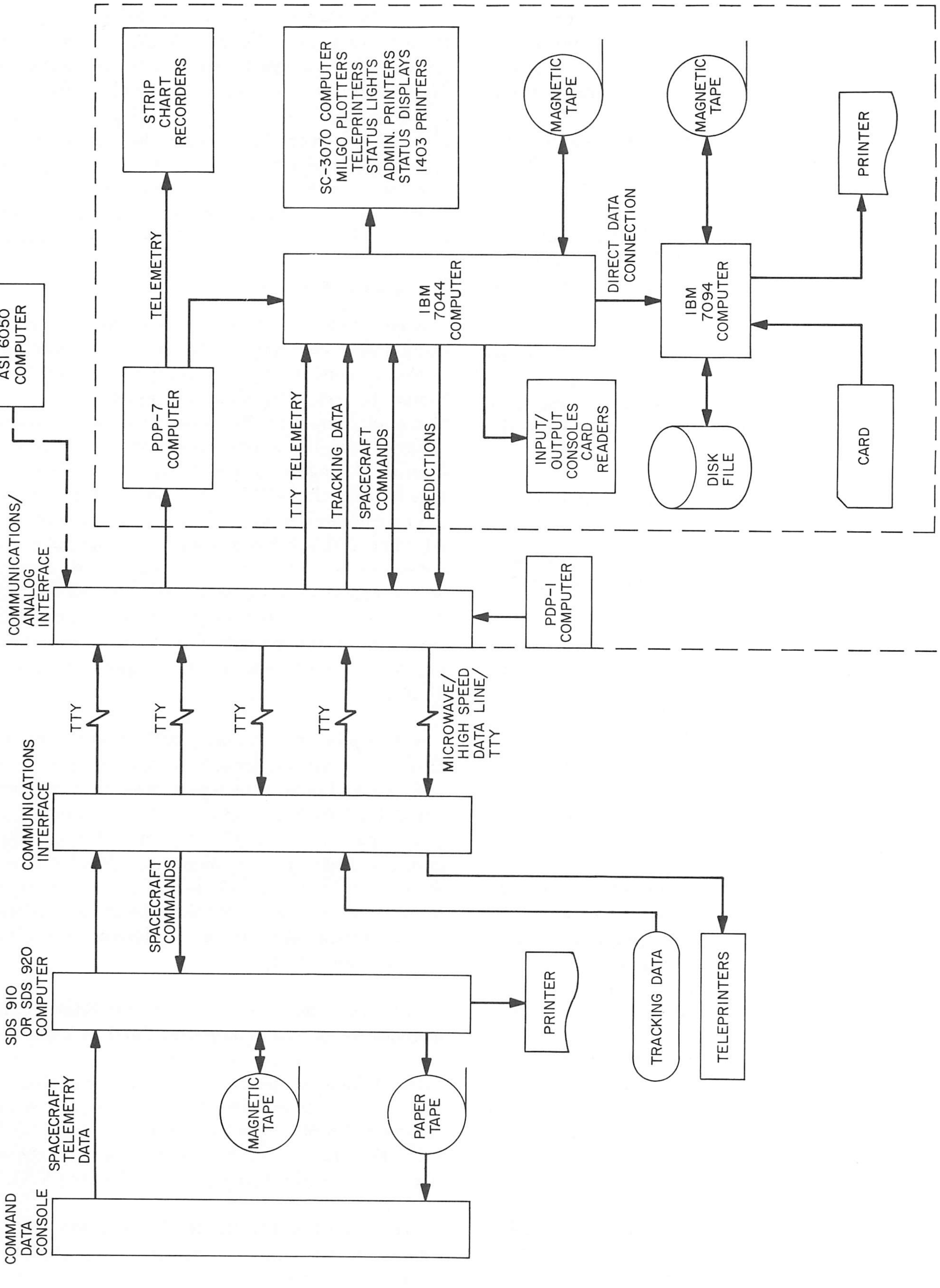


Fig. VII-2. Overall interaction of SFO computers

Table VII-1. Space flight operations computer system

Computer	Number at each location	Location
SDS 920	2	DSS 11, Goldstone, Calif. DSS 42, Tidbinbilla, Aust. DSS 51, Johannesburg, Republic of S.A. DSS 61, Robledo, Spain DSS 71, Cape Kennedy, Fla. DSS 72, Ascension island
SDS 910	1	DSS 51, Johannesburg, S.A
PDP-7	2	SFOF, Pasadena, Calif.
IBM 7044	2	SFOF, Pasadena, Calif.
IBM 7094	2	SFOF, Pasadena, Calif.

combination is shown in Fig. VII-2; a second computer combination could operate in parallel with the first. Figure VII-3 shows the SFOF peripheral equipment for generating off-line tabular listings and plots.

1. SDS 920 computer. The SDS 920 computer within the on-site data processing system served as the interface between the command data console (CDC) and the transmission lines to JPL. In this capacity, the SDS 920 computer was used to perform the following functions:

(1) Receive spacecraft telemetry data via the CDC.

(2) Edit, encode, and provide output telemetry data to the interface of the transmission lines, on high-speed data lines (HSDL), and TTY circuits.

(3) Receive spacecraft command messages from JPL via TTY, generate a command tape, and retransmit command data back to JPL.

(4) Receive spacecraft command data from the CDC, encode it, and transmit the data to JPL via TTY.

(5) Simultaneously search and verify the command tape for any specified burst number or major or minor sequence number.

(6) Perform average alarm functions on the spacecraft telemetry data for display on site, and transmit to the SFOF via TTY.

A second SDS 920 computer was required for *Surveyors V, VI,* and *VII* to perform the following:

(1) Input an alpha-particle-subcarrier oscillator (SCO) data stream at 2200-bits/sec.

(2) Input a clock pulse, synchronized to the 2200-bits/sec data stream.

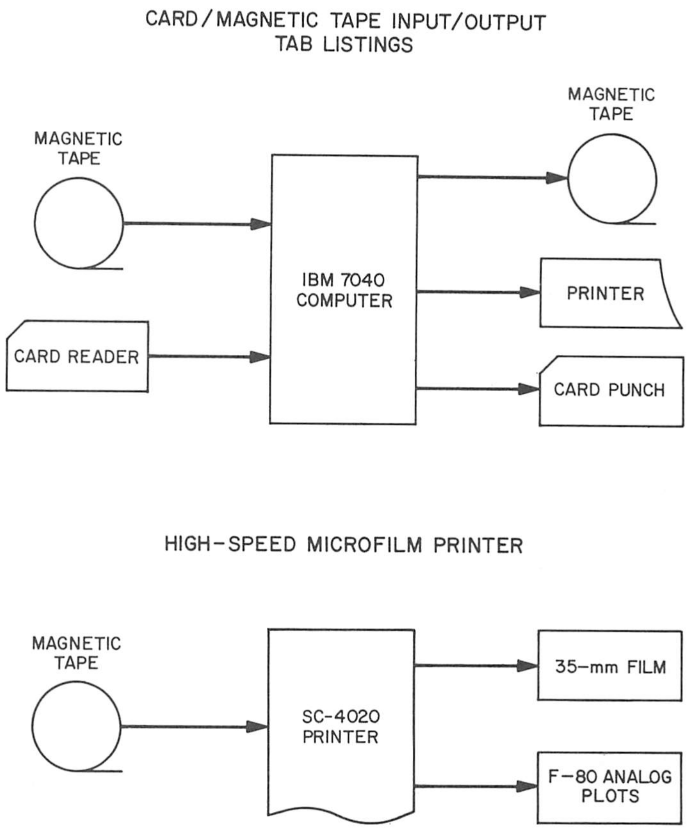


Fig. VII-3. The SFOF peripheral computer equipment input/output

(3) Input a proton-particle-SCO data stream at 550-bits/sec.

(4) Input a clock pulse, synchronized to the 550-bits/sec data stream.

(5) Accumulate alpha and proton spectra for parity-correct and parity-incorrect data.

(6) Submit the accumulated spectra, preceded by the required NASA communications network headers on the TTY.

2. PDP-7 computer. Spacecraft telemetry data, from the on-site data processor were transmitted directly into the PDP-7 computer in the SFOF. It was the function of the PDP-7 computer to:

(1) Receive, format, and assign parameter identifications to all pulse-code-modulated (PCM) telemetry data words and pass these data to the IBM 7044 computer via the 7288 subchannel. Various data modes were selectable.

(2) Recognize and respond to the manual input of data identifications.

(3) Accept, identify, and provide the gyro-speed word outputs.

(4) Recognize the command-enable and command-reject interrupts.

(5) Sample the ground instrumentation/test words, when available, at specified intervals.

(6) Accept and prefix the command words.

(7) Recognize the frame-sync-pattern detection interrupts associated with PCM data and test the validity of their occurrences.

(8) Sample the time words at specified times.

(9) Provide an output of up to 16 PCM parameters for analog displays for each PCM data mode received.

(10) Provide an output of up to eight selectable PCM parameters to the TPS bulk printer (for TPS diagnostic use only).

(11) Log on magnetic tape, all spacecraft data entering the PDP 7 computer.

(12) Recognize and provide the output of the 13 words of television identification data, (TVID) to the 7288 subchannel.

(13) Interrogate the 7288 subchannel and respond to subchannel dropout.

3. *PDP-1 computer.* Although the PDP-1 computer was not considered a part of the SFO complex, it served an important function in simulating the spacecraft flight missions.

The primary function of the PDP-1 computer was to generate spacecraft telemetry data for simulating missions. A secondary function was to generate an analog tape to be used by the PDP-7 computer or Deep Space Stations to simulate a mission.

Spacecraft telemetry data modes were preassigned for the PDP-1 computer. Accordingly, the computer was used to:

(1) Read a digital magnetic tape containing tables of spacecraft telemetry data.

(2) Time-correlate the data.

(3) Generate signals that were transmitted to the PDP-7 computer or to the Deep Space Stations for mission simulation.

(4) Generate an analog tape containing telemetry data for simulated flight (optional).

(5) Prepare command tapes for CDC control.

(6) Respond to TV camera linkup in the SFOF to generate TV identification telemetry data.

(7) Simulate the DSIF in verification of any commands and confirmation of TV commands.

(8) Generate a confirmation command paper tape from card input.

(9) Respond to SM/SS linkup in the SFOF to generate SM/SS and engineering telemetry data.

4. *IBM 7044 computer.* Tracking data, station reports, and TTY telemetry from the DSIF were transmitted through a 7288 subchannel into the IBM 7044 computer via the communications center. Telemetry data were also transmitted to the IBM 7044 computer from the PDP-7 computer via the 7288 subchannels. The IBM 7044 computer processed incoming data, stored it on disk and magnetic tape, and generated digital and analog displays. Functionally, the IBM 7044 computer was required not only to respond to the above inputs but to communicate with the IBM 7094 computer and the user areas (spacecraft performance analysis, flight path analysis, space science analysis, and data processing).

The functions performed by the IBM 7044 computer are listed below. Several of these functions could occur simultaneously and the IBM 7044 computer was required to respond to all of them according to a priority scheme. The functions are:

(1) Input spacecraft telemetry data received from the PDP-7 computer.

(2) Input TTY tracking and TTY telemetry data received from the communication center.

(3) Input and respond to the orders entered at the user area input/output console.

(4) Read cards from the user area card reader and pass them to the IBM 7094 computer, or process the cards for IBM 7044 computer control as required.

(5) Display data and equipment status to the user areas via the administrative printers.

(6) Interrogate the IBM 7094 computer and pass, or take information and/or requests for displays.

(7) Make plots (Milgo, SC-3070 or teleprinter printouts) and turn spacecraft telemetry status lights on or off.

(8) Execute recovery routines.

(9) Execute diagnostic routines to test the various input/output devices.

5. IBM 7094 computer. User programs such as orbit determination, telecommunications, and others requiring large-volume calculations were performed by the IBM 7094 computer. The system monitor program that was stored in the core of the IBM 7094 computer was interrogated by the IBM 7044 computer to turn on and execute the user programs stored on the disk. The IBM 7044–7094 computer linkup was established via this program. Data transfer between the two computers or disk was done by monitor program subroutines, or by interrogation of the monitor to establish the necessary linking.

6. ASI 6050 computer. The ASI 6050 computer was used to perform the following simulation functions:

(1) Simulate the SDS 920 computer in the processing of alpha–proton particle data streams.

(2) Generate alpha–proton accumulated spectra data for SSAC training.

(3) Accept alpha–proton data streams and perform functions (1) and (2).

b. Computer program descriptions. Computer user programs were those programs which operated in the IBM 7094 computer. They performed large-volume calculations and data processing. Mission support programs were those programs required for mission control. They could be run by the 7094, the SDS 910–920, the PDP-7, PDP-1, or the 7044 computers. Mission-independent programs were run by the 7094 or 7044 computer. The simulation programs were run in the 7094, PDP-1 or ASI 6050 computer.

Brief descriptions of the functions of each computer program are given in the following paragraphs. The cognizant agency, JPL or HAC, is indicated.

1. Flight-path analysis and command programs. The FPAC programs are identified and discussed in the subsequent paragraphs.

Tracking data processor, JPL. This program was used to process tracking data stored on the disk from a raw form to one that was usable by the orbit data generator. This program also eliminated tracking data not in proper format nor within established consistencies.

Orbit data generator, JPL. This program was used to edit the tracking data and prepare a data file for the orbit determination program.

Orbit determination program, JPL. Spacecraft orbit and tracking predictions were computed by this program from analysis of the tracking data or from a given set of spacecraft injection conditions.

Trajectory, JPL. This program was used to compute the spacecraft trajectory and the target body impact parameters.

Trajectory processor, HAC. This program was used to compute positions of the planets, sun, moon, and Canopus with respect to the spacecraft.

Injection condition evaluation program, booster model, HAC. By this program the injection conditions of the spacecraft were determined as a function of launch times.

Midcourse and terminal guidance program, HAC. This program was used to determine the spacecraft midcourse maneuver and compute the terminal maneuver required.

Tracking data prediction program, JPL. This program was used to compute doppler, view periods, occultation times, and transmitter voltage-controlled oscillator (VCO) frequencies for the tracking stations.

2. Spacecraft performance analysis and command (SPAC) programs. The SPAC programs are identified and discussed in the subsequent paragraphs.

Telecommunications, JPL. This program was used to compute signal-to-noise ratio margins in the DSIF receiver as a function of trajectory and spacecraft component parameters for all *Surveyor* data and command modes.

Automatic gain control calibration, JPL. This program was used to compute Deep Space Station calibration curves for station automatic gain control (AGC).

Postlanding attitude determination, HAC. This program was used to compute spacecraft attitude after landing and other attitude-related data.

EDPLOT *JPL.* This program was used to print and plot raw spacecraft telemetry data obtained from a magnetic tape that was recorded by the PDP-7 computer.

3. *Mission support programs.* The mission support programs are identified and discussed in the subsequent paragraphs.

Spacecraft command program, HAC. This program was used to generate spacecraft commands and prepare them for transmittal.

Command, verify and transmit, HAC. This program served to verify that the spacecraft commands, transmitted via the CDC, fulfill the spacecraft requirements. Also, the commands received from the DSIF were compared against those commands transmitted to the DSIF.

Surveyor on-site computer program, JPL. This program was used to process spacecraft telemetry and command data and transmit these data to the SFOF. This program operated on either the SDS 920 or SDS 910 computer.

On-site accumulated spectra, JPL. This program was used to compress the alpha particle (2200 bits/sec) and proton particle (550 bits/sec) data streams, accumulate the spectra, and transmit the data to the SFOF via TTY.

EDIT, *JPL.* This equipment consisted of the mission-independent and -dependent editors. The mission-independent editor separated the incoming data stream on the disk into tracking, telemetry, command, and TTY buffers for processing by other programs. The mission-dependent editor decommutated frames of spacecraft telemetry and stored the data on magnetic tape. Command data was processed and stored on the disk.

IBM 7094 computer monitor program, JPL. This program was used to establish communications between the IBM 7044 and the IBM 7094 computers, provide the linkup between these computers, and control transfer of data between the two computers and the disk. The program also served to respond to or interrogate the IBM 7044 computer for status and input/output, execute a priority scheme, bring user programs from disk to core, keep track of user program running time, and store (and retrieve) on the disk interrupted programs.

4. *Space science analysis and command (SSAC) programs.* The SSAC programs are identified and discussed in the subsequent paragraphs.

Television command generation program, JPL. This program was used to read cards containing English text to control spacecraft television system movement, interpret the text, and transform the text into a form usable by the spacecraft command program to generate TV commands.

Alpha–proton particle data processing, JPL. This program was used to process accumulated alpha–proton particle spectra to provide detector operation information.

5. *Simulation programs.* The simulation programs are identified and discussed in the subsequent paragraphs.

IBM 7094 computer telemetry-data tape generator for telemetry processor, JPL. This equipment was used to read punched cards containing tables or simulated spacecraft data and generate a magnetic tape formatted for input to the PDP-1 computer telemetry processor.

IBM 7094 computer telemetry-plot program, JPL. This program was used to read punched cards containing tables of simulated spacecraft data and to generate a magnetic tape for plotting spacecraft telemetry data for the SC 4020 plotter.

SIM 94, JPL. This equipment was used to compute metric data to simulate a spacecraft trajectory and to generate a magnetic tape. The magnetic tape is read by the PDP-1 computer to punch a paper tape containing the metric data.

PDP-1 computer telemetry processor, JPL. This equipment was used to read a digital magnetic tape containing tables of spacecraft telemetry data.

PDP-1 computer SM/SS processor, JPL. This equipment was used to respond to linkup of the SM/SS in the SFOF to generate SM/SS and engineering telemetry data.

PDP-1 television camera processor, JPL. This equipment was used to respond to camera linkup in SFOF to generate television identification telemetry data.

ASI 6050 computer alpha–proton particle processor, JPL. This equipment was used to generate alpha–proton accumlated spectra data for SSAC training.

c. *Computer program documentation.* A controlled system of documentation describes the complex system of computer programs. A brief discussion of each computer program document is included in the following paragraph.

A list of all user and mission support programs was generated and maintained by the data processing project engineer. This list required SFOD approval. Programs listed were designated as either user or mission support programs. The list contained program name and a brief statement of the intent and function of each program.

The computer programs were designed and developed in accordance with the guidelines of the following documents:

1. *Surveyor project computer program development plan.* This document provided the master guidelines for program development. A separate development plan was issued for each mission. The development plans included the names of programs which required corrections or modifications, the nature of the modifications, anticipated completion date, documentation, delivery dates, computer time, and facilities required to support the plan. The plans also included the conditions to which the modified programs would be subjected.

2. *Function design specification.* This was the prime document describing program functional content and control.

3. *Request for program.* This document served to furnish detailed information to the programmer for the program design.

4. *Computer program specification.* This specification served as a means of providing a technical statement from the programmer indicating the manner in which the request for program was to be translated into a computer program.

5. *Program test specification.* This specification defined a series of program acceptance tests whose successful completion was required before the program achieved operational status.

6. *User operating instructions.* These instructions specified the procedures to be used for initiating and operating the desired program on the computer.

The documentation also included a test record for each program. Magnetic tape numbers and the test results for the programs were contained in the record. All program tapes, whether fully certified or not, were retained in the central tape library. Only the personnel designated by the data processing project engineer were authorized access to the tapes. Duplicate copies of the program and data tapes were made and checked out for the IBM 7044, 7094, and PDP-7 computers. This procedure ensured a backup of magnetic tapes in case of a damaged tape. Copies of control cards were made, stored on magnetic tape, and checked out by operating the programs.

Several changes were implemented in the computers and computer programs before the *Surveyor V* project. This major transition was aided by the previously established control procedures and documentation.

d. Computer programs development, testing, and operations. The computer program operations at the SFOF were performed by personnel of the flight support group (FSG), the ECPO, and space data groups. The FSG operated the FPAC and SSAC alpha scattering computer programs, ECPO operated the SPAC programs, and the space data (SD) group operated the science command preparation programs. Operations of the on-site data processing system and the on-site alpha scattering programs were performed by station personnel.

The personnel of the FSG, ECPO, and SD groups participated in computer program development as necessary to learn the program operations. A data controller, provided by the FSG, was responsible for coordinating and controlling the data processing system activities. Also, a specific individual from the FSG was designated for the project to function as flight support head.

The flight support head was responsible for assuring to the data processing project engineer that all FPAC programs were properly integrated into the SFOF system. During flight training tests and spacecraft flight operations, the flight support head coordinated the computer operations activities in the FPAC.

The ECPO group assisted in the various stages of IBM 7094 computer program development and operation of the SPAC programs and in verifying the telemetry calibration coefficients, and the IBM 7040 computer displays. During training the flight operations, ECPO coordinated those activities required for executing the IBM 7040 display requirements.

Space data personnel performed equivalent activities for SSAC as ECPO performed for SPAC.

The data controller was the focal point of the DPS. The implementation of program development, training exercises, and flight operations were assigned to the individual holding this position.. The data controller was accountable to the data processing project engineer for executing the functions of the DPS during program development and the critical phases of a flight, as shown in Fig. VII-1. During training exercises and flight operations the data controller was accountable to the DSN project engineer for executing the DPS functions in support of the mission profile.

Computer programs were modified when new capabilities were required to accommodate spacecraft design changes, or to interface with other hardware changes. Tests were conducted to verify the adequacy of the changes.

A certification test was conducted after changes were implemented and verified. If the program functioned successfully, it was accepted by the cognizant engineer, technical area director, and the data processing project engineer. If any functions were not successful, liens were put on the program until expedited corrective action was accomplished, and efforts were initiated to perform another certification test. No programs were entered into the SFOF complex without completion of successful certification testing. Validation of the master program tape had to be conducted 1 wk before each launch.

The following changes were made in the computer and program configurations before the *Surveyor V* mission:

(1) Combinations processors, consisting of a series of Univac 490 computers, were put into use to transmit TTY data throughout the NASA communications network at 2400 bits/sec. Four hardware lines were previously used for each DSIF to transmit TTY data throughout the network and into the IBM 7044 computers.

(2) The 7044–7094 computer system was redesigned. This involved an interconnection between the 7044 computer and communications for transmitting and receiving TTY data.

(3) All IBM 7094 computer programs were modified to interface with the redesigned systems.

(4) An alpha scattering program was developed for use in the backup computer (SDS 920) at the Deep Space Stations.

(5) An ASI 6050 computer program was developed to simulate alpha scattering operations at JPL.

(6) An IBM 7094 computer program was developed for processing the alpha scattering accumulated channel counts and to compute histograms and spectra of the chemical elements.

3. Mission-Dependent Equipment

Mission-dependent is defined as being designed, procured, and used specifically for the *Surveyor* project in support of mission operations. The command and data handling console and spacecraft TV-GDHS constitute the two basic categories of mission-dependent equipment. These systems and the performance of the mission-dependent equipment during the *Surveyor* missions are discussed in subsequent paragraphs of this subsection.

a. Command and data handling console.

1. Original design concept. The design concept for the CDC as presented in the *Surveyor* proposal was submitted by HAC in December 1960 and did not change significantly during the life of the program. The CDC was envisioned to consist of a command system, demodulator, data handling system, television monitor with photography, and engineering and scientific data displays. The original plan proposed locating the lunar mission director (LMD) and the lunar operations director (LOD) at the JPL control center in Pasadena, Calif. The terminal, or landing phase of the spacecraft trajectory was planned so that it could be directed from Goldstone. Plans suggested that the LMD and LOD direct operations from that station during the terminal descent phase and for the first several days of lunar operations because from that station the TV display and engineering data return could be viewed directly in real-time. It was further proposed that during the actual lunar operations, the operational activities of each Deep Space Station would be under the supervision of a station operation chief (SOC). Each SOC would be supported by the Deep Space Station manager for facilities operations and administrative support. The SOC would be in charge of the Deep Space Station, the CDC, and its operators at all times. This organization structure was changed early in the program to that shown in Fig. VII-1.

A CDC was to be provided with the system test equipment assembly (STEA) at HAC and AFETR for test and checkout of all spacecraft flight and test models. Each Deep Space Station would be provided with a CDC. The intent of this proposal was to assure, to the maximum extent possible, that the same spacecraft interfaces would exist during flight as were seen during test; therefore, the total spacecraft/CDC response would be known. At Goldstone, the console would be housed in a building; overseas, the console would be located in an air transportable van. It was anticipated that the DSIF would supply recording equipment, such as oscillograph recorders, tape recorders, and a time code generator.

The proposal specified an SOC and three operators: one command operator, one television operator, and one operator with dual responsibility for the engineering and scientific data positions. The original CDC design concept proposed a four-bay console plus one rack for the auxiliary equipment (in addition to sufficient storage and desk space).

The command system was envisioned as an electronic digital encoder–subcarrier modulator with manual input keys. The system was to have three modes of operation: manual, semiautomatic, and emergency. A keyboard device called a digikey was to be used as the manual entry mode for commands. In addition, paper tape punches could be used for rapid command transmissions.

It was planned to make use of a frequency modulator with feedback demodulator to provide sufficiently high signal-to-noise ratios from the spacecraft for TV and scientific data during the lunar mission phase.

The PCM data handling system was proposed to make use of a digital decommutation system. The concept of engineering data display, whereby certain spacecraft telemetered functions would be routed to meters for operator monitoring, was proposed and these displays were a part of the operating console of the CDC.

Real-time television monitoring was planned to permit spacecraft camera focusing and positioning with a minimum expenditure of vehicle power. It was planned to use a 10-in.-diameter, selective-erase, direct-view storage tube. Another picture tube would be mounted with a swingaway 35-mm camera to photograph it. The primary method of recording TV pictures was planned to be by means of magnetic tape; however, the photographic method would be used as a backup to provide more rapid availability of photographs for study at the station.

2. *Final design concept.* The basic concept of the CDC remained the same through the final design. However, there were some differences in the detail mechanization of the subsystem. The final CDC concept did incorporate three major equipment additions to the original design concept. These consisted of the system tester/spacecraft simulator, the on-site data processor (OSDP) concept, and the incorporation of equipment for near-real-time processing of alpha scattering scientific data.

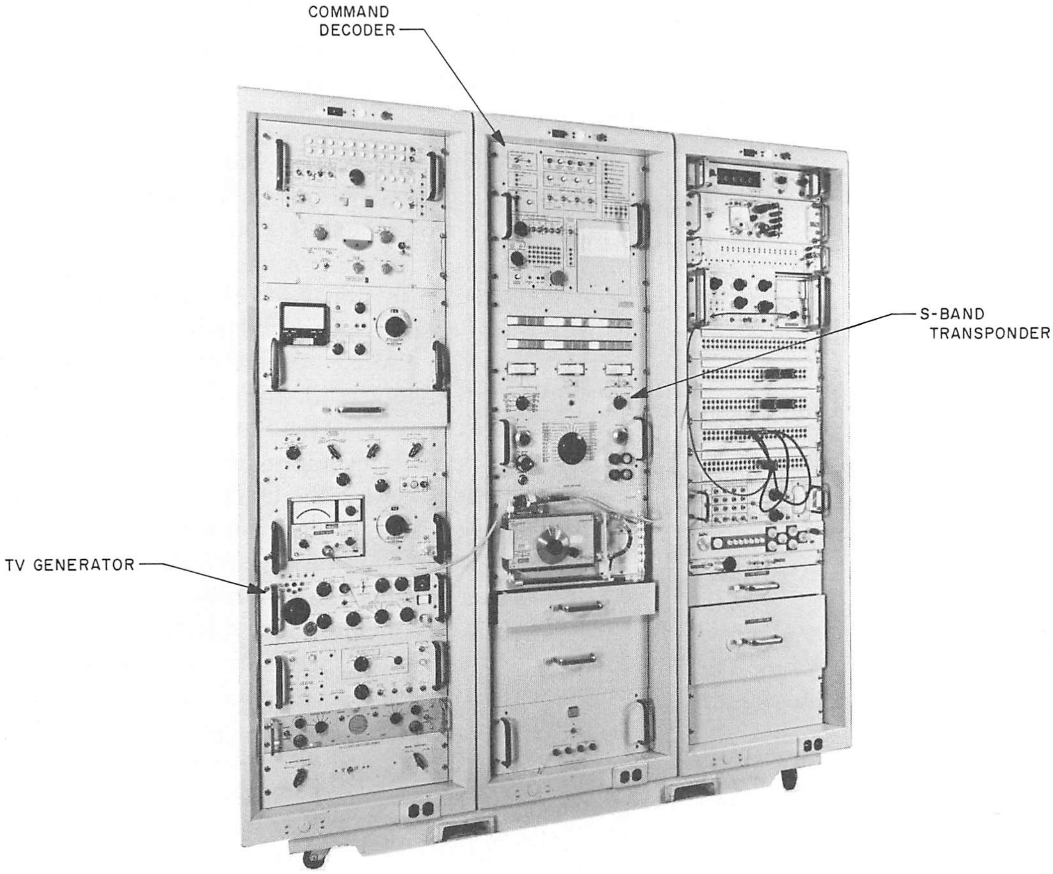
A design was proposed for a spacecraft simulator which would (1) generate signals comparable to those in the spacecraft and (2) accept and encode commands. The simulator was to be installed with each CDC to facilitate compatibility testing and system checkout. After additional discussions and negotiations, a much less elaborate tester/simulator was designed and fabricated that used an S-band transponder with some telemetry, command, and TV equipment.

The OSDP equipment addition resulted from a major change in philosophy in that the concept of central control was from the SFOF in Pasadena, Calif. The original concept was to provide sufficient capability within the CDC so that analyses of spacecraft actions necessary for real-time reaction could only be observed at the CDC. The new central control concept meant that the operation of each CDC would, with this addition, then be under the direct control of the SFOD. In order to provide the SFOF with sufficient information for the SFOD to make immediate decisions, the OSDP was designed to process spacecraft telemetry data, allowing only selected information to be transmitted to the SFOF via TTY and high-speed data lines. Additional functions of this OSDP system were: (1) alarm monitoring of the incoming spacecraft telemetry, (2) receipt of spacecraft command data from the SFOF, followed by preparation and verification of spacecraft command tapes and sequences in conjunction with the SFOF, (3) real-time transmittal to SFOF of commands transmitted to the spacecraft during the mission.

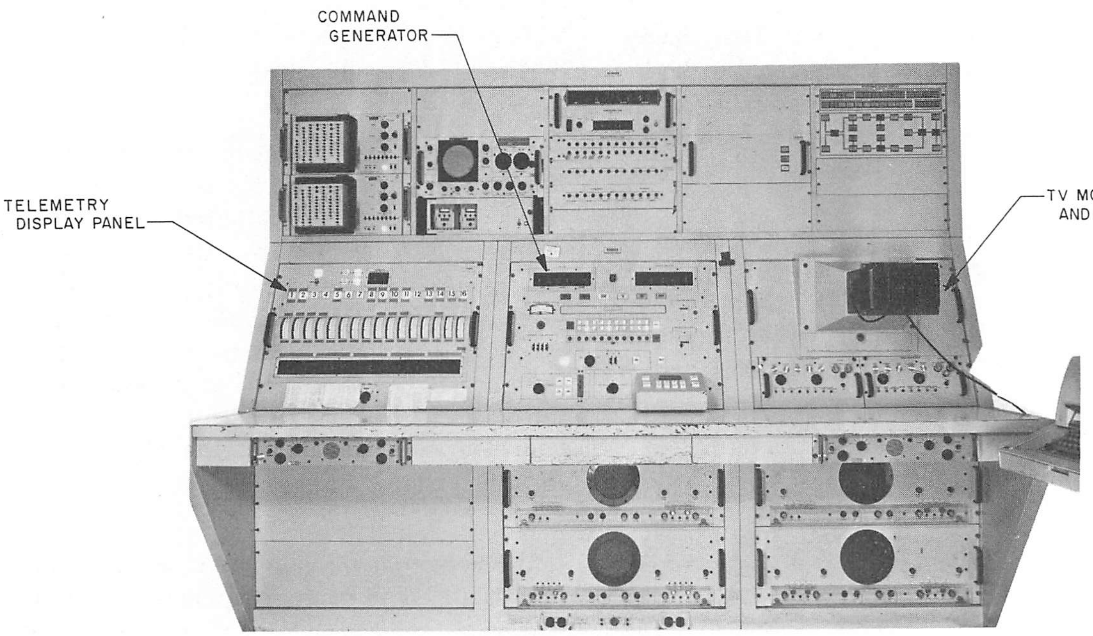
The major modifications for this function were made only to those CDCs used at the DSIF and not those for STEA. The changes consisted of adding telemetry converters, command buffers, paper tape punches, reader and TTY punch couplers, input/output typewriters, patch panels, and test equipment.

When the decision was made to add the alpha scattering instrument to the spacecraft, the proposed concept for accumulation of alpha scattering data was to use tape recording and then retrieve the data during postflight analysis. The capabilities of the on-site backup computer were used for the accumulation of near-real-time alpha scattering data for immediate transmittal to the SFOF. Additional CDC equipment to interface with the computer consisted of two PCM bit synchronizers plus test equipment for checkout of the alpha scattering data processing equipment.

3. *Final configuration.* The final configuration of the CDC included 16 racks of equipment, two input/output typewriters, and an SOC operations console. Figures VII-4 to VII-6 depict a full view of the CDC, the equipment



COMMAND DATA CONSOLE



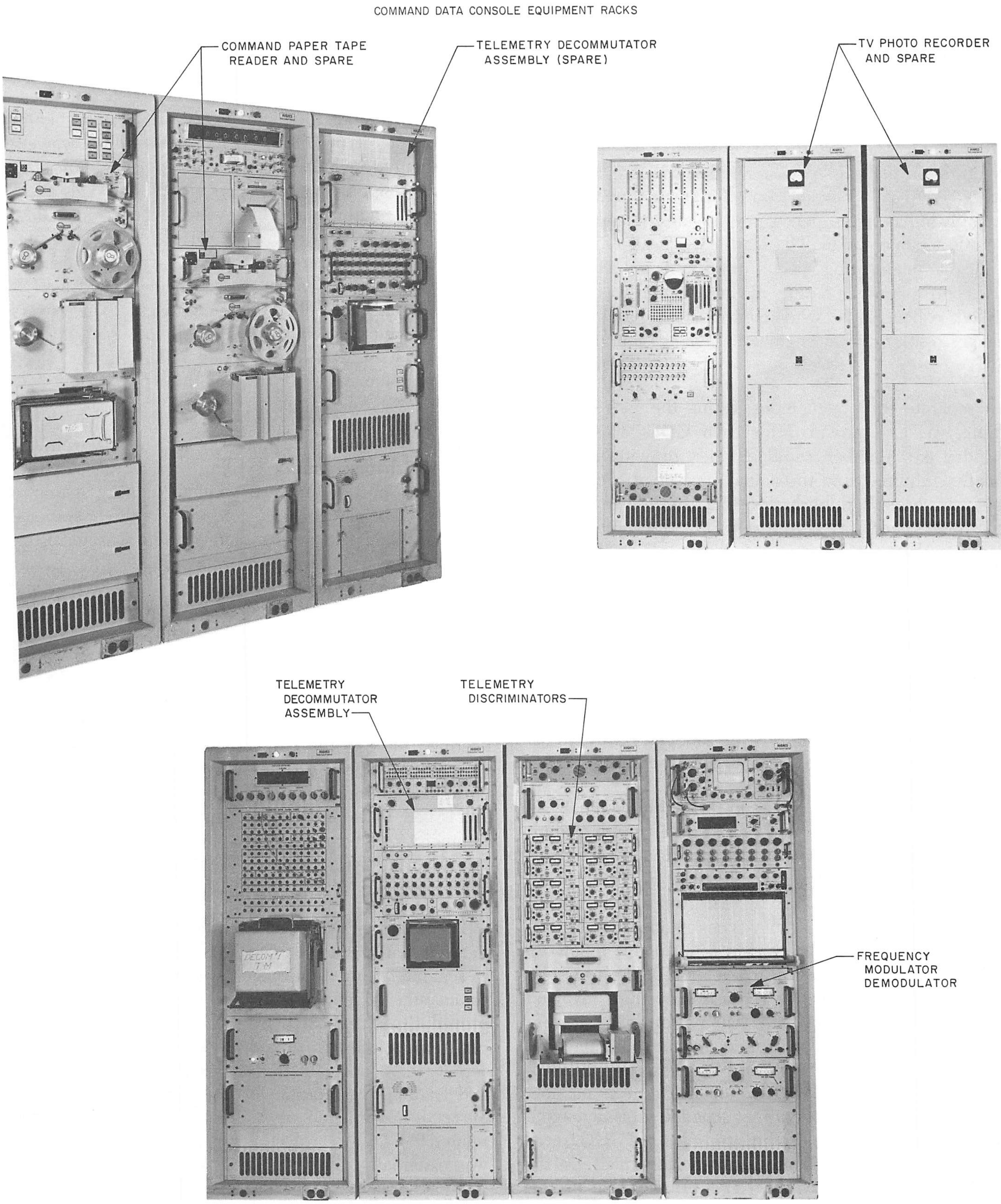


Fig. VII-4. The CDC equipment for DSIF

layout in each rack, and the floor plan for the Goldstone CDC. During the program 12 CDCs were fabricated, 6 for STEA, and 6 for DSIF use. The configuration of the CDCs used for STEA is described in Section IX-B-1.

4. *Configuration, quality, and reliability control.* The master index was the primary specification for CDC configuration control. A master index was prepared for each CDC to show current control item part numbers, in assembly indenture sequence, per the latest released engineering documentation. The master index also listed the applicable engineering specifications and procedures for each control item. Updating and publication was accomplished concurrently with control item part number changes resulting from engineering change authorization (ECA).

A separate equipment index, listing the as-is configuration, was also provided for each CDC. This listing included part number, serial number, drawing revision, quantity required, and the actual content of each CDC. This was normally published to identify the hardware of each mission and served as a record of the actual equipment configuration for each mission.

The retrofit summary index was another document used in maintaining CDC configuration control. This provided the history of all part number changes.

Other documents, used as follow-up mechanisms to maintain configuration and point out discrepancies, included modification confirmation notices, ECA status sheets, and CDC configuration accounting sheets.

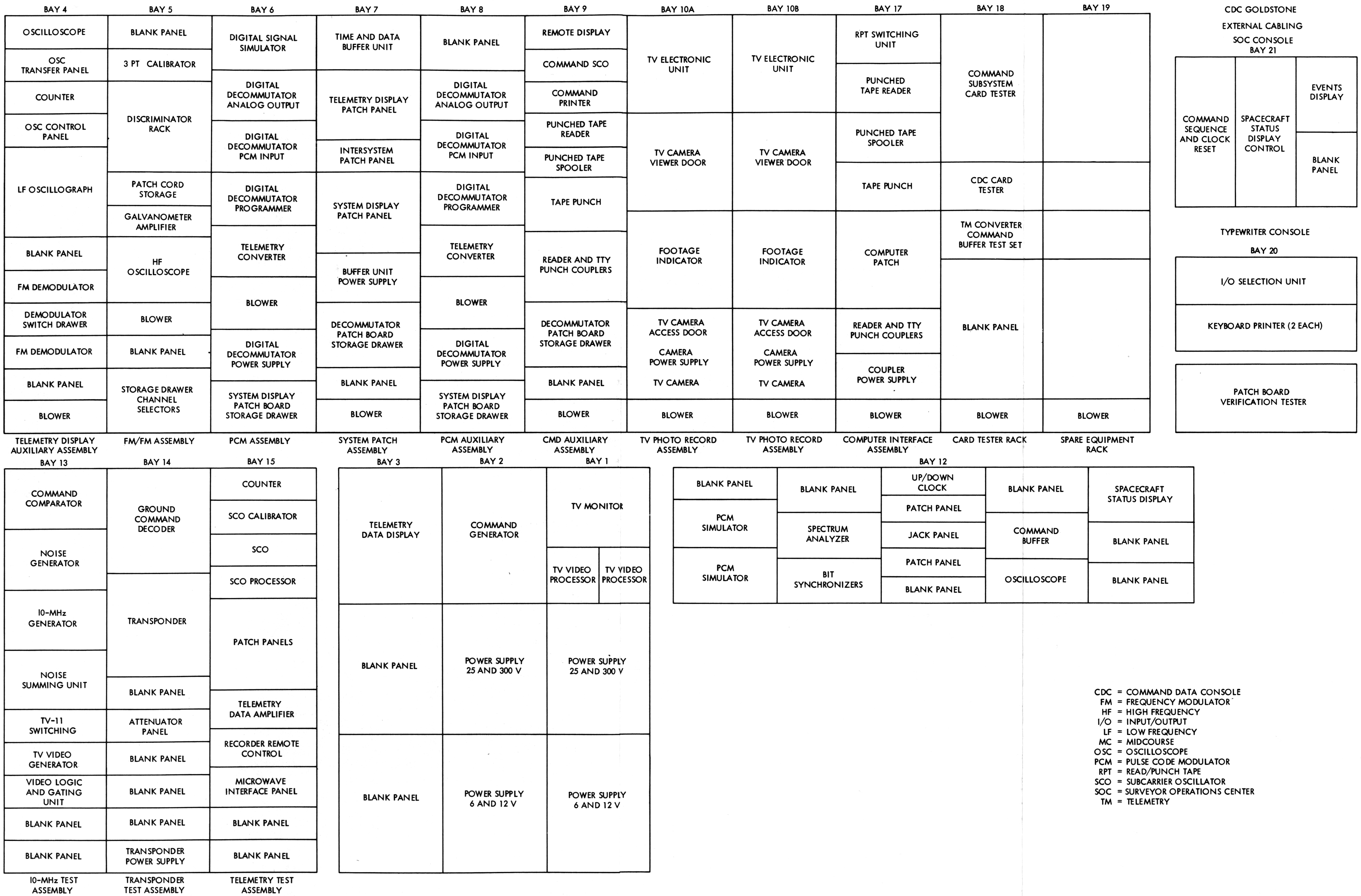
The CDC equipment was controlled under the HAC change control activity as well as the trouble/failure report (T/FR) system. Approved modifications were incorporated on nondelivered CDCs at the factory and on CDCs by modification kits. Several procedures were set up for submitting emergency-type engineering change requirements. These all provided a means for follow-up and subsequent acceptance or rejection of the proposed change. The TWX change request procedure involved transmitting the ECR by TTY. The use of this procedure was limited to emergency situations where a change was necessary to prevent (1) holding of a mission, (2) jeopardizing the system capability to perform a mission, (3) delaying of test or operations schedules, (4) damaging equipment, or (5) slippage of mission-directed CDC freeze dates. Another procedure used was identified as the TWX change notice system. The primary use of this procedure was to correct minor errors or to make hardware conform to released engineering documentation. The TWX change notices were used only when a major modification had been approved for incorporation and minor errors in the change needed rapid correction.

A system of deviation waivers was established to document the disposition for all open DSIF/CDC items before a mission. The types of discrepancies for which waivers were prepared included: (1) test values not within tolerances, (2) equipment modifications approved and installed, but formal modification kits were not yet available, (3) equipment modifications that varied from the released engineering, (4) other hardware vs available released engineering documentation discrepancies, (5) open ECR and T/FR problems believed to be significant. The waivers were required of each DSIF/CDC before each mission and each item was dispositioned either by the Deep Space Station manager or by joint JPL/HAC action.

Three types of patchboards were used in each DSIF/CDC. A control specification was written for each and a formal change control system was established that included inspection and sealing under quality assurance control. These patchboards were used in the operational tests preceding and during each mission. The telemetry panel overlays also were controlled so that the proper telemetry measurement was observed on each display and the calibration matched that of the spacecraft.

A 10-volume set of operations and maintenance manuals was developed for the CDC. The manuals provided operating instructions, specifications, maintenance procedures, and theory of operation for CDC equipment. Since engineering changes affected contents of the manuals, an interim data sheet system was established to provide up-to-date information until change pages were published and issued. Since much of the equipment in the CDC was commercial test equipment, the manufacturer handbooks were reprinted for inclusion in the operations maintenance manuals.

The history of approved ECAs and T/FRs written per quarter year are depicted in Fig. VII-7. The equipment is separated into four major subsystems of telemetry, command, TV and test equipment. The number of approved ECAs, indicated by the shaded bars, reached a peak during the first quarter of 1966 reflecting the high activity of spacecraft testing, mission operations tests, and training preparation for the first launch. After the *Surveyor IV* mission of mid-1967, the change activity related to CDCs decreased and, for the latter part of 1967, was essentially nil. The T/FR system was initiated on CDC equipment in the last quarter of 1964. The high peak indicated during the first quarter of 1965 resulted from the preparatory CDC test activity for shipment of



CDC = COMMAND DATA CONSOLE
FM = FREQUENCY MODULATOR
HF = HIGH FREQUENCY
I/O = INPUT/OUTPUT
LF = LOW FREQUENCY
MC = MIDCOURSE
OSC = OSCILLOSCOPE
PCM = PULSE CODE MODULATOR
RPT = READ/PUNCH TAPE
SCO = SUBCARRIER OSCILLATOR
SOC = SURVEYOR OPERATIONS CENTER
TM = TELEMETRY

Fig. VII-5. Equipment layout in CDC racks

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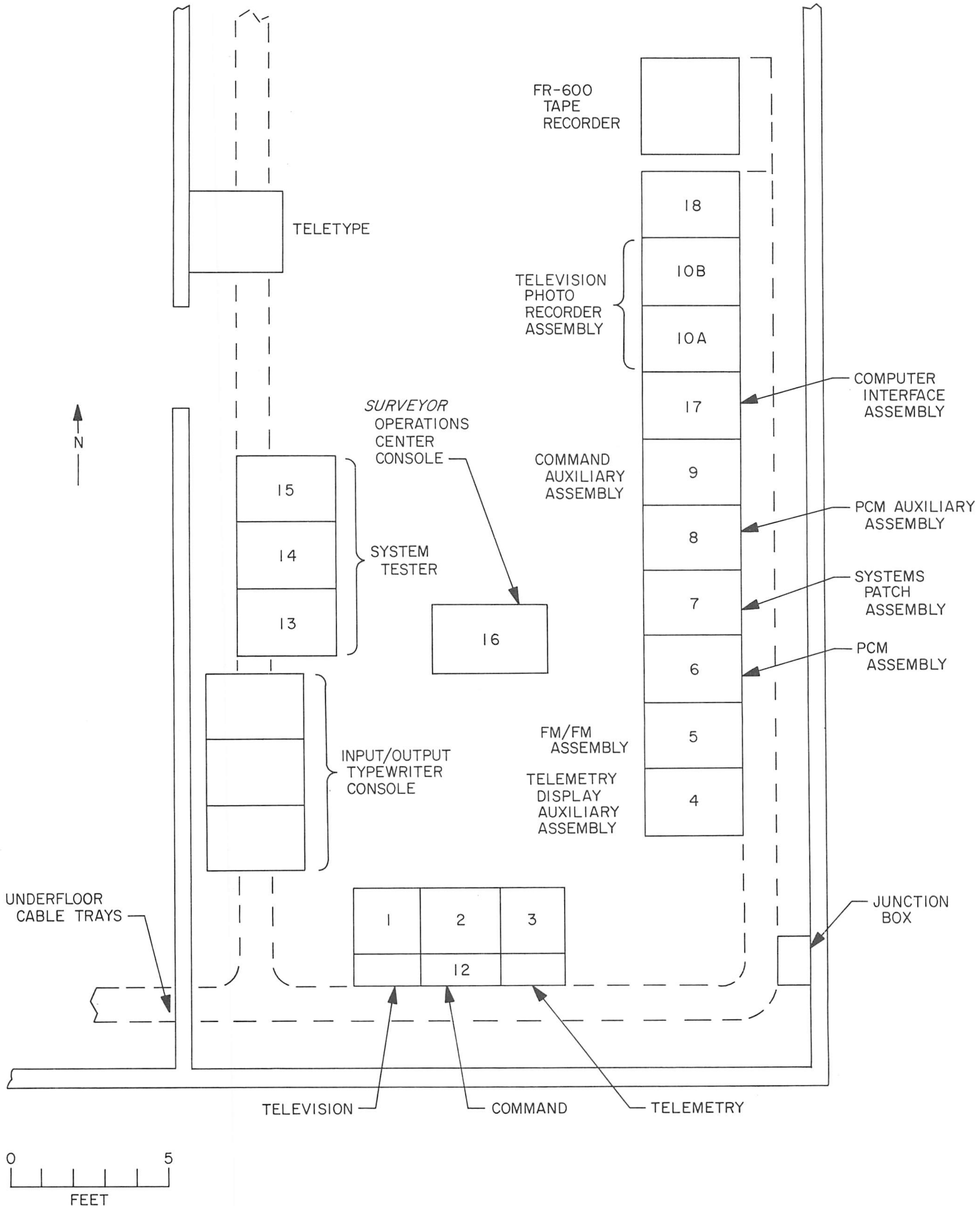


Fig. VII-6. Floor plan of CDC 1 at DSS 11

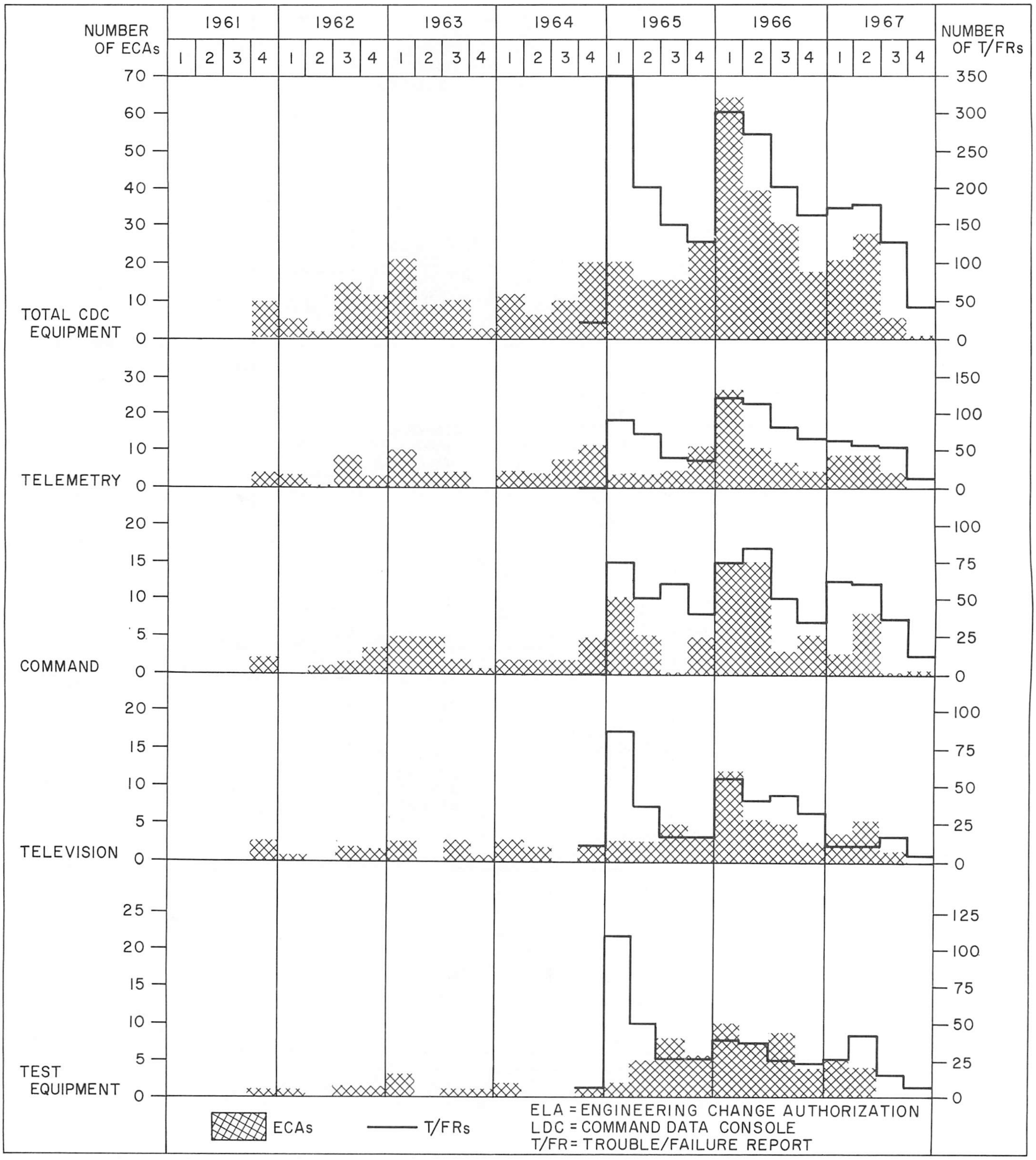


Fig. VII-7. History of ECAs and T/FRs for CDC

DSIF units overseas. The second peak in the first quarter of 1966 resulted from intensive spacecraft and mission operations testing before the *Surveyor I* mission.

5. *Test and operations.*

Objectives, plans, and procedures. A manual of operating instructions was prepared to provide a consistent policy for operating CDCs at Deep Space Stations. These instructions were grouped into three categories: (1) those pertaining to HAC/DSIF/*Surveyor* Project office interface areas, (2) HAC only (in-house or on-site), and (3) DSIF only. The manual included policies on items such as logistics, failure reporting, configuration control, and organizational structure.

Activities associated with the spacecraft and CDC operations at the stations were divided into two separate functions designated as mission control, and maintenance and operations (M&O). Mission control was defined as those activities associated with the control of the spacecraft, analysis of spacecraft data and determination of corrective actions in the case of a nonstandard situation. Maintenance and operations was defined as those activities associated with maintenance, routine testing of the CDC, CDC countdown, and operation of the CDC equipment during a mission.

The general ground rule established by JPL was that CDC installation, checkout, compatibility testing, and mission operations for the first mission would be accomplished by on-site HAC personnel. After the first mission, local station personnel would be trained to take over the M&O function, with HAC personnel retaining the mission control responsibility. A plan for the phaseout of HAC M&O personnel and training of local engineers and technicians to take over this responsibility was negotiated between HAC, the JPL *Surveyor* project office, and the DSN. It was also agreed that final phaseout of HAC M&O personnel would coincide with CDC selloff and the complete transfer of the M&O responsibility to the local DSS manager. This objective was not achieved in that HAC M&O personnel were phased out of DSSs 11, 51, and 61 before CDC selloff, and formal transfer of M&O did not occur at these stations.

An integrated system of test procedures was prepared to document the unit, subsystem, and system level readiness tests for determining functional operation and readiness for major tests and missions. The three principal levels of testing were as follows:

Unit tests. These tests were performed on the relatively independent units and were preparatory to performing the subsystem tests on a group of units. The unit testing of commercial test equipment was usually performed in the calibration laboratory.

Subsystem tests. These were the key CDC tests for obtaining both qualitative and quantitative data for determining proper operation and readiness.

System tests. These tests made use of the system tester and were designed to verify proper operation of the composite CDC/DSIF system at the deep space stations. The first CDC fabricated was designated as the operational prototype unit. This unit was subjected to a number of unit, subsystem, and system tests to verify test procedures, investigate potential problem areas, and check the effectiveness of engineering improvements proposed for incorporation in the production units.

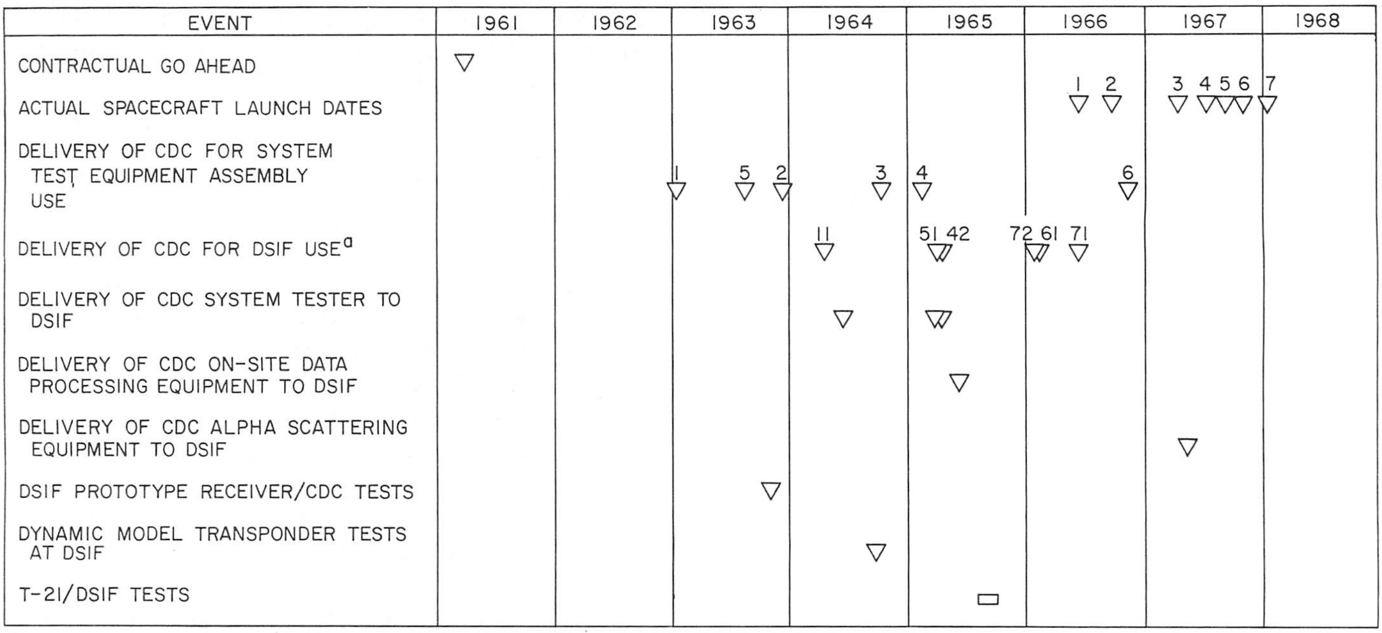
Since all CDCs fabricated for the *Surveyor* Project were assigned to operational use, a unit was not available for engineering development or investigative testing. During the project, a need was evident for a laboratory where deficiencies could be investigated. A policy was established to use DSIF CDCs for development laboratory functions. The development test request bulletins (DTRB) system was established to control and keep testing uniform. In all, 23 DTRBs were run during the *Surveyor* program. Tests defined by DTRBs were run primarily at Goldstone, but this activity was not limited to that site. For further discussion on this subject, see Section XI.

Major milestones. A schedule of major milestones for the project is shown in Fig. VII-8. This schedule shows the CDC deliveries and some of the major tests performed during the development phase.

Early test and operational problems. During initial operational use of the CDCs the following significant problems were discovered:

Erratic telemetry bit error rate. The telemetry bit error rate was erratic, exceeding specification limits. Laboratory tests determined that external transients and radio-frequency interference (RFI) influenced the bit error rate. Minor changes in the decommutator grounding and improved alignment procedures for certain circuits resulted in consistent and satisfactory bit error performance.

Television generator instability. Stability problems were experienced early in the program with the TV generator. Also, the TV generator did not include a crosshatch pattern, for use in calibration linearity of the TV recording



[a]NUMBERS DENOTE DEEP SPACE STATIONS CDC = COMMAND DATA CONSOLE DSIF = DEEP SPACE INSTRUMENTATION FACILITY

Fig. VII-8. Schedule of CDC major milestones

system. A new design was implemented using digital techniques instead of analog to improve the generator's stability, and crosshatch pattern was incorporated.

Tape reader instability. The command system used a model RR-50-R punched paper tape reader manufactured by Remex Electronics. Both electrical and mechanical components, such as clutches, brakes, and head alignment in this unit were difficult to maintain. A survey of commercial tape readers revealed that a new model, RR-4050-B, by Remex, incorporated many desirable design features and was interchangeable with model RR-50-R units. These new units were procured, tested, and installed in the CDCs. A slight, but significant, decrease in the command tape reader problems resulted.

Noisy 35-mm film images. The 35-mm film images developed in the television system of the CDC contained a considerable amount of noise due to power supply ripple and other causes. The noise was never completely eliminated although improvements were made through improved filtering in the high- and low-voltage power supplies of the TV recorder and a revision in the grounding.

Television horizontal sync jitter. The photographic records produced by the CDC exhibited considerable TV horizontal sync jitter. This occurred at strong signal-to-noise ratios due to circuit instability problems, and at threshold signal-to-noise ratios due to the effect of noise on the sync circuits. A redesign of the sync portion of the video processor was made that resulted in significant video improvement.

False triggering of video circuitry. During intermittent carrier on/off operation, in 200-line video, the video circuits triggered on noise during the carrier off time. An oscillator, keyed on during absence of carrier, was used to prevent false triggering. The presence of a carrier immediately turned off this oscillator to allow video signals to be received properly.

Erratic command printer. This problem was not fully solved. The command printer is an electromechanical device requiring considerable maintenance for which procedures were developed to alleviate the problem. In addition, a redundant method of logging transmitted commands reduced the significance of this problem.

b. Spacecraft television ground data handling system.

1. Design objectives and considerations. The spacecraft TV-GDHS evolved out of the consideration of two major requirements. One of these requirements resulted from a desire to extract as much scientific information as possible from *Surveyor* and other project television pictures. The other requirement resulted from a mission operations concept change. The original plan specified decentralized operational control of the television experiment. This was to be implemented by an SOC and a TV experiment team at each receiver site. The later concept called for a centralized TV experiment team. These two requirements resulted in two major, and several contributive, design objectives for the system. The major objectives were as follows: It was desired to build a system that would be capable of translating video signals from a spacecraft to photograph images, with little or no photometrically or photogrametrically introduced distortion. This system would serve to accurately record the video signals and accompanying telemetry on film and magnetic tape, and to translate the telemetry to engineering units. It was also desired to build a system which could supply near-real-time data, in the form of pictures and telemetry to the operations team, and the science team. These data would serve to allow control of the spacecraft and missions in an efficient manner. Other objectives were the construction of a system:

(1) As nearly spacecraft independent as possible.

(2) Easily operated and maintained.

(3) Able to store and retrieve photographic material and produce additional photographic material such as duplicate negatives, enlargements, and other products.

(4) Able to produce photographic images from magnetic tape recordings made in real-time.

2. *Design specifications, implementation plans, and major milestones.* In 1962 a study contract was awarded to Fairchild Camera & Instrument Corp. for putting together a set of functional specifications based on the requirements known at that time. These requirements covered functional and operational considerations. At the conclusion of the study contract, JPL assembled a task team to develop a set of functional specifications from the Fairchild specifications (the JPL FOT series) for use in implementing the system. The major milestones in the development of the spacecraft TV-GDHS were as follows:

(1) Determination of need for a spacecraft TV-GDHS (1962).

(2) Study contract let to Fairchild (1962).

(3) JPL design task team organized (June 1963).

(4) Implementation contract let to General Precision, Inc. (January 1964).

(5) Installation of the interim configuration (started July 1965).

(6) Interim configuration accepted (September 1965).

(7) Installation of the final configuration (started January 1966).

(8) Final configuration accepted (April 1966).

(9) *Surveyor I* launch (May 1966).

(10) Delivery of final document (December 1966).

System implementation began with the awarding of a cost-plus-incentive contract to General Precision, Inc. (GPI) in January 1964. The most salient feature in the implementation of the system was the complexity of the system and the tight interrelationships between equipment performance and operational constraint.

The functional specifications provided clear requirements for equipment and system performance during real-time operation, i.e., during actual receipt of spacecraft signals. The specifications did not clearly define the requirements for system testing, setting up system calibration to a particular spacecraft, and system operational usage, both real- and non-real-time. These latter requirements were covered lightly or were only implicit. Most of the operational problems encountered in this system were a result of this deficiency.

The original functional specification called for several functions and items in the photo processing subsystem that were later dropped from the contract. These included rectification, custom enlargement (ability to rotate the image within the frame during an enlarging operation), a focus analyzer, and an identity copier. However, the identity copier was subsequently implemented by a separate contract with Applied Optics and Mechanics, Inc.

It was originally intended that installation of the on-site subsystem be made in the SFOF as a preliminary configuration in order to gain experience with the equipment. The on-site equipment would then be moved to Goldstone for installation of the scan converter and its breadboard drivers in the SFOF as an interim configuration. Installation of the rest of the system would then complete the final configuration.

The preliminary configuration was never delivered because of delays in the design and manufacture of the equipment. Even though the schedule for launching *Surveyor I* was slipped, it became apparent that elimination of the delivery of the preliminary configuration could optimize mission support. Consequently, only the interim and final configurations were delivered. Installation of the interim configuration began in July 1965 and it was accepted in September 1965.

Installation of the final configuration was started in January 1966 and final acceptance was on April 30, 1966. The *Surveyor I* was launched in May 1966.

The facilities were furnished by JPL for the installation of the system. These included space, special darkroom areas, power, water, special deionized water, sewer lines, etc. The requirements for the facilities were generated by GPI as part of the contract.

3. Design description. The system consisted of two major parts: TV-11 (Goldstone), and TV-1 (SFOF).

Equipment at TV-11. The configuration of TV-11 used for the *Surveyor VII,* as shown in Fig. VII-9, consisted of an on-site data recovery (OSDR) subsystem and an on-site film recorder (OSFR) subsystem. The RF signal was injected into the system at the interface between the DSS 11 receiver and the OSDR. At this point, the S-band spacecraft signal had been converted to an intermediate frequency of 10 MHz. The OSDR further converted the signal to 5 MHz. In the 600-line-scan mode this 5-MHz signal was transmitted to the SFOF via a microwave link and recorded on the DSS 11 FR-800 videotape recorder. This signal was demodulated to supply the video baseband signal to the film recorder and the DSS 11 FR-1400 tape recorder. In 200-line-scan mode the OSFR converted the signal to 500-kHz and 70-kHz. The system was designed to transmit the 500-kHz signal to the SFOF over the microwave link; however, during *Surveyor I,* the 70-kHz signal was used because only the interim configuration at the SFOF was committed at that time. The 70-kHz signal was demodulated to supply the video signal to the film recorder. In both modes, the OSDR recovered the TVID, generated vertical and horizontal sync signals, and provided time that was translated from the NASA time code for use by the film recorder. The film recorder exposed film with the video image, an internally generated electrical gray scale, the raw TVID in bit form, and time and other identifying characters in readable form. The film was sent to TV-1 for processing.

Equipment in the SFOF (TV-1). This equipment consisted of a media conversion data recovery (MCDR) subsystem, a media conversion film recording (MCFR) subsystem, a media conversion computer (MCC) subsystem, a scan conversion subsystem, an FR-700 videotape recorder (later FR-1400 and HW-7600 tape recorders), a display subsystem, and a photographic, storage and retrieval subsystem. Figure VII-10 is a functional block diagram for TV-1.

At the SFOF, the MCDR accepted the signal from TV-11 and processed it in essentially the same manner. In addition, the MCDR also provided the computer with the raw TVID for conversion to engineering units. The MCFR exposed film in the same format as the OSFR with the addition of readable TVID as processed by the computer.

The MCFR had two exposure stations (optics boxes). After exposure, film from one of the boxes was fed directly to the Kodak Bimat processor where the film was laminated with Bimat imbibed film as a continuous process. The continuous processing provided a controlled development negative and a high contrast positive transparency. The negative was then used to produce strip contact prints. The negative was chipped and filed. The film from the other optics box and from TV-11 was wet processed and put into storage as archival data (for the *Surveyor I* mission the TV-11 film was wet processed commercially). The converted signal was processed by a subsystem which became known as the video display and driver subsystem (VDDS). This included the scan converter and the drivers to the console displays and the quick-look paper camera located in the SSAC area.

The consoles displayed slow-scan television, Radio Electronics Television Manufacturing Association (RETMA) scan television, scope A video signal presentation and selected digital data of the TVID. The quick-look paper camera had the capability of providing paper prints within 10 sec of receipt of the video image. In the SSAC area there was a status and TVID wall board display unit that was driven by the media conversion computer. This displayed the spacecraft camera parameters within 1 sec after receipt of the video image.

The photographic storage and retrieval subsystem provided the capabilities of producing the strip contact prints, mentioned above, and making 8 × 10 in. enlargements of selected frames upon demand. Subsequent to *Surveyor I,* additional capabilities (which were part of the original design) were implemented. These consisted of the ability to make copies of mosaics, to make duplicate-roll negative for the use of the Public Information Office, and for distribution to the scientific community.

4. Design changes subsequent to Surveyor I mission support. Many design changes were made to the system after initiation of *Surveyor* mission support. These changes are described below under four categories. Most of the major equipment modifications were direct results of the experience gained from the *Surveyor I* mission. Changes to improve operational efficiency or stability are discussed in the subsequent paragraphs.

Microwave link, center frequency change. The center frequency of the RF signal transmitted to the SFOF from

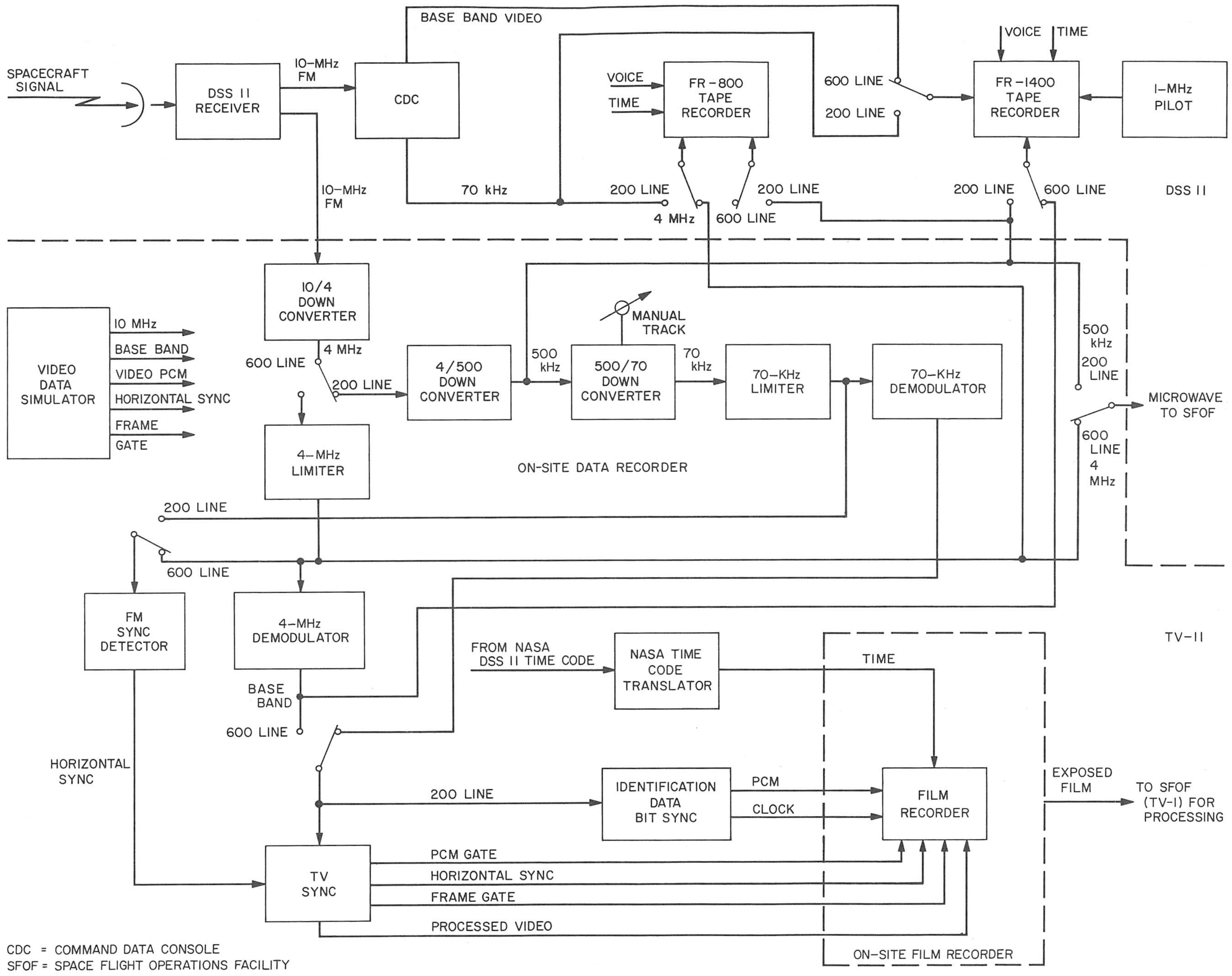


Fig. VII-9. Functional block diagram of TV-11 recording real-time mode, *Surveyor Project*

TV-11 which was recorded on the FR-800/700 tape recorders was changed between *Surveyor I* and *II* missions from 5 to 4 MHz. It was found that the sync tip frequency (6.25 MHz) was very close to the upper limit of the capabilities of the microwave link and of the FR-800/700 tape recorders. This change increased the capability for recovering TV sync.

Video amplifier replacement. The video amplifiers in the system were replaced between the *Surveyor II* and *III* missions. The original amplifiers were found to have sufficient long-term dc drift to seriously degrade countdown calibration. A high crossover distortion was also evident. Calibration of these amplifiers required opening the equipment and making a screwdriver adjustment. This was too time consuming for efficient operation.

Video data simulator change. The video data simulator was changed between the *Surveyor II* and *III* missions to allow independent control of the various parts of the generated waveform. The redesign improved the simulation of spacecraft parameters, and reduced the time required to go between the 200- and 600-line modes of operation.

Real-time computer program change. The real-time system computer program was found to be awkward to use in a few instances, and changes were initiated subsequent to the *Surveyor I* mission to correct these poor characteristics.

Conversion from Bimat to wet processing. Between the *Surveyor III* and *IV* missions, the Bimat processor was removed from the system and wet processing was substituted. This decision was based on several factors:

(1) The paper cameras in SSAC proved to be adequate for producing the quick-look mosaics.

(2) The project scientist was able to grant some relief on the delivery of real-time products as a result of operational experience.

(3) The Bimat-imbibed material was expensive, short-lived, and long procurement times caused a high wastage factor.

(4) The Bimat processor had design and reliability problems which were not solvable within the time and resource constraints.

The wet-processed products adopted for use were produced on a more timely basis and were more acceptable to the wide variety of users.

Separation of the video display and driver subsystem. In the original system concept, the MCFR was the source of the driving signals for the scan converter and the video displays. Because of the phased delivery of the system, these signals were generated by a breadboard through the first mission. After *Surveyor I,* the decision was made to change the original concept of a single-source MCFR to a two-source concept. The breadboard was cleaned up and combined with the scan converter to become the VDDS. The other source, or parallel path, then became the MCDR/MCFR. This redundant system proved very useful and beneficial to real-time support.

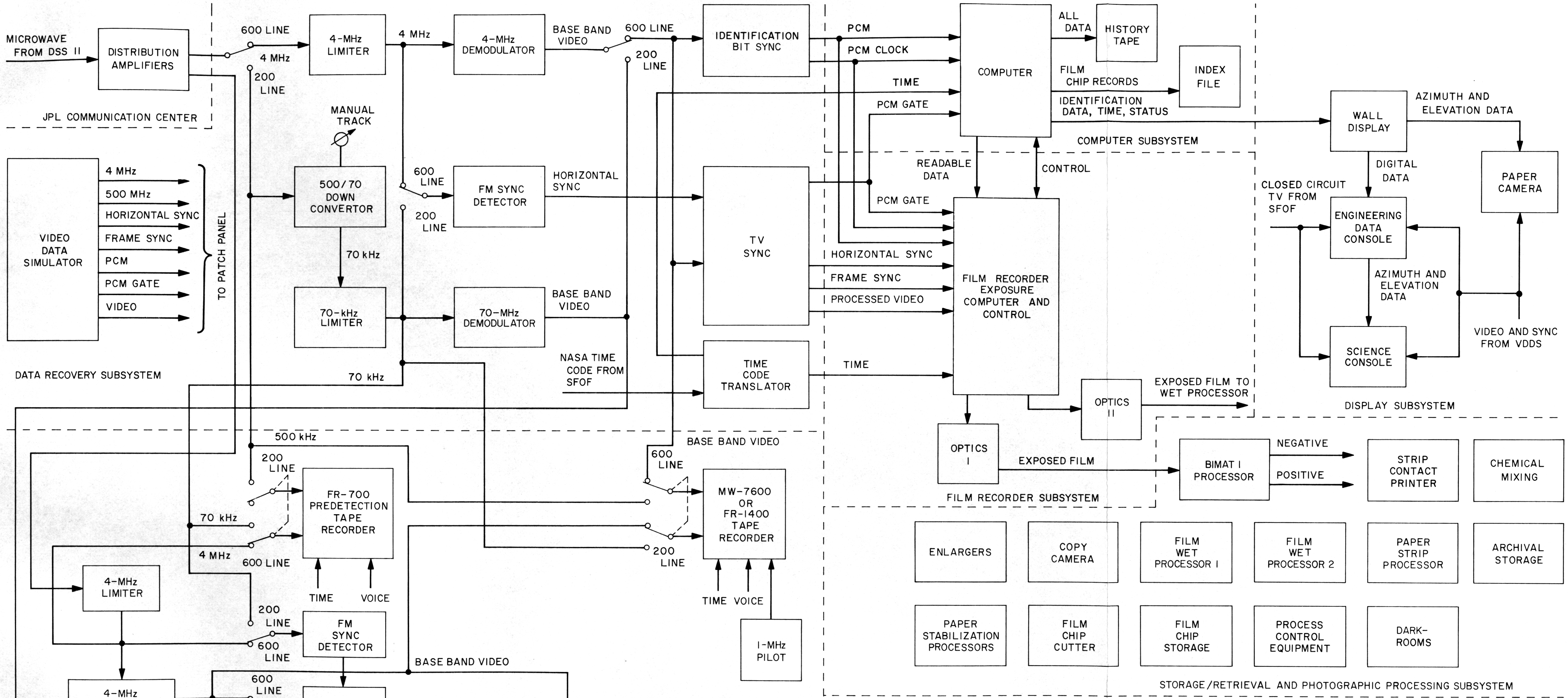
Patch panel modification. Between *Surveyor II* and *III* missions, the patch panels in the OSDR, the MCDR, and the VDDS were changed in both type and configuration. The BNC connectors used in the original patch panels were extremely inconvenient to use, in that signals had to be interrupted to make measurement or tees had to be used which cluttered the panels to the point of making them almost unusable. The new connectors (trumpeter type) provided for alternate loads and sources, monitoring, and automatic termination where required. A much more operable system resulted.

Addition of RF isolation amplifiers. RF isolation amplifiers were added to the OSDR, MCDR, and the VDDS between the *Surveyor II* and *III* missions to provide a more operational system. This provided constant loads to the RF feedlines from outside the system, eliminating the many operational problems that had been caused by changing-load conditions (resulting in improper terminations); isolation of RF loads, so that failures in one area would not affect other areas; reduction of noise pickup; increased versatility of the system, and better control over signal amplitudes.

Addition of FR-1400 and HW-7600 magnetic tape recorders. The original design of the system included an FR-1400 tape recorder within TV-1. This was not implemented until after the *Surveyor I* mission. Between *Surveyors II* and *III,* the HW-7600 tape recorder was added to the system. This provided improved time-base stability to the base band recordings, and redundancy.

Changes to meet increased loads on the system. A second wet film processor was added to the system to meet the following increased processing loads and to provide some redundancy:

(1) The removal of the Bimat processor.



NOTES:

BIMAT PROCESSOR USED ONLY FOR *SURVEYORS I* AND *III*. FILM WAS OTHERWISE WET PROCESSED.
EXCEPT FOR THE BIMAT PROCESSOR, THE SYSTEM IS SHOWN AS CONFIGURED FOR *SURVEYOR VII*.
NOT SHOWN IS THE PATCHING CAPABILITY THAT ALLOWED THE TAPE RECORDERS, THE FILM RECORDER, THE SCAN CONVERTER, AND THE ANALOG DISPLAYS TO BE DRIVEN BY EITHER THE DATA RECOVERY SUBSYSTEM OR THE VIDEO DISPLAY AND DRIVER SUBSYSTEM.

CCTV = CLOSED CIRCUIT TELEVISION
RETMA = RADIO ELECTRONICS TELEVISION MANUFACTURING ASSOCIATION
SFOF = SPACE FLIGHT OPERATIONS FACILITY
SSAC = SPACE SCIENCE ANALYSIS AND COMMAND
VVDS = VIDEO DISPLAY AND DRIVER SUBSYSTEM

Fig. VII-10. Functional block diagram of TV-1 real-time recording mode, Surveyor Project

(2) An added requirement for a duplicate negative roll for the Public Information Office to use, delivered in near-real-time.

(3) An added requirement for a duplicate negative roll for science use delivered in near-real-time.

(4) The vastly increased number of video frames received in real-time from the spacecraft over that contemplated during design of the system.

Changes to match the changes in spacecraft characteristics. Each spacecraft had different characteristics, some of which necessitated equipment changes over and above designed operating range capabilities. Below is a summary of the spacecraft characteristic changes requiring spacecraft TV-GDHS changes:

Characteristic	*Surveyor*						
	I	*II*	*III*	*IV*	*V*	*VI*	*VII*
Reverse scan	No	No	No	No	Yes	Yes	Yes
Overscan	No	No	No	No	No	Yes	No
Over deviation	No	—	No	Yes	Yes	No	No

For most video display devices reverse scan simply required reversing the leads to the vertical deflection coils. This was not true for spacecraft TV-GDHS film recoder since the image was only a portion of the total frame. The frame also contained both human-readable and machine-readable data relating to the image. Additionally, the reversal necessitated changes in the sensing for the cathode ray tube (CRT) protection circuit.

A 10% overscan was used on the *Surveyor VI* mission. Circuitry in the film recorders had to be changed to accommodate this increase in horizontal size.

Surveyor IV and *V* spacecraft had cameras in which the video signal deviated the FM signal in excess of the original specification such that the signal representing white in the pictures fell outside the RF bandpass of the system and the video tape recorders. Two principal changes were made to the system to minimize the problem. The center frequency of the FM signal was shifted 500 kHz by receiver tuning at DSS 11 on the receiver feeding TV-11, and one of the bandpass filters was bypassed in both the OSDR and the MCDR.

During the second lunar day of the *Surveyor V* mission, the demodulation video portion of the TV signal (sync was not affected) was about 50 db below normal because of damage to the camera during the lunar night. Approximately 50 db of gain was added to the processed video signal (video stripped of sync). All calibration was lost but usable pictures resulted.

The television synchronizer was designed to operate in three modes, a high mode for high signal-to-noise ratio, operating on the baseband signal and two lower modes for medium and low signal-to-noise ratio, operating in the RF domain. The high mode was the only one that could be used for FR-1400 tape playbacks. Playback of the FR-1400 tapes recorded at DSS 42 during the *Surveyor I* mission revealed a lower signal-to-noise ratio than could be tolerated by the TV synchronizer in the high mode. A change was made in the TV synchronizer of the MCDR to incorporate some of the more sophisticated techniques, such as the digital comb filter and special gating, used in the two lower modes.

5. *Tests and operations.*

Objectives, plans, and procedures. Before acceptance of the final configuration, while using the accepted interim configuration, a considerable amount of effort was expended by the entire mission lunar operations team to develop acceptable countdown and operating procedures and to certify the system for use. During the last 4 mo before launching *Surveyor I*, the final configuration was installed and checked out.

Since operations had been tested using only the interim configuration, the final system supported *Surveyor I* on a noninterference basis. In spite of the very limited operational experience of the personnel manning the final configuration, the relatively unknown response of the system, and incomplete calibration, the final configuration contributed very substantially to the support of *Surveyor I*. This also provided the experience which was used as a basis for modifying the equipment, the computer programs, and the manning concepts to make the spacecraft TV-GDHS more operational.

The final documentation, which was to be delivered shortly after equipment delivery, was behind schedule. The last items arrived 8 mo after receipt of the equipment. Unfortunately, the last document delivered was one of the most important. This was the utility manual for the film recorders. This created serious problems in that training of new personnel was difficult. The preliminary adjustment procedures not only were inadequate, but led to an inordinate amount of time required for repair and adjustment.

Controlled systematic unit test procedures were not generated by the contractor. Maintenance procedures were given in the utility manuals. Selected portions of the acceptance test procedures were used as the unit test procedures but in many cases these were not adequate.

Accurate system adjustments were required before real-time operations. Accurate calibration during the real-time countdown was required to achieve the accuracy of the recorded data required by the project and stated in the functional specification. This was implicit in the functional specifications, but the *Surveyor I* mission revealed the necessity of better test equipment, the modifications, and better countdown procedures.

Tests and personnel training. The on-site equipment and most of the SFOF real-time equipment were subjected to *acceptance tests* at contractor facilities. The tests were repeated within JPL facilities after installation. The remainder of the system was subjected to acceptance tests only at JPL.

Numerous RF compatibility tests, run during the fall of 1965, included engineering and operational tests with the T-21 spacecraft to determine total system characteristics. These tests resulted in the development of a procedure to reverify RF characteristics of the total system before each mission.

The spacecraft TV-GDHS and personnel participated in several types of *operations tests.* Before the *Surveyor I* mission, these tests included personnel training tests (designated as class A tests), system performance tests, (designated as class B tests), and flight simulation tests, (designated as class C tests). The class A tests served a dual purpose of supporting SSAC in their training for lunar operations, and in training the spacecraft TV-GDHS personnel.

Operational problems. The contractor's original estimate of the number of personnel required to run the system (40) proved remarkably accurate. However, before the *Surveyor I* mission it was difficult to justify this number. It was determined that approximately 25 persons were adequate for the periods of 3–4 mo between missions.

From an operational point of view, the thought and effort that went into the photo processing subsystem during implementation were inadequate. Consequently, a great amount of effort was required after the *Surveyor I* mission to develop compatible processes, ferret out the problems, and set up operational procedures. As a result of the insufficient development of this subsystem, four types of problems occurred during the missions.

The first type of problem was unsatisfactory photometric control. Photoprocessing is not a finite science. Although the requirements in the functional specifications were well delineated, the implementation left much to be desired. Throughout the program, repeatability (or consistency) was a difficult characteristic to obtain despite the rather elaborate controls. The problems of photometric control have been traced to characteristics of the sensitometers, densitometers, the film processors, and the recording film itself.

For any given photo product, there was a number of combinations to consider. These involved: developer, temperature, time of development, and exposure. But since only one or two processors were available and there were four different products required, only one or two of the process parameters could be varied for a compatible operational system to result. The search for the proper combination was extremely time consuming. By the time of the *Surveyor V* mission, a fairly good combination had been developed. However, it was not considered to be optimum.

A separate problem from film mottling revolved around the recording film (S.O. 337) and the wet film processor Pakorol G-17. The film was high-speed, medium- to high-contrast, CRT recording film which exhibited high mechanical pressure sensitivity. The film processor was a type in which the transport rollers formed a wringer type squeegee action. This combination proved to be the source of mottling. Keeping the developer transport rollers extremely clean reduced, but did not eliminate, this problem. Changing the processor, the recording film, or both, appears to be a solution. After the problem source and the apparent cures were determined, sufficient time was not available to implement a change.

Production of duplicate negative rolls was the fourth type of the photographic subsystem problem. The identity copier was delivered late in the program due to the delays associated with letting a separate contract for its implementation. Subsequent problems associated with making the identity copier operational precluded its use for real-time, or near-real-time support of the *Surveyor* missions. The lack of this equipment precluded the production of the six or seven duplicate negatives required for delivery to the scientific community as originally planned. The strip contact printer was used to produce the duplicate near-real-time negatives and those which made up the data package for the *Surveyor VII* mission. Duplicate negatives for the data packages of the other missions were produced at various agencies of the government.

c. Simulation system. The simulation system was designed and implemented to provide realistic simulation of the spacecraft and its responses for use in operational tests. The final system, involving several configurations, was the result of several changes and additions. The last addition was the alpha scattering data simulation configuration.

The final simulation system consisted primarily of the SDCC in the SFOF augmented by certain *Surveyor* mission-dependent equipment, the system tester portion of the CDC at Deep Space Stations, and the interconnecting ground communications circuits.

The simulated data were generated in the SDCC based upon tracking and telemetry data requirement packages specifically prepared for each test. Separate simulation system configurations were used for the generation of telemetry, tracking, and science data. Telemetry and tracking data could be provided simultaneously. The computer in the SDCC did not have the capacity to handle the additional task of providing video data. Due to constraints on resources, the video data simulation system configuration was significantly different in that peripheral equipment was used in addition to the computer. Those operational tests which involved use of both types of data were constrained to allow for reconfiguration. The various configurations are described below:

1. *Telemetry simulation.* The first portion or configuration of the simulation system implemented was for T&DS. Simulated spacecraft PCM telemetry data were generated by using the telemetry data simulator program in the PDP-1 computer in the SDCC (Fig. VII-11). Individual analog and digital signal values vs mission time were provided for input to the PDP-1 computer from a magnetic tape deck. The computer performed the functions of commutation, analog-to-digital (A/D) conversion, PCM formatting, and bit rate selection to output a nonreturn-to-zero (NRZ) PCM bit train identical to that provided by the spacecraft signal processing system. The PDP-1 computer output was routed to a bank of SCOs comparable to those in the spacecraft. The composite PCM/FM signal was recorded on FR-600 magnetic tape along with voice anotation for tape location identification and time code signals. These tapes were provided to each Deep Space Station and AFETR for playback to the CDC system tester or, in the case of AFETR, the station telemetry data monitoring equipment. During tests with DSS 11, the PCM/FM signal was routed to the station in real-time via the 96-kHz microwave link. Later, data phones were added to the system in the SDCC at DSS 61, DSS 42, and AFETR. The PCM PDP-1 computer output could then be routed directly to these stations during operational tests via the communications lines. After this capability was added, the PCM/FM recordings were used only as backup for these stations.

The SCO bit rate selection during tests was controlled by the commutator test coordinator in correlation with direction of command transmission reported over the operational voice command net. The test coordinator also had a limited capability for the control of signal value changes in real-time.

The time-correlated signal values were developed at HAC for each mission and provided on IBM cards for input to the SFOF 7094 computer for preparation of the magnetic tape used to input the data to the PDP-1 computer (Fig. VII-12). Generation of telemetry signal values required a detailed analysis of each spacecraft subsystem as a function of the designated trajectory and the standard sequence of events. Computer programs were used for the power and thermal signals. All other signals were generated manually. Each value was then converted to a telemetry voltage by application of the appropriate calibration coefficients.

A technique was developed to provide real-time response to midcourse and terminal descent maneuvers generated during the test. This involved the hand computation of certain telemetry signal values in accordance with the maneuver parameters provided by FPAC and expeditious preparation of a special PDP-1 computer magnetic tape. The telemetry data package also contained a graph of the station-received signal level as a function of time for the specific trajectory simulated. This was used at the Deep Space Stations to adjust an attenuator in the system tester to provide realistic signal input levels to the station receiver.

2. *Tracking data simulation.* Simulated tracking data were generated in the SFOF 7094 computers before each test or test phase, based on established launch conditions. The process used several computer programs in sequence, as shown in Fig. VII-13. The resulting data consisted of individual tracking data points for each

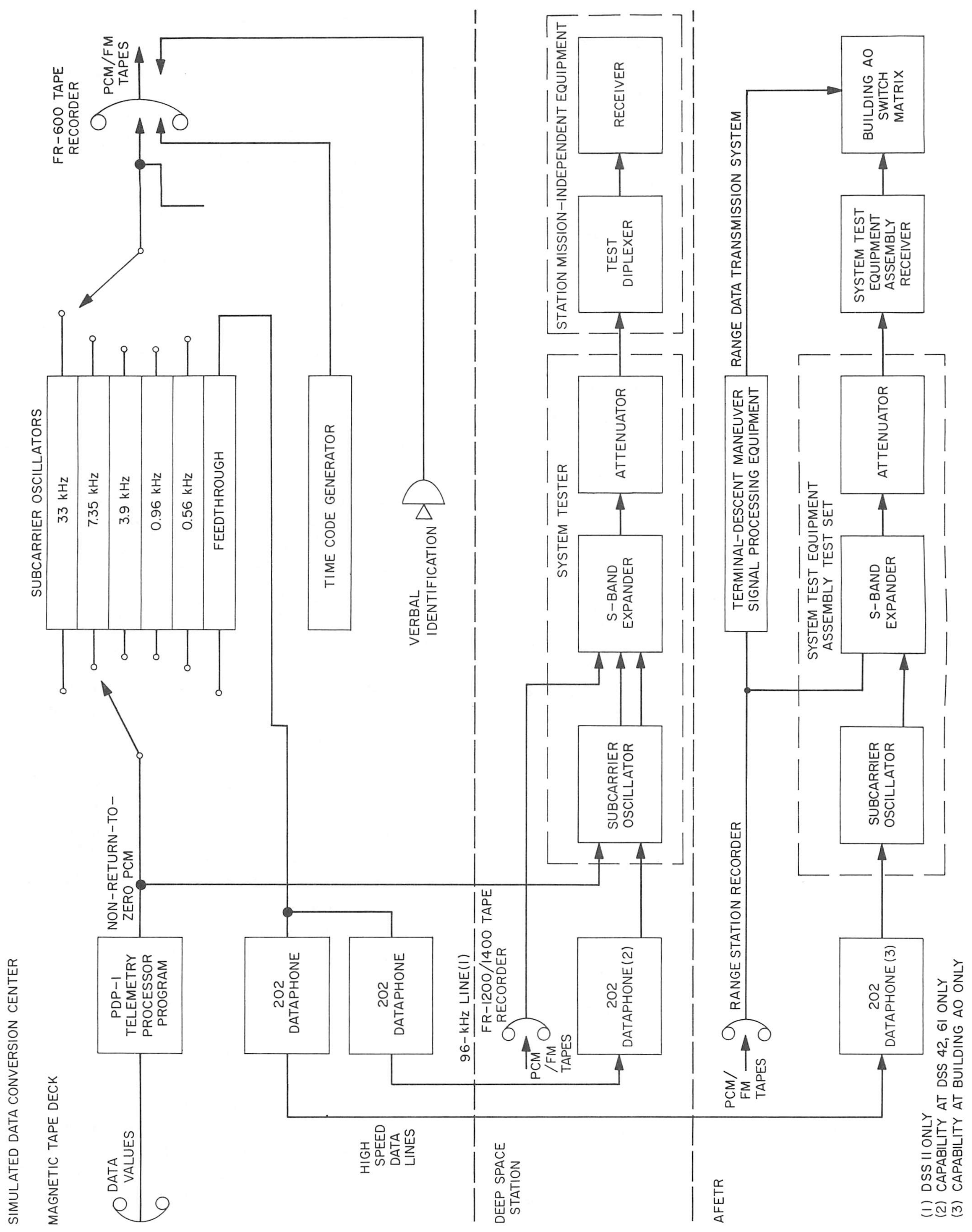


Fig. VII-11. *Surveyor telemetry simulation system*

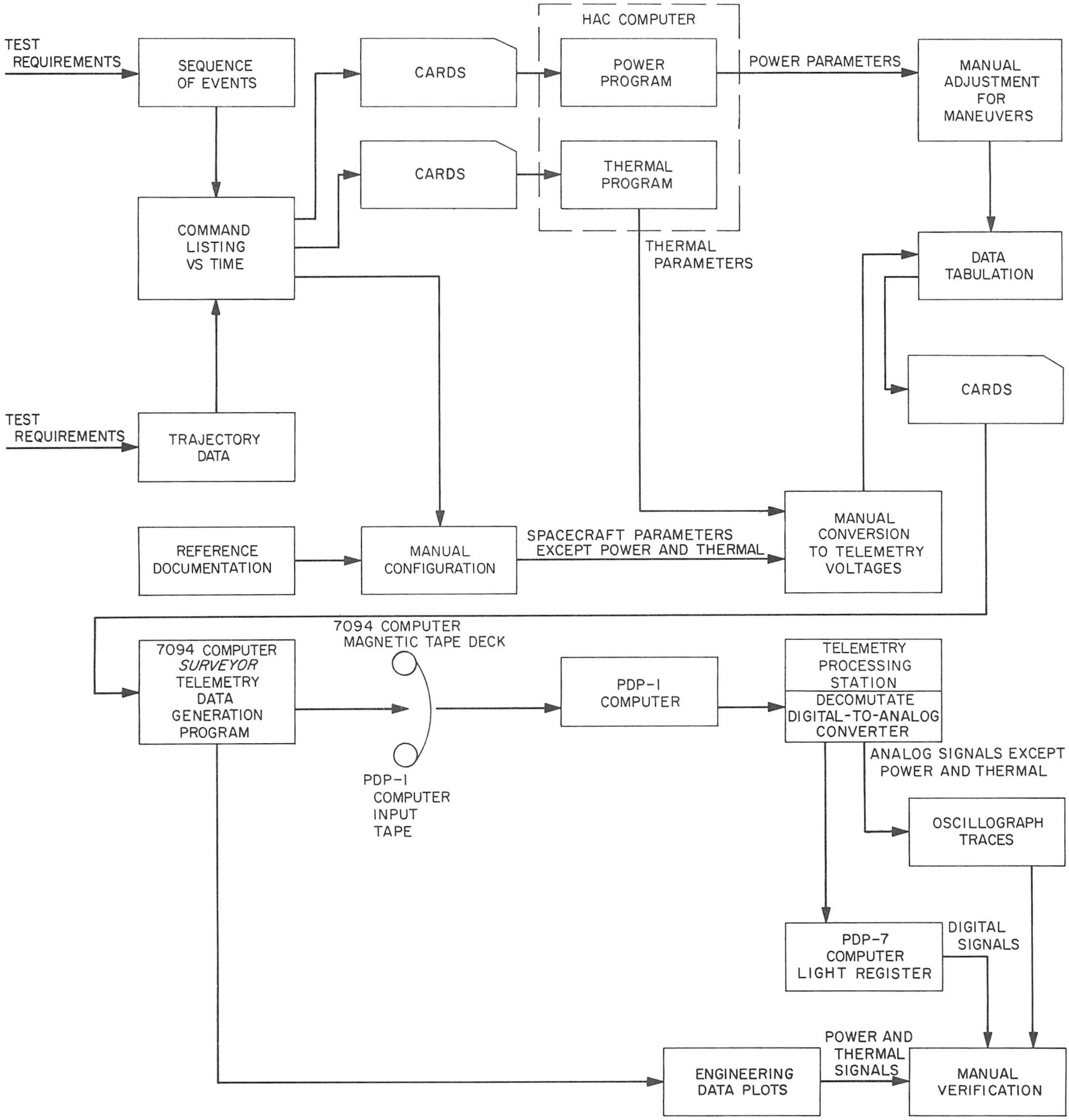


Fig. VII-12. Simulated telemetry data preparation and checkout sequence

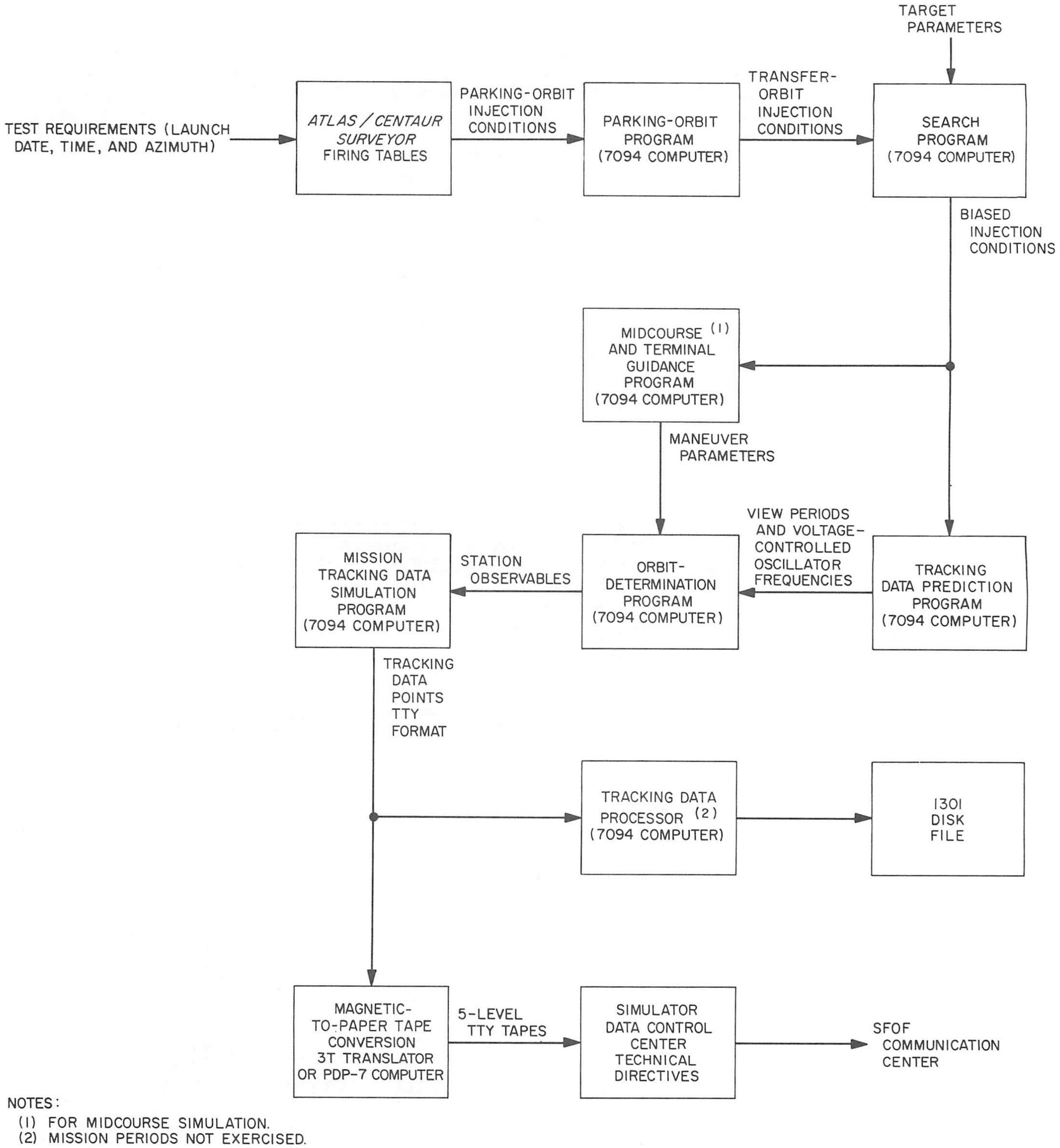


Fig. VII-13. Simulated tracking data generation sequence

designated station. The data were in TTY format identical to that generated by the station tracking data handling equipment. During a test the data were transmitted by the SDCC TTY equipment to the SFOF communications center for distribution into the operational system or, for those portions of the mission not exercised during the test, the data were loaded directly into the 1301 disk file from magnetic tape. The TTY data output from the SDCC could either be routed to the appropriate station for playback to the SFOF or directly to the SFOF DPS. Both options were used at different times depending upon individual test requirements. Post-midcourse tracking data could be generated to reflect the actual midcourse maneuver simulated during a test; however, this procedure necessitated a break of several hours in the test.

3. *Science data simulation.* The original television simulation system configuration was intended to provide a minimum system for use in training the SSAC personnel. This first configuration consisted of: a prototype spacecraft camera, peripheral equipment for driving the camera, interim configuration of the spacecraft TV-GDHS data recovery subsystem and scan converter, science and engineering consoles in SSAC, and closed-circuit television in the SFOF. The camera could be located either within a circular panorama or on the roof of the SFOF. The performance and benefits derived from the version were marginal.

Changes in requirements and design were implemented. The revised configuration (Fig. VII-14) used the PDP-1 computer to generate commands and to produce realistic data associated with the camera condition. It was, in effect, a closed-loop system. This system could be used internally to the SFOF (by simulating the functions of the CDC, the station, and the spacecraft) or expanded to include DSS 11, the CDC, and TV-11. This provided the capability of operating the MOS in the standard mission configuration.

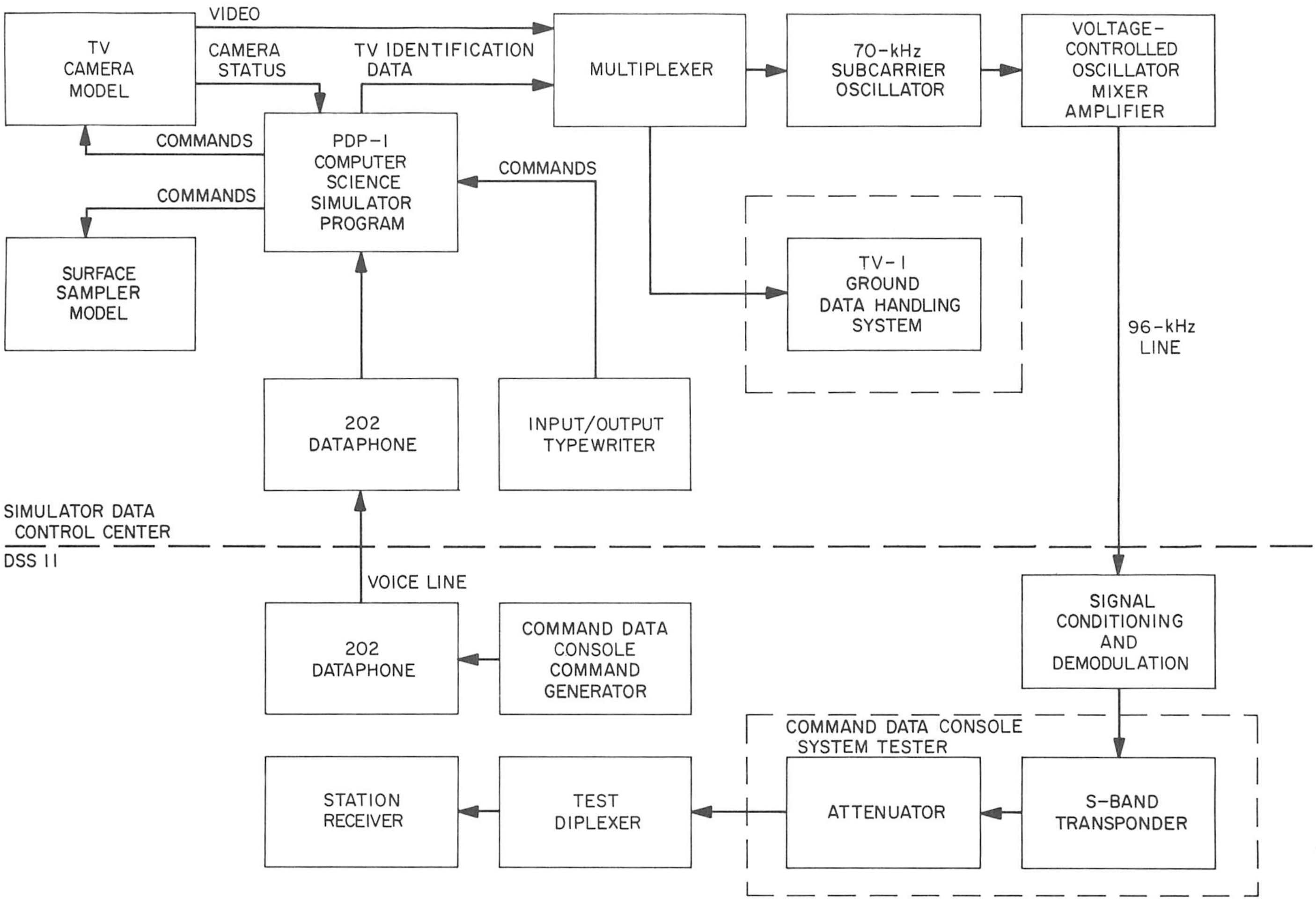


Fig. VII-14. Television/surface sampler simulation system configuration

Between *Surveyors II* and *III*, a model of the SM/SS was added with its interface equipment. This configuration provided a realistic simulation of the operational constraints and the capabilities of *Surveyor III, IV,* and *VII* spacecraft.

Alpha scattering data simulation was provided by using an instrument model located in one of the JPL environmental facilities. The PCM data were transmitted via data phone circuits to the SDCC to modulate appropriate SCOs. The signals were multiplexed with the normal engineering telemetry generated by the PDP-1 computer. The composite FM was transmitted via the 96-kHz link to the CDC system tester at DSS 11. FR-600 recordings were provided to the overseas stations. For simulation within the SFOF only, the alpha scattering data were applied directly to the SDCC ASI 6050 computer. This computer performed data accumulations in the same manner as the station SDS-920 computers and output TTY formats to the communication processor (Fig. VII-15).

4. DSIF simulation. In addition to simulating the spacecraft, the Surveyor simulation system could simulate functions of the DSIF and CDCs. Simulation included command recovery of the simulated spacecraft data in a format comparable to the CDC. It could also simulate command log TTY messages, normally transmitted by the DSIF, in communications processor compatible formats. These capabilities were used primarily for internal SFOF tests.

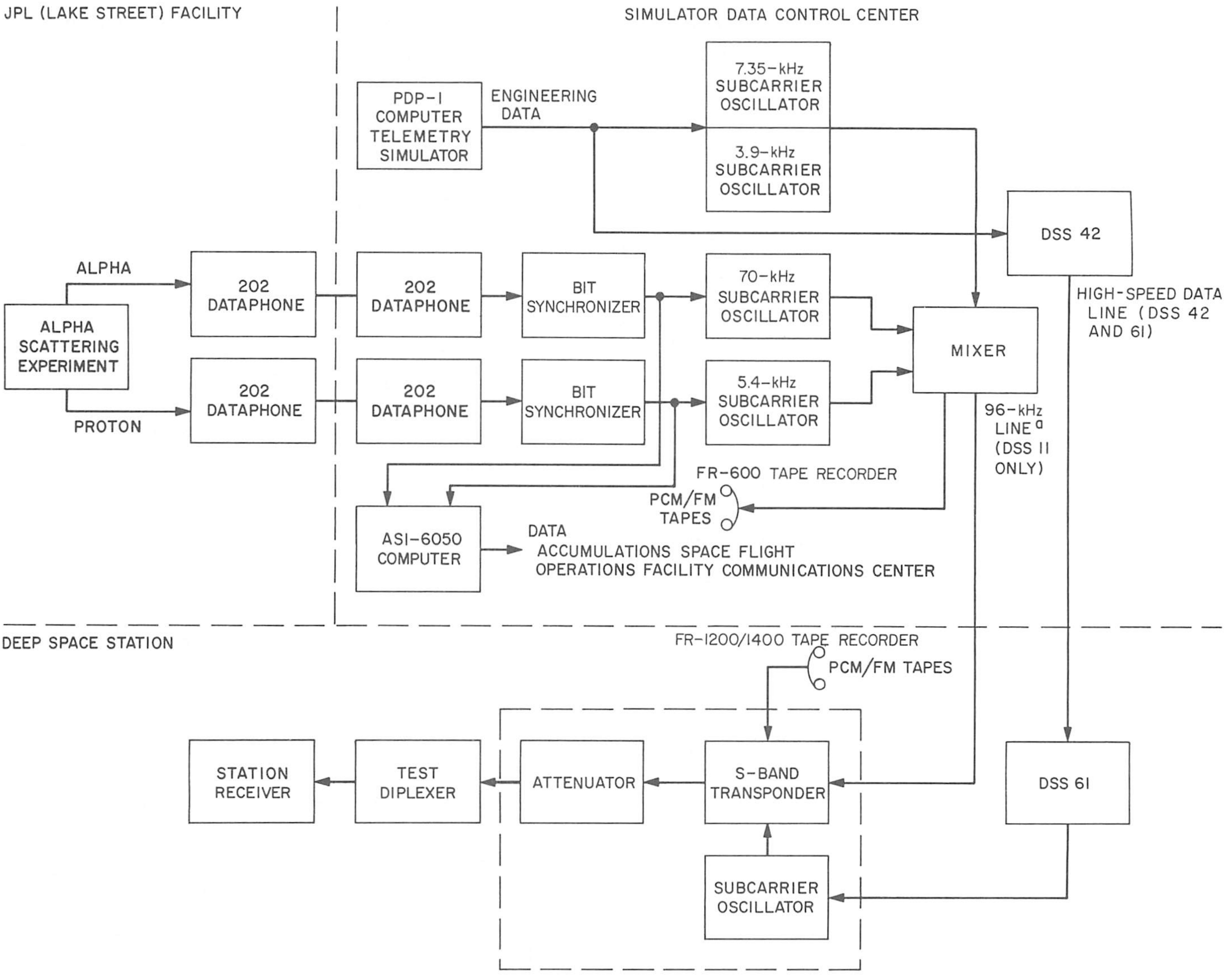


Fig. VII-15. Alpha scattering experiment simulation configuration

4. Mission Operations System Testing

The MOS test program basically consisted of two categories of tests: equipment engineering compatibility test and operational tests. This section will define the concept behind these categories of tests.

a. Equipment engineering compatibility tests. The primary objective of the MOS equipment engineering compatibility tests was to verify that all equipment necessary to the MOS were electrically and functionally compatible with each other and with the spacecraft, and were within specified tolerances. A brief description of the more significant tests performed in pursuit of this objective is given in the following paragraphs:

1. Initial compatibility tests of a composite CDC/DSIF/spacecraft system at HAC. The major items of the system tested were a JPL feasibility model DSIF receiver, a Goldstone production model CDC, a breadboard spacecraft transponder, and a simulated DSIF transmitter. The primary objectives of these tests were to determine overall functional compatibility between the CDC and the feasibility model DSIF receiver, and to determine the system changes required to correct significant deficiencies and incompatibilities.

2. Compatibility tests for DSIF/CDC at Deep Space Stations. A series of DSIF/CDC compatibility tests was conducted upon initial delivery of each CDC to a Deep Space Station. The objectives of these tests were as follows:

(1) Determine that the CDC, system tester, and DSIF interfaces were compatible and were as defined in the *Surveyor*/DSN interface description.

(2) Ensure that all applicable and measureable *Surveyor* telecommunications parameters agreed with values specified in the db allocation and margin summary for telecommunications system document and other applicable specifications.

(3) Demonstrate and certify that the CDC and DSIF telecommunication and instrumentation equipment were capable of supporting subsequent operational tests and space flight operations.

(4) Demonstrate and retest necessary changes required to correct deficiencies and incompatibilities.

3. Dynamic model/DSIF compatibility tests at DSS 11. A spacecraft transponder scheduled for launch in an early *Atlas/Centaur* flight was tested for compatibility with the DSIF. The transponder was located in a screen room about 6 mi from DSS 11 with its output radiated into the DSS 11 antenna.

4. Compatibility tests at Goldstone for T-21/MOS. The T-21 spacecraft was operated at Goldstone over the 9-wk period from August 9–October 8, 1965. The tests were generally separated into three categories: engineering compatibility, functional compatibility, and operator training.

The primary objective of the test was to demonstrate the engineering and functional compatibility between a flight-equivalent spacecraft, and the *Surveyor* MOS and DSN system, and to provide operational training for the MOS and DSN personnel.

The secondary objectives of the test were to establish the validity of spacecraft simulation techniques involving the system tester and magnetic tape recordings, to verify spacecraft telecommunications system design parameters, and to validate the RF STEA as a simulated Goldstone duplicate-standard S-band transmitter and receiver system.

The T-21 spacecraft was mounted on a ground transport vehicle and installed in an RF-shielded enclosure approximately 6 mi from DSS 11. Sufficient AGE was provided to install, check-out, and maintain the T-21 spacecraft in an operating condition. The objectives of the engineering or functional compatibility and operator training phases of the T-21 tests are described below:

Spacecraft/DSIF/CDC verification test. This test was run to demonstrate that there were no gross system incompatibilities that might preclude the successful accomplishment of the engineering compatibility tests.

Radio frequency compatibility tests. These tests were run to demonstrate the ability of the DSIF S-band system to communicate with the spacecraft for all spacecraft RF system configurations.

Command compatibility test. This test was run to demonstrate the ability of the DSIF/CDC to command the spacecraft in both spacecraft receiver operating modes, automatic phase control (APC), and automatic frequency control (AFC).

Telemetry compatibility tests. These tests were run to demonstrate the ability of the DSIF and CDC equipment to receive, detect, process, display and record spacecraft telemetry data in all spacecraft telemetry modes.

Television engineering compatibility tests. These tests were run to obtain the necessary data for a quantitative assessment of the spacecraft TV system performance and of the effects of the ground TV processing and recording subsystems on the quality of the TV pictures.

Operator training tests. During the latter part of the T-21 test period, tests were conducted that were specifically for training representatives of the operations crews at DSSs 11, 42, and 51. For the early part of the test, a sequence of events was used, exercising all possible modes of the spacecraft, generally in the order that they occur in a mission. Emphasis was placed on spacecraft–DSIF–CDC interaction rather than data evaluation. For the latter part of the tests, a real sequence of events was used on a condensed time scale. These tests were run from the SFOF using the SFO staff and procedures. The tests were arranged such that personnel from each of the three stations participated in a sample of all the tests and thus were exposed to a number of spacecraft DSIF situations.

The operator training tests proved more useful for both DSIF and SFOF than originally anticipated. The importance of this kind of training for all projects could not be too strongly emphasized.

5. Spacecraft TV-GDHS/DSIF compatibility tests at DSS 11. The spacecraft TV-GDHS equipment was received from the contractor in July 1965 and installed at DSS 11. In September 1965 a series of RF system tests was conducted. These tests measured the 600- and 200-line performances of the total RF/TV-GDHS, including rate response and FM improvement tests. Also, FR-800/1400 tape recordings and playback were accomplished, showing resultant rate response, FM improvement, and tape residual noise.

b. Operational tests. During the design, development, and operation of the MOS, a comprehensive system test program was carried out to verify the functional compatibility of the various system elements, validate operational procedures, and ultimately, to demonstrate the operational readiness of the total system to support each mission. The operational test program was adequate in all respects, fulfilling the basic requirements for verifying the operational readiness of the MOS before each mission.

1. Test concept, objectives, and plans. During the development of the MOS, the test concepts, requirements, objectives, and techniques were developed, and a master test plan was prepared and published. A preliminary issue of this document was available for the start of operational testing which was a little more than two years before the first launch.

Originally, three major test classifications were established to satisfy the basic requirements of the test program. These major classifications were:

(1) Class A: facility internal tests.

(2) Class B: SFO/DSN functional compatibility tests.

(3) Class C: operational tests.

In general, accomplishment of the tests was intended to be successive, progressing through a sequence combining larger elements of the system through complete system integration and validation. The final test served to simulate the total mission by exercising all system elements simultaneously (Fig. VII-16). The test classifications were defined during the early portion of the operations phase.

After the first two missions, the test designations were changed to: training, compatibility verification, and operational readiness test (ORT). However, the objectives and requirements for the tests remained consistent with the A, B, and C classifications.

2. Test coordination. To provide a realistic test environment and the necessary test support, a test coordination activity was established under the SFOD. The primary responsibility of this group was to perform those functions required to provide realistic simulation during the test which were not a normal part of the mission operations. The group consisted of personnel located at the SFOF and the Deep Space Stations during test periods.

The test coordination group functioned within the SFOF to provide simulated spacecraft telemetry, tracking, and video data that were correlated with the transmission of spacecraft commands and other operational functions. In addition, the group provided the initial test conditions and other data which would normally be available during an actual mission. The group also provided data that simulated certain operational positions required to establish realistic interfaces in those tests that exercised less than the total system. The SFOF group maintained communications with station test coordinators during tests and directed their activities as required to ensure time correlation for the test support activities.

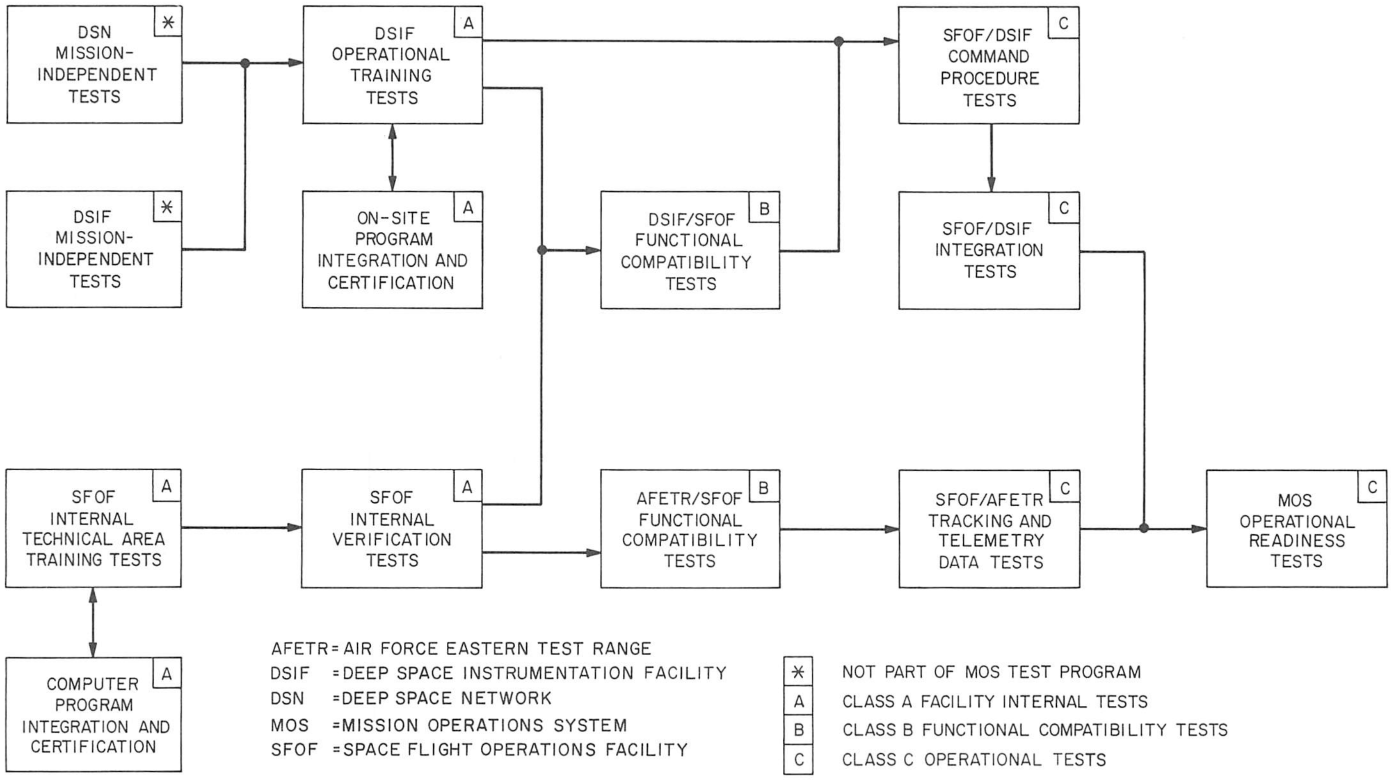


Fig. VII-16. Relationship of MOS tests

The station test coordinators were responsible for simulation data tape playback and for simulating realistic RF signal strength within the ground system in time correlation with the mission progress. Station test coordination requirements were greatly reduced after the first two missions. At that time the capability to transmit simulated telemetry data in real-time from the SFOF Deep Space Stations, except DSSs 51, and 72, was provided. Later this capability included the AFETR. A very significant function performed by the test coordination group was the development and injection of anomalous data to exercise the operational system in nonstandard situations. This included simulation of spacecraft failures, large injection errors, and reduced ground system performance.

3. *Class A-facility internal tests.* The objective of these tests was to ensure that the equipment, software, and personnel in the Deep Space Station and at the SFOF had the required capability and were prepared to support subsequent testing and missions. These tests were conducted under the direction of the cognizant local manager and to the extent required to assure satisfactory operational capability. In some cases, extra facility support was required to provide realistic interfaces. The major part of class A testing occurred before the first mission, beginning in the SFOF about two years before the first launch. It began at DSS 11, 42, and 51 about one year before the first launch. As operational experience was acquired and class B testing was begun, class A testing decreased accordingly until after the first mission. Similar testing was accomplished at DSS 61 and 72 as these stations became operational. Subsequent testing of this type was accomplished before each mission in varying degrees, depending upon the unique requirements for the mission. These subsequent tests were designated as flight training or lunar training tests and were directed toward familiarizing operational personnel with the critical phases of each upcoming mission.

4. *Class B-functional compatibility tests.* The primary objective of the class B test series was to verify the total SFO/DSN system performance in accordance with the specified functional requirements and that these functional requirements were compatible with the spacecraft data configuration, including command, telemetry, tracking, and science. A secondary objective was to verify that all *Surveyor*/DSN hardware and software interfaces were compatible. The total system was exercised in various configurations and under a variety of conditions which could occur during space flight operations. The successful accomplishment of these objectives enabled the appropriate elements of the MOS to be declared capable

of supporting the mission and provided the basis upon which a system configuration freeze could be established. Successful class B tests were not repeated. However, major system configuration changes which occurred throughout the operational phase of the program required individual compatibility tests to be run between missions through *Surveyor VI*.

5. *Class C-operational tests.* The primary objective of the class C test series was to exercise the elements of the *Surveyor* SFO/DSN system (personnel, hardware, and software) to the maximum extent feasible, within the limitations of the simulation capability; and to ensure compliance with the *Surveyor* SFO plan. The tests were designed to progress through a logical sequence and to accomplish the following:

(1) Utilize the full complement of personnel, facilities, equipment, and software to conduct simulated mission phases in a realistic manner, and within the normal mission time constraints.

(2) Emphasize standard mission operating procedures and selected nonstandard situations regarding spacecraft and ground systems performance.

(3) Require handling, processing, and interpretation of the full range of mission data under conditions of normal and degraded communications.

(4) Establish the operational readiness of the SFO/DSN system for each *Surveyor* mission.

Satisfactory completion of the class B tests was a prerequisite for each facility before participation in class C testing. Class C testing was extensive during the 2-mo period preceding the first mission, both in terms of the quantity of tests conducted and in the phases of the mission simulated. Subsequent class C test activity was decreased, as a result of increased operational experience, to a single operational readiness tests preceding each of the last four missions. These latter tests exercised the MOS in the configuration required for each mission and consisted of simulating only the critical mission phases of launch, midcourse, and terminal descent. Following each ORT, a mission operations readiness review was conducted, to make an evaluation of the T&DS, the mission operations equipment, and the SFO system launch readiness.

5. Performance During Missions

This subsection describes the performance of the basic elements (i.e., personnel, computer programs, and equipment) of the MOS during the seven *Surveyor* missions.

*a. **Personnel performance.*** Personnel performance was consistently good in all areas of the MOS throughout each mission. The reactions of personnel reflected the high degree of readiness attained from the premission operational training tests. However, as to be expected in a complex and extended exercise such as a *Surveyor* mission, random human errors did occur. The built-in system of organizational redundancies prevented these errors from seriously affecting programs, equipment, or the spacecraft performance.

The capability of MOS personnel to cope with nonstandard situations was demonstrated many times during the seven missions. The majority of these situations were minor and were resolved by using documented procedures for nonstandard situations or by the use of engineering know-how. Examples of the four more paramount nonstandard situations that put practically all resources and capabilities of the MOS personnel to task are given in the subsequent paragraphs.

1. *Surveyor II.* The activities of the MOS, following the failure of a vernier engine to thrust during the *Surveyor II* midcourse correction, were almost entirely nonstandard. The mission control function was taken over by project management. The MOS personnel advised project management of alternate courses of action, provided qualitative judgment of the risks involved, assessed the MOS capability for implementing the various actions, and then implemented the actions to carry out project management decisions. The performance of the MOS personnel during this mission was quite adequate even though the spacecraft failed to achieve a soft landing on the moon.

2. *Surveyor IV.* A NASA decision to conduct a vernier engine static firing experiment after touchdown was made after the launch of *Surveyor IV*. Consequently, extensive MOS planning and coordination effort was required during the transit phase. Action had to be taken to schedule this activity, develop the command sequence, and provide necessary data to DSS 42 personnel to allow familiarization before the first posttouchdown pass. The MOS personnel accomplished this task in a very proficient manner. Unfortunately, due to a *Surveyor IV* spacecraft failure during the terminal descent, a postlanding vernier engine static firing was not attempted until the *Surveyor V* mission.

3. *Surveyor V.* A spacecraft vernier propulsion system helium leak occurred after the *Surveyor V* midcourse correction, resulting in an extensive analysis of the problem. The possible solution for the problem was developed

by a special task force team of MOS personnel. The task force effort culminated in the generation of a completely new terminal descent sequence that allowed *Surveyor V* to be soft landed on the lunar surface. Additional midcourse corrections were also performed in connection with the special effort to correct the leak and to ultimately achieve the desired vernier propellant weight and ullage conditions for the terminal descent.

4. Surveyor VII. Following a failure of the deployment mechanism of the alpha scattering instrument by which the instrument could not be lowered to the lunar surface, the MOS personnel devised a means to forcibly lower the instrument by using the SM/SS. Following tests at HAC and at JPL, by which the feasibility of the method was demonstrated, the devised method was applied and the alpha scattering instrument was successfully deployed.

b. Computer and computer program performance. There were a number of computer problems during the flight operations. Some were corrected by work-around procedures and others by replacing peripheral hardware.

The overall support was adequate for SPAC, FPAC, and SSAC operations. FPAC program computations were completed in the times specified for the sequence of events. SPAC and SSAC programs were occasionally delayed but with no major impact upon the operations. Operations personnel reacted expeditiously to the various problems and, consequently, the problems were more of a nuisance than catastrophic.

Telemetry data processing was excellent for all missions with relatively few problems encountered. The selection of telemetry data was achieved by switch control, providing rapid response to the data selection from one computer string to the other.

The **SOCP** underwent extensive testing before being put into operation at the Deep Space Stations. As a result, relatively few problems occurred during the missions. All changes were thoroughly verified, and the programs were certified before use in operations.

An interface problem between the IBM 7044 and 7094 computers caused considerable inconvenience during the *Surveyors I* and *II* missions. The problem was corrected through software and hardware changes and the computer and computer program performance during *Surveyor III* and *IV* missions was excellent.

The 7044/7094 computer interface timing was not completely resolved for *Surveyor* missions *IV* and *VII* to support the computer orbit determination program (**ODP**). This problem caused some aborts for the **ODP** runs for *Surveyor VI* and considerably more for *Surveyor VII.* Most of these problems were solved by clearing and restarting the IBM 7094 computer.

Major software and hardware design and philosophy changes were made to the IBM 7044 and IBM 7094 computers for the *Surveyor V* mission. Although training tests were minimal before the flight, personnel adapted quickly to the new design. There were a number of software and computer system problems involving the IBM 7044 computer core memory and drum storage and 7044/7094 computer interface timing that caused input card information transmission problems from the 7044 to the 7094 computers.

For *Surveyor V*, an on-site alpha scattering program was used at DSSs 11, 42, and 61 on the backup telemetry and command processor (TCP) and an IBM 7094 computer program was used at the SFOF. Some operational deficiencies in the program were corrected after *Surveyor V* with excellent performance during the last two missions.

The SC 3070 printers, administrative printers, and card readers caused considerable problems throughout the missions. Many computer runs had to be aborted due to incorrect input caused by the card readers. The SC 3070 printer problems caused loss of some telemetry and IBM 7094 computer outputs. The administrative printer problems would cause the loss of an operating area. In cases such as these, the operations were resumed at a different operating area.

c. Mission-dependent equipment performance.

1. Command data console performance. The 12 CDCs played an important role in the success of the *Surveyor* test programs and mission operations. The DSIF CDCs had sufficient built-in redundancy to prevent the problems that arose during the mission from having direct adverse effects. The most significant factor in reducing the number of failures was the extensive test program established for the CDCs. In addition to the development test program a prescribed series of unit and subsystem tests were required for each CDC before each mission, along with the periodic maintenance. The level of testing generally exceeded that of other DSIF equipment; however, the alignment and calibration during these tests served to make the subsequent mission relatively free from trouble.

During the seven missions of *Surveyor*, over 800,000 commands were transmitted to the spacecraft, over 87,000 TV frames were received, and more than 2,300 hr of alpha scattering data were accumulated.

Table VII-2 lists the CDC equipment problems by station and mission. This shows a complete list of problems and does not differentiate as to whether they occurred during the countdown or tracking operation or as to the problem severity. In general, the missions with lengthy lunar operations and the stations participating in these operations experienced most of the equipment problems. This was true except for DSS 61 where significantly fewer problems occurred. This is attributed to the relative newness of the CDC at DSS 61 compared to that at DSS 11 and DSS 42. The CDC at DSS 61 had been installed less than 6 mo prior to the first mission, whereas those at DSSs 11 and 42 were installed 25 and 13 mo before the first mission, respectively.

2. *Spacecraft TV-GDHS performance.* The spacecraft TV-GDHS was highly redundant by design. In particular, data were recorded in several forms and at several locations, to assure availability of data for postflight evaluations. This redundancy, plus built-in versatility, contributed to the capability of the spacecraft TV-GDHS to support video operations on all missions. At times, some of the nonoptimum backup modes provided the needed support. A general system performance rating for each mission is given in Table VII-3. The rating is given in terms of expected performance, from fair to very good (*Surveyors I* and *VII*). The expected performance for each mission was usually greater than for the previous mission.

Table VII-2. Equipment in which CDC problems occurred during missions

Station	*Surveyor*						
	I	*II*	*III*	*IV*	*V*	*VI*	*VII*
DSS 11	Tape reader, command printer, decommutator, and video processor	Command decoder	Elapsed time meter, telemetry amplifier, low-frequency oscillograph, switching unit (RPT), and noise generator	None	Television generator, elapsed time meter, input/output typewriter, command generator, telemetry panel, tape punch	Television monitor, command generator, command printer, and television photo recorder	None
DSS 42	Television monitor, tape punch, elapsed time meter, telemetry display, tape reader, and command generator	Command printer	Television photo recorder low-frequency oscillograph, oscilloscope, decommutator, tape reader	Decommutator and SOCP console	Patch panel, command generator, low-frequency oscillograph, decommutator, command subcarrier oscillator, television monitor, tape punch, command printer, and television photo recorder	Television monitor camera, command printer, tape reader, command generator, and low-frequency oscillograph	None
DSS 51	Decommutator and sequence counter	Tape punch	Low-frequency oscillograph and command printer	None	None	Command printer	None
DSS 61	Television photo recorder and television generator	Command printer, tape reader, and decommutator	Command generator and power supply	None	Command comparator	None	None
DSS 71	None	None	None	None	None	None	None
DSS 72	None	Decommutator	None	None	None	Not applicable	None

Table VII-3. Spacecraft TV-GDHS performance

Surveyor	Performance vs expected performance	Major accomplishments	Major detractions
I	Very good	1. Provided much more data and photo products than expected 2. Provided good real-time scan conversion for the public Information Office	1. Not well calibrated 2. Undermanned 3. Quantity of data overwhelmed system 4. Inability to make duplicate roll negatives 5. The major portion of system not operationally ready and not committed
II	Not applicable (no lunar operations)		
III	Fair	1. Relatively smooth prepass countdowns	1. Rash of equipment failures 2. Inability to make duplicate roll negatives 3. Operations not well coordinated 4. Film mottling 5. Data record keeping not well organized
IV	Not applicable (no lunar operations)		
V	Good	1. Recovered usable video in spite of spacecraft anomaly on second lunar day 2. Provided limited quantity of duplicate roll negatives	1. Film recorder camera failures 2. Film mottling 3. Excessive shading 4. Data record keeping better but not up to needs. 5. Film logistics problem
VI	Good	1. Kept up well under overload due to extremely large number of video frames from spacecraft	1. Film mottling
VII	Very good	1. Well coordinated operations	1. Film mottling 2. A few data recovery equipment failures

Table VII-4 gives a summary of the spacecraft TV-GDHS products with comments. In addition to the products shown in the summary, the system provided the following:

(1) Developed CDC 35-mm original negatives for all missions.

(2) Produced a positive and duplicate negative of the CDC 35-mm film of *Surveyor I.*

(3) Provided real-time displays consisting of slow-scan presentation, scope A presentation, identification display, and a scan-converted display.

(4) Provided the drive for the paper camera quick-look display.

(5) Provided FR-700 videotape recordings and FR-1400/HW-7600 tape recordings.

(6) Provided a digital tape dump of the chip index file used by the science ground data handling group to produce film chip record catalogs.

(7) Provided a calibration report as follows:

(a) *Surveyor I*: None.

(b) *Surveyor III*: Extraction methods only.

(c) *Surveyor V*: Same as *Surveyor III* with additional data on shading, limited number of density measurements and a frequency-to-density calibration envelope.

(d) *Surveyor VI*: Same as *Surveyor V* on an improved format and an increased amount of data.

(e) *Surveyor VII*: Same as *Surveyor VI* but on an extensively revised format and comprehensive data.

B. Tracking and Data System

1. Introduction

The T&DS for the *Surveyor* Project consisted of selected facilities of AFETR, GSFC, and DSN. Each of these facilities was composed of various elements capable of tracking a spacecraft or launch vehicle, recovering telemetry or video data, generating and sending commands, processing and displaying data, and providing orbit determination data. Elaborate communication networks were included for the collection and distribution of data between the elements of the T&DS and its various central control points.

The T&DS, with the exception of certain mission-dependent equipment, was a mission-independent element of MOS. The capabilities of the T&DS enabled the MOS to assess and evaluate mission status and performance, implement appropriate command sequences required to maintain spacecraft control, and perform desired spacecraft operations during launch, during transit, and on the lunar surface.

The normal sequence of mission events divided naturally into two phases: the near-earth phase (describing the immediate prelaunch activity and approximately the first hour of flight from liftoff) and the deep-space phase (describing the events from initial acquisition by a Deep Space Station about 30 min after launch through the transit and lunar phases to the end of mission).

Each of these two phases placed different requirements on, and required different performance from, the supporting elements of the T&DS and for that reason are treated separately in the following pages.

a. Near-earth phase. In the near-earth phase, the facilities of AFETR and GSFC were employed with the emphasis on tracking the *Centaur* launch vehicle and spacecraft to provide early trajectory prediction data in real-time in case of an abnormal injection when nominal predicts were unlikely to be of much value. This provided the DSN with acquisition information based on the actual flight trajectory.

Telemetry data during this phase were helpful to launch vehicle and spacecraft designers in evaluating performance during launch. Key events (mark events) that were determined from the telemetry received were used in the near-real-time determination of the normality of the flight.

The seven *Surveyor* missions involved three direct ascent trajectories and four parking orbit trajectories. In the parking orbit missions, injection occurred between the south Atlantic and east Indian oceans. Each launch posed different near-earth problems for the T&DS in providing reliable coverage and communication.

b. Deep-space phase. In the deep-space phase, the DSN facilities provided continuous tracking coverage of the spacecraft so that a communication link between the SFOF and the spacecraft was maintained at all times. A stream of telemetry or video data, a command and command verification data stream, and a tracking data stream flowed over this communication channel.

With telemetry, video, command, and tracking data continuously available in a readily assimilable form, mission controllers in the SFOF were able to establish the spacecraft orbit and to control spacecraft function to achieve designated mission objectives.

c. Testing. To insure the successful launch and flight of each *Surveyor* spacecraft, much preparation and compatibility testing were needed to establish the operational readiness state. Thus, operational readiness tests were performed before each mission to optimize the probability of mission success.

The readiness of the mission operations personnel to support the *Surveyor* Project resulted from over 3 yr of training exercises and actual operations conducted jointly by JPL and HAC. Tests for each mission included compatibility tests of the spacecraft and the ground systems, communications tests between the Deep Space Stations and the spacecraft, communications tests between the Deep Space Stations, and AFETR, and the SFOF, communications tests within each facility, and operational tests of all equipment within each facility. Simulated anomalies were introduced into various systems to prove the abilities of personnel and equipment to provide corrective measures.

d. Performance. The difficulty of establishing a meaningful overall standard of mission support to compare with the actual support provided is due to the large number of independent areas of support involved. The larger the scope of the evaluation, the more general the evaluation becomes until the only criterion available is acceptable support or unacceptable support.

It is significant that, although data were lost on some occasions, there was no deleterious effect on the missions, and the majority of the data were later retrieved

Table VII-4. Spacecraft TV-GDHS products summary

Surveyor	Lunar day	Total television commanded frames	Recorded in real-time at TV 1	Original negatives: Playback DSS 42	Original negatives: Playback DSS 61	Original negatives: No. of sets Bimat process	Original negatives: No. of sets wet process	Strip contact print: No. of sets per day of operation	Strip contact print: Post-flight	Positive transparencies and duplicate negatives: Producing agency	Positive transparencies and duplicate negatives: No. of sets positive	Positive transparencies and duplicate negatives: Public Information Office negative	Positive transparencies and duplicate negatives: Engineering data reduction negative	Positive transparencies and duplicate negatives: No. of data package sets negative	Mosaic copies: No. of mosaic	Mosaic copies: No. of copies	Mosaic copies: No. of negatives	Prints Polaroid	Total No. of 8 × 10 in. enlargements
I	1	~10,300	~10,300			1	2[a]	1		AMS	2	—	—	6	none			none delivered	~2000
				150		1	1		2 sets	JPL	1 bimat[b]								
										JPL	1								
	2	~300	~300			1	2[a]	1		AMS	2			6					
										JPL	1 bimat[b]								
II	Not applicable (spacecraft failure)																		
III	1	6369	~6100			1	2	5–7[e]	~2000 frames	GSFC	2			7	127	1016	254	~20 touchdown pass	~2500
										JPL	1 bimat[b]	1[c]							
				~300[d]		1	1		5 sets	GSFC	2			7					
	2	40	40			1	2	5		JPL	1 bimat[b]	1[c]			none				
IV	Not applicable (spacecraft failure)																		
V	1	18,056	~16,500				3	6–7[e]	~2000 frames	JPL	1	1	1		182	1456	364	~300 touchdown pass ~100 special order	4620
										MSC				7					
				~2000	~650		2		6 sets	JPL MSC	1	1	1	7					
	2	1043	1043					7		JPL	1	1	1		none			~700	~100
										MSC				7					
	3	6	6				3	7		JPL	1	1	1	7					
VI	1	30,065	~25,600				3	6	~2000 frames	JPL La RC[f]	1	1	1	7	283	2264	566	~300 touchdown pass ~100 special order	3719
				~3700	~1800		2		6 sets	JPL La RC	1	1	1	7					
VII		20,993	~15,000				3	7–8[e]	~2000 frames	JPL	1	1	1	7	162	1134	324	~300 touchdown pass ~200 alpha scattering special order ~50 8×10 alpha scattering special enlargement ~100 special order	6619
				~3400	~2600				7 sets	JPL	1	1	1	7					

[a]1 set of original negatives commercially developed.
[b]Bimat positive of limited usefulness.
[c]Public Information Office duplicate negative produced from bimat positive (negative produced was of only fair quality).
[d]DSS 42 plus DSS 61.
[e]Depending upon mode of operation.
[f]Langley Research Center.

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from recorded data sources. This was due in part to the extensive redundancy built into all systems, to the substantial performance margins at which most systems operated, to the high reliability built into all equipment, and to the thoroughness of personnel training. Although the T&DS performance data given in this section are not indicative of the number of faults and anomalies encountered during the various missions, they are a reasonably straightforward indication of the dependability of the T&DS in support of mission operations.

In the deep-space phase, which includes postlanding lunar operations, the Deep Space Stations, in conjunction with the SFOF and GCF, fully met their support obligations.

On the basis of meeting obligations, the dependability of the T&DS in support of the *Surveyor* missions was excellent.

2. Interfaces and Management

The AFETR, GSFC, and GCF portions of the T&DS interfaced with the Launch Vehicle System, managed by Lewis Research Center. The AFETR and GSFC interfaces were between the tracking stations, the launch vehicle C-band transponder for tracking data, and the launch vehicle VHF telemetry system for vehicle performance and guidance data. The interface between the GCF and the Launch Vehicle System was minor and was usually considered as part of the MOS support.

The AFETR, GSFC, and DSIF portions of the T&DS interfaced with the Spacecraft System, managed by JPL. The AFETR and GSFC interfaces were between the tracking stations or ships and the spacecraft S-band telemetry system or the launch vehicle VHF telemetry channel carrying spacecraft data for confirmation of various important spacecraft events. The DSIF interface with the Spacecraft System was between the network of ground tracking stations and the spacecraft S-band telemetry and transponder system for transmitting commands to the spacecraft, receiving telemetry and video data from the spacecraft, and tracking the spacecraft by means of a coherent two-way communications channel.

The entire DSN (SFOF, DSIF, and GCF) interfaced with the MOS, managed by JPL. The personnel, facilities, hardware, software, and services needed for the SFOD and his supporting staff to control the deep space phase of the mission from the SFOF were provided to the MOS through this interface.

There were additional important interfaces between the T&DS and the SFOF, which was the control and data processing point for the DSN. The AFETR interface with the SFOF transferred trajectory and frequency data for a short period in the near-earth phase of the mission. The DSIF interface with the SFOF provided personnel, facilities, and services needed to control the mission support activities of the Deep Space Stations. The ground communication facility interface with the SFOF was a hardware interface consisting of the terminations of the external communications circuits linking appropriate tracking stations of the DSIF, AFETR, and GSFC to the SFOF. It also provided a terminal for the SFOF internal communications system, which connected various user areas to the SFOD and to each other for operational purposes, and for the SFOF closed-circuit television system.

The T&DS manager was responsible to the project manager for coordination and direction of all activities relative to the T&DS. The management structure employed for planning the utilization of T&DS resources is shown in Fig. VII-17.

The JPL/AFETR operations manager, under the authority of the T&DS manager, was in charge of the interfaces with the Kennedy Space Center, AFETR, MSFN GSFC Data Operations Branch, and NASA Communications network in the near-earth phase and coordinated the support required from these elements of the T&DS. The interface with the Launch Vehicle System was also managed by the JPL/AFETR operations manager through the *Centaur* missions office of Kennedy Space Center–Unmanned Launch Operations.

The DSN manager, under the authority of the T&DS manager, was responsible for managing all SFOF, DSIF, and GCF interfaces. The DSIF and GCF interfaces with the SFOF were managed by the track chief and the communications chief, respectively, under the authority of the DSN manager. The DSIF operations manager appointed a project engineer for each area of planning engineering and system data analysis to work with the DSN project engineer and the project MOS manager or his representative to define and implement support required from the DSIF. Thes SFOF operations manager appointed a project engineer to define and implement communications support in the SFOF. Facilities and data processing support in the SFOF was provided as a normal service in response to project requirements and was not covered by a specifically designated project engineer.

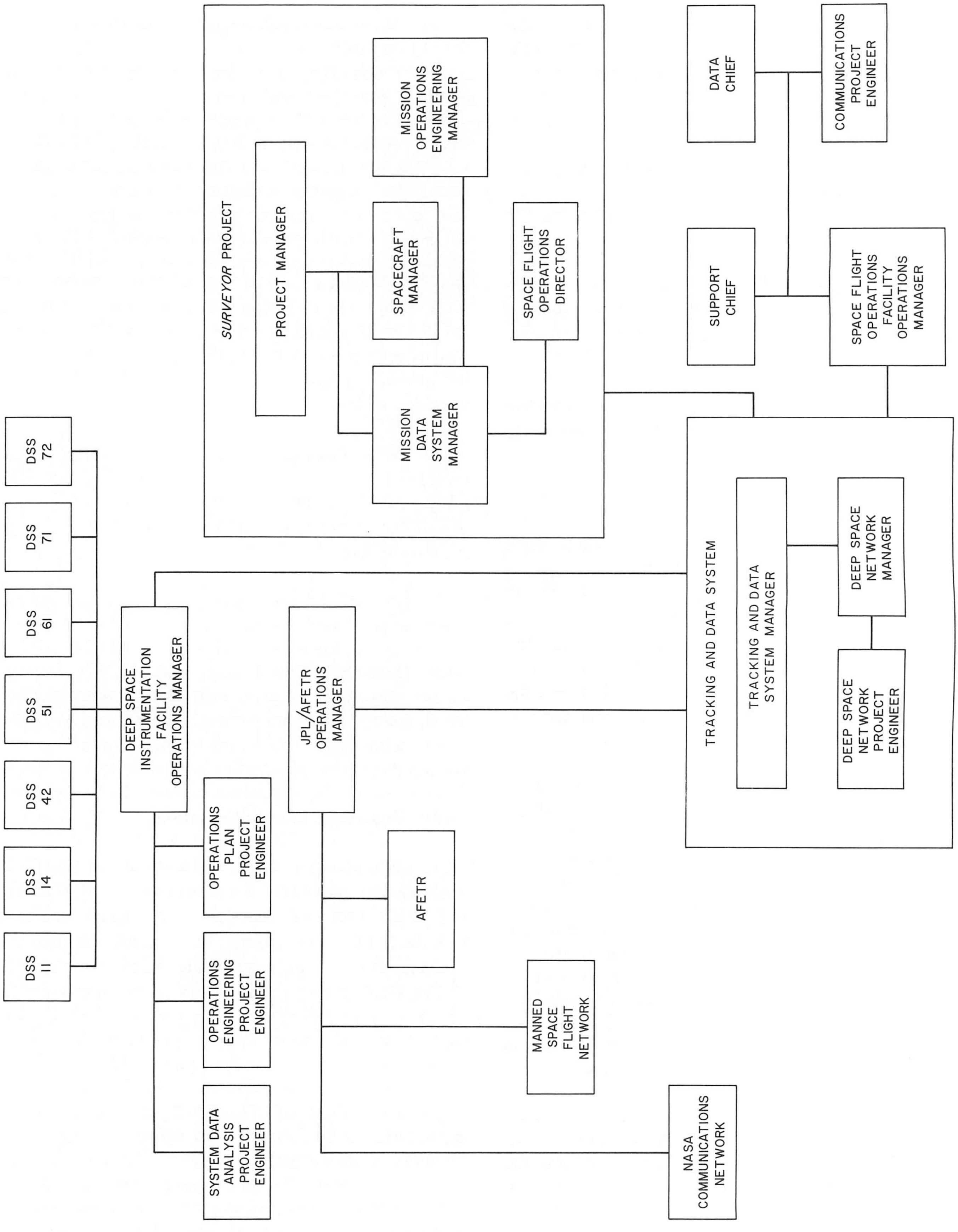

Fig. VII-17. Management structure for the T&DS

The authority of the T&DS manager and DSN manager applied to *Surveyor* support activities only. The line organizations of the AFETR, GSFC, DSIF, and SFOF facilities took care of all the routine activity required to enable each facility to meet the requirements levied on the T&DS. The support requirements are discussed in Refs. VII-1 and VII-2.

When the project entered an operational phase, the T&DS organization took on the configuration shown in Figs. VII-18 and VII-19 to permit it to support the project in both the near-earth and the deep-space phases. Before launch and in the near-earth phase (Fig. VII-18), the JPL/AFETR operations supervisor monitored the mission progress and the activities of all elements of AFETR, MSFN, and the NASA Communications Network (NASCOM) that had been committed to support a particular mission. He kept the T&DS manager and mission advisor informed to permit them to make their inputs to the mission director on matters affecting the decision to launch or subsequent conduct of the mission. In the deep-space phase (Fig. VII-20), the T&DS and DSN managers acted in an advisory capacity to the project, while the DSN project engineer and the chiefs of the various operational areas in the DSIF and SFOF actively supported the SFOD in the conduct of the mission.

3. Near-Earth Phase

a. Requirements. The primary T&DS objectives during the near-earth phase of the *Surveyor* missions were to provide:

(1) The initial acquisition station of DSIF with prediction data.

(2) Launch vehicle and spacecraft data for evaluation of the near-earth phase of the flight.

(3) Orbital information for an early analysis of the trajectory.

Those requirements that were specifically needed to meet these objectives were defined as class I. Requirements of lesser priority were placed in class II or III. The class I requirements are presented in Subsection VI-A-1 for direct-ascent and parking-orbit missions.

b. Configuration. The T&DS was comprised of facilities of AFETR, MSFN, and DSN. A list of the facilities which were available to support the *Surveyor* missions and their capabilities is presented in Section VI-I-2. A sample of the area coverage capabilities of the various stations and the corresponding station coverages in terms of time from launch are in the same section. Section VI presents station coverages only for the injection locations shown. The coverage capability of some stations changes radically with changing injection locations, especially in the case of parking orbit missions. For direct-ascent missions, the injection flight-path angle strongly influenced some stations' coverage capabilities.

Centaur C-band metric tracking data were transmitted to the AFETR Real-Time Computer System (RTCS for use in orbital computation, generating DSN prediction data and reformatting to decimal-type data. The reformatted and computed data were then transmitted to building AO for retransmission to the SFOF via the NASCOM as shown in Fig. VII-20. In addition to a receive and record capability, the AFETR and MSFN telemetry stations retransmitted certain channels of *Centaur* vehicle data and *Surveyor* spacecraft data in real-time to Kennedy Space Center (building AO, DSS 71, and building AE). Vehicle data were analyzed at building AE. Spacecraft data, being provided to the spacecraft engineers in building AO for analysis, were also processed by the CDC and TCP at DSS 71 and retransmitted to the SFOF (as shown in Fig. VII-21) for use by the SPAC team. A backup data path existed via 202 dataphone from building AO to the SFOF.

Before *Surveyor III*, a system was implemented to transmit *Surveyor* real-time telemetry data from the MSFN station at Carnarvon, Australia to the SFOF. The 30-ft antenna and unified S-band system at Carnarvon were used to receive, record, and demodulate the spacecraft telemetry signal. The 550-bit/sec data stream was then transmitted from Carnarvon to DSS 42, using 202 data sets and a NASCOM voice/data circuit. At DSS 42 the output of the data set was transmitted to the CDC, where it entered the normal system for processing and transmission to the SFOF.

c. Testing. The AFETR facilities were constantly being tested. The preparation for launch and near-earth phase support of each *Surveyor* mission included compatibility tests of the spacecraft and the ground systems; communications tests between the ground stations and the spacecraft, between facilities, and within each facility; and operational tests of all equipment within each facility. Subsections A-4 and B-4 contain a description of the tests conducted in participation with the other elements of the T&DS.

d. Mission performance. The extent to which the near-earth elements of the T&DS carried out their functions

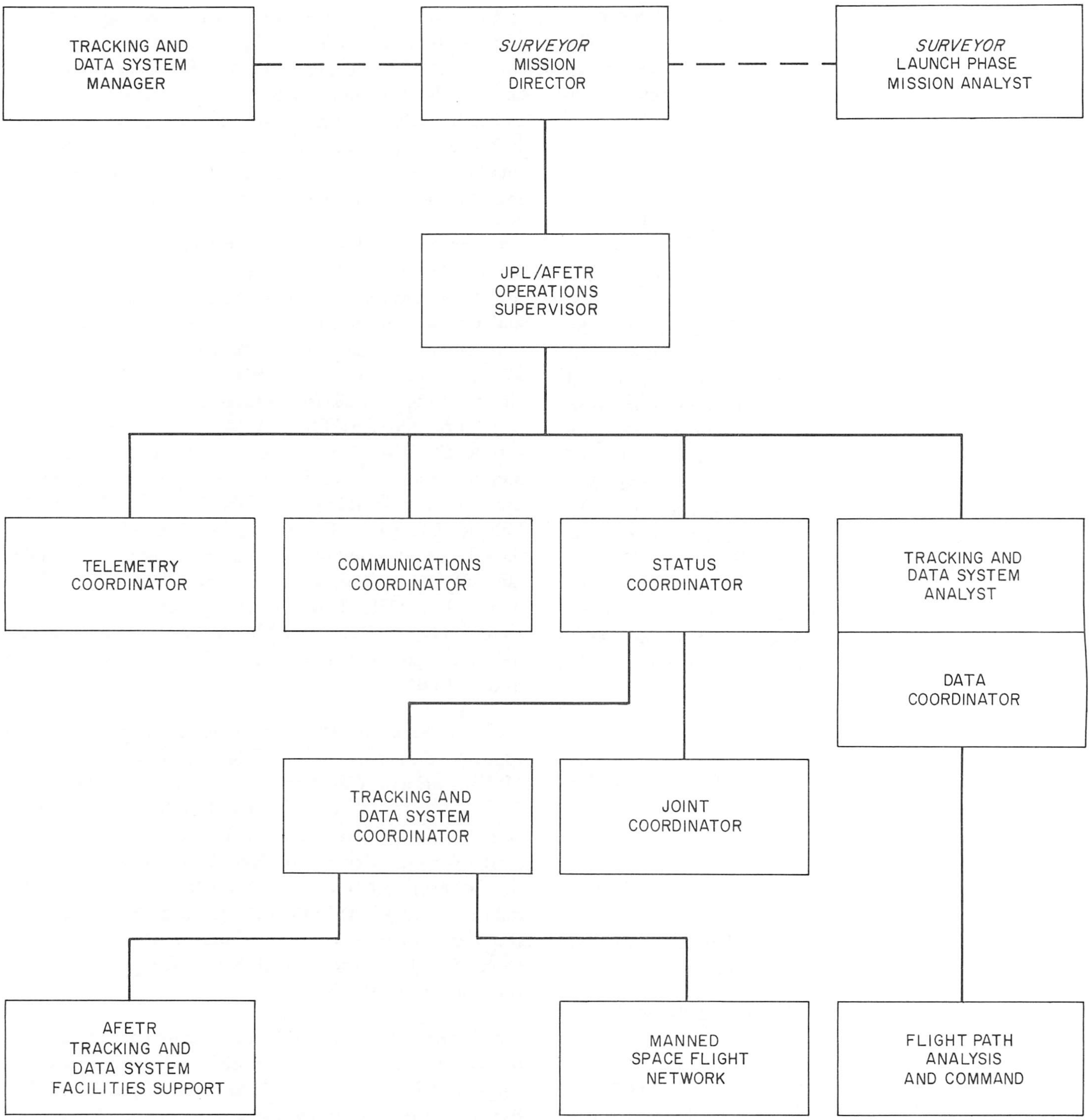

Fig. VII-18. Operations organization for the T&DS in the near-earth phase

is most easily seen by reviewing the general support provided as discussed below. Since the coverage capability and hence the actual coverage of most of the T&DS stations varied from mission to mission, no attempt is made here to list the actual coverage intervals on a station-to-station and mission-by-mission basis.

1. Metric tracking data (C-band). In general, all class I requirements were met, with actual coverage equalling or exceeding the estimates.

At AFETR and MSFN stations intermittent tracking conditions were occasionally caused by unfavorable

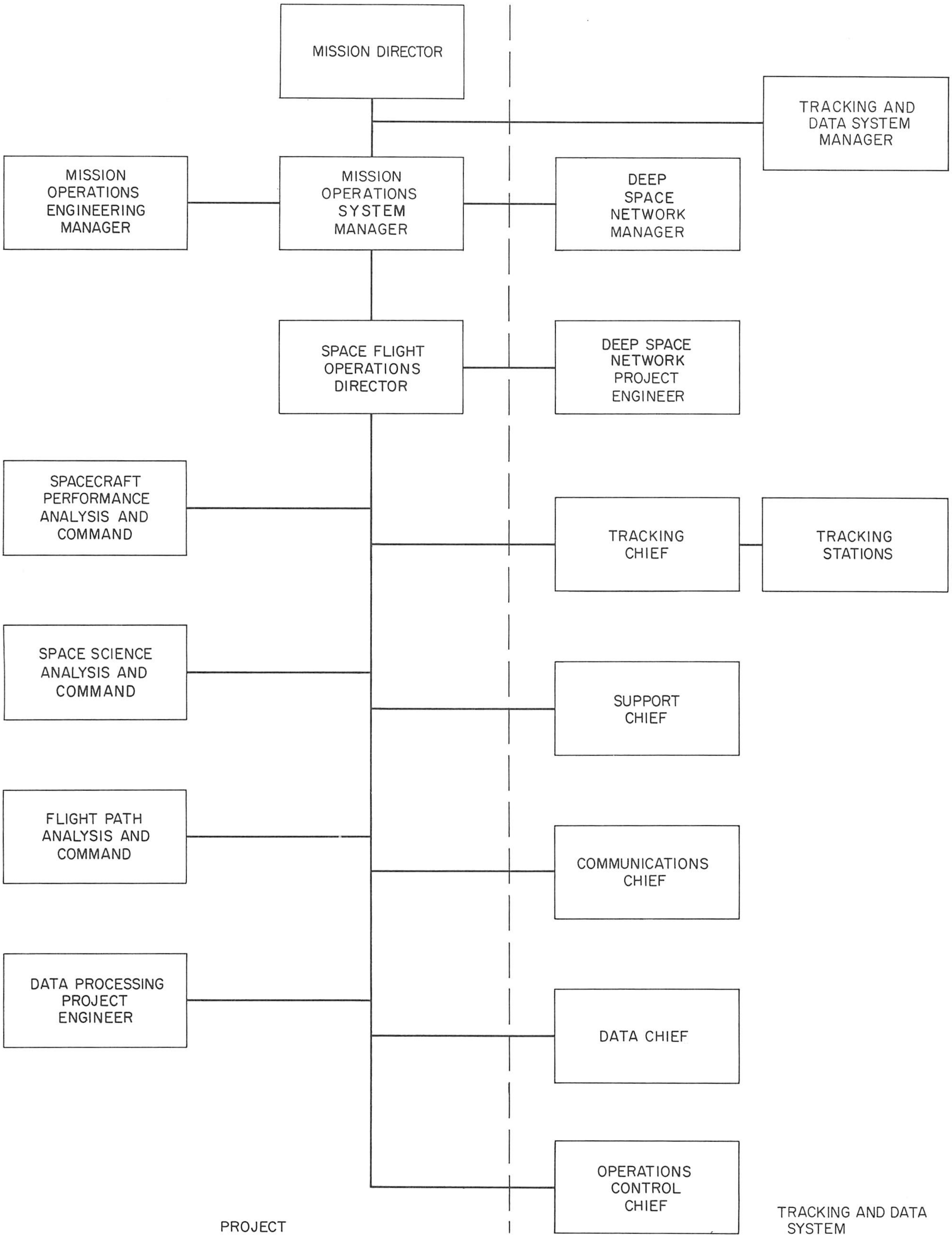

Fig. VII-19. Operations organization for the T&DS in the deep-space phase

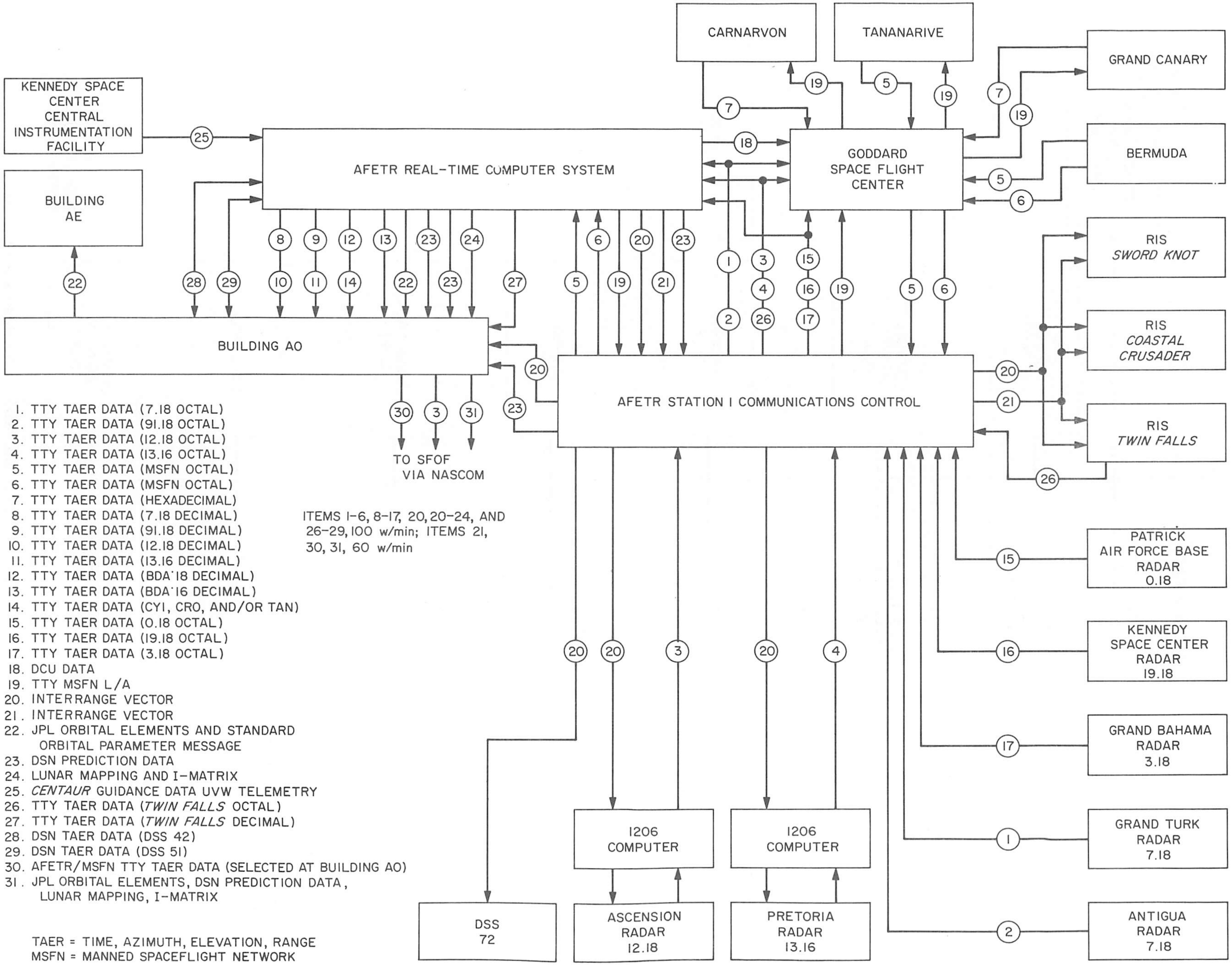

Fig. VII-20. *Surveyor* metric data flow

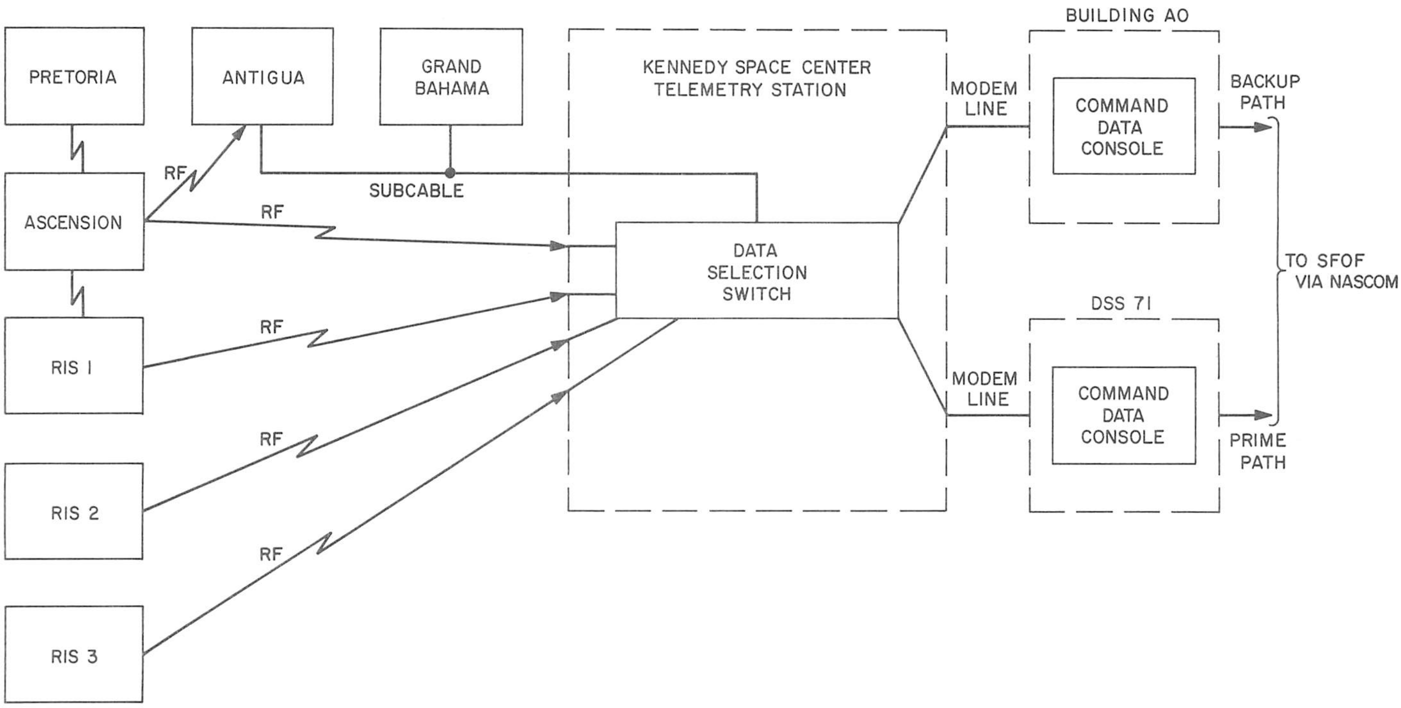

Fig. VII-21. Surveyor real-time spacecraft telemetry data transmission system

station-to-vehicle aspect angles or unexpected vehicle instability.

Equipment difficulties, procedural problems, and operator errors were also experienced occasionally. These problems were invariably corrected before the succeeding launch and there were no instances of repetitive failures.

On early missions some difficulties were encountered in delivering correct predict information to the MSFN stations from the GSFC computers. This was improved on later missions, as was the integration of the MSFN with other elements of the T&DS.

2. *Atlas/Centaur telemetry (VHF).* The requirements were met or exceeded with continuous and substantially redundant data being provided. In most cases, land station coverage was supplemented with two or more Range Instrumentation Ships (RIS). On the *Surveyor II* and *III* missions, range telemetry aircraft were used as needed to fill gaps in the coverage.

The MSFN stations also performed well, and the types of problems experienced meeting the C-band tracking requirements were notably absent. Between the MSFN and AFETR stations, all mark events were reported in real-time, or as near-real-time as possible.

3. *Spacecraft telemetry (S-band).* All primary S-band systems were used on a limited commitment basis since the *Centaur* vehicle was not roll-stabilized and the aspect angles could not be predicted. However, a sufficient amount of redundant coverage was available to fill these gaps in most cases and the requirements were met with minor exceptions.

For all *Surveyor* missions, initial acquisition of the spacecraft by the DSN occurred as scheduled. DSS 72 on Ascension island performed the initial two-way acquisition and commanding of the spacecraft, and provided early two-way tracking data for orbit determination on the *Surveyor IV* mission.

On several missions, where the trajectory made DSS 42 the initial acquisition station, the MSFN used the upper sideband station at Carnarvon, Australia, and was able to provide additional S-band telemetry, in real-time, before the DSS 42 view. In these cases the AFETR coverage from the RIS in the Indian ocean carried over into the Carnarvon view, giving continuous coverage from high power "on."

4. *Surveyor real-time data transmission and computation.* The quality of both real-time telemetry data retransmitted and the output of computed trajectory data from the RTCS depended primarily on the quality of input data received from the supporting stations. There

were no instances where data were lost or degraded after reaching the AFETR. However, inputs to TEL-4 and RTCS varied from mission to mission. In some cases transmission of the data from downrange was hampered by high frequency communication circuit outages. Operational procedures also gave difficulty, as did equipment problems on some occasions. However, all requirements on all missions were met with minor exceptions.

The data path between DSS 71 and the SFOF worked well except during the *Surveyor V* mission when TCP computer problems were encountered shortly before launch which made DSS 71 data unusable. The backup data path via 202 Dataphone from building AO to the SFOF was used, and no data loss resulted from the DSS 71 computer outage.

Surveyor VII real-time data transmission and computation is the best example of total support in this area. On this mission, the RTCS provided a total of eight orbits, including a parking orbit from Antigua data, a second and third parking orbit from *Centaur* guidance telemetry data, a theoretical transfer orbit using Antigua data plus nominal second burn, two actual preretro transfer orbits using Pretoria data, a postretro orbit using Carnarvon data and a spacecraft orbit from DSS 42 data. For real-time telemetry data transmission on *Surveyor VII*, the backup path for real-time data from building AO to SFOF using a 202 Dataphone was activated and provided good data, but was not used during flight because of the high quality of data via the prime DSS 71 path. In addition, two new satellite communications circuits from the Ascension island station of AFETR provided improved reliability of communications from downrange. These circuits were implemented by NASA in cooperation with AFETR. One voice-grade circuit extended from DSS 72 to JPL via GSFC. The 202 data sets were used to transmit 550 bits/sec *Surveyor* data direct to JPL from DSS 72 and stations farther down range. Good data were received in the SFOF TPS via this circuit; however, they were not processed because of the high quality of data coming in via the normal TEL-4 DSS 71 path. A second satellite circuit was used to transmit downrange metric teletype data from DSS 72 to Kennedy Space Center. This circuit was checked and provided good data. The system for transmission of real-time telemetry data from the Carnarvon, Australia MSFN station to SFOF via DSS 42 was also activated for *Surveyor VII*. The system performed well, as on previous missions, and good data were received in SFOF from Carnarvon acquisition until DSS 42 switched to processing its own data, a period of approximately 14 min.

4. Deep Space Phase

a. Requirements. The obligations placed upon the T&DS in the deep space phase of mission support rested primarily upon the DSN, that is the Deep Space Stations of the DSIF, the SFOF, and the GCF.

The DSIF requirements were:

(1) To receive, record, and deliver good metric data (either one-way, two-way, or three-way doppler) to the station communication terminal for transmission to the SFOF.

(2) To receive, record, and deliver good telemetry data to the station communication terminal for transmission to the SFOF.

(3) To provide two 7044–7094 computer combinations of DSS 11, to deliver video data to the station communication terminal for transmission to the SFOF.

(4) To deliver good command data from the station communication terminal to the spacecraft antenna terminal.

The GCF requirements were:

(1) To provide a good high-speed data communication circuit between the SFOF and each of the stations.

(2) To provide good voice communications circuits between the SFOF and the stations.

(3) To provide good teletype communications circuits between the SFOF and the stations.

(4) To provide good microwave communications between the SFOF and Goldstone stations.

The SFOF requirements were:

(1) To provide a telemetry processing station for high speed data processing.

(2) To provide a communications processor for teletype data processing.

(3) To provide two 7044–7094 computer combinations for telemetry and tracking data processing.

(4) To provide supporting services for project MOS staff to conduct the mission.

The deep space phase tracking and telemetry requirements are described in greater detail in Tables VII-5 and VII-6. It was required that commands could be sent to the spacecraft at any time from acquisition by a Deep Space Station to the end of the mission during times that the spacecraft was visible from DSSs 11, 42, 51, 61, and 72.

Table VII-5. Tracking (metric) requirements

Mission segment	Sampling rate	Data[a] required
Spacecraft separation to first midcourse maneuver	1 sample/min except, from initial DSIF acquisition to launch plus 1 hr, 1 sample per 10 sec	Doppler (two-way and three-way) and angle
First midcourse to touchdown	1 sample/min	Doppler (two-way and three-way)
Midcourse and terminal maneuvers executions	1 sample/sec and transmit data at 1 sample/10 sec	Doppler (two-way and three-way or one-way)
Touchdown to end of mission	1 min sample rate at: 1 hr after 10 deg elevation, 1 hr centered around max. elevation and 1 hr prior to 10 deg elevation at station set for DSSs 11, 42, 51, and 61	Doppler (two-way and three-way) and angle

[a]Inflight data presented by teletype page print and teletype print; postflight data presented by magnetic tape (FR 1400).

The extent to which the foregoing mission support was required was as follows:

Mission segment	Support requirements
Preflight testing	Facility usage and personnel
Transit phase	24 hr/day
First lunar day	24 hr/earth day
First lunar night	24 hr/earth day
Second lunar day	24 hr/earth day
Second lunar night	24 hr/earth day
Third lunar day	8 hr/earth day

The basic support plan required support as shown above; however, the requirements for support following touchdown were renegotiated during the missions.

Inherent in the requirements for tracking telemetry and command capability was a requirement for recording, data processing, and data display at all the tracking stations of the DSIF and at the central control point, the SFOF at JPL. The recording requirement at the DSIF included magnetic tape recordings of telemetry, video, and digital data as well as punched paper tape, and oscillograph strip chart records of analog data. Punched paper tapes of all commands transmitted to the spacecraft via the tracking stations were also required. In the SFOF, log tapes were required of all data that entered the input/output system, as well as a magnetic tape of all telemetry entering the TPS. Input/output devices in the various mission areas of the SFOF were required to communicate with the *Surveyor* computer programs in the DPS.

Table VII-6. Telemetry requirements

Mission segment	Subcarrier oscillator frequency (kHz)	Data rate (bits/sec)
Prelaunch to DSIF acquisition	3.9	550
DSIF acquisition to end of mission	70	2200
	33	4400
	7.35	1100
	5.4	550
	3.9	550
	0.960	137.5
	0.960	17.2

A complex and versatile communications system, of worldwide scope, was required to coordinate the activities of all the tracking stations and to transfer data between them and the SFOF. In addition, an equally complex internal communications network was required in the SFOF to coordinate the activities of all personnel involved in supporting a mission, to display essential data of many kinds as required, and to make certain selected data available to the Public Information Office for release.

Certain other requirements, such as use of facilities, working space, power, light, water, and photographic, reproduction, and calibration services, were incidental to the main requirements, but were accounted for in planning mission support.

b. Configuration

1. Deep space instrumentation facility. The following Deep Space Stations provided prime support to the *Surveyor* Project during transit phase and lunar operations. Each was equipped with an 85-ft parabolic reflector, maser amplifier, and full S-band tracking and communications system.

Station	Location
DSS 51	Johannesburg, Republic of S.A.
DSS 61	Robledo, Madrid DSCC, Spain
DSS 42	Tidbinbilla, Canberra DSCC, Aus.
DSS 11	Pioneer, Goldstone DSCC, Calif., U.S.A.

Other Deep Space Stations which provided special support during various *Surveyor* missions were:

Station	Location
DSS 71	Cape Kennedy, Fla., U.S.A.
DSS 72	Ascension Island, South Atlantic ocean
DSS 12	Echo, Goldstone DSCC, Calif., U.S.A.
DSS 14	Mars, Goldstone DSCC, Calif., U.S.A.

Near-earth support was provided by DSSs 71 and 72 with 4 and 30 ft antennas, respectively. Backup to DSS 11 during critical phases such as midcourse maneuvers was provided by DSS 14 with its 210-ft antenna.

The locations of DSSs 11, 42 and 61 and their coverages are shown in Fig. VII-22.

The block diagram for DSS 11 is shown in Fig. VII-23. This diagram is typical for all Deep Space Stations supporting *Surveyor,* except that other stations had neither a TV-GDHS nor a microwave system, and DSSs 51 and 72 do not have a full TCP. The spacecraft signal was received in phase-locked receivers working in conjunction with a low-noise maser amplifier. The receivers had two outputs, a wideband video output and a phase-detected telemetry output. The outputs of each receiver appeared at the receiver switch matrix, where the output of either receiver could be selected for input to the CDC and TV-GDHS.

The CDC supplied engineering, telemetry, command confirmation, and alpha scattering data to the TCP. Engineering data that had been decommutated in the CDC were edited and formatted for transmission to the SFOF, via high speed data lines and teletype. Selected engineering data channels were displayed locally. Command confirmation data were time tagged and formatted for TTY transmission. Alpha scattering data were decommutated within the TCP, and spectral totals were accumulated which were then transmitted via TTY to the SFOF.

For command purposes the uplink to the spacecraft was provided by a 10-kW transmitter. The transmitter exciter was phase modulated by a command subcarrier that was generated in the CDC, in accordance with TTY or voice instructions from the SFOF.

The spacecraft carried two S-band transponders (one was redundant) permitting the uplink and downlink radio paths to be coherent, although on different frequencies.

The two-way doppler data thus available at each tracking station, together with angle and frequency data, were returned to the SFOF by TTY where they were used for orbit determination purposes by FPAC.

2. *Space flight operations facility.* The block diagram for the SFOF is shown in Fig. VII-24. Telemetry data (other than alpha scattering data) entered the TPS over high speed data lines (digital form) or the 96-kHz line (analog form) from Goldstone. Data were recorded and provided as input to the bit synchronizer (the 96-kHz data first had to be run through a discriminator). The bit synchronizer output was run to a PDP-7 computer, and after decommutation and time-tagging, these data were recorded on digital tapes and provided as input to the 7044 computers. Selected data were converted by digital to analog converters for display on strip chart recorders in the user areas.

Teletype data (alpha scattering and tracking data) came into the SFOF communications processor and were passed to the 7044 computers via a wideband line.

The 7044 input/output computer accepted data from the TPS and the communications processor via its 7288 data subchannels. These data were logged on magnetic tape and placed on a disk file where they were accessible by the 7094 computer. The 7044 computer could convert these input data to engineering units and display them in real-time in any format desired. Displays included TTY output via the communications processor, high speed bulk printers, and plotters. Control of this computer and its outputs was via input/output consoles, administrative printers, and card readers located in the user areas.

User programs were run on the 7094 computer. These were lengthy programs for detailed analysis of spacecraft parameters and flight trajectories. The programs were controlled through the 7044 computer. Output from the 7094 computer was on magnetic tape which was processed on an off-line 7040 computer to generate listings, prints,

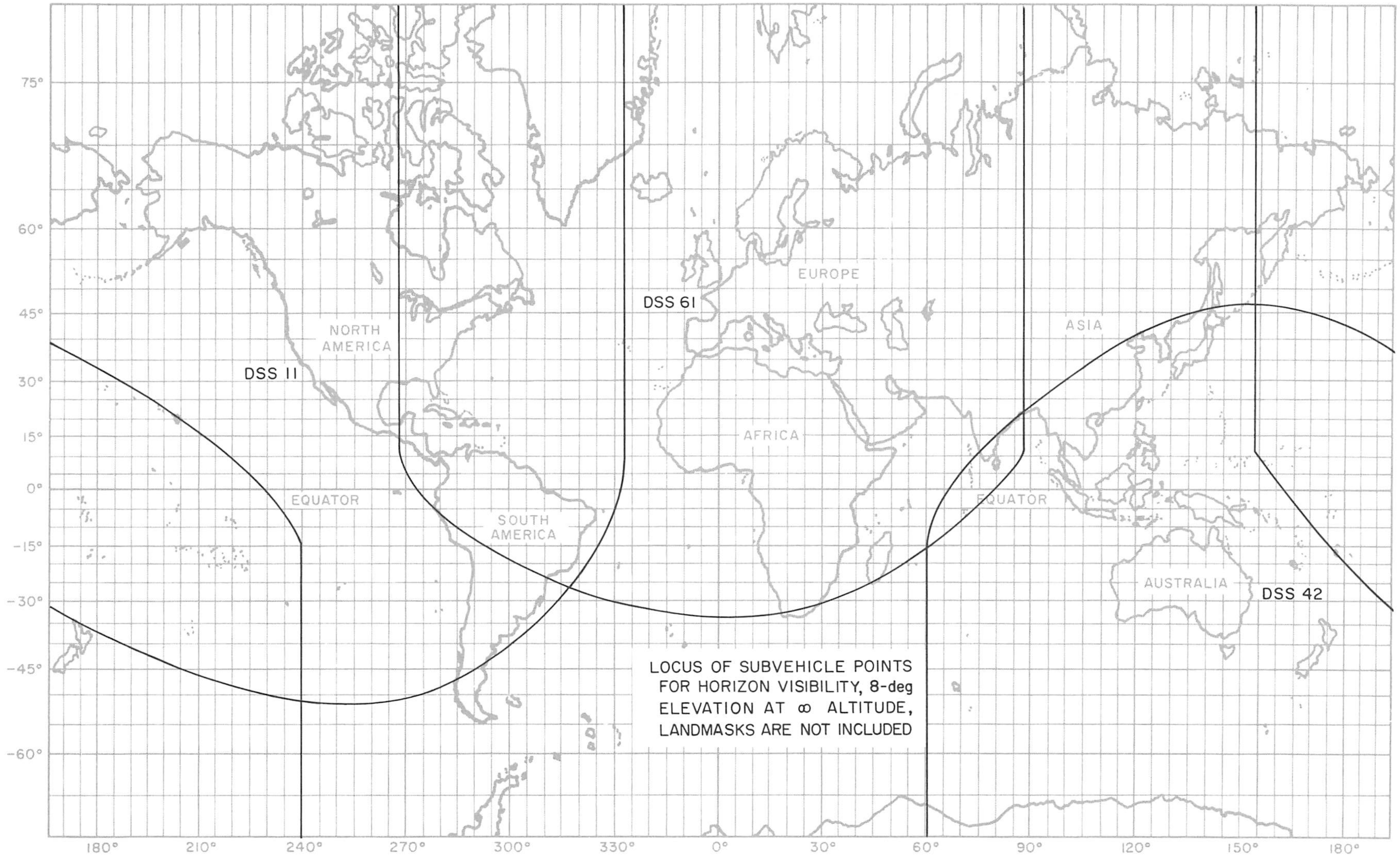

Fig. VII-22. Locations and coverage for DSSs 11, 42, and 61 and plots. The 7044–7094 computers communicated via single instructions over a direct data connection between the computers, but the bulk of data was passed via the disk files. Refer to Subsection A-2 for a description of the computer programs.

Certain types of data that required no processing also entered the SFOF. One of these was TTY command confirmation and average alarm data, which were merely routed by the communications processor to appropriate teletype page printers. Automatic gain control data from DSS 11 were transmitted directly to AGC recorders in the user areas.

3. *Ground communications facility.* The DSN/GCF part of the NASCOM supporting *Surveyor* was configured to provide real-time voice, teletype, high-speed data, and some video and analog data between DSN tracking stations and the SFOF, using portions of the NASA communications network shown in Fig. VII-25. It consisted, basically, of one voice line used for station control and commanding purposes, four TTY circuits used to carry various types of data, and one high-speed data circuit carrying spacecraft telemetry for each prime station.

In addition, DSS 11 was configured with one additional voice line, one AGC (analog) line, one wideband (96 kHz) channel for baseband data, and one wideband video channel (6 MHz) used for real-time video from the spacecraft.

Communications to DSS 72 consisted of one voice, one high-speed data, and two TTY circuits which were provided by satellite.

The network configuration described above was used during all missions with the exceptions that follow.

Teletype circuits. Surveyor I, II, III, and *IV* missions were supported by point-to-point TTY NASCOM/GCF circuits. *Surveyor V, VI,* and *VII* missions were supported using the communications processor systems through computers at Canberra, London, GSFC, and the SFOF.

High-speed data circuits. Additional high-speed data circuits carrying telemetry data from Carnarvon to DSS 42 were added for *Surveyors III, VI,* and *VII.* For *Surveyor VII* only, an additional high-speed data circuit and one TTY circuit were added from AFETR Station 12, at Ascension island. The high-speed circuit was routed to the SFOF via GSFC communications satellite circuit and carried downrange high-speed telemetry data. The TTY circuit was routed via GSFC communications satellite to building AO at Cape Kennedy and carried downrange metric data.

Wideband circuits. On the *Surveyor IV* mission, DSS 14 baseband telemetry data were transmitted through the DSS 11 CDC via the 96-kHz wideband channel to the SFOF during critical midcourse and terminal phases only. For *Surveyors V, VI,* and *VII,* the wideband 96-kHz channel was reconfigured from DSS 11 to DSS 14 during midcourse and terminal phases to allow DSS 14 data to move directly to TPS.

*c. **Testing.*** To ensure that all the various elements of the T&DS were ready to support an upcoming mission, a comprehensive program of equipment, personnel, and overall T&DS testing was developed. For each mission the same cycle of testing was repeated although, in later missions, the amount of testing was somewhat curtailed due to the increasing network activity and improved proficiency of the participating personnel. The purpose of the tests and their function in preparing the T&DS for supporting a *Surveyor* mission is described in the subsequent paragraphs.

1. *Configuration verification tests.* The configuration verification tests were performed by each Deep Space Station immediately prior to the station operational readiness testing phase, except on the *Surveyor V* mission as noted below. The tests were conducted in accordance with a well-documented test format and the test results were forwarded by teletypewriter exchange to the operations engineering project engineer at JPL for evaluation. This verified that station performance and configuration were in accordance with the network configuration document and the interface agreement. The tests included checkout of subsystem elements of each station as well as an overall system performance test. All forms of communications with the SFOF, including TTY, high-speed data, voice, and microwave communications were checked in representative data transfer tests. The CDC was also verified for operational status as a separate entity and as part of the overall tracking station system.

2. *System readiness verification tests.* For the *Surveyor V* mission, a system readiness verification test was introduced to verify that all elements of the DSN *Surveyor* configuration were ready to enter the operational readiness tests and flight support phases. This innovation was necessitated by the decision to reduce the amount of

Fig. VII-23. Block diagram of DSS 11 data handling

Fig. VII-24. Block diagram of SFOF data handling

HS	HIGH SPEED
HF	HIGH FREQUENCY
TAT–1	TRANSATLANTIC TELEPHONE CABLE
TAT–3	TRANSATLANTIC TELEPHONE CABLE
COMPAC	COMMONWEALTH TRANSPACIFIC CABLE
PT	PART TIME SERVICE
FDX	FULL DUPLEX
HDX	HALF DUPLEX
VOICE/DATA	ALTERNATE VOICE OR DATA WHICH CAN BE USED FOR VOICE, ANALOG DATA, OR HIGH-SPEED DATA. NOMINAL USE OF THE CIRCUIT IS UNDERLINED.

· — — - LANDLINE OR MICROWAVE FACILITIES

—·—·—·— SUBMARINE CABLE

············ HIGH-FREQUENCY RADIO CIRCUITS

JPL DSN STATION ID NUMBERS

AREA	0	1	2	3	4	5	6	7	8	9
1 GOLDSTONE		PIONEER	ECHO	VENUS	MARS					
2	COMMUNI-CATIONS	MCR No. 1	MCR No. 2	DSIF NET CONTROL						
4 AUSTRALIA		WOOMERA	CANBER							
5 SOUTH AFRICA		JOBURG								
6 EUROPE		SPAIN								
7 AFETR (AMR)		JPL LAUN								

NASCOM ROUTING INDICATORS

AREA	0	1	2	3	4	5	6	7	8	9
1 GOLDSTONE		JGLD	JGLD	JGLD	JGLD					
2 JPL	JJPL	JJPL	JJPL	JJPL					ETL	SAF
4 AUSTRALIA		AMOJ	AMBE							
5 SOUTH AFRICA		LJOB								LMTS
6 EUROPE										
7 AFETR (AMR)		LCAP								

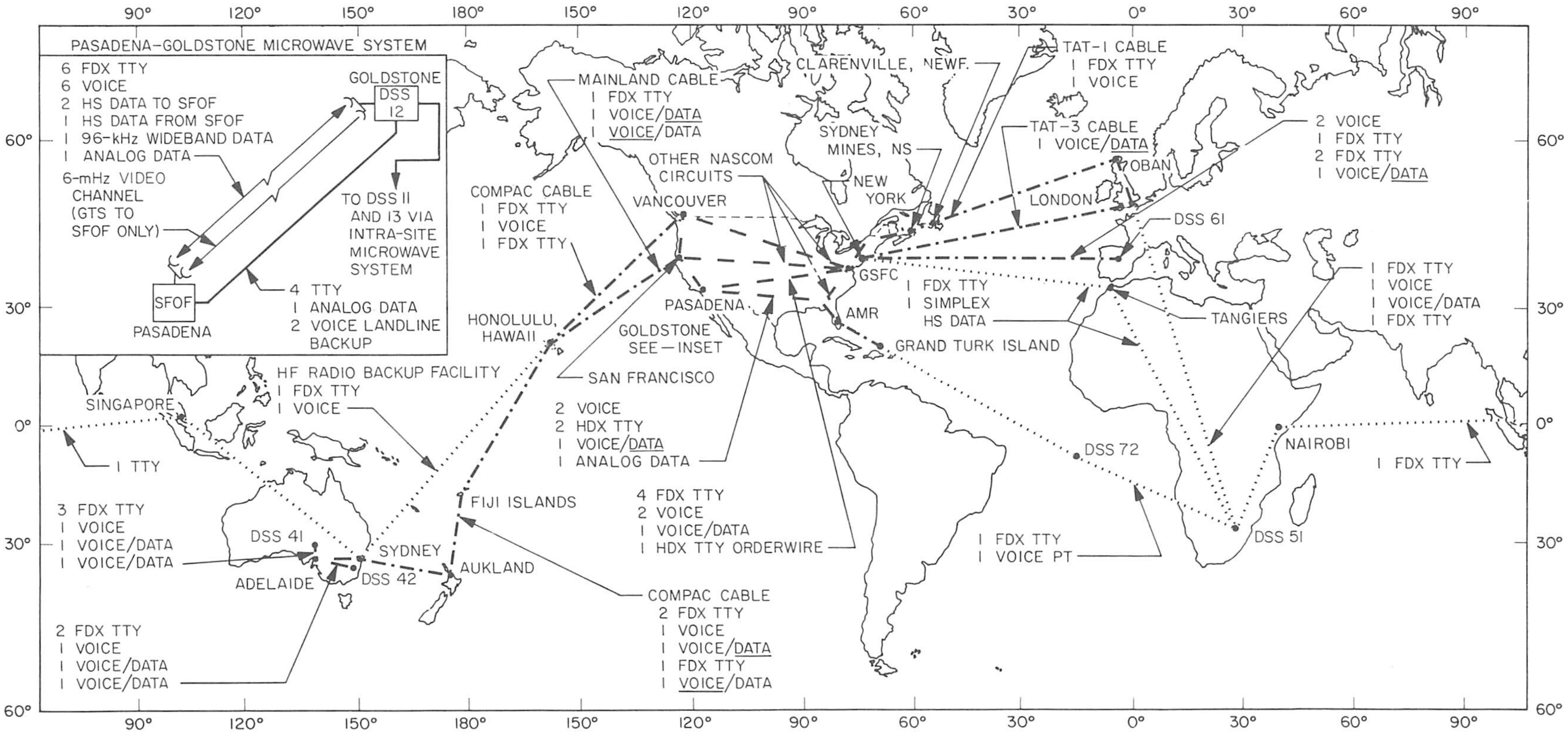

Fig. VII-25. The DSN ground communication network project operational testing to a single operational readiness test finishing only 36 hr before launch. Thirty-six hours was not considered sufficient time to solve any problems revealed by the operational readiness test.

Another highlight of *Surveyor V* prelaunch testing was the introduction of the communications processor/7044 computer redesign data processing system for operational *Surveyor* support, and the system readiness verification test was designed to validate the new interface thus introduced. These tests emphasized data transfer between each Deep Space Station and AFETR and included the essential operational elements of the SFOF including the communications processor. On *Surveyors VI* and *VII*, the system readiness verification test was not needed since these two missions were preceded by extensive second lunar day activity on previous *Surveyor* missions.

3. *Operational readiness tests.* The operational readiness tests were conducted shortly before launch operations to confirm readiness of a complete MOS to enter the mission support phase (refer to Subsection A-4). In earlier missions a simulation of the complete mission was conducted. In later missions the operational readiness tests were limited to the initial acquisition, midcourse maneuvers, terminal descent, and preliminary lunar operations phases of the mission. The DSN participated fully in the operational readiness tests not only to support the mission but also as a final checkout of all DSN elements. The support provided was the same as that provided for the actual mission with the addition of simulation data transmitted to the Deep Space Stations from the SFOF.

All elements of the MSFN and AFETR were involved in supporting the operational readiness tests. A typical flight azimuth was selected and simulated spacecraft telemetry data were provided in the form of magnetic tapes played back by the various downrange stations during the times when they would have view for the particular azimuth in use. Simulated tracking data from the MSFN and AFETR were received by the RTCS at Kennedy Space Center, where they were reformatted and retransmitted to the SFOF. The RTCS also used the data to calculate orbital elements and DSN prediction data and to make other computations used to define the orbit. These messages were then transmitted to building AO for retransmission to the SFOF.

4. *Initial acquisition tests.* Before the launch of *Surveyor I*, the DSN personnel expended considerable effort to develop a reliable procedure for making the initial acquisition of the spacecraft by the DSIF.

A series of tests was made at DSS 11, DSS 42, and DSS 51 to obtain experimental data on the best procedure for initial acquisition and typical times involved in carrying out the procedure. For this purpose a *Surveyor* S-band test transponder was installed in a helicopter and a flight plan was developed that simulated angular rates, view periods, RF power levels, and frequency offsets to be expected from the spacecraft in a real mission.

Although doppler rates could not be simulated, care was taken to see that each acquisition attempt was made independent of previous attempts, so that the operator had to undertake a frequency search to lock both downlink and uplink communications in each attempt.

Previously, an extensive series of aural acquisition tests had been made on a large group of receiver operators, to determine the minimum signal-to-noise ratio required for reliable aural acquisition of the zero beat note in the DSIF receiver under dynamic frequency conditions. These tests showed that for the rates expected in a mission (approximately 0.5 kHz) the signal-to-noise ratio in a 2-kHz bandwidth should be higher than −8-db. This condition was satisfied in the helicopter tests described above. The test program resulted in an acquisition procedure that allowed the acquiring station to proceed systematically from the first one-way RF contact on the acquisition aid antenna, to two-way auto tracking on the high-gain main-beam antenna with commanding capability, within the time constraints set by the spacecraft based on thermal and power considerations. The tests showed that time of 3–4 min were required to reach two-way auto track on the main antenna, after first visibility above the local horizon.

d. Mission performance. The deep space support for the seven *Surveyor* missions was provided per the standard configurations previously described. Support was provided for all operational tests, and continuous tracking and data processing coverage was provided by the DSIF from shortly after each launch until the end of the first lunar day activities or the end of the mission, whichever came first. Adequate support was provided for subsequent activities on succeeding lunar days.

There were some minor anomalies during the seven missions, but they caused no significant loss of data supplied to the project. In most cases these deficiencies were rectified or significantly reduced by the end of *Surveyor VII*. In particular, the quality of the magnetic tape recordings (especially video tapes) provided by the DSN was improved. In the early missions, these were unsatisfactory due to recording equipment problems and interface problems. These problems proved unusually difficult to correct, partly because they could not be readily detected in real-time. Considerable effort was devoted to developing a tape validation procedure, whereby tapes received at the SFOF were immediately subjected to a spot check and anomalies reported back to the stations concerned for corrective action. By the *Surveyor VI* and *VII* missions, the value of this system had been demonstrated, and tape recording problems had been reduced to an insignificant level.

1. Deep Space Instrumentation Facility. The Deep Space Stations which supported the *Surveyor* Project provided full and/or adequate support for the seven missions including prelaunch tests. Some station hardware problems occurred at various stations during their support of the missions; however, the problems were either minor in nature or covered by redundant equipment. The ability of the DSIF to cope with numerous ground system and spacecraft anomalies was demonstrated throughout the execution of the seven missions. Table VII-7 shows the support provided by the DSIF, in terms of total tracking system hours expended, number of commands sent, the number of pictures received, and the total number of telemetry system hours expended.

The tracking and telemetry hours expended far exceeded the actual mission duration, due to the considerable amount of redundancy employed by the DSIF to assure adequate reliability. For this reason, none of the down time shown in the tracking and telemetry sections had any significant effect on the mission.

2. Ground communications facility. The GCF provided all required support for the seven missions. Communication lines to each prime station included four teletype one voice, and one high-speed data, plus the wideband microwave circuits to and from Goldstone. In general, the GCF support for all the missions was excellent. Communications outages on circuits from DSS 51 and DSS 72, caused by radio propagation conditions, were a recurring problem. Other random circuit outages with other stations also occurred. Maximum support was always provided by NASCOM to restore the circuits as quickly as possible. The communication processors, which were implemented into the GCF before *Surveyor V*, provided adequate support even though numerous problems did occur. Special precautions to minimize the probability of communication processors problems were taken during critical operational periods. Table VII-8 shows the support provided by the GCF in terms of the percentages of total circuit hours expended for each type of service. Considerable redundancy was used to provide the high level of support shown.

3. Space Flight Operations Facility. The SFOF performance was excellent. The DPS used for *Surveyor I–IV*

Table VII-7. Summary of reported DSIF support for *Surveyor* missions[a]

Type of support	Surveyor						
	I	*II*	*III*	*IV*	*V*	*VI*	*VII*
Tracking							
Total time, hr	599	86	611	125	657	650	631
Down time, hr	25.1	0.6	5.4	1.3	1.2	3.3	6.0
Commands							
Sent correctly	86,177	1,542	57,107	1,232[b]	104,906	164,867	138,060
Sent incorrectly	1	1	2	Nil	5	36	1
Down time, hr	Nil	Nil	0.5	Nil	1.5	Nil	Nil
Video							
Pictures received by command	10,419	Nil	6,326	Nil	18,006	29,952	20,993
Pictures lost	Nil	Nil	Nil	Nil	98	34	5
Down time, hr	Nil	Nil	Nil	Nil	0.75	6.0	6.2
Telemetry							
Total time, hr	599	86	611	123	624	634	605
Down time, hr	19.4	Nil	0.2	Nil	6.4	Nil	0.2

[a]Includes through first lunar day operations only.
[b]Includes commands after loss of spacecraft signal.

Table VII-8. Summary of GCF support of *Surveyor* missions[a]

Station	Type of circuit	No. of circuits	Surveyor						
			I	II	III	IV	V	VI	VII
DSS 11	TTY	4	100.0	100.0	100	100	99.9	99.7	99.96
	Voice	2	100.0	100.0	100	100	99.9	99.9	100.00
	High-speed data/AGC	2	100.0	100.0	99.7	100	98.9	98.3	99.94
	Microwave (96kHz)	1	99.0	100.0	100	100	98.3	99.9	100.00
	Microwave (6MHz)	1	99.0	100.0	100	—	100	99.1	99.85
DSS 42	TTY	4	97.9	97.8	99.7	99.9	98.9	99.2	99.32
	Voice	1	100.0	100.0	99.8	100	100	99.9	99.82
	High-speed data	1	100.0	100.0	99.3	99.6	99.6	99.7	99.66
DSS 51	TTY	4	93.3	100.0	94.6	95.0	99.0	92.7	97.11
	Voice	1	86.3	94.8	93.5	94.0	99.6	88.8	95.86
	High-speed data	1	89.4	100.0	97.3	95.7	100	94.6	91.35
DSS 61	TTY	4	NA[b]	100.0	98.3	99.6	98.1	98.5	99.68
	Voice	1	NA	100.0	99.1	99.4	99.7	99.9	99.79
	High-speed data	1	NA	100.0	99.6	100	99.5	99.9	99.88
DSS 71	TTY	3	100.0	99.8	100	100	100	100	100.00
	Voice	1	100.0	100.0	100	100	100	100	100.00
	High-speed data	1	100.0	100.0	100	100	100	100	100.00
Building AO	TTY	3	99.7	99.8	93.1	99.2	100	100	100.00
	Voice	2	100.0	100.0	100	99.2	100	100	100.00
	High-speed data	1	100.0	100.0	100	100	100	100	100.00
DSS 72	TTY	4	NA	94.8	95	97	99.2	NA	NA
	Voice	1	NA	97.0	100	100	100	NA	NA
	High-speed data	1	NA	100.0	100	99	100	NA	NA

[a]Satisfactory performance is given in percent of total circuit time.
[b]Not applicable.

remained unchanged and performed well. Some minor hardware or software problems did occur but these problems did not detract from mission support. The 7044 redesigned communications processor system was first used for *Surveyor* flight support during *Surveyor V*. The DPS operated during this mission with higher reliability than in any previous *Surveyor* mission. During *Surveyor VI*, the 7044 computer was shared with *Mariner V* during lunar operations. A separate 7044 computer mission-dependent system was required for each of the spacecraft which required sharing a 7044 computer. A high degree of coordination was required between projects and the DSN operations control center scheduling office to obtain current information on 7044 computer share capabilities and requirements. No current 7044 computer system was available for sharing between *Surveyor VI* and *Mariner IV*, which caused difficulty in scheduling shared computer usage, but enough resources were available to meet requirements from the projects.

For *Surveyor VII*, the major change to the DPS was the reconfiguration of the 7044W computer and the 7094V computer into a new operational W combination. This combination was checked out before the operational readiness tests and was used for support during the flight. The DPS operations were quite reliable throughout the flight and lunar operations. Although some problems occurred during the missions, the DPS provided its committed support at all times.

The new SFOF display system was used to support *Surveyor VII*; several interesting and useful displays were developed to support real-time operations.

Before the launch of *Surveyor VII*, a configuration for patching the scan converter output throughout the SFOF was developed and checked out. This helped to eliminate the problems encountered during earlier missions when a large number of real-time requests for patches was made just after touchdown, creating confusion and degrading the scan converter output signal.

Table VII-9 shows the support provided by the SFOF in terms of total data processing machine time expended and the effecting mission outages.

C. Summary of Significant MOS/T&DS Technical Activities

The *Surveyor* Project served as the initial space project to have its telecommunications at S-band. When the project was initiated in 1960, those people responsible for the management of the frequency spectrum recommended that there be no further L-band projects such as *Ranger* and *Mariner* Venus 1962, but that future projects such as *Surveyor* should be planned at S-band. In 1960, the first flight of *Surveyor* was planned for the fall of 1963. When it became apparent that this objective could not be met (in fact, *Surveyor I* was not launched until May of 1966), the first project to actually use S-band in flight was *Mariner* Mars 1964 (Ref. VII-3).

The *Surveyor* Project provided the first spacecraft design that was almost wholly dependent on ground station commands from the DSN for its in-flight activities.

Table VII-9. Summary of SFOF support for Surveyor missions, hr

	Surveyor						
	I	II	III	IV	V	VI	VII
Total mission duration	336	43	413	62	373	415	464
IBM 7044 computer							
Total usage	470	88	497	106	449	463	732
Effective outage	3.75	Nil	Nil	Nil	0.2	2.5	1.75
IBM 7094 computer							
Total usage	367	70	500	106	443	374	491
Effective outage	3.75	Nil	Nil	Nil	2.2	1.1	1.1
Telemetry processing station							
Mission operations	564	68	496	173	396	432	618
Other services	93	Nil	175	Nil	132	149	179
Effective outage	3.75	Nil	Nil	Nil	Nil	0.75	0.1
Support services							
Total services	414	45	413	62	503	393	464
Generator power usage	85	66	101	87	95	87	96
Generator power outage	3.75	0.3	Nil	Nil	Nil	Nil	1.6

It was significant that there was not much redundancy in the network for commanding at any single station. Thus, time-critical command activities were backed up by having on-line other stations rather than having redundancy, such as two transmitter chains at each station. An example of the latter case is the MSFN configuration for *Apollo*.

The *Surveyor* Project was the first deep space project to make extensive use of high-bit rates requiring extensive use of high-speed data lines from the Deep Space Stations to the SFOF, as a primary mode for the conduct of spaceflight operations. A considerable amount of equipment was reconfigured and experience gained in operations at bit rates up to 4400 bits/sec from Goldstone and up to 1100 bits/sec from the overseas stations through the ground communication system into data processing systems in the SFOF. Before this project, teletype circuits served as the primary means for transferring data and for the conduct of spacecraft flight operations.

The *Surveyor* Project was the first deep space project to make extensive use of real-time high-speed (greater than 50 bits/sec) data processing in the 7044-7094 computer system in the SFOF. This required, at times, two such combinations to be fully operational for extended periods, especially during the transit flight to the lunar surface. The support for this requirement was successfully provided, requiring, however, considerable man-hours of hardware and software development and complex operational activities.

The *Surveyor* Project philosophy of highly centralized control of space flight operations required high reliability of communications circuits from the Deep Space Stations to the SFOF. This reliability was achieved by providing considerable redundancy culminating in the first operational use of a communication satellite over the Atlantic ocean for deep space data acquisition.

The development of the *Surveyor* spacecraft required use of equipment and facilities for compatibility testing culminating in an extensive series of tests at Goldstone with the T-21 flight configuration test vehicle. The availability of these facilities and the T-21 spacecraft also provided a basis for extensive training of network personnel in the acquisition problems and varied communications procedures required by the design of the *Surveyor* spacecraft.

The T-21 test vehicle was used at Goldstone test facilities to provide operational training for the Space

Flight Operations personnel and supporting personnel of the DSN.

The *Surveyor* Project was one of the first to use DSS 71 at Kennedy Space Center. This station, located in the vicinity of the launch pad and the checkout facilities, was used in the final tests of compatibility between the flight spacecraft and the DSN.

The *Surveyor* Project made first use of DSS 72 on Ascension island for near-earth telemetry coverage, for tracking for early orbit determination, and for filling coverage gaps between other deep space stations.

The *Surveyor* Project provided mission-dependent equipment at each of the Deep Space Stations for the functions of sending commands and processing telemetry and video data from the spacecraft. This equipment underwent considerable compatibility testing where it interfaced mission-independent or network equipment. Initially, it was a source of many interface problems not only in the hardware area, but in documentation, operations, and procedures. Because of the extensive use of mission-dependent equipment throughout the T&DS, procedures were developed and documented for the interface structure, configuration control, and operations of both the mission-independent and mission-dependent elements of the project. These highly developed procedures have been applied to the support of all subsequent flight projects.

The *Surveyor* Project provided the personnel to maintain and operate the mission-dependent equipment on-site pending the training and transfer of responsibility to the on-site personnel. The project also provided on-site personnel for spacecraft control and data analysis should there be at any time a catastrophic failure in communications between the Deep Space Stations and the SFOF in Pasadena.

The DSN provided the support of its highest-performance station, DSS 14, to provide better signal-noise magnetic tape recordings of the strain gage telemetry measurements during spacecraft touchdown. The higher performance capability of this station was also used to return 4400 bits/sec telemetry during trajectory correction maneuver of the spacecraft. The *Surveyor* spacecraft radiating from the moon provided an excellent far-field source for the 210-ft antenna pattern measurement.

Surveyor Project provided the first opportunity of demonstrating the capability to perform automatic data quality comparison in real-time on telemetry data received simultaneously from two Deep Space Stations. This was accomplished by the telemetry processing system using two PDP-7 computers in the SFOF.

Surveyor Project provided, for the first time at JPL, go-no-go spacecraft subsystem status indicators for use by the SPAC group.

Surveyor Project pioneered the use of computers in the DSN to generate and verify commands in real-time for transmission to the spacecraft.

Surveyor Project was the first flight project wherein video information was received, retransmitted, processed, and displayed in real-time for operational decision making.

Surveyor Project was the first to provide real-time recordings on film of video data transmitted from the moon.

Surveyor Project used the full capabilities of the 6-MHz wideband microwave line from DSS 11 to the SFOF.

Surveyor Project and *Mariner* Venus 67 were the first projects to utilize a computer string and the JPL communications processor in the multiproject usage mode.

Surveyor Project, during the terminal descent phase of the missions, provided in the SFOF real-time plotting and display of spacecraft altitude vs time-to-touchdown.

The SFOFs new real-time displays in the mission control room were first used by *Surveyor VII*.

Surveyor Project was the first space flight project that required the real-time transmission of spacecraft telemetry data from the near-earth phase network (i.e., from the AFETR ship and land stations and in the later missions from the MSFN stations at Carnarvon to DSS 42). It developed closer working relationships among the three supporting networks. This integration was achieved to a degree never before required for the support of a space flight program.

Surveyor Project was the first to use real-time simulation of maneuvers with coordinated tracking and telemetry data, video simulation, and the use of high-speed telemetry simulation data to overseas stations in real-time by using the outgoing side of the high-speed data line to each of the Deep Space Stations.

During the last few *Surveyor* missions, the computer associated hardware and software that was used for Spacecraft System testing was also used for providing data to the spacecraft performance analysis area. Presentation of data in the same formats was useful in monitoring the specific characteristics of each spacecraft as established during system test.

The data from the Spacecraft System test computer utilized the first commercial wideband hardware communications line in southern California.

Surveyor Project provided the first occasion for decommutation of PCM telemetry data using a combination of hardware and software. This function was accomplished in the telemetry processing station of the SFOF and permitted automatic recognition of spacecraft data mode change in the SFOF when high-speed data from DSS TCP was being processed.

During the last two missions, *Surveyor* first used a cathode ray tube device for the SFOF by the space flight operations personnel. This device displayed spacecraft data in plots, frame dumps, command logs, and spacecraft status after processing by the Spacecraft System test computer. Up to 500 different formats were selectable under the control of the users.

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VIII. Launch Vehicle System and Interface with Spacecraft

The *Atlas/Centaur* vehicle, this nation's first liquid hydrogen/liquid oxygen high-energy space booster system, was developed and first utilized operationally for the *Surveyor* missions, and required the application of new propellant technology. It established a remarkable record of achievement in its first application by successfully launching and accurately injecting into lunar transfer orbit each of the seven *Surveyor* spacecraft. Since development of the *Atlas/Centaur* was conducted in parallel with *Surveyor* spacecraft development, problems encountered in vehicle development had a direct influence on the overall *Surveyor* Program. The flight tests conducted in the *Centaur* research and development program fully qualified the vehicle for *Surveyor* missions (including demonstration of a second-stage, two-burn capability for parking-orbit missions). The launch vehicle program included the development of new test and launch facilities, new ground support equipment, and special spacecraft interface systems.

A. *Atlas/Centaur* Vehicle Development

A chronological history of the *Centaur* Project through the launch of the first developmental vehicle (AC-1) is given in Table VIII-1.

Table VIII-2 provides the significant test objectives, configuration features, and results of each *Centaur* flight. The AC-6 flight on August 11, 1965, successfully demonstrated *Centaur* operational readiness for *Surveyor* direct-ascent missions. The AC-9 flight on October 26, 1966, successfully demonstrated two-burn indirect-ascent capability. Satisfactory performance of the improved *Atlas* SLV-3C was successfully demonstrated during its initial usage as the first stage of the AC-13 vehicle on the *Surveyor V* mission.

1. Early *Centaur* Project History

In 1956, General Dynamics/Convair (GD/C) performed analytical studies for the United States Air Force (USAF) relative to launching high-altitude satellites for early warning of missiles, surveillance, communications, and similar missions. As a result of these studies, a GD/C proposal was presented to the USAF in December 1957 for the development of a launch vehicle which would use a new pressure-fed *Centaur* second stage using liquid hydrogen (LH_2) and liquid oxygen (LO_2). A modified *Atlas* missile was to serve as the first stage. The design of the second stage featured an integral tank-structure system similar to that already developed for the *Atlas*, with jettisonable insulation panels for the liquid-fuel tanks.

Table VIII-1. Early milestones of the *Centaur* project

Date	Event
8/28/58	*Centaur* Project originated by ARPA[a]
10/17/58	Contract awarded to Pratt & Whitney Aircraft by ARDC[b] for development of LH_2/LO_2[c] 15,000-lb-thrust engine
11/19/58	Contract awarded to GD/C[d] by ARDC for 6 upper-stage vehicles and 1 battleship tank (*Centaur*)
5/15/59	Contract awarded to GD/C by ARDC for 9 sets of guidance equipment
6/30/59	Responsibility for *Centaur* development transferred by ARPA to NASA
7/28/59	Contract awarded for construction blockhouse, Launch Complex 36, AFETR[e]
9/29/59	Contract awarded for construction of Complex 36 launch platform and service building
6/30/60	*Centaur* Project assigned to Marshall Space Flight Center by NASA
11/6/60	First testing of dual liquid-H_2/O_2 engine (LR-115) at Pratt & Whitney Aircraft
11/15/60	Checkout started on first *Centaur stage* (C-1) at San Diego
2/1/61	*Atlas* booster 104D delivered to AFETR
3/1/61	*Centaur* stage C-2 delivered to AFETR
10/31/61	*Centaur* stage C-1 delivered to AFETR
2/9/62	First hot firing of a *Centaur* test stage at San Diego
5/8/62	First development flight, *Atlas* 104D/*Centaur* C-1 (AC-1)
5/15/62	Congressional and NASA review of *Centaur* Program initiated
10/8/62	*Centaur* Project transferred from Marshall Space Flight Center to Lewis Research Center

[a]Advanced Research Projects Agency.
[b]Air Research and Development Command.
[c]Liquid hydrogen and liquid oxygen.
[d]General Dynamics/Convair.
[e]Air Force Eastern Test Range.

Table VIII-2. *Centaur* research and development test summary

Launch date	Launch vehicle (*Atlas/Centaur* serial numbers)	Payload	Ascent mode	Flight test objectives	Configuration features	Results
5/8/62	AC-1 (104D/C-1)	None	*Centaur* suborbital coast for demonstration of engine restart	Demonstrate *Atlas/Centaur* structural integrity and separation, nose fairing and insulation panel jettison, propellant management and propulsion system performance (including restart), flight control, and obtain flight data on all systems	*Centaur* tank cylindrical section skin gage 0.011 in. aft, 0.010 in. forward, spot- and seam-welded Zero-*g* separator in liquid hydrogen (LH_2) tank antislosh baffles in liquid oxygen (LO_2) tank, propellant utilization system installed Jettisonable short nose fairing and lightweight jettisonable insulation panels No ground chilldown of RL10A-1 engines, steering closed loop by *Centaur* guidance system	*Centaur* structural failure 54 sec after liftoff resulted in total vehicle destruction Experience and data obtained on vehicle assembly cryogenic propellant, handling, and launch *Atlas* performance satisfactory to flight termination

Table VIII-2 (contd)

Launch date	Launch vehicle (*Atlas/Centaur* serial numbers)	Payload	Ascent mode	Flight test objectives	Configuration features	Results
11/27/63	AC-2 (126D/2B)	None	Direct	Demonstrate *Atlas/Centaur* structural integrity and separation, *Centaur* propulsion system start and burn to programmed duration, and performance of the flight control system Obtain data on guidance system accuracy, performance of hydrogen peroxide (H_2O_2) systems, and launch-on-time capability, fixed azimuth	*Centaur* tank cylindrical section skin gage 0.016 in.,[a] fusion-welded, cap and gores chem-milled Nonjettisonable heavyweight insulation panels, short nose fairing, and heavyweight slosh baffles in LO_2 tank; no LH_2 zero-*g* separator or propellant utilization system liquid-helium chilldown of RL10A-3 engines[b] Open-loop steering with commands from autopilot to programmed vehicle attitude Electronic programmer[c] 1.5- and 3-lb attitude control and 50-lb vernier/ullage engines Zero-thrust T-section duct downstream from LH_2 vent valve	Successful; all flight objectives were met
6/30/64	AC-3 (135D/1C)	None	Direct	In addition to objectives listed for AC-2, demonstrate: structural and thermal integrity, guidance system integrity, and jettisoning of the nose fairing and insulation panels; obtain data on feasibility of boost pump operation after main-engine cutoff without burp pressurization of propellant tanks	Jettisonable *Surveyor*-type (long) nose fairing, lightweight jettisonable insulation panels, heavyweight slosh baffles in LO_2 tank; no zero-*g* separator or propellant utilization system Guidance steering and *Centaur* engines same as AC-2 Four 1.5-lb and two 3-lb attitude control engines, and four 50-lb vernier/ullage engines Zero-thrust T-section duct downstream from LH_2 vent valve	Partially successful; *Centaur* hydraulic pump shaft failed, causing premature engine shutdown at main-engine-start + 154 sec All flight objectives met except full-duration burn time, a boost pump experiment, and guidance system integrity (which were partially satisfied)

[a]Cylindrical tank skin gage, AC-2 through AC-5: 0.016 in., AC-6 and on: 0.014 in.
[b]Liquid-helium chilldown, AC-2 and on.
[c]Electronic programmer, AC-2 through AC-4.

Table VIII-2 (contd)

Launch date	Launch vehicle (*Atlas/Centaur* serial numbers)	Payload	Ascent mode	Flight test objectives	Configuration features	Results
12-11-64	AC-4 (146D/4C)	Mass model (2070 lb)	Parking orbit	In addition to objectives listed for AC-3 (except feasibility of boost pump operations and programmed main-engine cutoff), demonstrate: guidance system ability to provide proper discretes and steering signals to flight control, proper propellant control by ullage and attitude control engines, *Centaur* propulsion system burn to guidance cutoff; obtain data on retromaneuver capability	Jettisonable redesigned *Surveyor*-type (long) nose fairing,[d] lightweight insulation panels with Thermalag on fairing and aft portions of insulation panels,[e] and airborne helium purge[f] of insulation panels. Heavyweight LO_2 tank slosh baffles Zero-thrust, baffled container downstream from H_2 vent valve Vernier engines removed and two 2-lb ullage engines added Other flight control engines same as AC-3 Guidance steering closed-loop during sustainer and *Centaur* phases, programmer-stored fixed vector for retromaneuver	*Partially* successful All flight objectives met except demonstration of propellant control by ullage and attitude control engines and demonstration of *Centaur* engine restart Retromaneuver capability only partially demonstrated
3/2/65	AC-5 (156D/6C)	*Surveyor* dynamic model (1411 lb)	Direct	Demonstrate operational readiness of *Centaur* for direct-ascent mission, including payload separation, through verification of the system structural and thermal integrity, guidance integrity, and system operational objectives of previous vehicles Demonstrate retromaneuver capability and electromechanical-timer performance on a single-burn mission	Lightweight jettisonable insulation panels and redesigned lightweight LO_2 tank slosh baffles. Zero-thrust baffled container downstream from H_2 vent valve Two 3-lb and four 1.5-lb attitude control engines; no vernier or ullage engines Payload separation system[g] Steering same as AC-4. Electromechanical timer,[h] retrovector supplied by guidance[i]	Unsuccessful *Atlas* fuel prevalve closed at approximately 0.88 sec after liftoff, resulting in premature booster engine cutoff Vehicle was totally destroyed after settling back into launcher and rupture of propellant tanks

[d]Redesigned *Surveyor*-type nose fairing (with thermal bulkhead), AC-4 and on.
[e]*Thermalag* on nose fairing and insulation panels, AC-4 and on.
[f]Provisions for airborne helium purge, AC-4 and on. Inoperable, AC-5 and on.
[g]Spacecraft separation system, AC-5 and on.
[h]Electromechanical timer, AC-5 and on.
[i]Guidance supplied retrovector, AC-5 and on.

Table VIII-2 (contd)

Launch date	Launch vehicle (*Atlas/Centaur* serial numbers)	Payload	Ascent mode	Flight test objectives	Configuration features	Results
				Obtain data on propellant utilization system performance, launch-on-time capability (variable azimuth) and injection accuracy by postinjection Deep Space Instrument Facility (DSIF) tracking of the *Surveyor* S-band transponder	Liquidometer and continuous capacitance probe propellant utilization system[j] *Atlas* booster engine system thrust uprated from 309,000 to 330,000 lb[k]	
8/11/65	AC-6 (151D/2D)	*Surveyor* dynamic model (2100 lb)	Direct	Same as AC-5	Lightweight jettisonable insulation panels modified to accommodate new, smaller LO_2 tank without slosh baffles[l] Steering, retromaneuver, and H_2 vent same as AC-5 No vernier or ullage engines. Attitude control engines same as AC-5 except 1.5-lb engine nozzles canted 25 deg off vehicle centerline *Atlas* booster and sustainer fuel prevalve actuators with hand valves mechanically locked in open position	Successful; all flight objectives met Theoretical aim point miss distance of 430 km would have required a 4.25-m/sec spacecraft midcourse correction Liftoff only 4 sec late; specific impulse I_{sp}: 434 sec; 1.8 deg/sec rotation rate imparted to spacecraft at separation: 1614 km spacecraft *Centaur* separation distance 5 hr after separation
4/7/66	AC-8 (184D/6D)	*Surveyor* mass model (1730 lb)	Parking orbit	Demonstrate operational readiness of *Centaur* for parking-orbit missions Objectives relating to performance of structural, thermal, and other systems (including engine restart) basically same as previous flights	Propellant management features including diffusers for energy dissipation at propellant return and helium pressurization line tank inlets, LH_2 tank slosh baffles, and balanced-thrust H_2 vent with exits on *Y-Y* axis	Partially successful Premature termination of *Centaur* second burn after engine ignition due to loss of H_2O_2 (hydrogen peroxide) fuel through attitude control engine manifold leak causing starvation of H_2O_2 driven boost pumps and H_2O_2 engine system

[j]Continuous capacitance probe, AC-5, -6, and -8.
[k]330,000-lb *Atlas* thrust, AC-5 and on.
[l]Liquid-oxygen slosh baffles removed from AC-6 and on.

Table VIII-2 (contd)

Launch date	Launch vehicle (*Atlas/Centaur* serial numbers)	Payload	Ascent mode	Flight test objectives	Configuration features	Results
				Demonstrate operation of dual electromechanical timer on a two-burn mission and *Atlas/Centaur* capability for launch-on-time, fixed azimuth	Two electromechanical timers. *Centaur* main engine I_{sp} increased from 433 to 444 sec by improved injector design, increased nozzle expansion ratio, and increased chamber pressure H_2O_2 engine system consisting of: four 3.5-lb engines canted 25 deg and two 6-lb engines for attitude control, four 50-lb vernier engines for propellant settling, and four 3-lb ullage engines for propellant retention and attitude control during coast	Short *Atlas* sustainer burn and high LO_2 residual Normal *Centaur* first burn and 1500-sec coast Spacecraft separation was accomplished
10/26/66	AC-9 (174D/8D)	*Surveyor* mass model (1740 lb)	Parking orbit	Same as AC-8 plus demonstration of H_2O_2 system performance and *Centaur* programmed supplemental booster steering to minimize structural loading; launch-on-time capability demonstration for a variable azimuth	Same as AC-8 plus addition of supplemental pitch and yaw programs in *Centaur* navigation computer Nonsegmented propellant utilization probes	Successful; all flight objectives met Theoretical aim point miss distance of 1012 km would have required a 6.54-m/sec spacecraft midcourse correction (miss only)

In the proposal, GD/C had chosen a pressure-fed propulsion system to preclude an expensive long-term cryogenic-pump development program. Shortly after presenting the proposal, however, GD/C became aware of the availability of a Pratt & Whitney Aircraft Co. (P&W) LH_2 pump and resubmitted a revised proposal for using a pump-fed propulsion system for the *Centaur* vehicle.

In conjunction with the newly formed Advanced Research Projects Agency (ARPA) of the Department of Defense who had assumed responsibility for cognizance of the *Centaur* vehicle and had reviewed the revised proposal, another revision was made by GD/C and a basic proposal agreement was reached in August 1958. This basic proposal agreement called for an LH_2/LO_2 propulsion system using two 15,000-lb-thrust engines with integral turbopumps which were to be government-furnished.

By October 1958, the Air Research and Development Center (ARDC) of the USAF received an order from ARPA for the development of this *Centaur* high-energy upper stage. In November, ARDC awarded GD/C a contract for six upper-stage flight test vehicles and one heavyweight tank structure for ground development testing. The first flight test was scheduled for January 1961, some 26 months after the contract was awarded. The contract stipulated that the development was to be done on a noninterference basis with the *Atlas* weapon system program. To conserve funds and expedite the program, readily available hardware and existing *Atlas* tooling and technology were used wherever possible.

During the same period, ARDC awarded a contract to P&W for the development of the *Centaur* engines with integral turbopumps.

The basic GD/C contract did not: (1) include the development of a guidance system, (2) specify a mission, or (3) specify performance requirements. The contract purpose was to demonstrate the feasibility of an LH_2/LO_2 propulsion system. In December 1958, ARPA specified a primary mission for *Centaur,* which was to place a satellite in a 24-hr equatorial orbit. This mission called for a 6-hr parking orbit and a three-burn capability. With a capability to perform under these stringent requirements, the *Centaur* could certainly be applied to other less demanding missions such as space probes and low earth satellites.

In May 1959, ARPA awarded GD/C a contract for guidance equipment to complete the contractual arrangements for developing the *Centaur.* On June 30, 1959, ARPA transferred the *Centaur* Project to the newly formed National Aeronautics and Space Administration (NASA). The original USAF contracts and *Centaur* Project Manager were retained, but operating under NASA direction through the USAF contract administrator.

In December 1959, the *Centaur* primary mission definition was reestablished as the *Advent* 24-hr communication-satellite launch. Shortly thereafter, the *Centaur* vehicle was considered for launching the newly defined *Mariner* and *Surveyor* spacecraft on NASA lunar and planetary exploration missions.

The *Surveyor* spacecraft was defined initially to weigh 2500 lb (including 200 lb of scientific instruments), and was to be soft-landed on the moon. There were to be seven spacecraft launched during the period 1963–66. Due to the unavailability of any other launch vehicle with sufficient payload capability, the *Centaur* was established for use with the *Surveyor* missions. The *Centaur* development program, as originally conceived, called for six flight tests at one-month intervals between January and July 1961.

In June 1960, the responsibility for the *Centaur* Project was delegated by NASA to Marshall Space Flight Center (MSFC) with retention of the USAF technical and contract-management teams. USAF participation in the *Centaur* development ended (except for inspection functions) in January 1962, when the *Centaur* Project Office was transferred from the Los Angeles Air Force Space Systems Division (AFSSD) to MSFC at Huntsville, Ala. The USAF contracts were converted to NASA contracts for administration by MSFC.

2. First Development Flight

The first flight test of a *Centaur* vehicle (F-1, later designated AC-1, for *Atlas/Centaur* flight 1) was conducted May 8, 1962. At approximately 54 sec after liftoff, in the region of maximum air turbulence, the nose fairing weather shield failed, which resulted in LH_2 tank rupture, buckling of the *Centaur,* and total vehicle destruction. Although the flight was unsuccessful, it provided a basis for redesign of vehicle hardware and improvement in the test program and procedures. Also, valuable experience was gained in performing *Atlas/Centaur* prelaunch operations.

After the AC-1 flight, an extensive evaluation was made of the entire *Centaur* program jointly by U.S. House of Representatives science and astronautics subcommittee, NASA/MSFC, and the various contractors. The evaluation resulted in a decision to continue the program with extensive contractual, technical, and organizational changes (Refs. VIII-1 and VIII-2).

In October 1962, NASA transferred the *Centaur* Project management responsibility from MSFC to its Lewis Research Center (LeRC), Cleveland, Ohio. LeRC organized a *Centaur* Project Office, whose first function was to perform a comprehensive technical and contractual review. The review verified the basic *Centaur* system approach; however, a number of requirements for design changes, ground tests, special fabrication innovations and techniques, development of new prelaunch procedures, and reorganization of the GD/C *Centaur* Project were initiated. In addition, LeRC redirected the program to develop both a single-burn, direct-ascent *Centaur* vehicle and a more difficult two-burn vehicle, which would have the capability to coast in parking orbit for up to 25 min.

Completion of the *Centaur* development program review by LeRC resulted in the immediate implementation of a number of changes and additions to hardware and evaluational testing. Significant changes, with implementation required before the next development flight (AC-2), were:

(1) Prelaunch chilldown of the LH_2 turbopumps to minimize *Atlas* postseparation chilldown requirements prior to *Centaur* engine start.

(2) Instrumented boilerplate nonjettisonable insulation panels and nose fairing (for AC-2 flight only) to provide design data for future flights.

(3) Addition of a second hydrogen vent valve to insure adequate venting capacity.

(4) Extended hydrogen vent fin to prevent hydrogen from entering the vehicle boundary layer and possibly igniting.

(5) Increased LH_2-tank skin thickness and fortified nickel-welding techniques for adequate structural integrity.

(6) Replacement of the AC-1 latch system for *Atlas/Centaur* interstage adapter separation with a GD/C-developed flexible linear shaped charge.

(7) Larger retrorockets on the *Atlas* for faster separation from *Centaur.*

(8) Redesign of the weather shield that failed on AC-1.

(9) Additional design verification; functional and acceptance testing of components, subsystems, and the total vehicle system.

3. Later Development Flights

The launch of the AC-2 vehicle on November 27, 1963, marked the first in-flight firing of the P&W RL10A-3 LH_2/LO_2 rocket and the first successful *Atlas/Centaur* test flight. Liftoff occurred about 243.5 min later than planned due to additional checks, ground-support hardware problems, weather, and extra time required for LO_2 topping of the *Atlas.* The flight verified the basic *Atlas/Centaur* launch vehicle system and provided a substantial quantity of data for design improvement, including changes to the insulation panel system. Panel redesign and test efforts were accelerated to have a satisfactory jettison system in time for the next test flight. As a result of a problem of insulation panels freezing to the LH_2 tank during jettison tests at GD/C, the helium purge gap between the insulation panels and the LH_2 tank was doubled and provision for in-flight purging was added.

The first attempt to launch the AC-3 vehicle on June 26, 1964 was aborted at $T-120$ min in the countdown due to a guidance problem. The vehicle was successfully launched on June 30 and the flight was normal through nose-fairing and insulation-panel jettison, *Atlas/Centaur* separation, and the *Centaur* main engine start (MES) sequence. At MES$+5$ sec, the *Centaur* engine lost hydraulic system pressure due to a broken pump shaft; the resultant vehicle roll rate caused liquid vortexing and subsequent loss of the LO_2 supply to the boost pump. Although terminated at about MES$+253$ sec, the flight was considered partially successful and the single-burn *Centaur* was considered operational, except for flight demonstrations of the spacecraft separation system and the *Centaur* retromaneuver.

The primary objective of the AC-4 vehicle was to launch on a fixed azimuth of 105 deg, roll to 102.5 deg and inject the *Centaur* and nonseparable 2100-lb *Surveyor* mass model into a 90-nmi circular parking orbit. After a coast phase of approximately 24.5 min, an additional experiment to demonstrate *Centaur* engine restart was to be accomplished.

Performance of all launch complex and vehicle systems was normal through the *Centaur* first burn and until fuel tank venting during the coast phase. A forward shift in the *Centaur* center of gravity had occurred at main engine cutoff (MECO), implying that the propellants were not settling. At fuel tank venting (MECO + 267 sec) the vehicle received a yaw torquing moment greater than the control recovery capability of the attitude control system. Continued venting of a mixture of liquid and gaseous hydrogen increased the tumbling rate and eventually forced the propellant to the ends of the tank, resulting in boost pump starvation and no engine restart. Significant accomplishments of the AC-4 flight were the satisfactory performance of the closed-loop *Centaur* guidance system and the use of optical alignment during countdown for maintaining gimbal block azimuth orientation.

After a textbook countdown, the AC-5 vehicle was launched on March 2, 1965, only 4.2 sec later than planned. The operation of the *Atlas* propulsion system was normal throughout start, thrust buildup, launcher release, and the 2-in.-motion portion of liftoff. Between 0.737 and 0.954 sec after 2-in. motion, the inlet pressures to the booster engine fuel pumps decreased from 66 to 6 psia, indicating a fuel prevalve closure after liftoff. This resulted in a loss of *Atlas* thrust. The vehicle settled back onto the launcher, rupturing the tanks and causing total vehicle destruction to occur at about $L+3.26$ sec. The launcher was destroyed and Launch Complex 36A received extensive damage. Since the fuel prevalve is used for ground test only, a hand valve, mechanically locked in the open position, was installed for future missions to preclude recurrence of the prevalve closure.

The first attempt to launch the AC-6 vehicle on August 10, 1965, was aborted due to inability to arm the range safety command unit during the countdown. The countdown for the second attempt (August 11) was according to plan, with no unscheduled holds; liftoff (2-in. motion) was accomplished within 4.5 sec of the planned time. The flight was completely successful and essentially concluded the *Centaur* research and development phase for the single-burn, direct-ascent *Surveyor* missions. The abort on August 10, provided a bonus demonstration of the vehicle's ability to be recycled from a fully tanked condition and launched within 24 hr.

4. Final Development Flights

Following the AC-4 flight test, several major modifications were incorporated in the AC-8 vehicle (Table VIII-2) to improve propellant management. After long factory and combined systems test cycles, setup of the AC-8 vehicle was completed at AFETR Launch Complex 36B on January 31, 1966.

Following two aborted launch attempts due to failure of the *Centaur* aft umbilical to eject properly and excessive winds aloft, the vehicle was finally launched April 7, 1966. The flight progressed satisfactorily through a 25-min coast phase between the first and second *Centaur* main engine firings, which successfully demonstrated liquid propellant management under near-zero-gravity conditions. Premature termination of the flight occurred shortly after second *Centaur* engine ignition due to loss of H_2O_2 propellant supply, which caused starvation of the boost pumps and attitude control engine system. Successful restart of the main engines was not achieved and vehicle stabilization was lost. Although separation of the mass-model payload occurred, the payload tumble rate was an excessive 24 deg/sec.

Postflight analysis indicated the H_2O_2 was lost through a leak in the H_2O_2 manifold at one of the attitude control engines. The manifold connection was redesigned to prevent this failure on following flights.

The last scheduled vehicle (AC-9) in the *Centaur* research and development program successfully demonstrated the operational readiness of the *Atlas/Centaur* to perform *Surveyor* parking-orbit missions. The AC-9 launch window simulated the conditions of a *Surveyor* mission, including a variable launch azimuth. The first launch attempt, on October 25, 1966, was aborted at $T-5$ min due to propulsion system instrumentation difficulties.

The second launch attempt, on October 26, 1966, was successful and resulted in satisfactory *Atlas* and *Centaur* performance through *Centaur* retromaneuver. Programmed steering commands from the navigation computer of the *Centaur* stage were used for the first time during the boost phase of flight to supplement the *Atlas* fixed pitch and yaw program. These trimming signals were employed to minimize vehicle structural loading from winds aloft. The first *Centaur* burn period was approximately 5 sec longer than expected due to early *Atlas* sustainer engine cutoff (SECO). However, the coast-orbit perigee altitude was 86.1 nmi and the apogee was 90 nmi compared to the planned circular orbit of 90 nmi, indicating excellent *Centaur* first-burn guidance and propulsion system performance. The coast period was 1457 sec and propellant management was accomplished very satisfactorily. *Centaur* second burn was terminated by a guidance discrete after 106.27 sec of burn time. Separation of the *Surveyor* mass model was nominal, and the resultant net payload tumble rate was about 0.64 deg/sec.

5. *Centaur* Payload Capability History

A history of the *Atlas/Centaur* payload capability and spacecraft weight for a direct-ascent mission (using the spacecraft and launch vehicle scheduled for the first *Surveyor* mission as an example) is shown in Fig. VIII-1.

When HAC was selected by NASA as the *Surveyor* Spacecraft System contractor on January 19, 1961, the spacecraft injection weight was constrained to a maximum of 2500 lb, based on an expected *Atlas/Centaur* payload capability at that time of 2518 lb.

By March 1962, as a result of an MSFC assessment of expected launch vehicle performance, the allowable spacecraft injection weight was reduced to 2100 lb for the first five *Surveyors* and was left at 2500 lb for the last two.

After the failure of the first *Centaur* development flight (AC-1) on May 8, 1962, and subsequent *Centaur* program reviews by MSFC and LeRC, and guidance/performance analyses by Space Technology Laboratory (STL), the payload capability of the launch vehicle was estimated as 2100 lb for direct-ascent missions and 2160 lb for parking-orbit missions. In January 1963, after transfer of the *Centaur* Project to LeRC, NASA directed that all seven *Surveyors* would be limited to a 2100-lb injection weight.

By August 1963 (and prior to the AC-2 development flight), changes initiated by LeRC to improve the payload capability of the launch vehicle (see Subsection A-2) had resulted in a 2340-lb payload capability estimate for

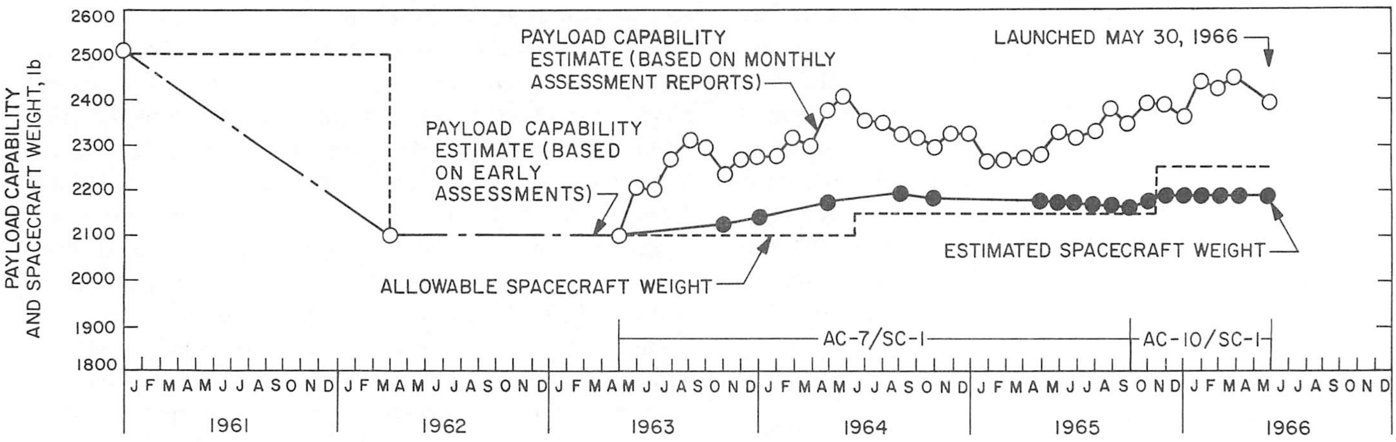


Fig. VIII-1. ***Atlas/Centaur*** **payload capability history (direct-ascent—*****Surveyor I*****)**

AC-7. At that time AC-7 was designated as the launch vehicle for the first *Surveyor* launch. Confidence that the launch vehicle could provide the payload capability necessary for the 2100-lb spacecraft was enhanced by the successful AC-2 flight.

A number of additional performance improvements were planned by LeRC, including uprating the performance of the *Centaur* main engines through improved nozzle design and the utilization of chem-milling in propellant tank manufacturing. These improvements resulted in a predicted 2417-lb payload capability for AC-7 by March 1964. Also, a payload capability of 2341 lb was predicted for *Atlas/Centaur* parking-orbit missions.

Since only a partial confirmation of *Centaur* performance was obtained during the AC-3 flight of June 30, 1964, and planned performance improvements were not fully implemented, the expected AC-7 launch vehicle payload capability declined after March 1964, and reached a low of about 2260 lb by February 1965. During this period, the allowable *Surveyor* spacecraft injection weight was increased to 2150 lb.

By March 1965, efforts by the performance and trajectory working group to improve accuracy and confidence in performance reporting, analytical technique improvements by GD/C and STL, and an LeRC weight saving program all combined to establish with high confidence a payload capability of 2272 lb for the AC-7 direct-ascent vehicle and 2412 lb for the AC-12 parking-orbit vehicle. During 1965, redirection was given to HAC that the first four *Surveyors* would be weight-limited to 2250 lb and the last three spacecraft would be limited to 2450 lb.

Implementation of improvements continued and AC-10 was launched with a payload capability of 2395 lb including allowance for a 3-sigma flight performance reserve of 175 lb. (The launch vehicle for *Surveyor I* was changed from AC-7 to AC-10 in October 1965. The actual injected weight of *Surveyor I* was 2193 lb, leaving a payload capability margin of 202 lb.)

In January 1966, GD/C was contractually directed to implement the use of the *Atlas* SLV-3C for the last three *Surveyor* missions. This change resulted in an approximate 200-lb gain in payload capability. The successful AC-9 development flight demonstrated the operational readiness of the *Centaur* in a parking-orbit ascent configuration, and the last three *Surveyor* missions were flown utilizing this ascent mode.

A summary of the launch vehicle payload capability, spacecraft injection weight, and payload capability margin for each of the *Surveyor* missions is given in Table VIII-3.

The payload capability improvement program was conducted by LeRC and GD/C throughout the *Centaur* and *Surveyor* programs. Some of the more significant items which contributed to the 600-lb gain in payload capability between March 1963 and the end of the *Surveyor* Program are presented in Table VIII-4.

B. Launch Vehicle System

The first stage of the *Atlas/Centaur* vehicle used for the seven *Surveyor* missions was a modified version of the *Atlas* used on many previous Air Force and NASA missions such as *Ranger* and *Mariner*. The LV-3C model was used for the first four *Surveyor* missions; the *Surveyor V* mission was the first *Atlas/Centaur* flight to

Table VIII-3. Summary of launch vehicle payload capability for *Surveyor* missions

Identification	Description						
Launch vehicle	AC-10	AC-7	AC-12	AC-11	AC-13	AC-14	AC-15
Spacecraft	SC-1	SC-2	SC-3	SC-4	SC-5	SC-6	SC-7
Launch vehicle payload capability,[a] lb	2395	2361	2413	2299	2618[d]	2615	2704
3 σ performance reserve,[b] lb	175	175	195	235[c]	255[e]	255	255
Spacecraft injected weight, lb	2193	2204	2281	2295	2217	2220	2289
Payload capability margin,[a] lb	202	157	132	4	401	395	415

[a]Per specified ground rules.
[b]Indicated 3 σ performance reserve was allowed for in determination of launch vehicle payload capability.
[c]Increase from performance reserve of previous direct-ascent mission (AC-7) due to addition of 60-lb contingency.
[d]Approximate 200 lb of the payload capability increase from previous parking-orbit mission (AC-12) due to introduction of *Atlas* SLV-3C.
[e]Increase from performance reserve of previous parking-orbit mission (AC-12) due to addition of 60-lb contingency.

Table VIII-4. Major *Atlas/Centaur* payload-capability improvement items

Item	Effectivity
Chem-milling utilized in tank manufacturing process	AC-2
Atlas booster engine thrust uprated from 309,000 to 330,000 lb	AC-5
Propellant utilization system improved and *Centaur* propellant loading system improved	AC-5
Reduced *Centaur* tank skin thickness, lightweight thrust barrel and forward bulkhead incorporated	AC-6
Centaur LO_2 tank slosh baffles removed	AC-6
Centaur LO_2 tank size reduced	
Nominal rating of *Centaur* main engine I_{sp} increased from 433 to 444 sec	AC-8
SLV-3C *Atlas* utilized	AC-13
Reduction in *Centaur* retromaneuver propellant blowdown time from 981 to 250 sec	AC-10

use the new *Atlas* SLV-3C, which had a 51-in. longer propellant tank section and increased thrust compared with the LV-3C. The increased length permitted tanking of approximately 20,000 lb of additional propellants. The *Atlas/Centaur* SLV-3C vehicle (with the *Surveyor* spacecraft encapsulated in the nose fairing) was 117 ft long and weighed about 325,000 lb at liftoff (2-in. rise).

The *Centaur* second stage, developed for the *Surveyor* Program, was first flown in an operational mission when it successfully injected *Surveyor I* into a lunar intercept trajectory on May 30, 1966.

The basic diameter of the *Atlas/Centaur* launch vehicles was a constant 10 ft from the aft end to the base of the conical section of the nose fairing. The configuration of the completely assembled vehicle (*Atlas* SLV-3C) is illustrated in Fig. VIII-2. Both the *Atlas* first stage and *Centaur* second stage utilized pressurized thin-wall, stainless-steel main propellant tank sections of monocoque construction to provide primary structural integrity and support for all vehicle systems. The first and second stages were joined by an interstage adapter section of conventional sheet and stringer design. The clamshell nose fairing was constructed of laminated fiber glass over a core of honeycombed fiber glass; it was attached to the forward end of the *Centaur* cylindrical tank section.

A description of each of the stages is presented in the following paragraphs. The functional sequence of events during the launch phase is discussed in Subsection F-2.

1. *Atlas* Description

Significant differences between the *Atlas* LV-3C and SLV-3C are noted in Table VIII-5. The *Atlas* LV-3C utilized the Rocketdyne MA-5 propulsion system, which burned RP-1 kerosene and LO_2 in each of its five thrust chambers.

The LV-3C model produced a total liftoff thrust of approximately 387,000 lb. The individual sea-level thrust ratings of the engines were: (1) two booster thrust chambers at 165,000 lb each; (2) one sustainer engine at 57,000 lb; and (3) two vernier engines at 670 lb each.

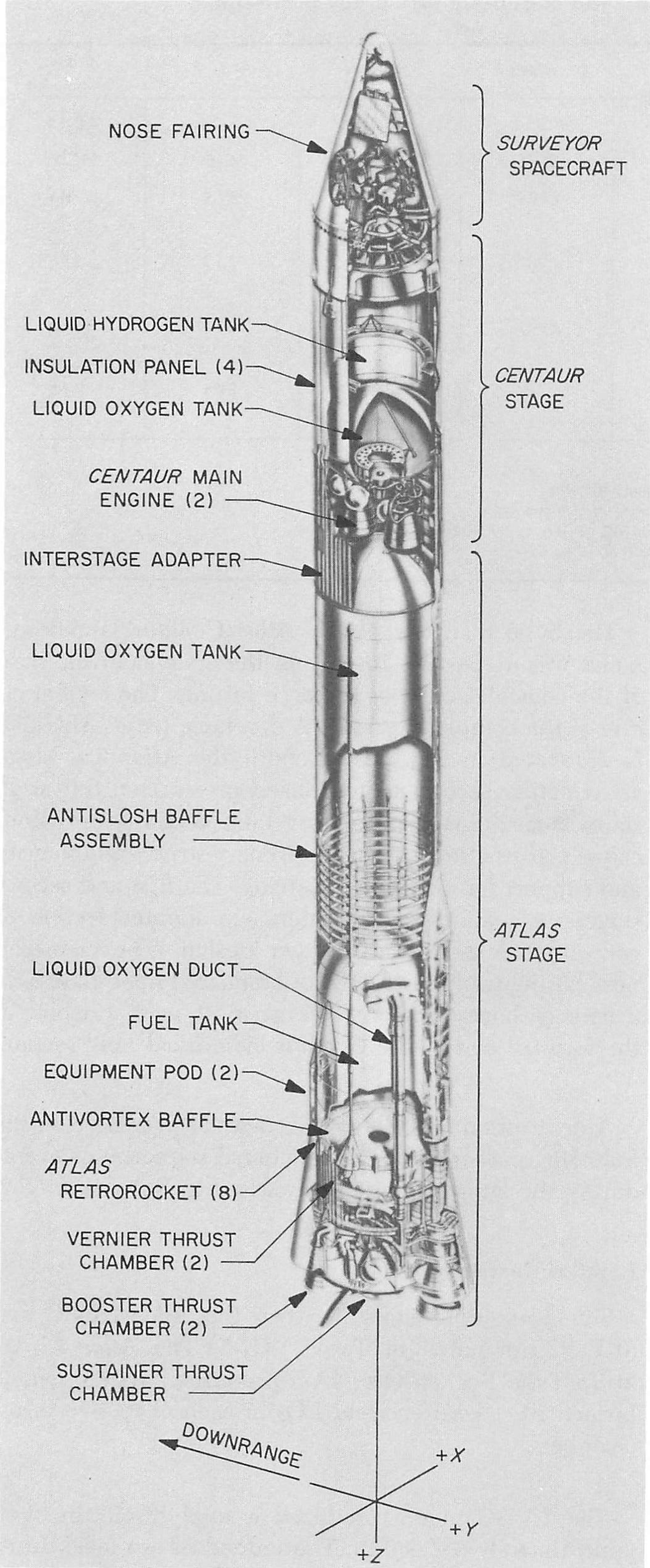


Fig. VIII-2. *Atlas* (SLV-3C)/*Centaur*/*Surveyor* space vehicle configuration

Table VIII-5. Differences between *Atlas* LV-3C and SLV-3C models

Item	Modifications incorporated in SLV-3C
Propellant tanks	Total length of *Atlas* tank section was increased 51 in. to provide 20,000 lb additional propellant capacity. Gross space vehicle weight (including spacecraft) increased from 303,000 to 325,000 lb
Engine thrust	Rated sea-level thrust was increased as follows (by raising the propellant regulator reference pressure to achieve increased propellant flowrates): Booster thrust chambers from 165,000 to 168,000 lb each; Sustainer engine from 57,000 to 58,000 lb; Total liftoff thrust from 387,000 to 395,000 lb
Booster engine firing-duration capability	Increased to 167 sec by increasing the usable capacity of the booster lube oil tank from 23 to 25 gal
Helium supply	Number of helium tanks was increased from 6 to 8 to provide additional pressurization gas for the larger propellant tanks
Propellant utilization system	System inhibited for first 13 sec of flight because manometers were not lengthened for new tanks and no longer extended to top of tanks
Autopilot	Rate gyro package mounting location was moved approximately 16 in. further forward for consistency with the dynamics of the longer vehicle structure. Appropriate gain changes were also made in the rate and displacement gyro amplifiers to accommodate the revised vehicle dynamic characteristics
Guidance	Insulation panel jettison command changed from BECO+30 sec to BECO+45 sec and nose fairing jettison command from BECO +56 sec to BECO+75 sec
Telemetry system	A second telemetry package, together with a second telemetry battery, was added to AC-13 and AC-14 to provide an increase in the number of measurements telemetered for evaluation of the SLV-3C
Electrical harnesses and equipment arrangement	Electrical harnesses were redesigned to eliminate splices and for standardization to facilitate installation of mission-peculiar equipment modules. Some equipment was also relocated for improved arrangement and for compatibility with the tank section extension

The *Atlas* SLV-3C was equipped with an uprated Rocketdyne MA-5 propulsion system, which provided a total liftoff thrust of approximately 395,000 lb. The individual sea-level thrust ratings of the engines were: (1) two booster thrust chambers at 168,000 lb each;

(2) one sustainer engine at 58,000 lb; and (3) two vernier engines at 670 lb each.

The *Atlas* was considered a 1½-stage vehicle because the booster section was jettisoned shortly after liftoff; this section weighed 6000 lb and consisted of the two booster thrust chambers together with a single booster turbopump for each propellant, with other equipment located in the aft section. Jettison took place about 2.4 min after liftoff for the LV-3C and 2.6 min after liftoff for the SLV-3C. The sustainer and vernier engines continued to burn until propellant depletion. A mercury manometer propellant utilization (PU) system was used to control mixture ratio for the purpose of minimizing propellant residuals at *Atlas* burnout.

Flight control of the first stage was accomplished by the *Atlas* autopilot, which contained displacement gyros for attitude reference, rate gyros for response damping, and a programmer to control flight sequencing until *Atlas/Centaur* separation. After booster jettison, the *Atlas* autopilot was fed steering commands from the all-inertial guidance set located in the *Centaur* stage. Vehicle attitude and steering control was achieved by the coordinated gimbaling of the five thrust chambers (2 booster, 1 sustainer, and 2 vernier) in response to autopilot signals.

The *Atlas* contained a VHF telemetry system which transmitted data well beyond *Atlas/Centaur* separation. A single telemetry package was utilized which operated on a frequency of 229.9 MHz (except for the *Surveyors V* and *VI* spacecraft, which also used a second telemetry package operating on a frequency of 232.4 MHz). For the telemetry system, two antennas were mounted on opposite sides of the vehicle at the forward ends of the equipment pods.

Redundant range safety command receivers and a single destructor unit were employed on the *Atlas* to provide the range safety officer with means of terminating the flight by initiating engine cutoff and destroying the vehicle. The *Atlas* destruct system was inactive after normal *Atlas/Centaur* staging occurred.

2. *Centaur* Description

The *Centaur* second stage was the first vehicle to utilize LH_2/LO_2 high-specific-impulse propellants. The cryogenic propellants required special insulation to be used for the forward, aft, and intermediate bulkheads as well as the cylindrical walls of the tanks. The cylindrical tank section was thermally insulated by four jettisonable insulation panels having built-in fairings to accommodate antennas, conduits, and other tank protrusions. Most of the *Centaur* electronic equipment packages were mounted on the forward tank bulkhead in a compartment which was air-conditioned prior to liftoff.

The *Centaur* was powered by two P&W constant-thrust engines, each rated at 15,000 lb thrust at vacuum conditions. Each engine could be gimbaled to provide control in pitch, yaw, and roll. Propellant was fed from each of the tanks to the engines by boost pumps driven by hydrogen peroxide turbines. In addition, each engine contained integral bootstrap turbopumps driven by hydrogen propellant. Hydrogen propellant was also used for regenerative cooling of the thrust chambers. Model RL10A-3-3 main engines were used to power the *Centaur* stage for the *Surveyor* parking-orbit flights. These engines were improved over the RL10A-3-1 models used on direct-ascent flights by providing for operation at lower NPSH (net positive section head) and by increasing the specific-impulse rating from 433 to 444 sec.

A PU system was used on the *Centaur* stage to achieve minimum residue of one propellant at depletion of the other. The system controlled the mixture ratio valves as a continuous function of propellant in the tanks by means of capacitive-type tank probes and an error ratio detector. The typical oxygen/hydrogen mixture ratio was 5:1 by weight. Special design features were incorporated in the hydrogen tank design for parking-orbit missions to ensure propellant control during the coast phase. These included: (1) an antiswirl/antislosh baffle located at the hydrogen level at the end of first burn, (2) diffusers for energy dissipation at the tank inlets of propellant return and helium pressurization lines, and (3) special ducting to provide balanced thrust venting of the hydrogen tank.

The second stage utilized a Honeywell all-inertial guidance system containing a navigation computer which provided vehicle steering commands after jettison of the *Atlas* booster section. Except for AC-10 and AC-7, a provision for windshear relief during the *Atlas* booster phase was added, wherein the navigation computer could also be used to generate incremental corrective pitch and yaw signals to supplement the *Atlas* fixed pitch and yaw program. The incremental pitch and yaw programs were selected and fed into the navigational computer before launch, based on predicted wind conditions. The *Centaur* guidance signals were fed to the *Atlas* autopilot until *Atlas* SECO and to the *Centaur* autopilot after *Centaur* MES. During flight, platform gyro drifts were compensated for analytically by the guidance system computer rather than by applying corrective gyro torquing signals. Subminiature rate gyros with higher response were used for the first time on the AC-13 flight for pitch, yaw, and roll control.

The *Centaur* autopilot system provided the primary control functions required for vehicle stabilization during powered flight, execution of guidance system steering commands, and attitude orientation during parking-orbit coast and following the powered phase of flight. In addition, the autopilot system employed an electromechanical timer to control the sequence of programmed events during the *Centaur* phase of flight, including a series of commands required on a parking-orbit mission.

The *Centaur* attitude control system provided thrust to control the vehicle during parking-orbit coast and after powered flight. For small corrections in yaw, pitch, and roll attitude control, the system utilized six individually controlled, fixed-axis, constant-thrust, hydrogen peroxide reaction engines. These engines were mounted in clusters of three, 180 deg apart, near the periphery of the main propellant tanks just aft of the interstage adapter separation plane. Each cluster contained one 6-lb-thrust engine for pitch control and two 3.5-lb-thrust engines for yaw and roll control. In addition, four 50-lb-thrust and four 3-lb-thrust hydrogen peroxide engines were installed on the aft bulkhead, with thrust axes parallel with the vehicle axis. (The 3-lb-thrust engines were not installed for the direct-ascent missions.) These engines were used to provide axial acceleration for propellant control during parking-orbit coast, to achieve initial separation of the *Centaur* from the spacecraft prior to retromaneuver blowdown, and for executing larger attitude corrections when necessary.

The *Centaur* stage utilized a VHF telemetry system with a single antenna transmitting through the nose fairing on a frequency of 225.7 MHz. The telemetry system provided data from transducers located throughout the second stage and spacecraft interface area as well as a spacecraft composite signal from the spacecraft central signal processor.

Redundant range safety command receivers were employed on the *Centaur*, together with shaped-charge destruct units for the second stage and spacecraft. This provided the range safety officer with means to terminate the flight by initiating *Centaur* MECO and destroying the vehicle and the spacecraft retrorocket. The system could be safed by a ground command, which was normally transmitted by the range safety officer when the vehicle reached orbital energy. (See Subsection C-3-b.)

A waiver, obtained from the AFETR safety office, permitted elimination of the inadvertent-separation system, which had been designed to provide for the automatic destruction of the *Centaur* and the *Surveyor* spacecraft in the event of premature spacecraft separation. The inadvertent-separation system was not flown on *Surveyor* missions.

The *Centaur* contained a C-band tracking system which included a lightweight transponder, circulator, power divider, and two antennas located under the insulation panels. The C-band radar transponder provided real-time position and velocity data for the range safety instantaneous impact predictor as well as data for early orbit determination and postflight guidance and trajectory analysis.

Significant differences between the *Centaur* stages used on direct-ascent and parking-orbit missions are detailed in Table VIII-6.

Table VIII-6. *Centaur* configuration comparison between direct-ascent and parking-orbit missions

Item	Modification for parking-orbit missions
Main engine specific impulse I_{sp}	I_{sp} upped from 433 to 444 sec by improving propellant injector design and increasing nozzle expansion ratio through a reduced throat area, an increased nozzle exit area, and an increase in chamber pressure from 300 to 400 psi
Hydrogen tank	Antiswirl/antislosh baffle located at LH_2 level at end of second burn Diffusers added for energy dissipation at tank inlets of propellant return and helium pressurization lines Special ducting added to provide balanced thrust venting of hydrogen tank
Programmer	A second timer added to provide for additional programmer events required for two-burn (parking-orbit) missions
Hydrogen peroxide engines	Four 3-lb-thrust engines added ot provide axial acceleration for propellant retention during parking-orbit coast
Pressurization system helium supply	Helium supply for in-flight insulation panel purge and LO_2/LH_2 tank pressurization increased from 4650 in.^{3} at 3000 psi to 7365 in.^{3} at 3300 psi

C. Launch Vehicle/Spacecraft Interface

1. Interface Description

The general arrangement of the *Surveyor/Centaur* interface is illustrated in Fig. VIII-3. The spacecraft was completely encapsulated within a nose fairing/adapter system in the final assembly bay of the explosive-safe facility at AFETR prior to being moved to the launch pad both for joint flight-acceptance composite testing (J-FACT) and for launch. This encapsulation provided protection for the spacecraft from the environment before launch as well as from aerodynamic loads and heating during ascent. An ablative coating (Thermalag) was applied aver the nose fairing and *Centaur* insulation panels to provide added thermal protection.

The spacecraft was first attached to the forward section of a two-piece conical adapter system of aluminum sheet and stringer design by means of three latch mechanisms, each containing a dual-squib pinpuller. The following equipment was located on the forward adapter: three separation spring assemblies, each containing a linear potentiometer for monitoring separation; a 52-pin electrical connector with a pyrotechnic separation mechanism; three pedestals for the spacecraft-mounted separation sensing and arming devices; a shaped-charge destruct assembly directed toward the spacecraft retro-rocket; a diaphragm to provide a thermal seal and prevent contamination from passing to the spacecraft compartment from the *Centaur* forward equipment compartment; and accelerometers for monitoring vibration at the separation plane.

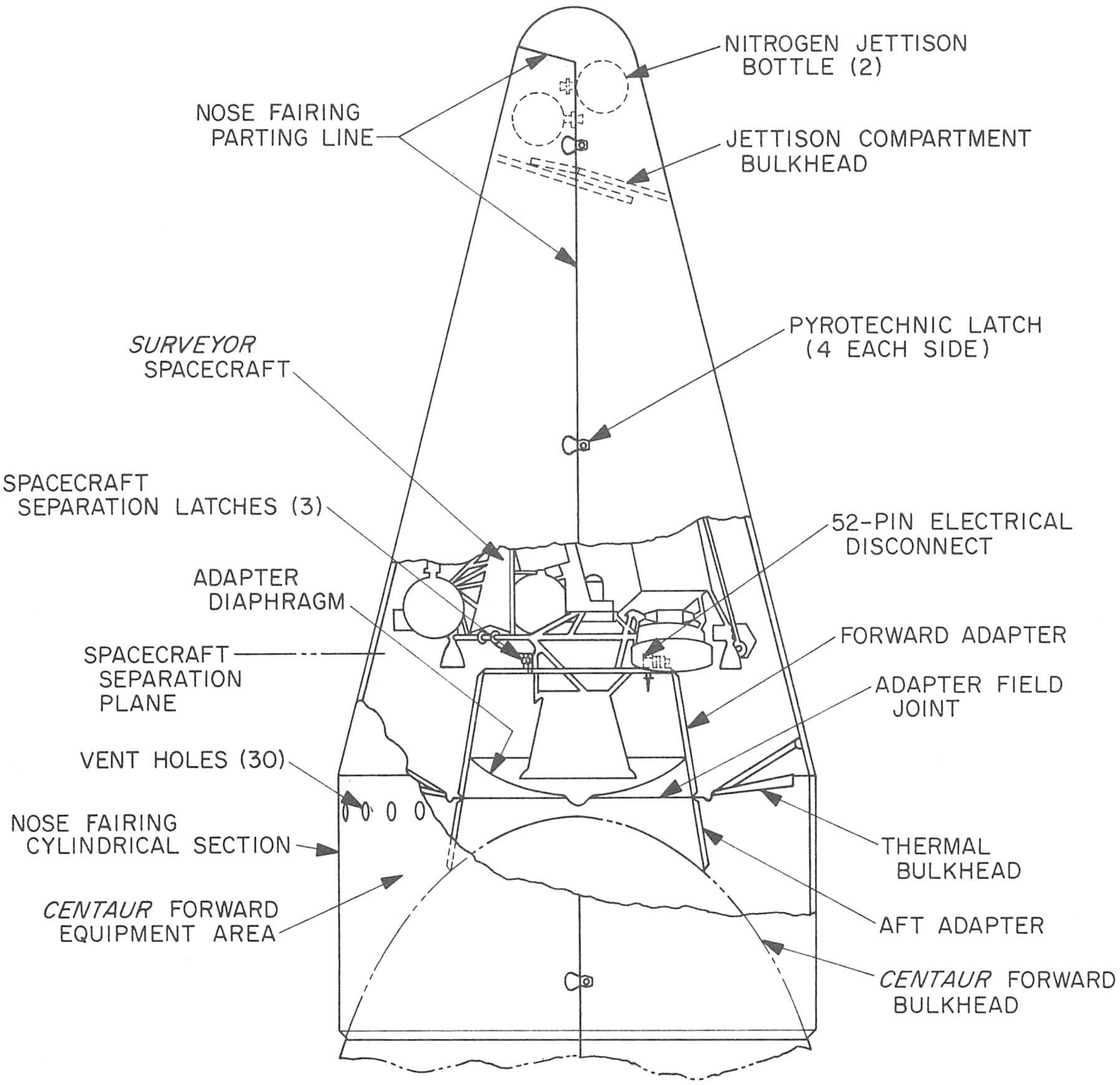


Fig. VIII-3. *Surveyor/Centaur* interface configuration

On the AC-13, -14, and -15 vehicles, two high-frequency accelerometers were located on the vehicle side of the separation plane just below the spacecraft attachment ring of the forward adapter section. One of these accelerometers was mounted in the radial direction near the spacecraft leg 1 attach point. The other was mounted in a longitudinal direction near the leg 3 attach point. The outputs of both accelerometers were telemetered continuously. On previous missions, one accelerometer had been mounted differently on the adapter, and four accelerometers had been installed on the spacecraft side of the separation plane.

The low-drag nose fairing was an RF-transparent, clamshell configuration consisting of four sections fabricated of laminated fiber glass faces and honeycomb fiber glass core material. Two half-cone forward sections were brought together over the spacecraft mounted on the forward adapter. An annular thermal bulkhead between the adapter and base of the conical section completed encapsulation of the spacecraft.

The encapsulated assembly was mated to the *Centaur* with the forward adapter section attaching to the aft adapter section at a flange field joint requiring 72 bolts. The conical portion of the nose fairing was bolted to the cylindrical portion of the fairing, its two halves attached to the forward end of the *Centaur* tank around the equipment compartment. Doors in the cylindrical sections provided access to the adapter field joint and the *Centaur* forward equipment area. The electrical leads from the forward adapter were carried through three field connectors and routed across the aft adapter to the *Centaur* umbilical connectors, the *Centaur* programmer, and the telemetry units.

Special distribution ducts were built into the nose fairing and forward adapter to provide air conditioning of the spacecraft cavity until liftoff. Seals were provided at the joints to prevent shroud leakage except through vent holes in the cylindrical section. Before launch, the shroud cavity was monitored for possible spacecraft propellant leakage by means of a toxic gas detector tube which disconnected at liftoff. Tubes were also inserted into each of the vernier engine combustion chambers to permit nitrogen purging for humidity control and leak detection until manual removal before the service tower was rolled away. On the *Surveyor V, VI,* and *VII* missions, the spacecraft alpha scattering instrument was also purged by means of a tube which disconnected from the nose fairing at liftoff.

The entire nose fairing was designed to be ejected by separation of two clamshell pieces, each consisting of a conical and cylindrical section. Four pyrotechnic pinpuller latches were used on each side of the nose fairing to carry the tension loads between the fairing halves. Nose fairing loads were transmitted to the *Centaur* tank through a bolted joint which also attached to the forward end of the *Centaur* insulation panels. A nitrogen bottle was mounted in each half of the nose fairing near the forward end to supply gas for cold gas jets to force the panels apart. Hinge fittings were located at the base of each fairing half to control ejection, which occurred under vehicle acceleration of approximately 1 *g* during the *Atlas* sustainer phase of flight.

2. Interface Management

The *Surveyor* spacecraft–*Atlas/Centaur* interface was managed and controlled by the *Surveyor* and *Centaur* Project Offices in accordance with operating relationships and procedures as outlined in the *Surveyor Project Development Plan.* The operating relationship for interface planning and design is depicted in Fig. VIII-4.

a. Development and implementation of interface requirements. During the early stages of the *Surveyor* Project, the task of establishing mutually acceptable requirements for the mechanical, electrical, RF, and operational integration of the *Surveyor* spacecraft and the *Atlas/Centaur* launch vehicle system was complicated by the concurrent development of the two systems. The interface requirements were first formalized in a HAC specification, dated October 10, 1962, which was used by GD/C for design studies of the interface hardware and software items.

The HAC specification, modified by agency review and negotiation between JPL and LeRC, was transformed into a JPL project document, *Surveyor Spacecraft Atlas–Centaur Launch Vehicle Interface Requirements,* first published in April 1963. This document defined the technical requirements and constraints necessary to ensure the mechanical, environmental, and electrical compatibility of the *Surveyor* spacecraft, the AFETR launch facilities, the combined systems test stand (CSTS), and the *Atlas/Centaur* Launch Vehicle System. In addition, the document reflects JPL/LeRC agreements for implementation of the technical coordination and responsibilities necessary for integrating the spacecraft, the launch complex, and the Launch Vehicle System, and included provisions for maintenance of a *Surveyor/Centaur* interface control drawing, a schedule for planning and continuous assessment of interface activity progress.

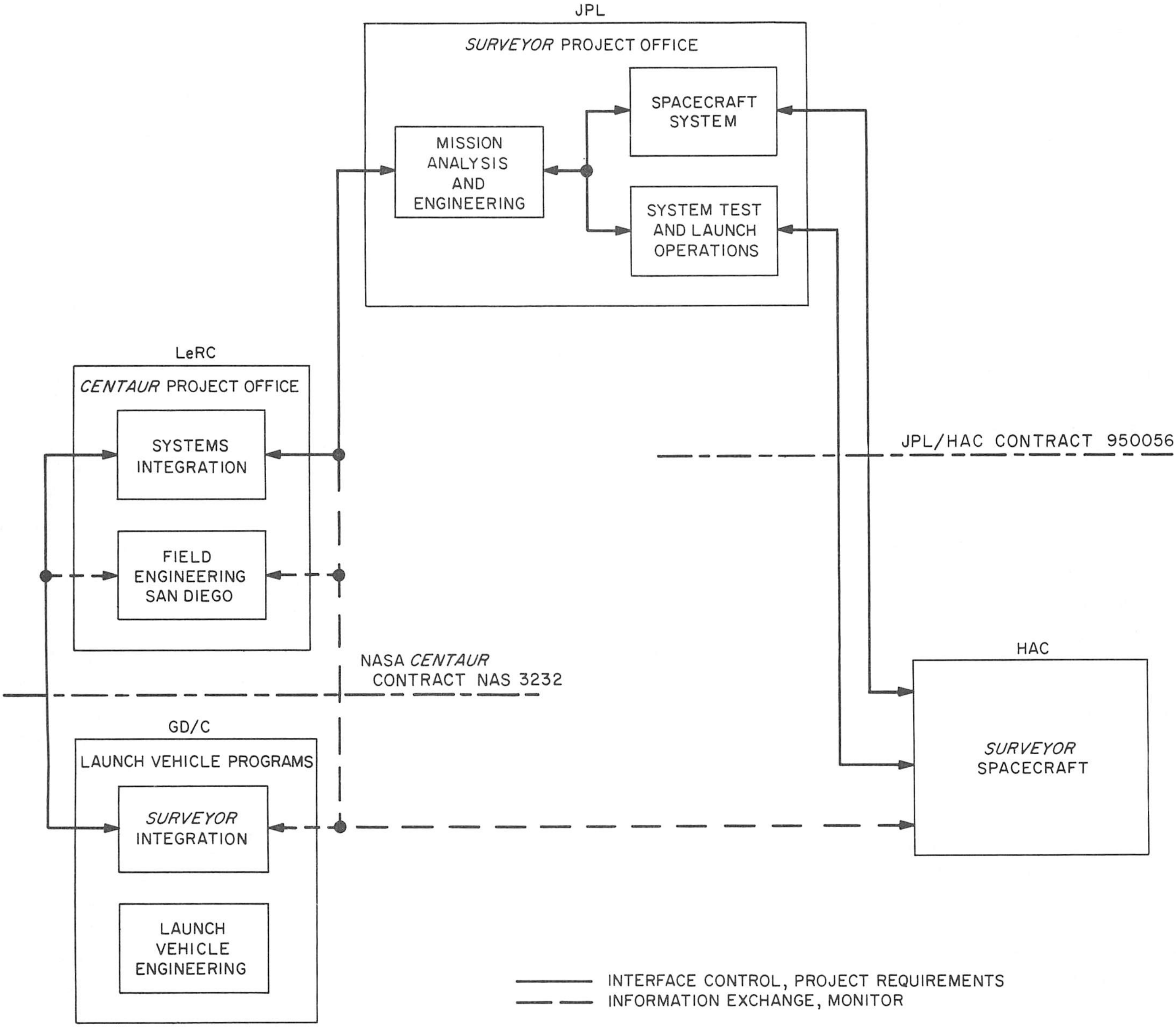


Fig. VIII-4. Spacecraft/Launch Vehicle System interface planning and design operating relationship

Implementation of the interface requirements was accomplished by contractual direction from JPL to HAC for the Spacecraft System and from LeRC to GD/C for the Launch Vehicle System.

b. Interface working groups. Close technical and schedule liaison and control of the electrical and mechanical interface were performed through the *Centaur/Surveyor* interface working group, whose specific responsibilities were to:

(1) Expedite the flow of detailed design criteria across the interface.

(2) Review status of items appearing on the project interface working schedule.

(3) Evaluate the technical ramifications of proposed changes which could affect the interface.

(4) Formulate, for project management consideration, technical and schedule recommendations for the resolution of incompatibilities or other problems which related to interface design, interface testing, and system operation.

The interface working group first convened on March 14, 1962. Membership consisted of technical representatives from JPL, LeRC *Centaur* Project Office, LeRC

Centaur project resident offices at San Diego, GD/C, HAC, and Kennedy Space Center (KSC). The KSC representative attended meetings only when agenda items were pertinent to launch operations. Special task groups and committees were employed to expedite solutions for particular interface problems such as RFI, instrumentation, *Surveyor* dynamic model integration, scheduling, combined systems test, and launch coordination. Action items were accepted only by JPL and LeRC and constituted a commitment for official Project Office response to the problem concerned.

Performance, trajectory, and guidance and control activities were assigned to the *Centaur/Surveyor* performance and trajectory working group. A separate panel was responsible for tracking, telemetry, and communications in support of the project elements.

c. Interface control drawing. The *Centaur/Surveyor* mechanical and electrical interface was outlined on a *Centaur/Surveyor* interface control drawing, originally issued on December 20, 1961. Following the May 1966 issue, responsibility for maintenance of the control drawing was transferred, through interface working group action, to JPL, where it was revised and issued on September 23, 1966, as JPL Drawing 142800. The last change was released on September 27, 1967. The interface control drawing reflected the following interface elements:

(1) Interface plane and structural outline of the spacecraft and forward elements of the *Centaur.*

(2) Nose fairing structural outline, including the air-conditioning ductwork and the payload dynamic clearance envelope.

(3) The spacecraft center-of-gravity envelope.

(4) Outline and location of all spacecraft/*Centaur* mating components.

(5) Electrical schematics showing connector part numbers, pin assignments, functions, and appropriate electrical characteristics.

(6) Location of ports for the toxic gas detector and spacecraft purge systems.

Changes to the interface, both physical and functional, were subjected to a rigid control and investigative procedure before being authorized by the *Surveyor* project manager for incorporation on the interface control drawing.

d. Interface schedules. Interface scheduling was accomplished through the *Surveyor/Centaur* interface committee and launch-operation working groups in accordance with official NASA flight schedules and within contractual boundaries.

Actions to establish a launch vehicle/spacecraft integration and documentation schedule in accordance with the *Surveyor* flight schedule were initiated during a *Surveyor* integration meeting at GD/C on October 5, 1961. Both HAC and GD/C furnished schedule estimates to JPL, who combined them into one project schedule which included details of launch complex design and readiness, and guidance and trajectory software. This combined schedule, presented during an integration meeting on December 20, 1961, included target dates for a launch-vehicle/spacecraft interface specification; an interface control drawing; and design release of the adapter, wiring harness, and nose fairing. In addition, dates were established for fabrication and delivery of a test nose fairing, test adapters, wiring harness, and mockups of the forward portion of the *Centaur* to be used by HAC for spacecraft testing.

Hardware delivery and test dates were also established for interface tests such as in-flight electrical disconnect, spacecraft/launch vehicle separation, nose fairing deployment, and vibration. The first test hardware items, delivered to HAC by GD/C on January 15, 1962, were the nose fairing and *Centaur* mockups.

Until February 1962, AFETR operation activities for flight spacecraft were estimated to require eight weeks from arrival of the first spacecraft and launch vehicle through launch. A second spacecraft was planned for arrival six weeks before the first launch. This spacecraft was to proceed through launch preparations in parallel with the first spacecraft and serve as a standby for assurance in meeting the first launch opportunity (August 1963). Program delays, launch vehicle experience in research and development flights, the addition of a combined systems test at San Diego and experienced gained in the T-21 prototype spacecraft test program combined to considerably alter the original estimates.

Updating and revision of the integration and documentation schedules continued and, on March 12, 1963, the integration and documentation schedule was approved and issued by the *Surveyor* Project Office. The schedule, published as an official project document, included *Surveyor* dynamic model hardware and documentation schedules in support of *Atlas/Centaur* research and development flights AC-5 and AC-6. Later issues included the *Surveyor* mass model hardware and documentation for support of *Atlas/Centaur* research and development flights AC-8 and AC-9.

In compliance with a JPL request, HAC submitted an operations schedule for SC-1 which conformed to a 100-calendar-day (74 workdays) time span from shipment of the spacecraft to the CSTS through launch. Through joint negotiations between JPL, LeRC, HAC, and GD/C, a spacecraft/launch vehicle integrated operations schedule was established and published on January 17, 1966. This schedule called for a 103-calendar-day (79 workdays) period for spacecraft operations beginning with spacecraft shipment to the CSTS, and 94 calendar days (70 workdays) from spacecraft/*Centaur* mate at CSTS through launch.

Planning dates for joint spacecraft/launch vehicle operations for each mission were established by counting backward from the launch date the estimated time between operations. This determined the target dates for the key milestones of spacecraft/*Centaur* mate at CSTS, demate at CSTS, mate for J-FACT at AFETR, and final mate for launch. These dates were adjusted to accommodate revised spacecraft, launch vehicle, and launch complex availability dates through interface liaison and joint meetings (launch operations coordination meetings). After spacecraft and launch vehicle activities began at AFETR, a launch operations working group further adjusted the schedule as day-to-day operations dictated.

A generalized flow sequence of spacecraft and launch vehicle combined systems tests and AFETR operations is shown in Figs. VIII-5 and VIII-6. A summary of significant spacecraft/launch vehicle milestones is shown in Fig. VIII-7.

The January 17, 1966 AC-10/SC-1 schedule, which was planned for 103 calendar days for the combined systems tests and the AFETR operations, was accomplished in 101 calendar days after a two-day-late arrival of the spacecraft at the CSTS.

The AC-10/SC-1 period, mate through demate, was planned for nine calendar days and was accomplished in five days by extending the workday and reducing the launch vehicle five-day data reduction, review, and buyoff period to four days. Later, this period was further reduced to three days for planning purposes by a decision to demate based on a quick-look test-data evaluation scheme. AC-11/SC-4 and AC-13/SC-5 proceeded through the combined systems test mate–demate period in two and three days, respectively.

The AC-10/SC-1 operations schedule also provided for a 12-calendar-day (10 workdays) AFETR J-FACT period (spacecraft/launch vehicle mate through the J-FACT and demate) which was accomplished in 10 days. The AC-7/SC-2 J-FACT period was planned for nine calendar days and was accomplished in eight days.

An *Atlas/Centaur* integrated propellant tanking test was planned and performed during the AC-10/SC-1 and AC-7/SC-2 J-FACT periods. The tanking test was performed to determine RFI susceptibility and thermal effects on the spacecraft due to the tanking of launch vehicle propellants. Data accumulated during these tests confirmed that no detrimental effects were likely to occur, and it was decided to conduct the launch vehicle tanking test after the J-FACT and demating of the spacecraft. However, a 60-min period was provided in the J-FACT countdown of the remaining missions to verify spacecraft electrical compatibility with the launch complex and vehicle propellant storage and handling systems, by exercising all tanking valves (without propellant) in lieu of a cryogenic tanking operation. There was ample time in the launch vehicle schedule after J-FACT for conducting the tanking test, and this procedure change enabled the planned J-FACT periods to be reduced from ten to four days. The AC-12/SC-3, AC-11/SC-4, and AC-13/SC-5 completed the J-FACT mate through demate operations in four calendar days. The AC-14/SC-6 and AC-15/SC-7 J-FACT operations were completed in three days.

Before the first *Surveyor* mission, launch complex 36A was prepared to accommodate the *Atlas* LV-3C in the direct-ascent launch configuration; Launch Complex 36B was prepared for the *Atlas* LV-3C in the parking-orbit launch configuration. Extensive modifications to both launch complexes were required to accommodate the new extended *Atlas* SLV-3C used for *Surveyors V–VII*. In addition, parking-orbit modifications had to be made to Launch Complex 36A for launching of any of the last three *Surveyor* spacecraft from that platform. The requirement for these launch-complex conversions and the scheduling of the last three *Surveyor* missions on two-month launch intervals resulted in the determination that two successive launches from the same launch complex (*Surveyors V* and *VI* from 36B) would yield the highest probability of meeting the launch dates. The schedule permitted 63 workdays for conversion of Launch Complex 36B and 64 workdays for 36A. In addition, the two successive launches from the same platform permitted only 18 calendar days between the first opportunity to launch AC-13/SC-5 and the J-FACT for AC-14/SC-6, including replacement of platform burnoff items.

Modifications to Launch Complex 36B began on April 17, 1967, immediately after the launch of *Surveyor III*. The SC-5 craft was mated to the AC-13 craft 95 days later. In the 18 days following the launch of *Surveyor V*, Launch Complex 36B was refurbished, and the AC-14 launch vehicle was erected, checked out, and made ready for J-FACT mate with SC-6. This 18-day period represented a significant reduction from previous planning estimates and met the most severe schedule constraint. Launch Complex 36A modifications began immediately after the launch of *Surveyor IV* on July 14, 1967, and there were no particular problems in accomplishing the modifications in time for the first mate of SC-7 and AC-15 on November 29, 1967.

The CSTS modifications required to accommodate the *Atlas* SLV-3C were also considered a scheduling risk since revalidation of the CSTS was conducted concurrent with a portion of the AC-13 launch vehicle checkout and test, due to the comparatively short period between the CSTSs for *Surveyors IV* and *V*. However, the modifications and test activities were accomplished, and the CSTS was ready for mating SC-5 and AC-13 in 61 days with no significant problems.

3. Interface Hardware Design, Development, and Qualification

General Dynamics/Convair, under contractual direction from LeRC, was responsible for the design, development, and qualification of the interface hardware except for the spacecraft propellant leak-detection system, alpha scattering instrument, vernier engine ground purge systems, and the spacecraft/*Centaur* 52-pin connector used for in-flight electrical disconnect.

Development testing of interface items, as well as the entire launch vehicle system, was accomplished by LeRC and GD/C under a rigid sequential test program consisting of design evaluation tests, design proof (qualification) tests, acceptance tests, and flight tests. This test program was detailed by GD/C in the *Centaur* unified test plan. A summary of major *Centaur/Surveyor* interface tests is given in Table VIII-7.

a. Nose fairing system. The *Surveyor* nose fairing was designed and developed by GD/C as an adaptation of the configuration planned for use with the *Advent* satellite. This configuration incorporated a cylindrical section which could be varied in length to accommodate different payload sizes. The *Surveyor* nose fairing design featured the following:

(1) Payload encapsulation capability prior to mating with the launch vehicle for protection of the spacecraft.

(2) Environmental protection of both the spacecraft and the *Centaur* electronic equipment area during the boost phase of flight.

(3) A low-drag profile to reduce aerodynamic heating and loads on the vehicle during ascent through the earth's atmosphere.

(4) A clamshell configuration to meet assembly, encapsulation, and jettison requirements.

(5) Built-in air-conditioning ductwork.

(6) Pyrotechnic separation latches.

(7) RF-transparent material.

(8) Thermal isolation of the spacecraft from *Centaur*.

The original design of the *Surveyor* nose fairing included provisions to enable sterilization of the spacecraft after encapsulation. The sterilization requirement was deleted in December 1962 (see Section XIII-C-7).

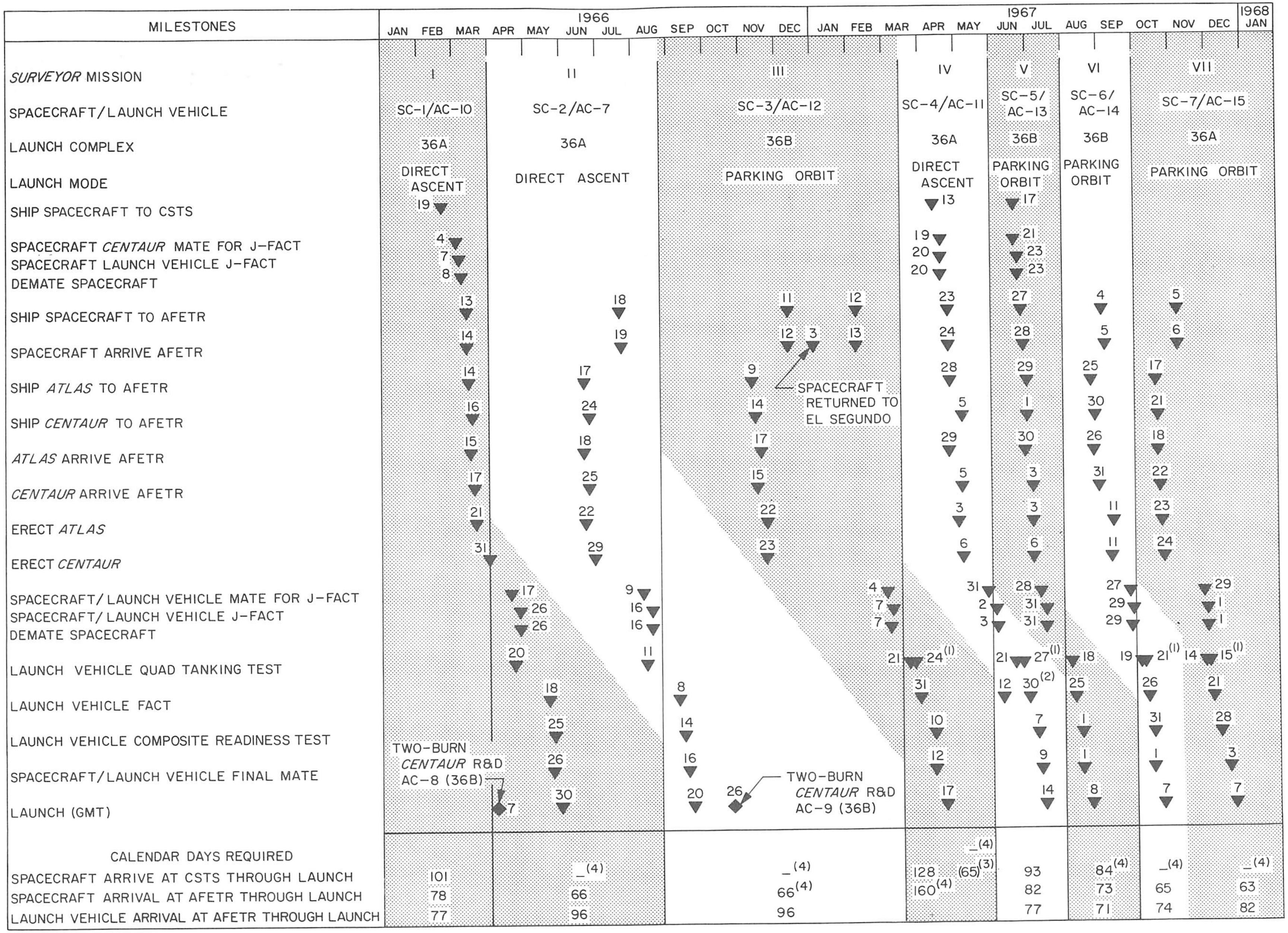
The *Surveyor* nose fairing consisted of a conical section, a cylindrical (or barrel) section, a thermal bulkhead, and a separation system. The nose fairing general arrangement is shown in Fig. VIII-8.

1. Nose fairing structure and separation system. The *Surveyor* nose fairing was approximately 22 ft long from the aft end of the cylindrical section to the tip of the nose dome of the conical section. The total jettison weight was approximately 2033 lb (AC-15). A thermal bulkhead extended between the base of the conical section and the payload adapter to provide thermal isolation between the payload cavity and the *Centaur* forward electronic equipment area. It had a split line in the vehicle *X-Z* plane to allow the fairing to be jettisoned in flight. Upon encapsulation, the two halves were sealed to prevent contamination and leakage into the spacecraft cavity (from external pressure during flight). Thermal criteria for design of the laminated fiber-glass skins were based upon inner and outer "glue line" temperatures of 300 and 1050°F. Aerodynamic heating was reduced by coating the outside surfaces of the conical and cylindrical sections with 121 lb of an ablative material called Thermalag. The fairing was essentially RF-transparent. Losses through the fairing did not exceed 6 db at frequencies below 5 GHz. Eight explosive bolt latches carried the tension loads between the nose fairing halves until unlatching occurred (before jettison).

Fig. VIII-5. Combined systems test sequence

SURVEYORS I AND II—LAUNCH VEHICLE PROPELLANT TANKING TEST CONDUCTED DURING J-FACT PERIOD WITH SPACECRAFT MATED.
SURVEYOR I SPACECRAFT RFI TEST CONDUCTED DURING J-FACT PERIOD.
ALTHOUGH OPTIONAL, A FACT WAS CONDUCTED FOR ALL *SURVEYOR* MISSIONS.

Fig. VIII-6. Generalized flow sequence, *Atlas/Centaur/Surveyor* prelaunch operations, AFETR



NOTES: (1) 2 TANKING TESTS PERFORMED FOR AC-12, AC-11, AC-14, AND AC-15.
(2) FACT PERFORMED TWICE FOR AC-11.
(3) SPACECRAFT RETURNED TO EL SEGUNDO FOR RETEST. LAUNCH VEHICLE IN STANDBY STATUS FOR ABOUT 2 MONTHS, 128 DAYS TOTAL FROM FIRST ARRIVAL, 65 DAYS FROM SECOND ARRIVAL AT AFETR.
(4) CST NOT PERFORMED WITH SC-2, SC-3, SC-6, AND SC-7.

Fig. VIII-7. *Surveyor/Atlas/Centaur* milestones

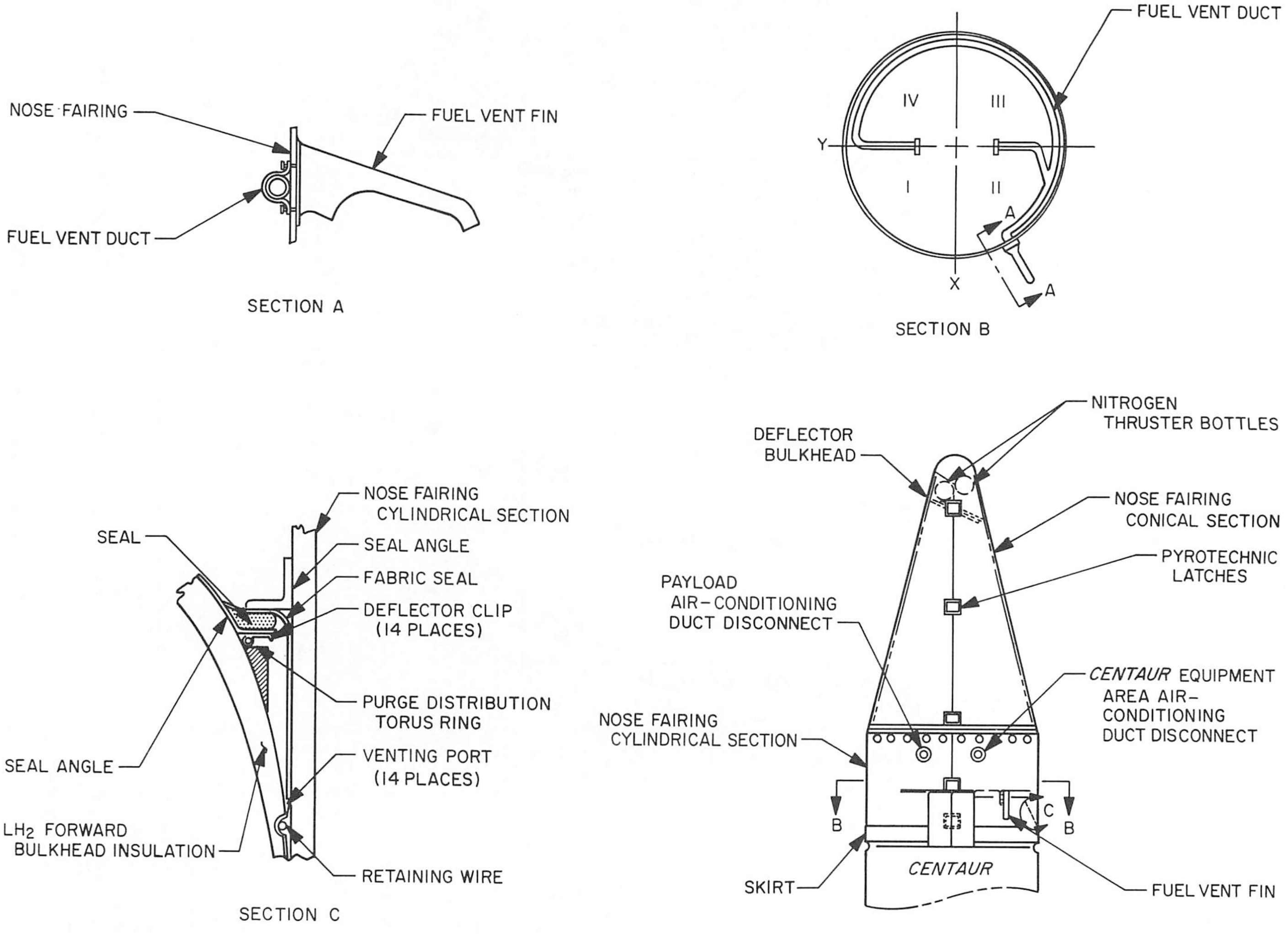


Fig. VIII-8. Nose fairing, general arrangement

Table VIII-7. *Centaur/Surveyor* interface test summary

Interfaces and associated tests	Test type	Test location	Test article description				
			Centaur hardware				Spacecraft hardware
			Tank	Payload adapter	Nose fairing	Other	
Ground tests—structural interfaces							
Mechanical structures integration							
Preliminary match–mate test	Design evaluation	HAC, El Segundo	Bulkhead mockup	Mockup	Gage		Predesign mockup
Formal match–mate test	Design evaluation	HAC, El Segundo	Bulkhead mockup	Test	Test		Test model (M-12)
Final proofing match–mate	Design evaluation	Combined systems test stand	AC-7 (3D)	AC-7	AC-7	All systems	Prototype (T-21)
Structural dynamics integration							
Payload adapter/*Surveyor* vibration and modal tests	Design evaluation	HAC, El Segundo		Test			Vibration model (S-1)
Forward bulkhead/adapter vibration transmissibility	Design evaluation	GD/C, vibration test tower	Test (3C)	Test		Dummy packages	Mass model (AC-4)
Atlas/Centaur/Surveyor modal tests	Design evaluation	LeRC, Plumbrook	Test (116/5C)	Test	Test	Dummy packages	Dynamic model (SD-4)
Ground tests—mechanical system interfaces							
Surveyor air-conditioning system development							
Formal *Surveyor* air conditioning	Design evaluation	HAC, El Segundo	Bulkhead mockup	Test	Test	Prototype unit	Thermal model (S-4)
Centaur/Surveyor air-conditioning system compatibility	Special development	Complex 36A	AC-5 (6C)	AC-5	AC-5	Operational unit	Dynamic model (SD-1)
Combined-system test stand, installation verification check	Special development	Combined systems test stand	AC-7 (3D)	AC-7	AC-7	Operational unit	Prototype (T-21)
Complex 36A (rebuilt), installation verification check	Special development	Complex 36A	Simulator	AC-8	AC-8	Operational unit	Prototype (T-21)
Complex 36B, installation verification check	Quad tanking	Complex 36B	AC-6 (2-D)	AC-6	AC-6	Operational unit	Dynamic model (SD-2)
Surveyor nose fairing jettison system development							
Nose fairing jettison tests (at sea-level conditions)	Design evaluation	GD/C, Pt Loma	Fixture	Test	Test		Geometrical envelope
Nose fairing unlatch tests, AC-3 configuration, single-burn design evaluation	Design evaluation	GD/C, Pt Loma	Test (3C)	Test	Test	Insulation panel purge Pyrotechnics	
Nose fairing jettison and cavity pressure transient tests (at simulated altitude conditions)	Design evaluation	LeRC space power chamber	Bulkhead mockup	Test	Test	Pressure instr	(1) Mast and solar panels (2) Mass model
Nose fairing jettison tests, simulated altitude, AC-6 configuration, final evaluation, single-burn design	Design evaluation	LeRC space power chamber	Bulkhead mockup	Test	Test	Pyrotechnics	Model of spacecraft omnidirectional antenna

Table VIII-7 (contd)

Interfaces and associated tests	Test type	Test location	Test article description				
			Centaur hardware				Spacecraft hardware
			Tank	Payload adapter	Nose fairing	Other	
Ground tests—mechanical system interfaces (contd)							
Nose fairing unlatch tests, AC-8 configuration, two-burn design evaluation	Design evaluation	GD/C, Pt Loma	Test	Test	Test	Oxidizer and hydrogen vent system	
Surveyor separation system development							
System tests							
Separation system test (hook-type latch, 55-71155)	Design evaluation	GD/C		Test		Test latch	Mass simulator (T-1)
Separation system test (hook-type latch, 55-71187)	Design evaluation	GD/C		Test		Test latch	Mass simulator (T-1)
Separation system test (pinpuller latch, 55-71190-1)	Design evaluation	GD/C		Test		AC-5 latch	Mass model (AC-4)
Separation system test (pinpuller latch, 55-71190-807)	Design evaluation	GD/C		Test		AC-6 latch	Mass simulator (T-1)
Component tests:							
Latch vibration test (hook-type, titanium, 55-71187)	Design evaluation	GD/C				Test latch	2100 lb AC-4, mass model
Latch vibration test (hook-type, hybrid, 55-71187)	Design evaluation	GD/C				Test latch	2100 lb AC-4, mass model
Latch vibration test (hook-type, titanium, 55-71187)	Design evaluation	GD/C				Test latch	2100 lb SD-4 dynamic model
Latch vibration test (hook-type, 4340 stainless steel, 55-71187)	Design evaluation	GD/C		Test		Test latch	2100 lb SD-4 dynamic model
Latch vibration test (pin-puller, 4340 stainless steel, 55-71190-1)	Design evaluation	GD/C		Test		AC-5 latch	1465 lb SD-4 dynamic model
Latch vibration test (pin-puller, titanium, 55-71990-1)	Design evaluation	GD/C		Test		AC-5 latch	2500 lb SD-4 dynamic model
Latch vibration test (pin-puller, titanium, 55-71190-1)	Design evaluation	GD/C		Test		AC-5 latch	2500 lb SD-4 dynamic model
Latch vibration test (pin-puller, titanium, 55-71190-3)	Design evaluation	GD/C		Test		AC-6 latch	2500 lb SD-4 dynamic model
Ground tests—electrical and RF system interfaces							
Centaur/*Surveyor* electrical and and RF integration							
Initial evaluation of electrical and RF interfaces	Special development	Combined systems test stand	AC-7 (3D)	AC-7	AC-7	EMI instr	Prototype (T-21)

Table VIII-7 (contd)

Interfaces and associated tests	Test type	Test location	Test article description					
			Centaur hardware				Spacecraft hardware
			Tank	Payload adapter	Nose fairing	Other	
Ground tests—electrical and RF system interfaces (contd)							
Detailed interference (RF) tests:							
Test 1: *Surveyor*/36A GSE, combined signature	Special development	Complex 36A	Electrical simulator	AC-8	AC-8	EMI instr	Prototype (T-21)
Test 2: *Atlas/Centaur* 36A GSE, signature	Special development	Complex 36A				EMI instr	
Test 3: *Atlas/Centaur* 36A GSE combined signature	FACT	Complex 36A	AC-7 (3D)	AC-7	AC-7	EMI instr	Electrical simulator
Test 4: Final *Atlas/Centaur/Surveyor* combined signature	J-FACT	Combined systems test stand	AC-10 (1D)	AC-10	AC-10	EMI instr	Flight (SC-1)
Surveyor safety systems development							
Surveyor destruct conical-shaped charge development	Vendor test	Vendor					Spacecraft retro case simulator
Toxic gas detection system evaluation and calibration	Design evaluation	Explosive-safe facility AFETR		AC-8	AC-8	Toxic gas sensor instrumentation	Spacecraft simulator
Postencapsulation emergency access procedures test	Design evaluation	Explosive-safe facility, AFETR			Wood mockup		Spacecraft geometrical relationship
Ground tests—ground handling and facilities interfaces							
Centaur/Surveyor ground handling equipment integration							
Formal match—mate test: Encapsulation and mating tests	Design evaluation	HAC, El Segundo	Bulkhead mockup	Test	Test	Prototype equipment	Test model (M-12)[c]
Towing tests	Design evaluation	HAC, El Segundo		Test	Test	Prototype equipment	2100 lb ballasted mass[c]
Surveyor/operational facilities integration							
Operational readiness verification	Validation	Combined systems test stand	AC-7 (3D)	AC-7	AC-7	Operational equipment	Prototype spacecraft (T-21)
Operational readiness verification	Validation	Explosive-safe facility, AFETR	Simulator	AC-8	AC-8	Operational equipment	Prototype spacecraft (T-21)
***Centaur/Surveyor* compatibility proofing**							
AC-4 mass model program							
Qualification, mass model structure (55-71177)	Design proof	GD/C, vibration laboratory		Test			Prototype mass model
Acceptance and combined-system testing	Acceptance	GD/C, dock 18	AC-4 (4C)	AC-4	AC-4	Flight instr	Flight mass model
Preflight checkout and launch operations	Flight	AFETR	AC-4 (4C)	AC-4	AC-4	Flight instr	Flight mass model

Table VIII-7 (contd)

Interfaces and associated tests	Test type	Test location	Test article description: Centaur hardware: Tank	Centaur hardware: Payload adapter	Centaur hardware: Nose fairing	Centaur hardware: Other	Spacecraft hardware
Centaur/Surveyor compatibility proofing (contd)							
AC-5 and -6 dynamic model program							
Retrorocket simulator structure only (64-01038)	Design proof (qualification)	GD/C, vibration laboratory				Dummy packages	Prototype structure
Retrorocket simulator and spaceframe	Design evaluation (verification)	GD/C, vibration laboratory		AC-5		Dummy packages	Prototype/SD-1
AC-5 retrorocket simulator only	Factory checkout (acceptance)	GD/C, Bldg. 5				Flight packages	Flight unit
AC-6 retrorocket simulator only	Factory checkout (acceptance)	GD/C, Bldg. 5				Flight packages	Flight unit
AC-5 retrorocket simulator and spaceframe	Flight acceptance	GD/C, vibration laboratory				Flight packages	Flight dynamic model (SD-1)
AC-6 retrorocket simulator and spaceframe	Flight acceptance	GD/C, vibration laboratory				Flight packages	Flight dynamic model (SD-2)
AC-5/SD-1	Interim	GD/C, dock 18 (combined systems test stand)	AC-5 (6C)	AC-5	AC-5	Flight packages	Flight dynamic model (SD-1)
AC-6/SD-2 combined systems testing	Interim	GD/C, dock 18 (combined systems test stand)	AC-6 (2D)	AC-6	AC-6	Flight packages	Flight dynamic model (SD-2)
AC-5/SD-1 preflight checkout and launch	Flight	Explosive-safe facility, AFETR	AC-6 (2D)	AC-5	AC-5	Flight packages	Flight dynamic model (SD-1)
AC-6/SD-2 preflight checkout and launch	Flight	Explosive-safe facility, AFETR	AC-5 (6C)	AC-6	AC-6	Flight packages	Flight dynamic model (SD-2)

Nose fairing loads were transmitted to the *Centaur* tank through a bolted joint which also attached to the forward end of the *Centaur* insulation panels. The joint contained a flexible, linear shaped charge for insulation-panel and nose fairing separation. Hinge fittings at the aft end of the cylindrical section permitted rotation of the nose fairing halves about this point at time of jettison. Separation of the nose fairing halves from the vehicle occurred at between 31 and 35 deg of rotation.

A positive impulse to rotate the nose fairing halves after latch release was provided by nitrogen at 2370 psig (AC-15) released from two thruster bottles located near the forward end of each conical section. A deflector bulkhead was provided to protect the payload compartment from gas impingement during jettison. A seal between the deflector bulkhead halves prohibited thruster bottle gases from entering the payload compartment until the bulkhead halves had separated completely. The nose fairing jettison system is outlined in Fig. VIII-9.

2. *Air conditioning system.* The encapsulated spacecraft was supplied with conditioned air or nitrogen during transit from the explosive-safe facility to the launch complex, during the preparation for hoisting, and continuously from mating until liftoff. HAC was responsible for spacecraft air conditioning during transit to the

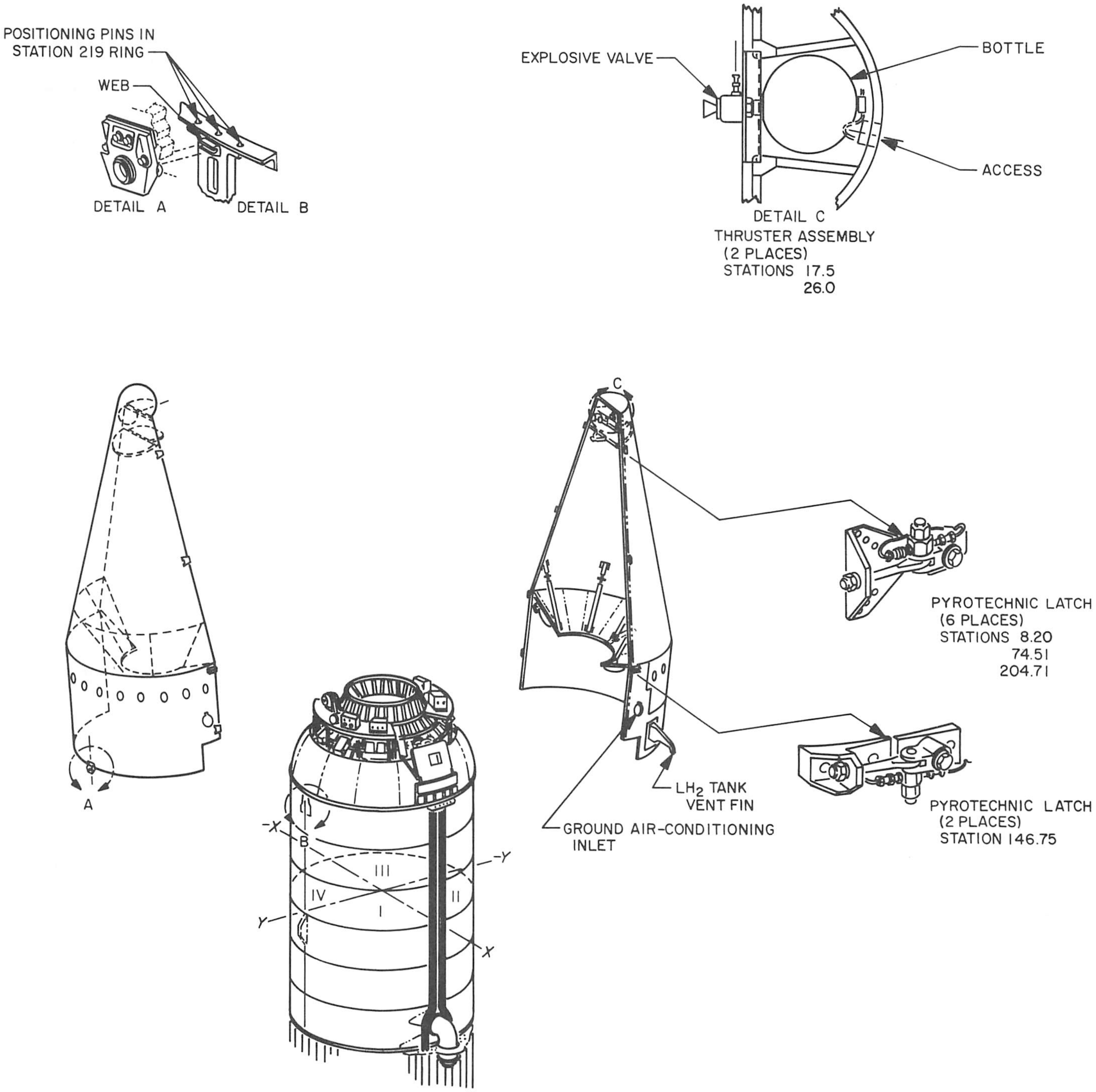


Fig. VIII-9. Nose fairing jettison system

launch complex. Thereafter, GD/C was responsible for supplying the encapsulated spacecraft with filtered air or nitrogen at 1000 $\pm$200 ft^3/min with a relative humidity of less than 60% at 65°F dry-bulb temperature. The temperature of the air at the inlet to the nose fairing cavity was maintained at 70 $\pm$5°F until two hours prior to simulated liftoff during the J-FACT or opening of the actual launch window. The air temperature was then raised to 85 $\pm$5°F.

The ductwork for distribution of the conditioned air was an integral part of the *Surveyor* nose fairing and attached to the support-equipment supply at a quick-disconnect located on the cylindrical section as shown in

Fig. VIII-10. Thirty-three 2.13-in.-diameter holes, spaced around the periphery of the forward end of barrel section, provided venting during prelaunch operations and atmospheric ascent to prevent sudden depressurization when the nose fairing was jettisoned. Honeycomb baffles were bonded in each of the 33 vent holes to prevent contamination of the nose fairing cavity from external sources. This baffling was removed from three of the vents to permit entrance of the alpha scattering instrument and vernier engine purge lines.

3. *Television targets and lights.* To permit spacecraft television camera testing and calibration after encapsulation, targets were painted on the interior surface of the nose fairing. Each target consisted of a pattern of black numerals on a white background in an area approximately 10 × 10 in. The targets were illuminated by explosionproof lights. Three targets were originally planned, two of which were used during *Surveyor I* prelaunch testing. To overcome deficiencies of insufficient illumination uncovered during the *Surveyor I* combined

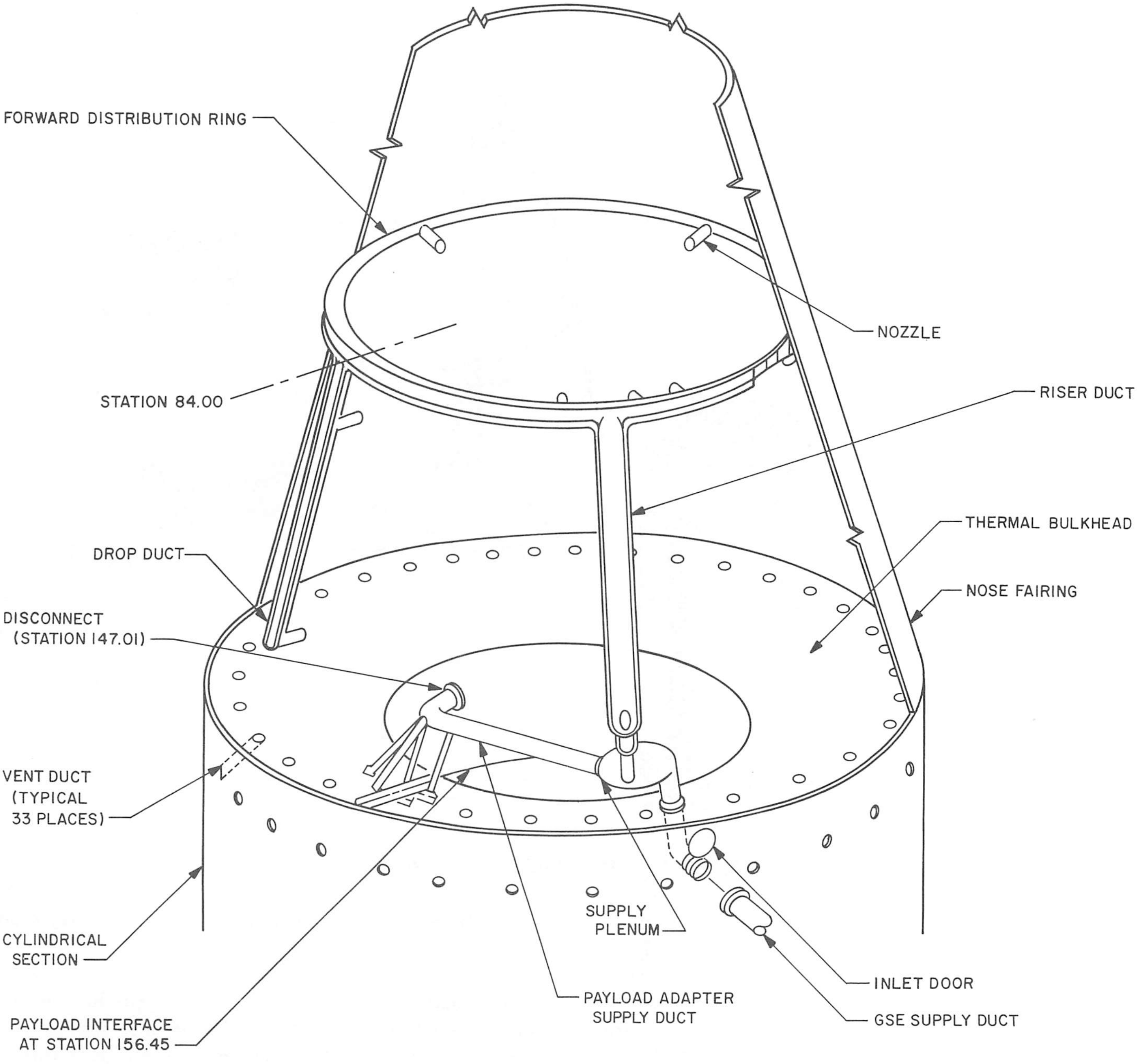


Fig. VIII-10. Spacecraft air-conditioning distribution system

systems test, use of these two targets was discontinued and another dual-lighted target was added for *Surveyor II* and subsequent models. Control of the television target illumination was exercised by HAC through the *Surveyor* operations console in the launch operations center.

b. Spacecraft/Centaur adapter and separation system. The spacecraft adapter provided a structural tie between the *Centaur* LH_2 tank forward bulkhead and the *Surveyor* spacecraft. The adapter transferred the spacecraft load from the three separate spacecraft attachment fittings to the mounting ring at *Centaur* station 172.45. The adapter structure was fabricated of anodized aluminum, weighed approximately 127 lb (AC-15), and was capable of supporting a 2500-lb payload.

The adapter contained an air-conditioning distribution duct (for thermal conditioning of the *Surveyor* retro motor) and a fiber-glass diaphragm to provide a thermal barrier between the *Centaur* stage and the payload compartment. The adapter also provided a mounting structure for the payload separation system components, a *Surveyor* in-flight electrical disconnect system, and a range safety destructor unit.

1. Adapter structure. The *Surveyor* adapter is shown in Fig. VIII-11. To facilitate spacecraft handling and mating with the *Centaur*, the adapter was fabricated in two portions which came together at a field joint (station 156.45). When assembled, the adapter was a semi-monocoque, truncated conical shell stiffened by circumferential frames and longitudinal stringers. The forward adapter was 28.13 in. long and 49.5 in. in diameter at its forward end (spacecraft mating surface). It was mated with the aft adapter through the 72-bolt field joint at its aft end. The aft adapter was 16 in. long with an aft diameter of 58.56 in. and a forward diameter of 53.16 in. (at the bolt circles). A 4.5-in.-diameter inspection port with a removable cover was provided in the center of the thermal diaphragm to permit inspection of the adapter interior and removal of any foreign material after encapsulation.

2. Spacecraft separation system. The spacecraft separation system consisted of (1) three mechanical separation latch mechanisms for securing the spacecraft to the upper payload adapter, (2) three sets of separation spring assemblies for providing initial spacecraft/*Centaur* separation, and (3) three linear potentiometers for separation measurement (Fig. VIII-12). Three sets of separation sensing and arming devices supplied by HAC and mounted on the spacecraft side of the interface completed the separation system. Bearing pads (or pedestals) for these devices were supplied and installed by GD/C on the payload adapter adjacent to the separation latches. (See Section XIV-D-2 for description.) The separation spring assemblies, linear potentiometers, and separation latch mechanisms (except the latch studs) were supplied and installed by GD/C. The latch studs were supplied by GD/C and installed on the spacecraft by HAC. Two design concepts for the spacecraft separation latch mechanisms were developed during the early part of the *Centaur/Surveyor* effort. The first concept, referred to as a hook-type latch, failed to operate satisfactorily during environmental testing early in 1965 and was abandoned in favor of the second design (known as the pinpuller latch). This latch mechanism successfully completed environmental testing and separation system test requirements. The pinpuller latch was used on all flights beginning with the *Centaur* AC-5 test flight. In addition to successful separation system tests, confidence that the separation system would not produce spacecraft tipoff rates in excess of the 3 deg/sec limitation (or cause spacecraft/*Centaur* collision) was gained by separation dynamics analysis performed by GD/C, JPL, and Lockheed Missiles and Space Co. The Lockheed analysis was performed under a separate contract with JPL.

The three separation latch mechanisms, except the latch studs, were installed on the upper face of the forward adapter section equally spaced on a 44.8-in.-diameter circle. Each separation spring assembly was mounted immediately adjacent to one of the latches on a 46.28-in.-diameter circle.

When the spacecraft was installed on the forward adapter, the three sets of previously installed separation springs were compressed and the spacecraft separation latches were secured. When the separation latch clevis pins were pulled by pyrotechnic squibs activated by *Centaur* programmer command, the compressed springs imparted a separation velocity of about 1 ft/sec to the spacecraft with less than 5 *g* shock and less than 3 deg/sec angular rotation.

3. Spacecraft electrical connector and in-flight separation mechanism. Except for the television target lights and retro-motor temperature sensor, all signal, power, and control circuits between the *Centaur* and the *Surveyor* (and between the GSE and the *Surveyor*) crossed the interface at the *Surveyor/Centaur* separation plane through a 52-pin electrical connector supplied by HAC. The connector receptacle portion was attached to the spaceframe by HAC and the plug portion was shipped to GD/C for attachment to a pyrotechnic separation mechanism. The separation mechanism, illustrated in Fig.

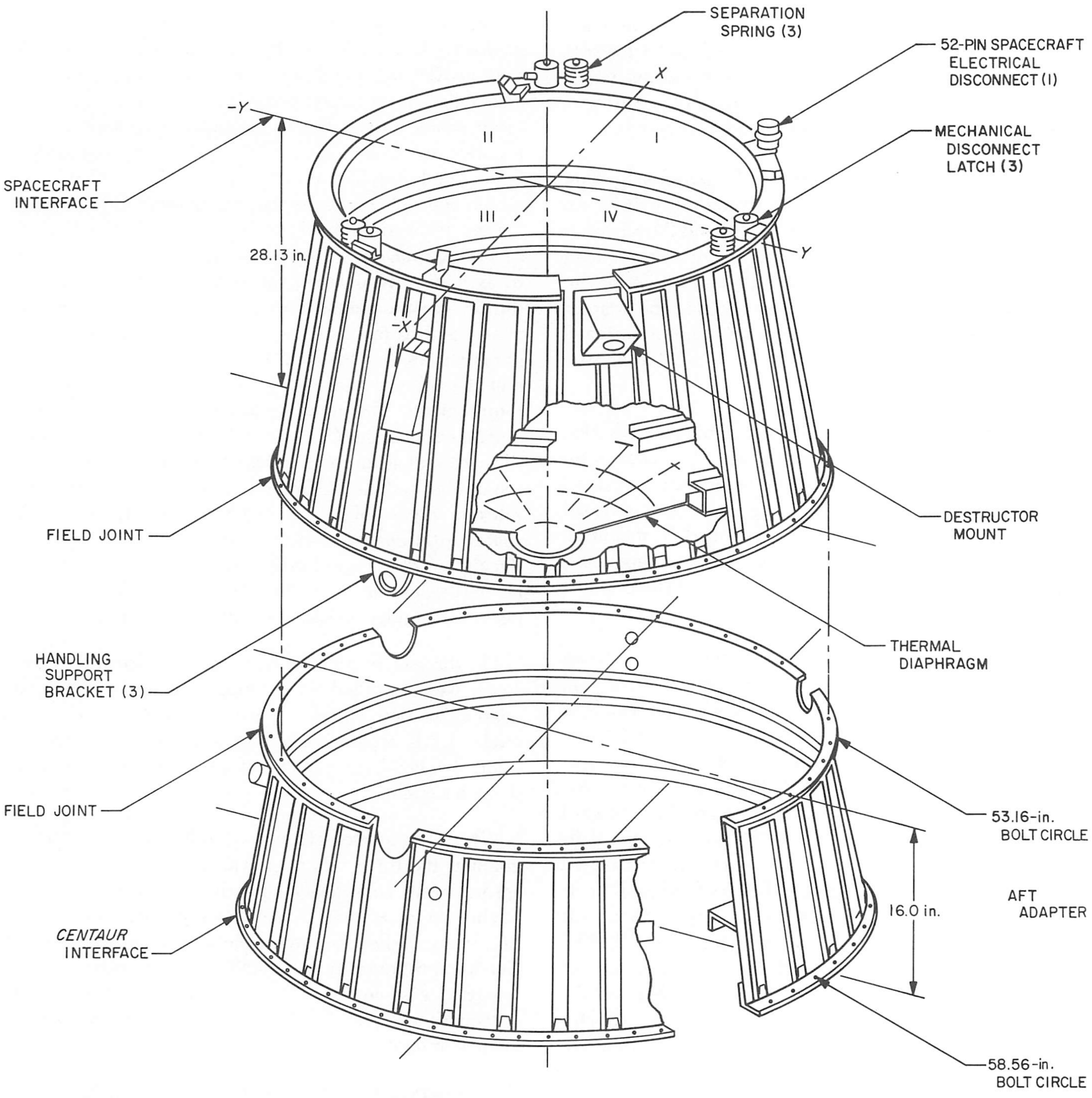


Fig. VIII-11. *Centaur/Surveyor* adapter

VIII-13, was mounted on the upper face of the forward adapter as shown on Fig. VIII-11. The circuits extended from the connector plug to the *Centaur* and GSE through an electrical harness attached to the exterior of the adapter, and then through another connector at the adapter field joint. Approximately 5.5 sec prior to spacecraft/*Centaur* separation, a *Centaur* programmer command fired the pyrotechnic charge in the mechanism, which released a coil spring to retract the connector plug and break the electrical connection between the spacecraft and the *Centaur*. The design of the separation mechanism would have allowed spacecraft/*Centaur* separation to occur even if the separation mechanism failed.

The spacecraft electrical connector receptacle and plug were joined during the spacecraft/forward adapter

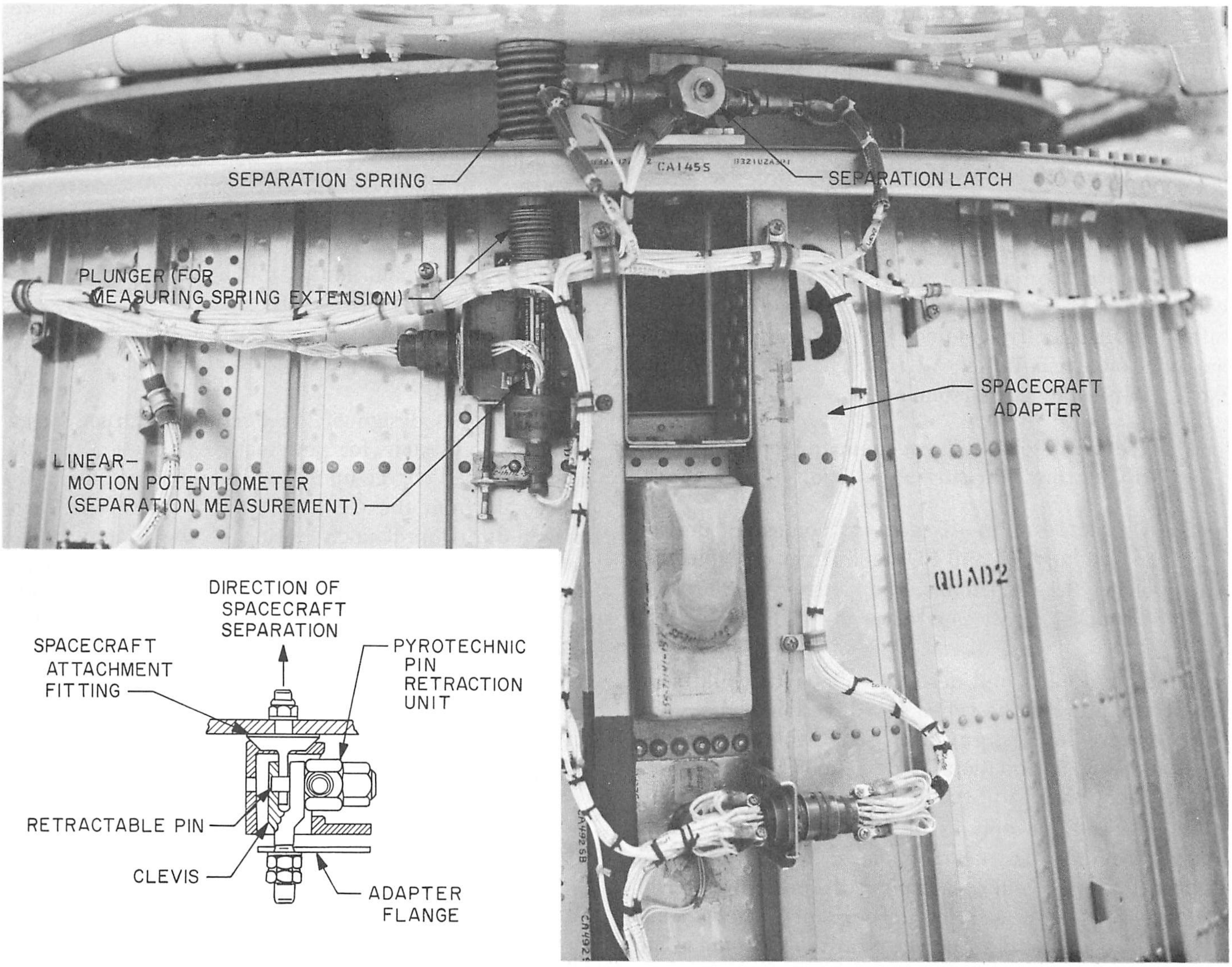


Fig. VIII-12. Payload separation mechanism and instrumentation

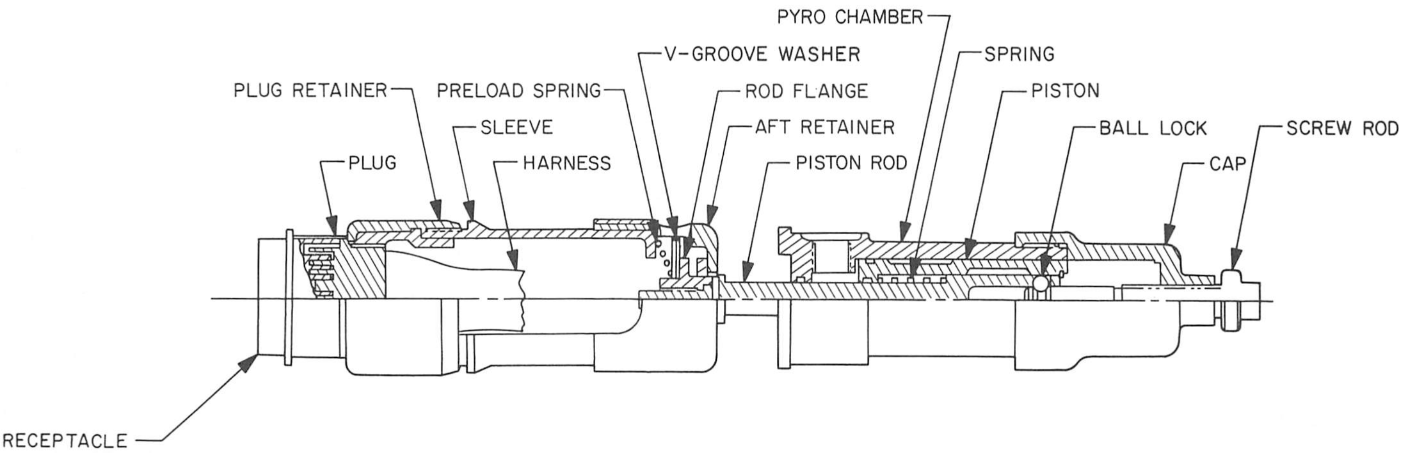


Fig. VIII-13. Electrical connector separation mechanism

mating operations. The electrical connector at the adapter field joint was joined during the spacecraft/*Centaur* mating operations. A motor-driven switch located at the unregulated battery voltage output of the spacecraft was used to interrupt spacecraft battery power to prevent accidental shorting of the connectors during mating operations.

4. *Surveyor destruct system.* The *Surveyor* destruct system was part of the *Centaur/Surveyor* range safety command system which was designed to meet the range safety requirements contained in the AFETR Range Safety Manual 127-1 (Ref. VIII-3). The functions of the range safety command system were:

(1) In response to an RF *arm* command, effect *Centaur* MECO, thus imposing a condition of zero thrust.

(2) In response to an RF *destruct* command, destroy the *Centaur* LH_2 and LO_2 tank structures, thus dispersing the propellants, and destroy the *Surveyor* retrorocket thrust capability.

The separate arm and destruct commands could have been transmitted to the vehicle by direction of the range safety officer who was provided visual displays of vehicle flight tracking information. If deviations from the prescribed flight path had become evident, this officer could have exercised the options of MECO only or MECO followed by the destruct command. When it became evident that the vehicle achieved its prescribed flight path or orbit, the range safety officer sent a *disable* command which removed all power from the *Centaur* range safety command system, thus preventing inadvertent MECO or vehicle destruction. The first-stage *Atlas* range safety command system was completely independent from the *Centaur* and *Surveyor* systems.

The range safety command RF carrier from the ground was received by one or both of two antennas mounted on opposite sides of the *Centaur* tank. The signal was conveyed from antennas through a junction to the range safety command receivers, where it was converted to 28-VDC commands. These commands were conveyed via relays to their destinations:

(1) The MECO command to the *Centaur* engine prestart circuits.

(2) The destruct command to the *Centaur* destructor and to the *Surveyor* safe–arm initiator (which initiated a pyrotechnic train to ignite the conical shaped charge).

(3) The RF disable command to the *Centaur* power changeover switches within the power control unit to remove power from the range safety command system.

During flight, power was provided by two batteries connected to the receivers and command circuits. The conical shaped charge was designed to destroy the *Surveyor* retrorocket by an explosive jet that was designed to bore a 2-in. hole through the retrorocket case and solid propellant (Fig. VIII-14). The *Surveyor* safe–arm initiator permitted electrical arming and either electrical or mechanical safing.

The original design of the *Surveyor* destruct system included provision for automatic detonation of the conical shaped charge upon inadvertent separation of the spacecraft from the *Centaur*. As a result of problems arising during prelaunch testing of the *Centaur* AC-8 vehicle and further analysis of the need for the automatic destruct feature in relation to range safety requirements, approval was obtained from the AFETR range safety office to remove the inadvertent-separation destruct system, and it was not utilized on *Surveyor* flights.

c. Flight instrumentation, telemetry, and command signals.

1. *Instrumentation and telemetry.* The *Atlas/Centaur* flight instrumentation and telemetry system was designed to obtain and transmit environmental data and system operational measurements for assessment of launch vehicle and spacecraft performance during the launch phase of flight. Before liftoff, a hardwire instrumentation system was designed to furnish measurements of key parameters during prelaunch testing and countdown.

The determination of flight environmental conditions as well as spacecraft performance was of major importance to *Surveyor* development. In recognition of the environmental data requirement, *Centaur* research and development vehicles, particularly those flights carrying dynamic and mass-model payloads (AC-5, AC-6, AC-8, and AC-9), were instrumented to obtain data on the environment which could have affected the *Surveyor* spacecraft as well as the launch vehicle.

Although the *Surveyor* S-band telemetry system transmitted on low power during the boost phase of flight, it was agreed between JPL and LeRC that the primary link for spacecraft data during this period would be provided by the *Centaur* telemetry system in order to assure satisfactory receipt of spacecraft data before injection.

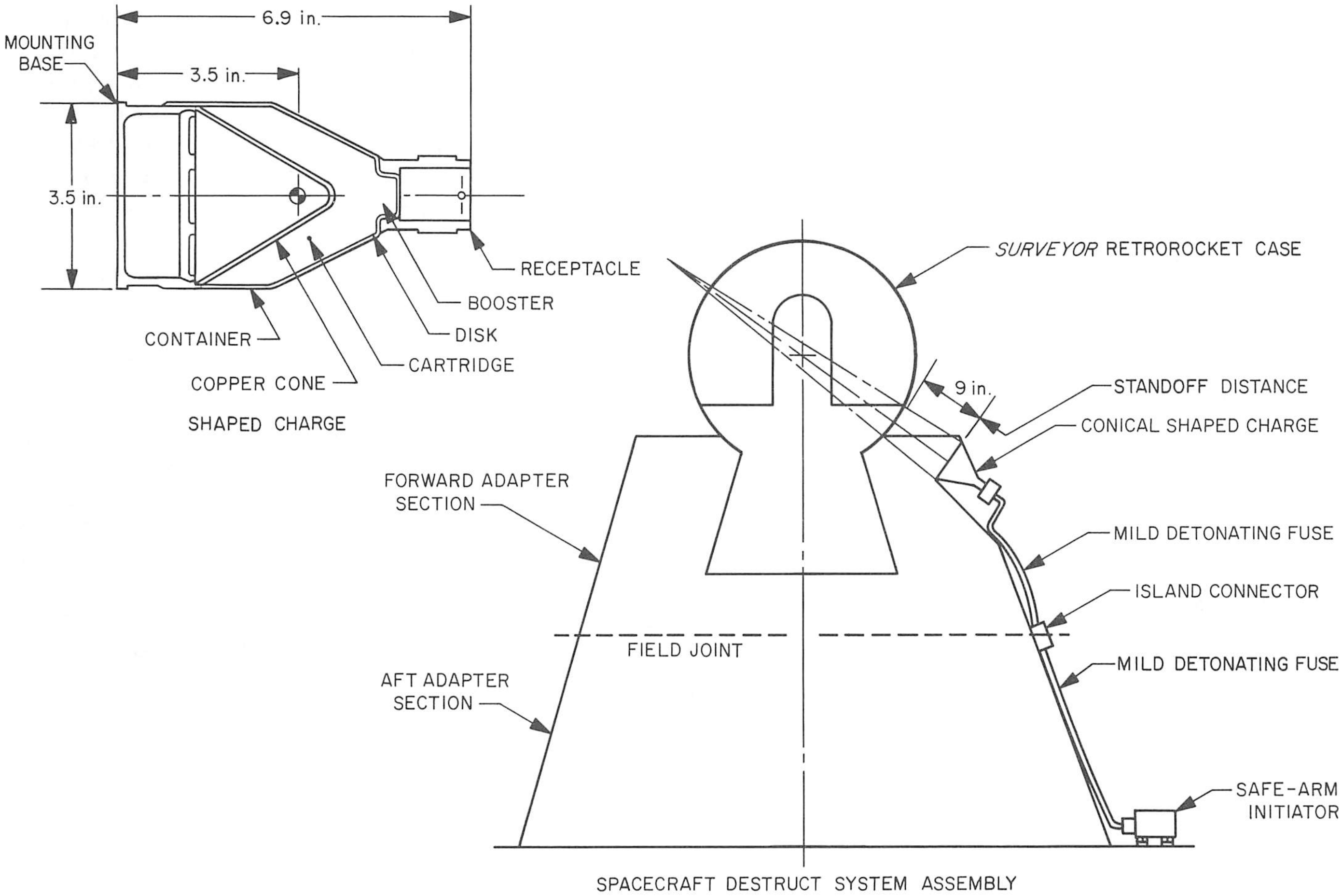


Fig. VIII-14. Spacecraft destruct system

Further negotiations, which considered the in-flight instrumentation and telemetry requirements for both the *Surveyor* and the launch vehicle system, resulted in the allocation of three *Centaur* telemetry channels for transmission of the spacecraft composite signal and payload area vibration measurements.

The *Centaur* telemetry system, provided to satisfy the combined spacecraft and launch vehicle requirements, was powered by the *Centaur* main battery during flight (from an external 28-VDC source prior to flight) and used one 18-channel telepak, one antenna, and associated wiring and connectors. One channel was used on all flights for continuous transmission of the spacecraft composite signal received from the spacecraft signal processor.

For *Surveyors I* through *IV*, the second channel provided continuous transmission of the data from one spacecraft-mounted accelerometer, and the third channel provided commutated (time-divided) data from three spacecraft-mounted accelerometers and one accelerometer mounted on the payload adapter. For *Surveyors V* through *VII*, two *Centaur* telemetry channels were used to transmit continuous vibration data from two accelerometers mounted on the payload adapter. For *Surveyor VII*, an extra-high-frequency accelerometer was installed on the payload adapter and transmitted over the *Atlas* telemetry system as a continuous measurement until *Atlas* separation. (See Subsection F-3-e for flight accelerometer locations and measurement data.)

At spacecraft electrical disconnect on all flights, the three *Centaur* telemetry channels assigned to the spacecraft were switched to transmit the spacecraft separation-potentiometer data.

2. *Command signals.* The spacecraft required four commands to prepare it for separation from the *Centaur*. These four commands were:

(1) A preseparation arming command to enable the spacecraft to act on the following three commands.

(2) An extend-landing-gear command.

(3) An unlock-omniantenna command.

(4) A transmitter high-power command, simultaneously with a preseparation arming off command.

These four commands were provided as discretes from the *Centaur* programmer through the spacecraft electrical connector. At 5.5 sec after issuance of the command to turn on the spacecraft high-power transmitter, the programmer issued a command to separate the spacecraft electrical connector, followed 5.5 sec later by the *Centaur/Surveyor* separation command. The sequence of the preseparation and separation commands is illustrated in Fig. VIII-15.

d. Interface testing. This subsection describes the special interface tests conducted during development of the spacecraft/launch vehicle interface.

1. Match–mate tests. Surveyor/Centaur match–mate tests were conducted jointly by HAC and GD/C at HAC, El Segundo, between May 7 and May 24, 1963. The tests consisted of full-scale mechanical assembly and mating checks of airborne and ground-support interface hardware. These tests were performed for the purpose of proofing the procedures and verifying the system design. Preliminary fit and procedure checks were first performed using the M-12 model spacecraft mockup, a mockup of the *Centaur* forward bulkhead and upper electronic-equipment tier, a prototype nose fairing, a prototype payload adapter with a spacecraft/*Centaur* separation system (using hook latches) and spacecraft electrical connector, the GTV-2 *Surveyor* transport trailer, and the necessary work stands and handling equipment.

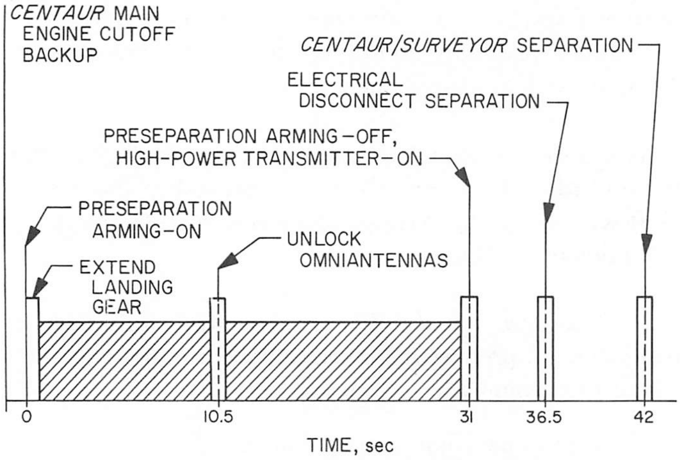


Fig. VIII-15. Spacecraft/*Centaur* separation sequence commands

The preliminary checks included mating the unencapsulated M-12 spacecraft and forward adapter to the positioned aft adapter on the *Centaur* forward bulkhead, assembling the nose fairing torus ring sections to the handling carts, positioning the nose fairing halves in the torus ring sections, encapsulating the spacecraft, and mating it to the *Centaur* (Figs. VIII-16 and -17).

The match–mate test was conducted using the prototype model T-21 *Surveyor* spacecraft. This was the first time that a full-weight spacecraft was encapsulated and installed in a flight position on the *Centaur*. During the test, clearance checks were made between the spacecraft and the *Centaur*, with the spacecraft in both the encapsulated and separated configuration, and a check was made of spacecraft television camera illumination. A dynamic road test of the *Surveyor* GTV-2 transport trailer was also performed during the match–mate test phase. For this test, the GTV-2 was loaded with the nose fairing and a 2100-lb weight (of the same center of gravity and bulk as a spacecraft) and was towed over a maneuvering course for a distance approximating the distance from the AFETR explosive-safe facility to the launch complex. Fifteen accelerometers (mounted at various locations on the trailer wheels, trailer bed, and spacecraft separation plane) gathered information for use in verifying spacecraft and trailer design loads.

Significant equipment and procedure modifications resulting from the match–mate tests were as follows:

(1) Redesign and stiffening of the separation sensing and arming device pedestals.

(2) Rerouting of the adapter wiring harness.

(3) Relocation of the adapter support-ring positioners.

(4) Revision of the forward adapter and torus-ring alignment procedures.

(5) Replacement of the hydraulic jacks on the GTV-2 transport vehicle with mechanical jacks.

(6) Addition of a steel top to the plywood bed of the GTV-2.

(7) Relocation of the torus-ring handling-cart alignment rollers to the outside of the alignment rail.

(8) Revised television target and light locations.

(9) Replacement of the GTV-2 steering mechanism with an improved cam-override type.

(10) Revision of encapsulation and mating procedures.

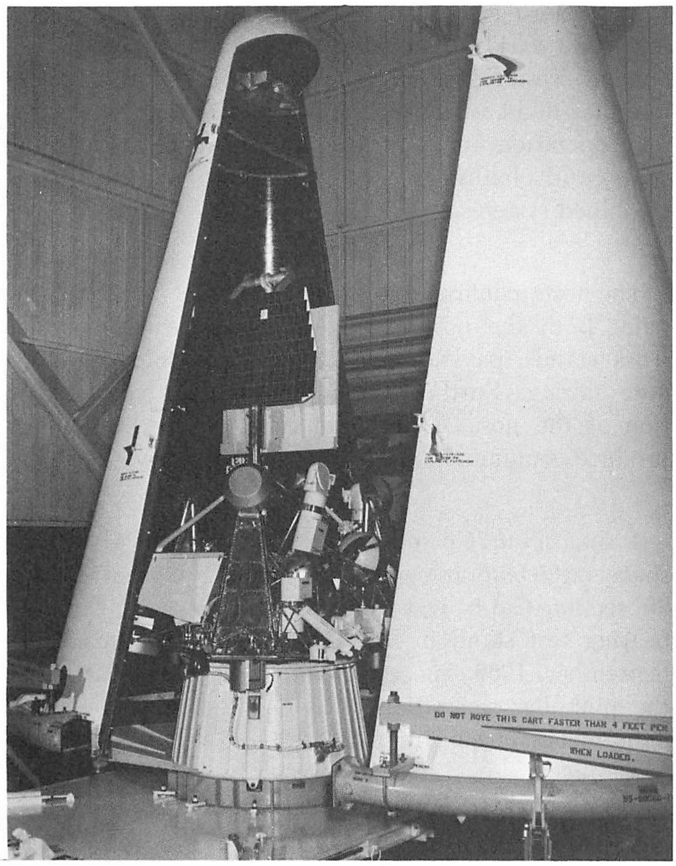


Fig. VIII-16. Spacecraft encapsulation

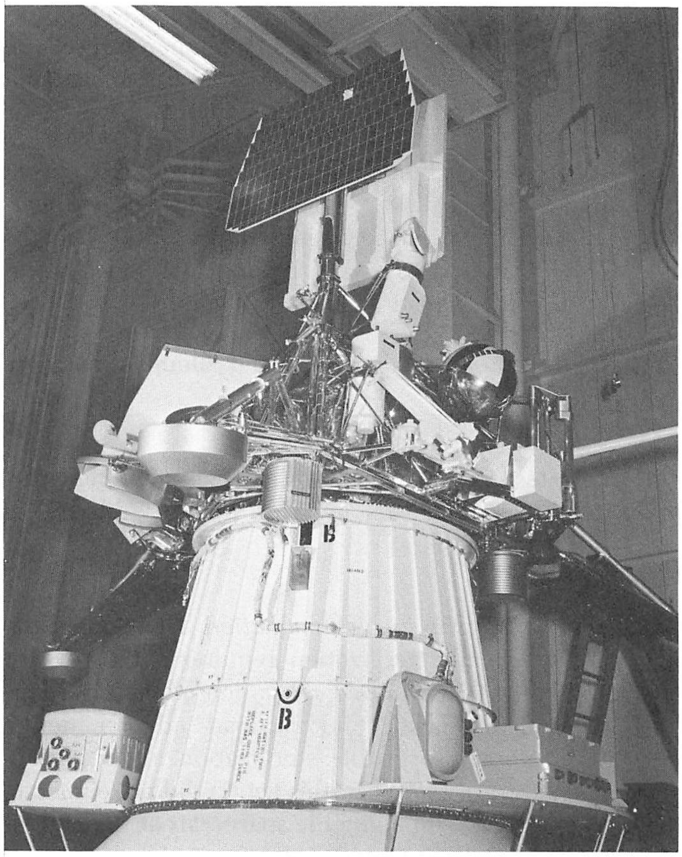


Fig. VIII-17. Spacecraft mated to *Centaur*

2. Air-conditioning tests. The *Surveyor/Centaur* air-conditioning tests were conducted jointly by HAC and GD/C at the HAC Space Environmental Laboratory, El Segundo, Calif., from September 23 through October 31, 1963. The objectives of the tests were to:

(1) Verify that the spacecraft air-conditioning equipment (designed by GD/C for installation at the AFETR launch complex and at the combined-system test stand) was mechanically compatible with the nose fairing.

(2) Verify that the equipment would properly respond to the air-conditioning requirements of the encapsulated spacecraft.

(3) Verify that the nose fairing and adapter ducting were properly sized and oriented.

(4) Determine the optimum location of the temperature control sensor.

(5) Obtain spacecraft thermal response data.

Following the air-conditioning tests, TV target illumination checkout and road tests were completed (in addition to those conducted during the match–mate tests). Except for the use of a HAC-supplied thermal model spacecraft and *Centaur* electronic-package air-conditioning units, the equipment used for the air-conditioning test was the same as that used during the match–mate tests. The encapsulated spacecraft was mounted on the *Centaur* forward bulkhead mockup and placed inside a special environmental chamber which was environmentally controlled by a separate air conditioner to simulate varying conditions expected at the AFETR.

The air-conditioning tests were successful and all of the test objectives were met. The more significant conclusions and equipment modifications were:

(1) The GD/C air conditioner was capable of satisfying the spacecraft requirements through the full range of expected ambient environment.

(2) A temperature sensor located in the air inlet duct near the entrance at the nose fairing was a suitable location for effecting temperature control.

(3) Modifications were made to the internal nose fairing duct to obtain satisfactory temperature conditioning of the spacecraft inertial reference unit and flight control electronic equipment.

(4) The thermal bulkhead was modified to prevent excessive backflow, which would have been characterized by higher temperature air from the spacecraft compartment flowing into the *Centaur* forward bulkhead area.

(5) Television-target locations and illumination were satisfactory.

Towing test results were satisfactory, including operation of the steering cam override feature added after the match–mate tests.

3. *Nose fairing cavity contamination.* Deletion of the *Surveyor* spacecraft sterilization requirement by NASA in December 1962 permitted relaxation of the nose fairing, thermal bulkhead, and payload adapter design features necessary to enable sterilization of the encapsulated spacecraft. However, this action did not eliminate the necessity for maintaining the spacecraft at the highest possible degree of cleanness at all times. After spacecraft encapsulation, contamination could accumulate from: trapped metal, paint, and epoxy particles in the nose fairing and payload adapter; particles entering the nose fairing cavity through the air-conditioning system; particles jarred loose at nose fairing jettison; ingestion into the nose fairing cavity of particles produced by sublimation of of the outer protective coating; and smoke emanating from the spacecraft separation squibs. Particle contaminants could occult the spacecraft sun sensor, cause loss of Canopus lock, degrade television operation, and be injurious to the operation of other critical guidance units.

Rigid specifications and procedures were established for controlling cleanness, including those involving cleanroom operation, cleaning materials and methods, equipment material, and operations and inspection.

Qualitative tests were run to establish the level of contamination of the spacecraft which might have developed during prelaunch and launch operations up to the point of nose fairing separation. These tests were conducted at the AFETR during December 1965, using a test nose fairing and spacecraft mockup for one phase and the AC-7 nose fairing for a second phase. The particle type and count obtained during these tests (as well as during separation system outgassing tests performed at various times during separation system development) resulted in improved cleaning procedures for interface flight hardware, improved air-conditioning system operation, more efficient separation latch squib seals, and cleanness-procedures modification to the combined systems test stand.

The tests confirmed the desirability of removing the paint from the nose fairing handling gear, cleanroom work stands, payload adapter, and the interior of the nose fairing. Further study determined that sublimation of the nose fairing protective coating would not produce contamination of the payload cavity.

4. *Spacecraft/Centaur separation system tests.* The spacecraft/*Centaur* separation system, as described in Subsection C-3-b, was system-tested on the GD/C air-bearing test stand in San Diego between January and September, 1965. Successful initial design evaluation of the pinpuller separation latch enabled removal of this item as a launch constraint against the *Centaur* AC-5 flight.

Design refinement of the separation switch and further system testing indicated satisfactory separation-system operation, and the system performed successfully on all *Surveyor* flights.

Testing of the total separation system was performed using the HAC T-1 full-scale model of the *Surveyor*, weighted and balanced for proper weight, moment of inertia, and center-of-gravity location. The model was suspended at its center of gravity by a knife-edge/airspring/air-bearing system to simulate a zero-gravity condition during separation from a fixed payload adapter. The test configuration is shown in Fig. VIII-18. Linear-motion transducers and oscilloscope recordings were used to monitor the movement of the three jettison springs. The separation event was determined by recordings of separation switch activation. Oscilloscope recordings were also made of the voltage and current at squib firing. Separation-latch loads were determined by strain-gage readings, and this information was used for later preload adjustment of the separation springs. A common 100-Hz triangular timing wave and a common 1-sec pulse was applied to all recorders.

The average axial separation velocity, measured with a side-view camera, was 8.91 in./sec. The pitch and yaw rates were well below the ±3 deg/sec allowable limits.

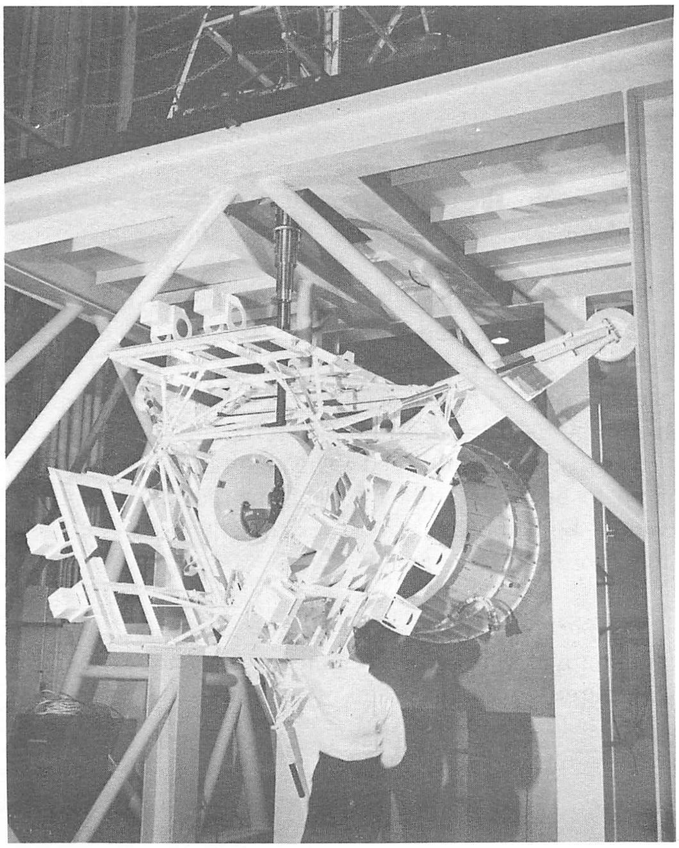


Fig. VIII-18. Spacecraft separation system test configuration

5. *Nose fairing jettison and pressure transient tests.* The *Surveyor* nose fairing jettison system was qualified for the *Surveyor* missions by 24 jettison tests conducted by LeRC and GD/C between mid-1963 and March 1966. The first of these was a series of full-scale jettison and seal-evaluation tests of the AC-3 nose fairing performed at the GD/C jettison test stand at Point Loma, Calif., before the AC-3 flight. The test stand permitted an average upward axial acceleration of 16.1 ft/sec^2 of a 3100-lb carriage. The jettisoned fairing halves were caught in an octagonal-shaped net having a point-to-point maximum dimension of 140 ft. These tests confirmed that the jettisoned fairing halves would clear the spacecraft envelope and *Centaur* forward equipment area, and that their trajectories would not result in collision with the *Centaur* or *Atlas* stages.

Also, before the AC-3 flight, nose fairing unlatch tests were conducted from a vertically mounted *Centaur* test tank filled with cryogenic propellants. These tests were conducted to verify that ice buildup in the area of the nose fairing hinges would not interfere with proper jettison.

Problems detected during the nose fairing jettison sequence of the AC-3 flight resulted in redesign of the nose fairing and jettison system. Postflight simulation of the AC-3 jettison sequence and jettison tests of the modified nose fairings were then conducted in the LeRC space power chamber both at sea-level pressure and at a simulated altitude up to 380,000 ft. These tests included investigations of an overlapped thruster bottle deflector bulkhead and a proposed removable nose fairing cap. In addition, measurements were obtained for determination of the pressure transients resulting from release of the thruster bottle pressure. An excessive pressure buildup within the nose fairing cavity at jettison could cause damage to the spacecraft solar cells, solar panel, and other equipment. A maximum pressure of 0.19 psi, which was in excess of the allowable 0.15 psi, was recorded at the bottom of the planar array.

Further modifications to the nose fairing and jettison system were made, and testing continued in the LeRC space power chamber. Loads and spring constants were obtained for the nose fairing hinge and *Centaur* station 219 attach ring, and additional nose fairing jettison pressure data was obtained. A nose fairing functional test, successfully completed in October 1964, demonstrated proper jettisoning of the redesigned nose fairing at a simulated altitude of 350,000 ft, and removed a constraint against the AC-4 flight test. An acceptable peak pressure of 0.15 psia, with a rise time of 5 msec, was measured along the roll axis immediately below the deflector bulkhead.

The evaluation test of the nose fairing (as configured for the AC-6 direct-ascent research and development flight test) was completed in mid-1965 at the LeRC space power chamber. This test successfully demonstrated that the nose fairing/interstage adapter flexible linear-shaped charge, separation latches, and thruster bottles would separate and jettison the nose fairing at simulated jettison altitudes and that all hardware would sustain the jettison shock loads without failure or fragmentation.

The last nose fairing jettison test was completed at GD/C on March 25, 1966, and successfully demonstrated proper operation of the balanced *Centaur* hydrogen vent system disconnect during nose fairing separation. The test was performed at sea-level altitude with a *Centaur* test tank loaded with cryogenic propellants and removed a constraint against the AC-8 flight.

6. *Electromagnetic interference tests.* Electromagnetic compatibility between the *Surveyor, Centaur,* and GSE

systems was confirmed during a series of electromagnetic interference (EMI) tests conducted at the CSTS and at the AFETR between May 1965 and March 1966, as discussed in Section X.

7. *Alpha scattering instrument purge system tests.* The alpha particle sources and backscatter detectors within the alpha scattering instrument carried aboard *Surveyors V* through *VII* would have been irreversibly degraded if subjected to moisture contamination. Consequently, a dry nitrogen purge of the instrument was required at all times after installation of the sources and detectors up to the time of liftoff. JPL and HAC were responsible for supplying, monitoring, and maintaining all elements of the purging system. The equipment was installed at the launch complex by LeRC and GD/C personnel as depicted in Fig. VIII-19.

Impoline tubing conducted the gaseous nitrogen (GN_2) from the GD/C-supplied disconnect on the nose fairing cylindrical section through the thermal bulkhead and up the inside of the conical section to a level opposite the alpha scattering instrument. A section of impolene tubing, coiled to obtain a spring action and fitted over a nipple at the instrument, completed the GN_2 flow path.

After encapsulation, and prior to mating the spacecraft to the *Centaur* on the launch pad, instrument purging was accomplished by mounting a GN_2 supply bottle, regulator, and flowmeter on the nose fairing torus ring.

Tests of the purge system using the torus ring installation were conducted during the *Surveyor V* combined systems tests. Personnel were trained in the details of making purge-line connections, and the requirement for a tubing-to-instrument pulloff force of 3.5 lb (+2.5, −1.0 lb) was established. This limitation on the force used to disconnect the tubing from the instrument prevented both damage to the instrument and interference with nose fairing jettison.

8. *Spacecraft vernier engine purge system tests.* Beginning with *Surveyor III* launch operations, a dry nitrogen purge system was used in conjunction with the toxic

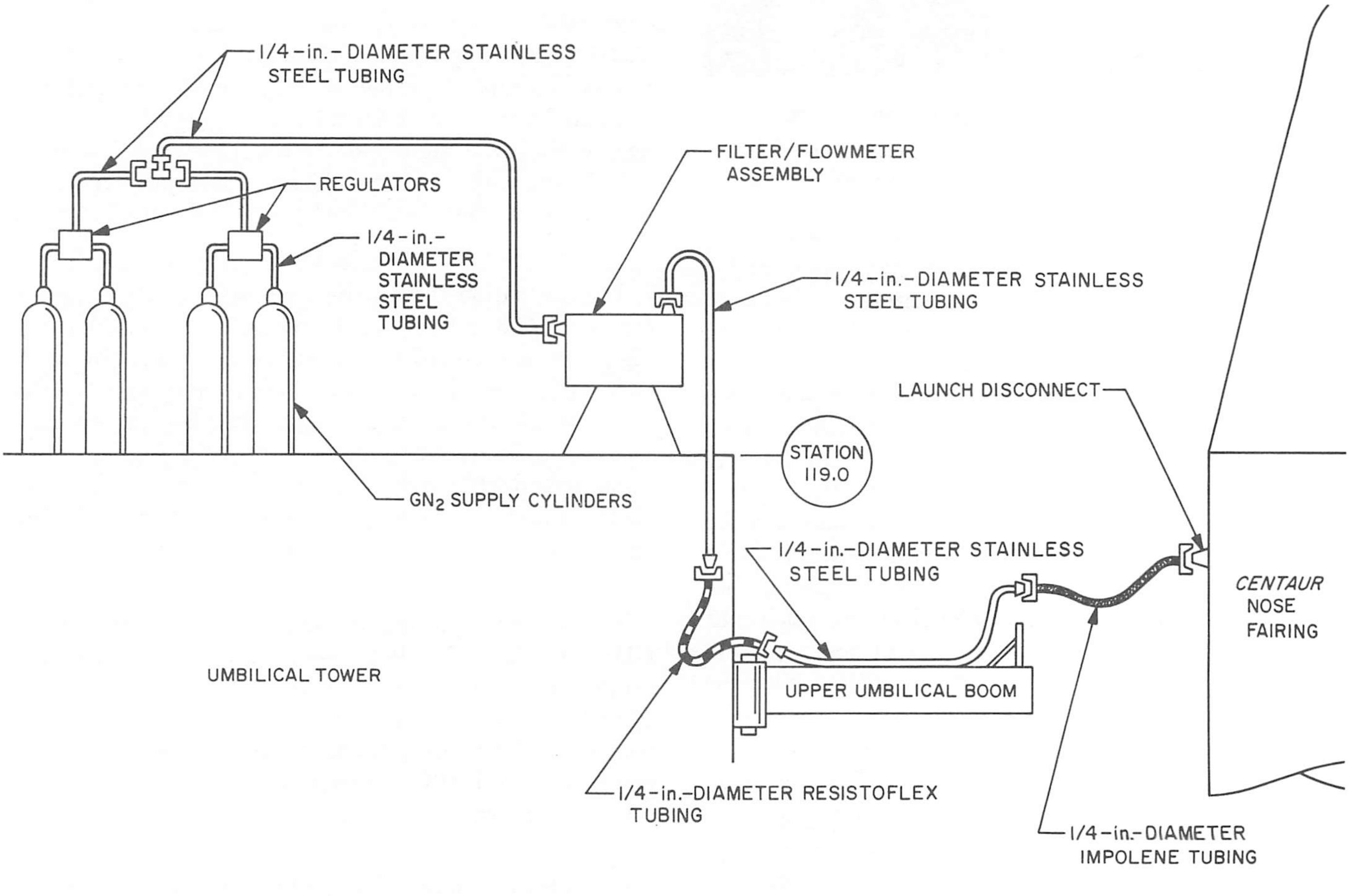


Fig. VIII-19. Alpha scattering instrument gaseous nitrogen purging system

gas detection system for purging the spacecraft vernier engine thrust chambers after encapsulation to ensure a dry atmosphere and prevent the formation of salts resulting from any propellant leakage. A spool-shaped flexible plug was inserted into the throat of each vernier engine nozzle. Two plastic tubes were routed from each plug through the thermal bulkhead and a vent port in the fairing cylindrical section to the toxic gas detector system. One tube connected the GN_2 supply to the engine nozzle and the other provided a vacuum sampling line between the engine and the gas detector. Prior to tower removal, at $T-175$ min in the launch countdown, the lines were pressurized to eject the plugs from the engine nozzles. The plugs and tubing were then manually threaded free of the vehicle. JPL and HAC furnished all fittings, tubing, regulators, filters, and a dewpoint indicator. Installation and removal of the plugs, tubing, and fittings was accomplished jointly by HAC and GD/C.

A plug diameter of 0.585 in. and an ejection pressure of 70 psi were established during plug blowout tests at HAC in January 1967. System installation, operation, and removal tests were accomplished during the encapsulation operations prior to the J-FACT of SC-3/AC-12. In addition, procedures for reinstallation of the purge system in the event of a launch delay exceeding 24 hours were prepared and tested prior to the launch of SC-4/AC-11.

9. Surveyor prelaunch emergency abort procedures. Emergency procedures were developed for downloading the spacecraft hypergolic propellants and high-pressure gases in the event of a propellant leak after the encapsulated spacecraft was mated to the *Centaur*. The procedure involved gaining quick access to the propulsion system through the nose fairing by an abort team to relieve propellant tank pressure and remove the propellants. Outlines of emergency access ports were painted on the exterior of the nose fairing to enable the abort team to cut the ports using pneumatically powered sabre saws. The decision to allow personnel to enter the service tower and perform the propellant downloading operation was to be made on the basis of the extent of propellant leakage as determined by the toxic gas detector system.

Calibration of the toxic gas detector system to enable determination of emergency actions was performed during the nose fairing cavity contamination tests. Emergency abort procedures, necessary equipment, and the location and size of the nose fairing access ports were formulated during tests at the AFETR using a plywood mockup of the nose fairing. These decisions were made preceding operations with the T-21 prototype spacecraft.

10. T-21 prototype spacecraft interface tests. Before the AC-10/SC-1 combined systems test stand and AFETR operations, the interface compatibility between the spacecraft and launch vehicle was demonstrated, and validation of the GSE was accomplished by joint testing, including simulated countdown and launch at the two locations using the T-21 prototype spacecraft. The combined systems test was accomplished using T-21 and the AC-7 launch vehicle, while the AFETR testing utilized the T-21, *Centaur* mechanical and electrical simulators, and Launch Complex 36A. (See also Table VIII-7.)

The *Centaur* mechanical simulator provided by GD/C supported the spacecraft in the service tower of Launch Complex 36A in lieu of mating the spacecraft to a *Centaur* vehicle. Steel I-beams and plywood decking supported the spacecraft, mounted in the torus ring, approximately 2½ feet above the normal *Centaur*-mated position. The support enabled operation of the RF link and connection to the air-conditioning ductwork and sensing lines.

The *Centaur* electrical simulator, also provided by GD/C, furnished the spacecraft with manually sequenced signals which were equivalent to those normally provided by the *Centaur* programmer. The electrical simulator included a junction box with telemetry output termination simulating the *Centaur* loads and necessary interconnecting cables and connectors.

D. Facilities and Support Equipment

1. Combined Systems Test Stand

a. Evolution. The *Surveyor* development concept included techniques for assuring maximum space vehicle system reliability by an extensive design verification test program. A major step which was implemented in the test program was the conductance of a combined systems test (CST). In this test a spacecraft and the launch vehicle were mated for the first time to test the mechanical, electrical, and RF compatibility of the combined system in a preliminary test of combined operation and countdown procedures. The CST was to be performed first with a prototype spacecraft, followed by tests with an actual flight spacecraft (only the SC-1, SC-4, and SC-5 spacecraft in addition to T-21 were tested).

A description of the CST and a proposal for a facility to support the test was presented to MSFC and GD/C by JPL on September 29, 1961. A technical working group composed of representatives from MSFC, JPL, and GD/C was formed to review the technical aspects of aligning JPL CST requirements with NASA testing concepts. Activities of the working group resulted in a proposal by GD/C on April 3, 1962, for a facility to accomplish a CST of an *Atlas/Centaur*/spacecraft complement. This original proposal included a 50 × 62 × 200-ft test tower for vertical erection of the entire vehicle, a separate 50-ft *Centaur* test tower for factory selloff, a control building, and spacecraft assembly and checkout areas. The ground support equipment and its interface was to be as near a duplicate of Launch Complex 36 as possible (including the actual launcher, swing booms, and blockhouse equipment). An alternate proposal was also considered which included two forms of an interim CST:

(1) The first proposal was one in which the entire vehicle, including the spacecraft, would be mated in a horizontal position (rejected by HAC and JPL).

(2) In the second proposal, the *Atlas* and *Centaur* would be mated in a horizontal position with the spacecraft in an adjoining area connected electrically to the *Centaur*.

In June 1962, GD/C was given the go-ahead to install an interim facility at San Diego in accordance with the second proposal. Since an interim CST would not provide a complete combined test, particularly in the areas of spacecraft/launch vehicle mating and RF compatibility, and since a permanent test stand (CSTS) would not be ready for the first two *Surveyor* flights, a proposal was made by JPL to LeRC on June 20, 1963, to conduct a joint operational compatibility test (J-OCT) at Launch Complex 36A utilizing the T-21 engineering prototype spacecraft and the AC-6 research and development vehicle. Such a test would demonstrate operational readiness of the spacecraft/launch vehicle/launch complex system before the first *Surveyor* mission. This proposal was not implemented due to program delays. Preliminary J-OCT of the *Surveyor/Atlas/Centaur* and support-equipment validation at the CST was accomplished using the T-21 spacecraft and the AC-7 launch vehicle. This was followed by a joint test at Launch Complex 36A using the T-21 spacecraft and a *Centaur* simulator (see Subsection C-3-d). The interim CSTS was utilized by GD/C for *Atlas/Centaur* compatibility testing of AC-5, AC-6, AC-8, and AC-9.

The go-ahead to proceed with design of the facility was issued to GD/C by LeRC in August 1963. By this time, the design had evolved from the vertical-test-tower concept to a smaller facility in which the *Atlas* would be installed in a horizontal position adjoining the vertically mated *Centaur* and spacecraft. Construction of the CSTS in San Diego began in February 1964 by GD/C. The facility was completed and accepted by NASA, including the installation and validation of facility and launch vehicle ground-support equipment (GSE), on January 29, 1965. The *Surveyor* GSE, consisting of the operations console, control rack, and system test equipment assembly (STEA), was installed and electrically connected by GD/C during March and April 1965 in time for the T-21 operation, which began on May 12, 1965.

Overall operational responsibility of the NASA CSTS was vested in GD/C. JPL and HAC were responsible for spacecraft activities, GD/C for launch vehicle activities (including conductance of the CST of the *Atlas/Centaur/Surveyor*). GD/C was also responsible for the joint spacecraft handling, encapsulation, and decapsulation operations. Operational support of the spacecraft operations was furnished by GD/C, including the provisioning and allocation of building space, power, communications, lighting, timing codes, air conditioning, calibration of test equipment, and maintenance services.

b. Operational configuration. The CSTS shown in Figs. VIII-20 and VIII-21, consisted of five functional areas: *Surveyor* assembly and test, *Atlas* first stage, test tower, an operations floor (similar to AFETR launch complex blockhouse) and the support areas.

The *Surveyor* assembly and test area (high-bay area), was utilized for spacecraft assembly, test, inert retro-rocket installation, and encapsulation–decapsulation activities. The high-bay area contained provisions for mounting a *Surveyor* test stand for positioning the spacecraft in azimuth and elevation during functional testing. This stand was used during the T-21 and SC-1 CSTs only. The spacecraft STEA and the programmed automatic telemetry evaluator (PATE) were located in a low-bay portion of the area during the T-21 and SC-1 tests. In order to effect more efficient utilization of this equipment, it was removed from the CSTS and installed in a trailer outside the building for SC-4 and SC-5 testing. Prior to the T-21 operations, several modifications were made to improve cleanness control in the spacecraft assembly area. The modifications included: (1) addition of sealants to the floor, wall, and ceiling; (2) installation of improved door seals; (3) addition of a GSE airlock with vacuum-cleaning equipment; and (4) the addition of a personnel airlock and access area with air shower, shoe cleaner, and clothing locker.

The *Atlas* first stage was installed horizontally on a test stand, as shown in Fig. VIII-20, and mated electrically with the *Centaur* through the interstage adapter by means of jumper cables. The test stand permitted gimbaling, automatic vehicle tank pressure regulation, and

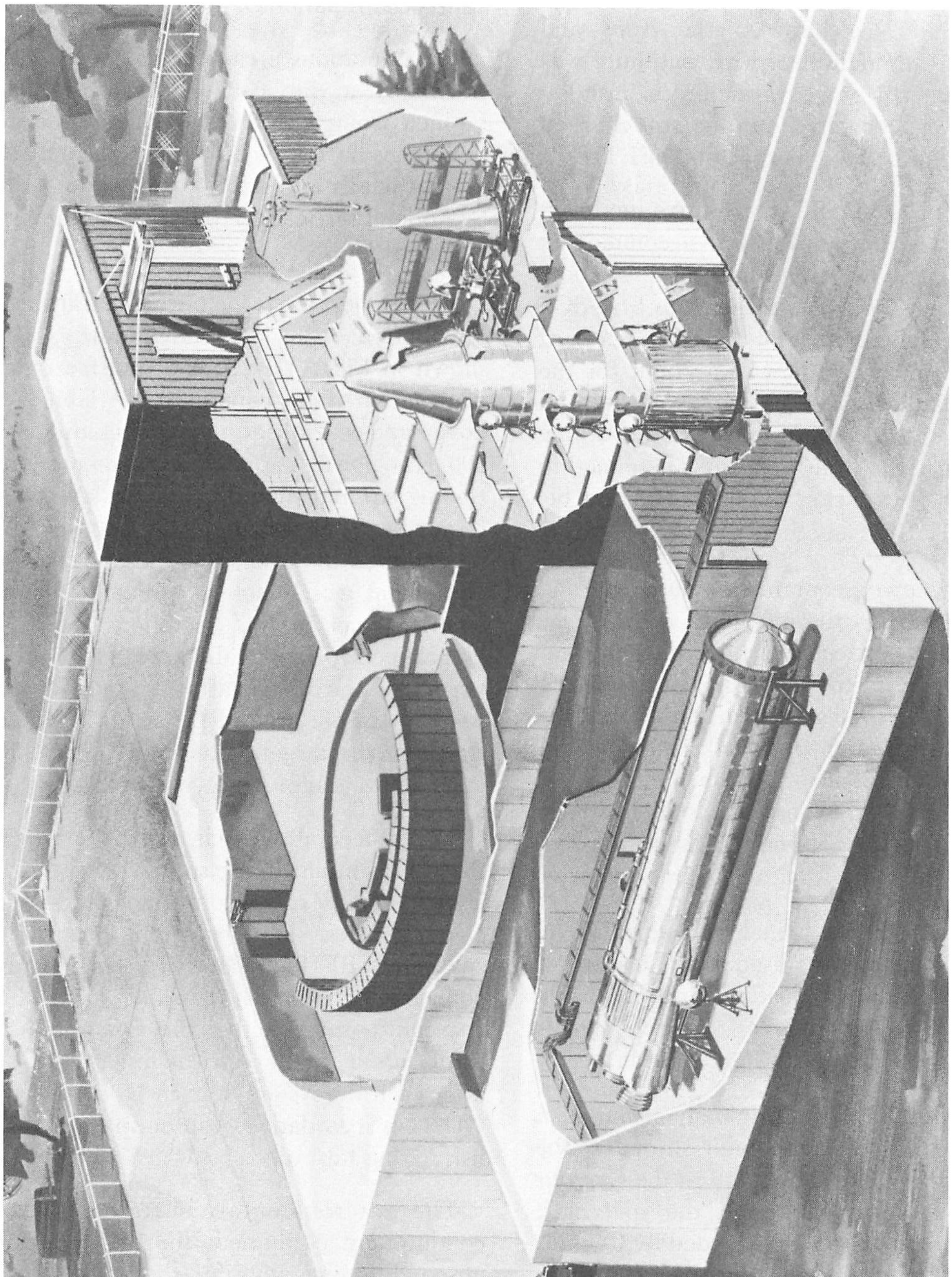


Fig. VIII-20. Combined systems test stand at San Diego the application of mechanical stretch to the vehicle should the tank pressure fail.

This configuration would not allow simulation of the RF or mechanical interface between the *Atlas* and the *Centaur*/spacecraft, though electrical continuity was maintained. The RF interface was verified after installation at the AFETR. The mechanical interface was checked with tooling fixtures and also verified at the AFETR.

Following the CST of AC-11/SC-4, extensive modifications to the *Atlas* area were accomplished to accommodate the SLV-3C of AC-13, used for the *Surveyor V* mission. (See Subsection B-1 for description of the *Atlas SLV*-3C.) These modifications included replacing the 60-in. forward support adapter with a 9-in. adapter to account for the 51-in. added length of the *Atlas* SLV-3C, replacing and relocating umbilicals and junction boxes, and reworking electrical terminations in the transfer room. These modifications were accomplished during the 61-day period between the CST of AC-11/SC-4 and the CST of AC-13/SC-5.

The 90-ft-high test tower contained equipment and facilities to erect and support the interstage adapter, the *Centaur* vehicle, and the spacecraft, including application of emergency stretch to the *Centaur*. Air conditioning of the *Centaur* electronics compartment and the *Surveyor* was provided through ductwork from the air-conditioning unit located adjacent to the building. Air conditioning was not supplied to the encapsulated spacecraft during the short period of time between exit from the system test area and initial mating operations.

During premate testing of the launch vehicle, a GD/C spacecraft electrical simulator was used in the test tower to simulate spacecraft electrical loading and to enable monitoring of the *Centaur* programmer's discrete signals which pass through the spacecraft/*Centaur* interface.

The CST concept for *Surveyor* originally included a provision for a spacecraft environmental shroud to be placed around the spacecraft when mated to the *Centaur* in the test tower, which would permit the spacecraft landing legs and omniantennas to be extended by *Centaur* programmer command during the testing sequence. This provision was later reduced to a test nose fairing, wherein the landing legs and antennas would not be deployed. This would permit access to the spacecraft during the plugs-out portion of the CST for the purpose of checking stray energy monitors and to replace squibs in the event a test rerun was necessary. As test requirements were refined, the requirement for a test nose fairing, was also deleted and a flight type nose fairing, using squib mufflers and stray-energy monitors, was used for all CSTs. During the CST, the *Centaur* programmer commands to fire the squib mufflers were verified by the telemetry system, and actual squib activation was verified after decapsulation of the spacecraft.

The operations area, located on the second floor, was essentially a duplicate of the blockhouse at AFETR Launch Complex 36. It contained the *Surveyor* operations console for monitoring and controlling spacecraft ground power and safety functions and all launch vehicle test operations except propellant tanking and the associated RFI test.

The support areas included the transfer room, terminal distribution room, and environmental control area as shown in Fig. VIII-21. The transfer room and terminal distribution room contained the launch vehicle ground power and control equipment, including the 28 V (DC and 400 Hz) power supplies, instrumentation, signal conditioning equipment and amplifiers, and commercial power terminals. There was no provision for emergency power operation in the event of commercial power failure. This equipment was remotely operated from the appropriate control panel in the operations room. The spacecraft launch control rack, also located in the transfer room, permitted the spacecraft operations console operator to remotely apply ground power to the spacecraft, control the television target lights, monitor ground power parameters, and monitor spacecraft voltage.

The spacecraft/launch vehicle electrical interface at the CSTS duplicated, insofar as practicable, the AFETR launch complex including power requirements, cable characteristics, shielding, grounding, etc. There was no spacecraft RF link, however. The pickup antenna was connected directly by hardline to the STEA when the spacecraft was in either the assembly area or mated to the *Centaur* in the vertical test tower. The CSTS operations wiring system was installed and checked out by GD/C in accordance with the spacecraft interconnections list, wiring tab runs, and CSTS electrical schematics.

The environmental control area consisted of air-conditioning equipment for the building and separate air-conditioning units for the *Atlas*/*Centaur* and spacecraft located adjacent to the building. These units did not have remote control capability. The spacecraft air conditioner supplied the encapsulated spacecraft (when mated to the *Centaur* in the test tower) at a relative humidity of less than 50% and at the temperature required to maintain the retro motor at 70 ±5°F. The building air-conditioning system provided filtered air of less than 50% relative humidity at a temperature of 70 ±5°F to the spacecraft operations area. The ground

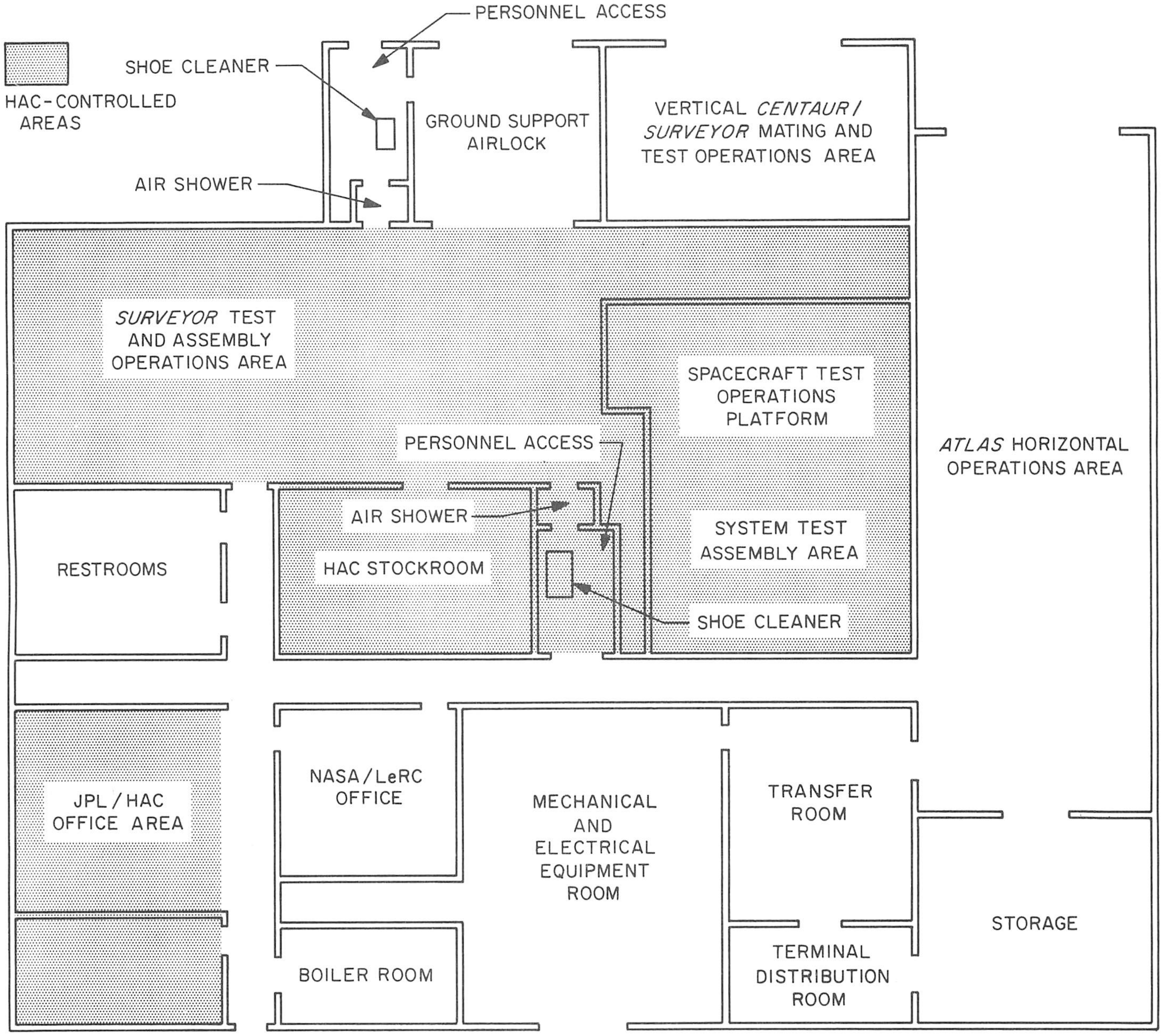


Fig. VIII-21. Combined systems test stand first floor plan handling equipment, torus ring, nose fairing mating fixtures, work stands, and hoisting slings were identical to those used at AFETR (see Subsection D-2-b), including a restraining device used during hoisting operations to prevent relative torsional deflections between the nose fairing and spacecraft.

2. Launch Complex

a. Launch complex evolution. Launch Complex 36 at the AFETR was originally constructed under the direction of the U.S. Corps of Engineers between July 1959 and February 1961 to support *Centaur* research and development test flights. The complex then consisted of a blockhouse and one launch platform with the necessary electrical and mechanical support systems. The blockhouse was a two-level structure, with test and control equipment installed on the upper level. The ground level was reserved for the equipment to control launch activities on a future second launch platform. In April 1962 a contract was awarded to GD/C to procure and install the GSE for support of the second complex, designated 36B.

The U.S. Corps of Engineers awarded a contract to a civilian firm to construct the launch facility on February 28, 1963. During that period the blockhouse was modified to permit the GSE for both platforms to be installed on the upper level, with the ground level being utilized for power supplies, cable terminations, etc. Because of funding and launch schedule interactions, construction of Launch Complex 36B and installation of the blockhouse equipment was halted in early 1965. Following the AC-5 explosion on the original platform on March 2, 1965, work on the platform for Launch Complex 36B was resumed on an expedited basis to enable the erection of AC-6 to be completed on May 27, 1965. Rehabilitation of the damaged complex was completed in time to permit erection of *Atlas* 174D (AC-7) on September 13, 1965.

b. Launch complex description. Launch Complex 36, shown in Fig. VIII-22, included the two launch platforms with their service buildings, service towers, and umbilical towers, a common blockhouse, and the support facilities.

1. Vehicle launcher and umbilical towers. The vehicle launcher, located on top of the two-level launch and service building, supported the vehicle on fixed geocentric coordinates with the Y axis of the vehicle at 105 deg east of true north for Launch Complex 36A and 115 deg east of true north for Launch Complex 36B. The launcher incorporated a holddown and release system which functioned automatically to release the vehicle at approximately $T-0.3$ sec, providing all prerequisites were met. The umbilical tower on each launch platform was also permanently fixed on top of the launch and service building and supported the vehicle umbilical booms and hydraulic retraction systems together with the launch vehicle and spacecraft air-conditioning ductwork, instrumentation and power circuits, GN_2 purging lines (including alpha scattering instrument purging system), and the spacecraft toxic gas detector system.

2. Service tower. The service tower at each launch platform provided for erection and assembly of the total vehicle and was used to service the vehicle during prelaunch operations. The towers, standing 179 ft tall on the platform of Launch Complex 36A and 209 ft tall on the Launch Complex 36B platform, had retractable work platforms at various levels to permit entrance to the vehicle access doors, pyrotechnics, and umbilical connections and furnished environmental protection from wind and rain. In addition, each tower provided a communication system, a fire protection system for the vehicle, and a supporting structure for the vehicle stretch system. Each tower was manually operated from a control room on the structure and was removed to a position some 300 ft away from the vehicle during the launch vehicle tanking test, the spacecraft RFI test, and the 60-min hold at $T-90$ min in the launch countdown.

Extensive modifications to the service towers, umbilical towers, and launcher systems were made to accommodate the *Atlas* SLV-3C, used for the *Surveyor V, VI,* and *VII* missions. (The *Atlas* SLV-3C was 51 in. longer than the LV-3C model used for the first four *Surveyor* missions; it also contained 20,000 lb more propellant and had redesigned guidance-equipment pods.) Modifications to Launch Complex 36B began immediately after the launch of *Surveyor III* on April 17, 1967, and included raising the umbilical tower boom supports, replacing the *Atlas* umbilicals, relocating the adjustable service-tower work platforms, altering the hydraulic and pneumatic systems, and replacing or lengthening the electrical circuits and ductwork. Launch Complex 36A modifications, which began after the launch of *Surveyor IV* on July 14, 1967, included the additional tasks of adding another deck to the umbilical tower for the *Centaur* upper umbilical boom supports, replacing the launcher, and relocating the fixed service-tower work platforms.

3. Launch and service building. The launch and service building of each launch platform provided facilities to house GSE for testing and launching of the *Atlas/Centaur/Surveyor* vehicles, including *Atlas/Centaur* propellant-loading equipment, pressurization systems, and a launcher

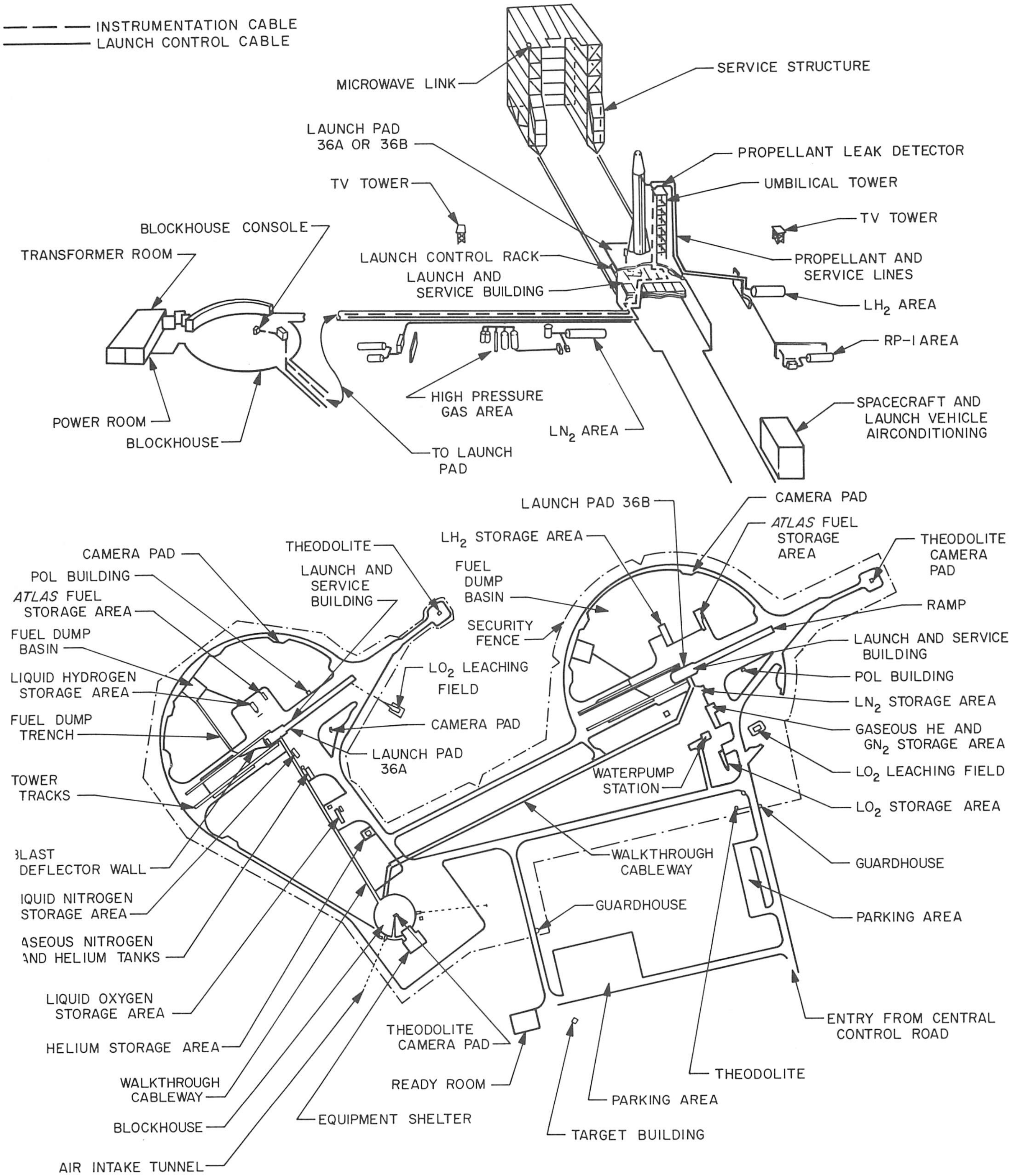


Fig. VIII-22. Launch Complex 36 configuration

stabilization system. The launcher stabilization system supplied 2500 psi hydraulic pressure to the launcher stabilization cylinders which supported and maintained the vehicle in a vertical position. Facility instrument air pressure was used to control the hydraulic pressure in the cylinders through a logic network, pressure control valves, and pressure switches. The instrument air source was changed to a different and higher pressure source and a direct pressure-indicating instrument was installed in the hydraulic system (to supplement the monitor light system) to prevent recurrence of the erroneous low-hydraulic-pressure indication which caused a 17-min delay in the countdown of AC-13/SC-5.

The transfer room, located on the second level of the service building, contained the launch vehicle equipment for controlling and operating the hydraulic, pneumatic, and pressurization systems; propellant loading; ground power supplies; and instrumentation. This entire equipment was operated remotely from the appropriate control panel in the blockhouse during test and launch operations. The spacecraft launch control rack was also located in the transfer room.

Prelaunch spacecraft control and monitoring signals were transmitted between the *Surveyor* operations console and the spacecraft via cabling through the transfer room, the umbilical tower, and three *Centaur* umbilical connectors. These umbilicals, together with the internal *Centaur/Surveyor* wiring scheme, are illustrated in Fig. VIII-23.

Industrial electric power was supplied (13.2 kV, 3-phase, 60 Hz) to the transfer room where it was transformed and switched for facility and vehicle usage. Prior to the

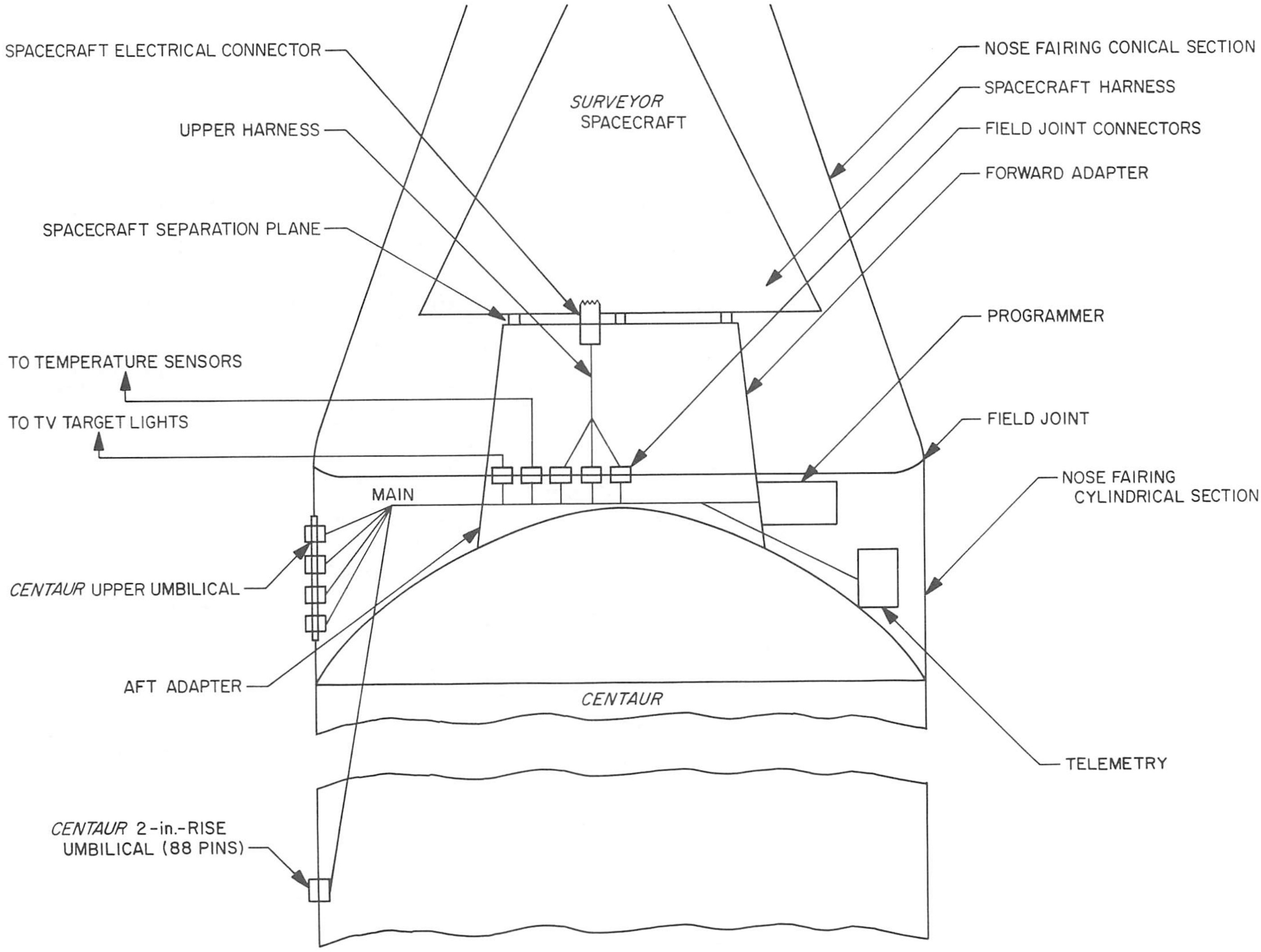


Fig. VIII-23. *Centaur/Surveyor* internal wiring scheme start of the launch vehicle tanking test and the final launch countdown, critical vehicle and instrumentation power was obtained from an AFETR central control area. The remaining power requirements continued to be supplied by commercial power. In the event of a commercial power failure, an emergency diesel generator located near the blockhouse automatically supplied power for noncritical circuits.

A 5-sec loss of commercial power occurred at $L+60$ sec following the launch countdown of AC-7/SC-2. Except for some loss of optical coverage, the power loss did not cause any significant problems since generator or critical power was being used.

During *Surveyor III* test and launch operations at AFETR, oscillations in the spacecraft ground power supply and monitoring circuits caused considerable difficulty in on-pad tests. The oscillations were reduced to an acceptable level for *Surveyor IV* and following missions by installation of additional wire paths for spacecraft battery charge and sensing line circuits between the launch control rack and the umbilical tower upper junction box on both launch platforms of Launch Complex 36. Other launch complex wiring changes made after initiation of *Surveyor* launch operations included the installation of additional wire shielding to reduce AC noise levels and the installation of additional wire paths and reassignment of electrical connector pins to conform to changing spacecraft functional requirements.

4. *Spacecraft RF link.* In addition to the hardline cabling and equipment installed at Launch Complex 36 for controlling and monitoring *Surveyor* test and launch operations, an RF link was installed by GD/C which permitted checkout and spacecraft countdown (independent of the launch vehicle) from the spacecraft checkout facility with the spacecraft located either in the explosive-safe facility or mated to the *Centaur* on both launch platforms. The RF link is depicted in Fig. VIII-24.

5. *Blockhouse.* The air-conditioned blockhouse was located approximately 800 ft from the center of Launch Platform 36A and 1350 ft from the center of Launch Platform 36B. It was a two-level structure of reinforced concrete providing blast protection for personnel and equipment by sloped walls 25 ft thick and a 12-ft-thick domed roof. The sole entrance was through a vault-type steel door which was sealed during the terminal portion of the launch countdown. Observation of the launch pad activities was through periscopes and a closed-circuit television system. All of the GSE for controlling, operating, and monitoring launch pad operations was located in the blockhouse. The GSE included a set of manually controlled *Centaur* guidance-system test and launch equipment, a computer-controlled launch set (which automatically tested and evaluated the launch readiness of the *Centaur* guidance system), and a spacecraft operations console. The launch set was installed in the blockhouse during the early part of 1967, and a satisfactory operational test was completed during a modified FACT of the AC-11 vehicle on June 12, 1967.

The spacecraft operations console, located in the upper level of the blockhouse, provided the means for monitoring spacecraft power, television-target illumination, propellant leakage, retrorocket safe–arm circuitry, and alarm circuits. The console also included a communications panel and permitted recording of spacecraft temperature sensor outputs.

The *Surveyor* operations console and launch control rack, configured for use with the *Surveyor* dynamic model, were first delivered by HAC to AFETR in December 1964 for installation by GD/C and operation by HAC in support of the AC-5 and AC-6 *Centaur* research and development flights with *Surveyor* dynamic models. The launch control rack and much of the cabling was severely damaged during the AC-5 fire and was subsequently replaced.

6. *Vehicle environmental control.* The environmental control system supplied filtered and dehumidified air (or GN_2) to the *Atlas, Centaur* and *Surveyor* spacecraft for heating and cooling purposes, as depicted in Fig. VIII-25. The *Surveyor* subsystem had two air-conditioning units, one of which was a backup unit.

The three systems, remotely controlled from blockhouse control panels, provided conditioned outside air to the *Surveyor* and *Centaur* prior to tanking cryogenics and to the *Atlas* thrust section prior to engine start for cooling and heating as required. Gaseous nitrogen was supplied to the *Centaur* and *Surveyor* after start of cryogenic tanking to maintain a dry atmosphere. It was also supplied to the *Atlas* thrust section to reduce the fire hazard during engine start. The *Atlas* equipment pods received cooling air during the entire countdown. The *Surveyor* air-conditioning unit provided the encapsulated spacecraft with filtered air of less than 60% relative humidity (based on a dry-bulb temperature of 65°F) through a filter which blocked out particles of greater than 10 μ in diameter. The airflow continued at a rate of

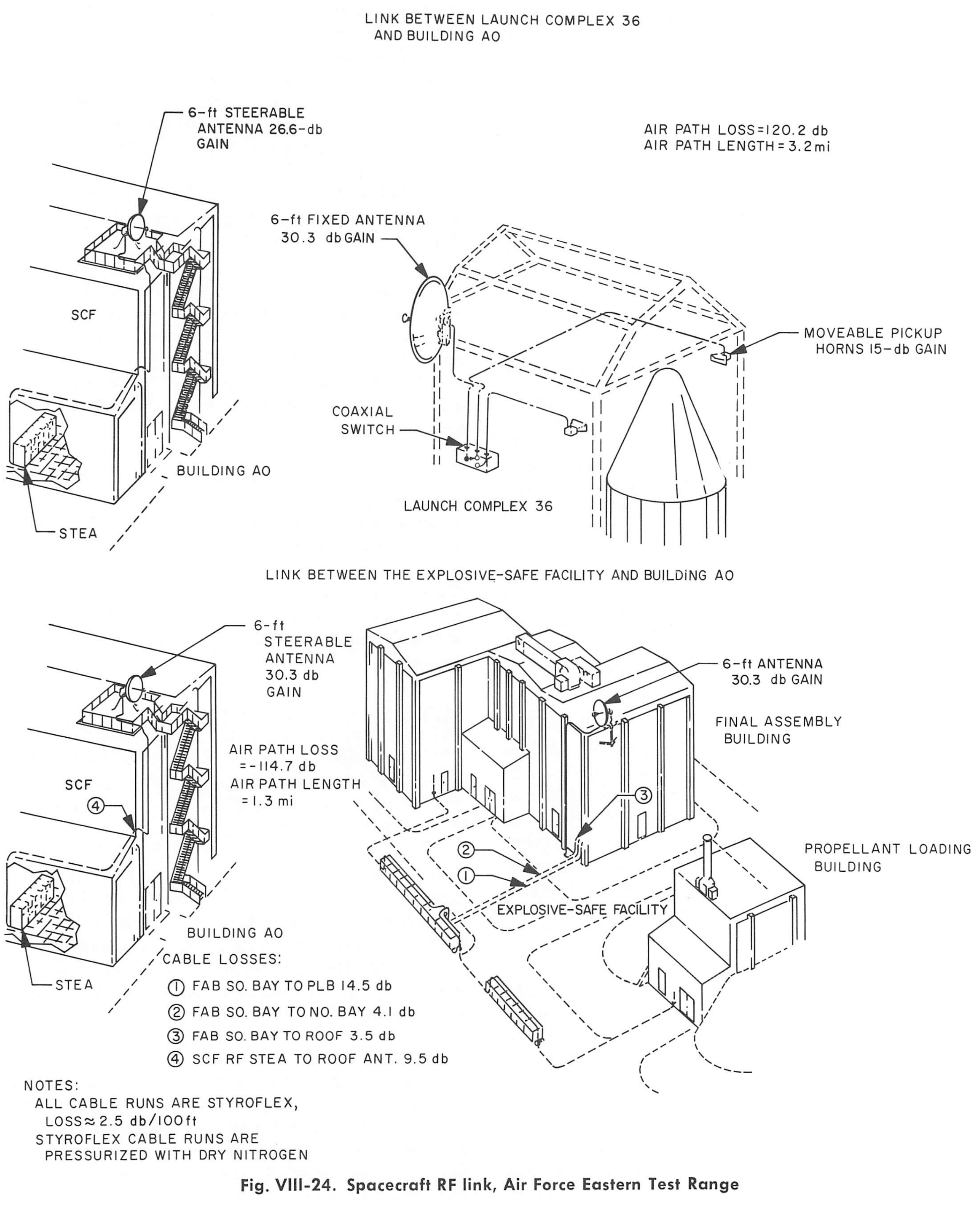


Fig. VIII-24. Spacecraft RF link, Air Force Eastern Test Range

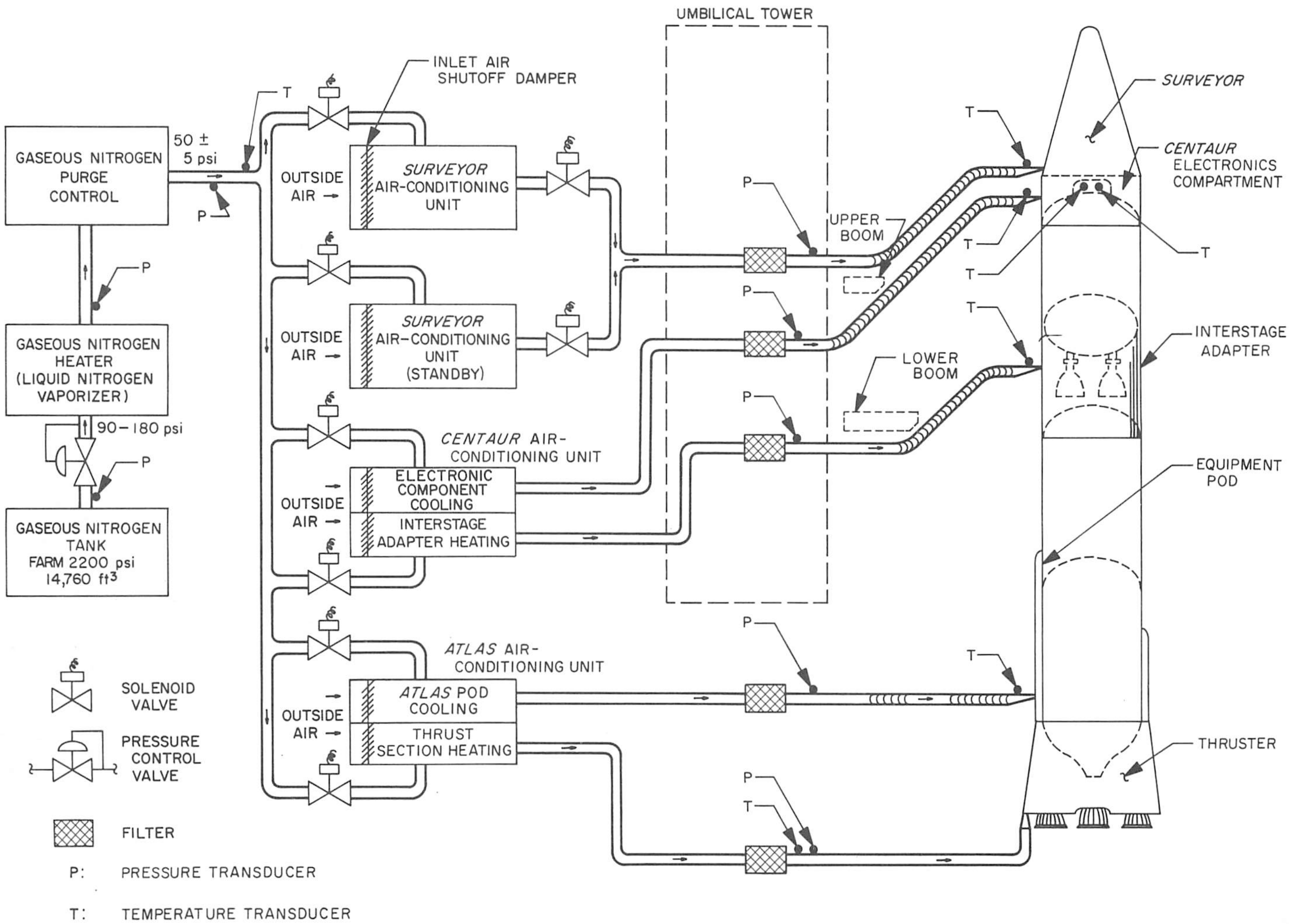


Fig. VIII-25. ***Atlas/Centaur/Surveyor environmental*** **control system, Launch Complex 36**

1000 ft^3/min from the time the encapsulated spacecraft arrived at the base of the service tower until it was ready for hoisting. The encapsulated spacecraft was without air conditioning during hoisting operations. The air-conditioning duct in the *Centaur* upper umbilical was connected during the mating operations, which permitted *Surveyor* air conditioning to be reestablished. The 70 ±5°F air temperature, required by the *Surveyor* retro motor, was continuously maintained until about two hours before simulated liftoff during the J-FACT, or opening of the launch window during launch operations, at which time the temperature was raised to 85 ±5°F to enable the spacecraft electronics and vernier engines to reach the desired temperature at liftoff without significantly raising the temperature of the retro motor. Performance of the environmental control system was satisfactory during prelaunch operations for all *Surveyor* missions. A minor problem occurred during the *Surveyor I* launch countdown when the temperature, measured in the vicinity of the retro motor, overshot the required 85 ±5°F and climbed to 92.2°F at $T-68$ min. However, the temperature was reduced to 86.5°F by $T-41$ min, and to 85.8°F by $T-3$ sec, and no detrimental effects were observed.

7. *Propellant storage and transfer systems.* The propellants (and gases) required for test and launch operations (except H_2O_2 and liquid helium) were stored at each launch platform. The liquid helium and H_2O_2 were brought to the launch complex by special trailers.

Liquid hydrogen. The liquid hydrogen (LH_2) storage and transfer system, essentially the same at both launch platforms, consisted of a 28,000-gallon storage tank, an LH_2 vaporizer for pressurizing the storage tank, an LH_2 flow control unit, and valves and transfer lines. The functional relationship of the major components of the LH_2 system are shown in Fig. VIII-26.

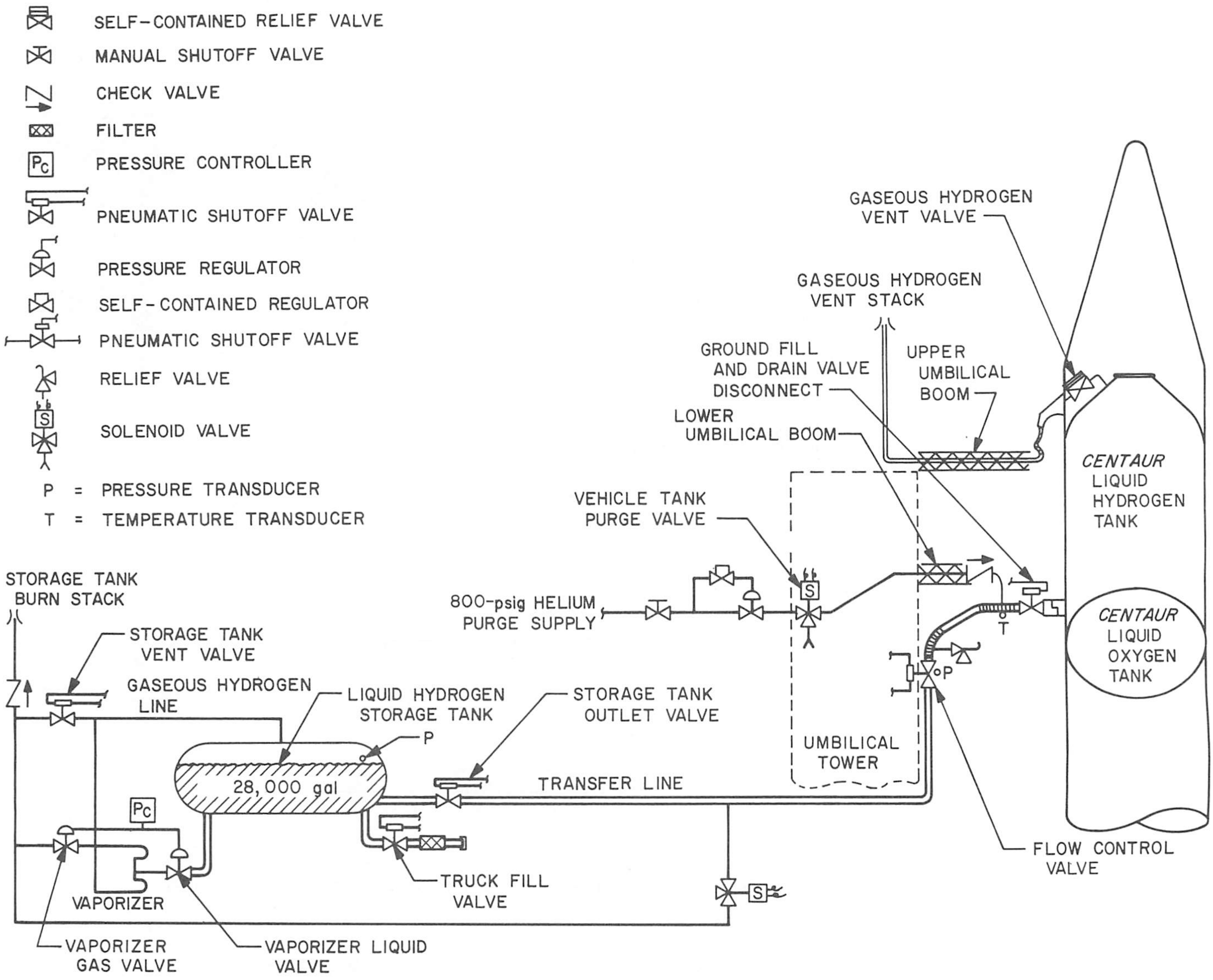


Fig. VIII-26. Liquid hydrogen transfer system, Launch Complex 36

The LH_2 transfer operation was accomplished during a launch vehicle tanking test prior to final mate of the spacecraft and during the launch countdown to load the *Centaur* LH_2 tank to 100.2% of the liquid volume required for flight and to maintain the 100.2% level until launch. The operation was performed in three separate and distinct phases:

(1) In the chilldown phase, the *Centaur* boiloff vent stack was purged with helium followed by pressurization of the storage tank to approximately 12 psig with gaseous hydrogen, then admitting the LH_2 into the *Centaur* tank. The LH_2 flowrate was controlled over a 20-min period at 10 gal/min to gradually chill the tank from ambient temperature to −420°F.

(2) After completion of the chilldown phase, the storage tank pressure was increased to 38 psig and the LH_2 flow was controlled between 10 and 750 gal/min until the LH_2 reached the 95% level in the *Centaur* tank and an automatic cutoff signal closed the LH_2 flow control valve.

(3) The topping phase was then initiated by reopening the flow control valve, reducing the storage tank pressure to 12 psig and controlling the flow of LH_2 between 10 and 50 gal/min to achieve and maintain the 100.2% level.

Liquid oxygen. The liquid-oxygen (LO_2) storage and transfer system at each launch platform was designed to deliver liquid oxygen to both launch vehicle stages from a common source (Fig. VIII-27). Each system consisted of a storage facility, a transfer unit, transfer lines, and associated hardware. Connection was accomplished to the first stage of the vehicle through launcher lines integral to the launcher assembly and to the second stage through an umbilical line installed on the lower umbilical boom. Launch Platform 36A utilized motor-driven pumps to transfer the LO_2 from the storage tanks to the vehicle. Launch Platform 36B utilized an LO_2 evaporative system to pressurize the storage tank and force the LO_2 into the vehicle. Each LO_2 transfer system was designed to deliver the prescribed volume of LO_2 to each stage; approximately 20,350 gal to the first stage, and approximately 2775 gal to the second stage. The initial LO_2 flowrate into each vehicle was low enough to allow all system components to reach cryogenic temperatures gradually. Rapid loading began upon conclusion of the cooldown phase, and continued until the liquid level reached the lower probe of the fluid level sensors. Rapid loading was followed by topping, which continued until the full level was indicated. The tanks were then continuously topped until loading was secured at $T-2.58$ min for *Atlas* and $T-1.25$ min for *Centaur*.

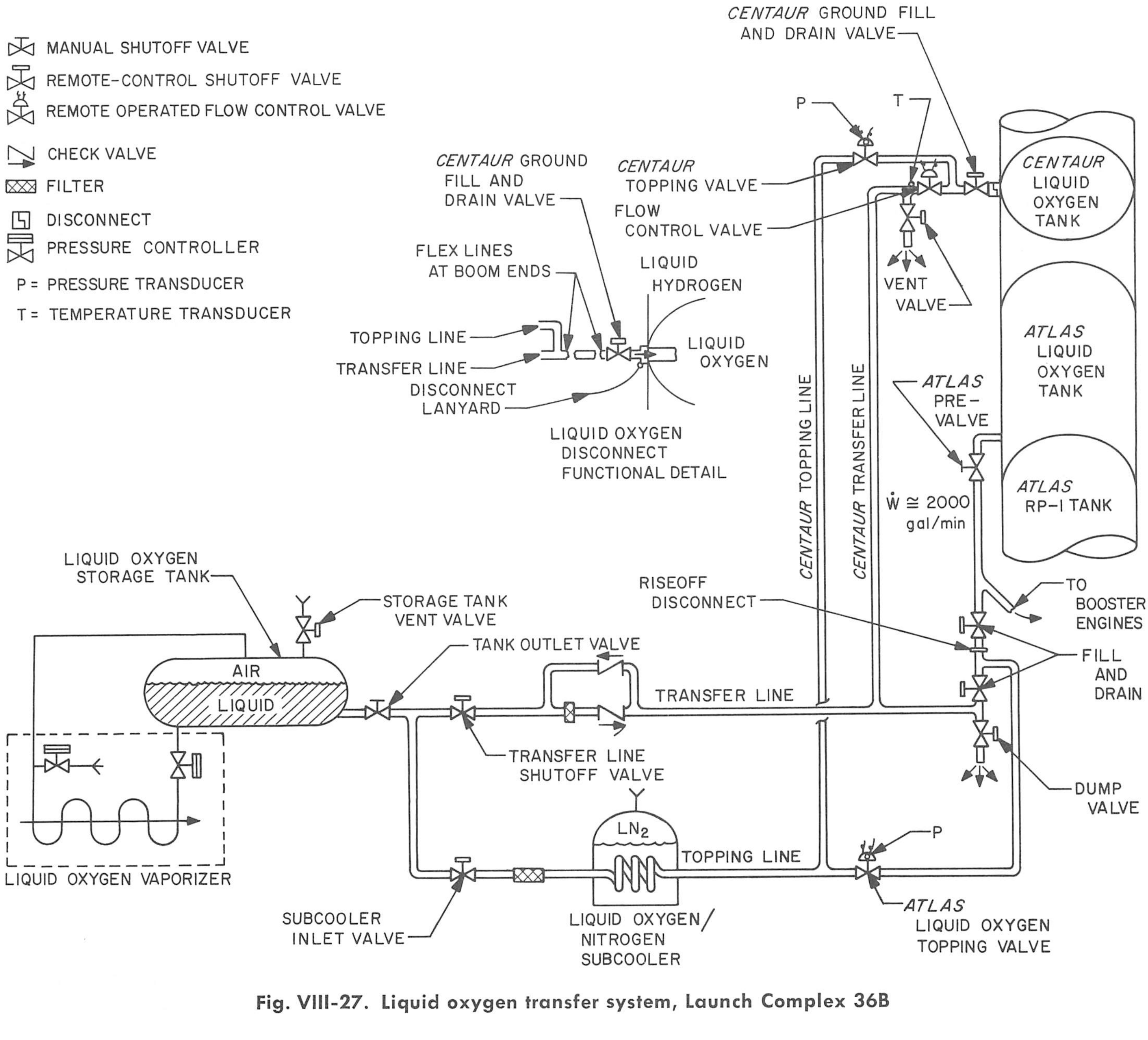


Fig. VIII-27. Liquid oxygen transfer system, Launch Complex 36B

Atlas fuel. The first-stage fuel transfer system was designed to furnish RP-1 fuel to the first-stage tank. The transfer system consisted of a storage tank, a transfer unit, and a cross-country line connecting the transfer unit to a launcher line integral to the launcher assembly. There were no significant differences between the RP-1 transfer systems at either launch platforms. The transfer unit contained two 500-gal/min pumps, a throttling valve to provide low flowrates for initial and final fill, a drain valve, a totalizer, and the necessary solenoids and regulators for equipment operation. The transfer system was designed to provide 12,675 gal of RP-1 fuel to the vehicle within 23 min against a maximum back pressure of 18.7 psig. The RP-1 fuel was filtered to prevent particles larger than 175 μ from entering the tank. Fuel tanking for the quad tanking test and for flight occurred 18 working days before launch, and two days before launch. A propellant level indication system, which provided for monitoring the fuel level through the use of hot wire point sensors, was used for tanking operations preceding those conducted on AC-13 (*Surveyor V*). This system was replaced by a sight gage for the tanking test of AC-13, and it operated satisfactorily for all remaining tanking operations.

Liquid and gaseous helium. The liquid-helium storage and transfer system at each launch platform consisted of a 1000 gal Dewar flask, a pressurization system, flow control equipment, and a vacuum-jacketed transfer line. Liquid helium at $-423°F$ was delivered to the *Centaur* main engine LH_2 turbopumps to (1) prevent atmospheric contamination of the engines at the beginning of final launch preparations, (2) purge any leaking LH_2 from the engines during tanking, and (3) prechill the engine turbopumps to $-310°F$ for the purpose of reducing the in-flight chilldown requirements from 20 sec to 5 sec before first *Centaur* MES.

Gaseous helium was brought to the launch complex in special trailers and transferred to storage tanks. The gaseous helium was used for (1) purging the LH_2 vent stack prior to tanking LH_2, (2) propellant tank pressurization, and (3) charging the *Centaur* airborne purge and tank pressurization bottles.

Hydrogen peroxide. The peroxide was tanked aboard the *Centaur* as fuel for the attitude control engines and for the boost-pump drive turbines. Both platforms had similar H_2O_2 transfer systems. No H_2O_2 storage was provided at either complex; instead, the monopropellant was brought to the platform in 300-lb drums as required. Normally, two drums were required for tanking. A specific weight of H_2O_2 was pressure-fed into the vehicle airborne storage bottle from the drum containers by a motor-driven pump controlled by the second-stage engine control panel in the blockhouse. The transfer system also included facilities for water flush and H_2O_2 containment and drainage in the event of a spill.

Liquid and gaseous nitrogen. Liquid nitrogen was stored in separate 28,000-gal storage tanks at each launch platform for use in subcooling the LO_2 provided to the *Centaur* during tank topping operations. The GN_2 storage tanks were common for both launch platforms. Before launch, GN_2 usage included (1) pressurizing the propellant storage tanks, (2) purging all electrical junction boxes located within 100 ft of the launch platform, (3) purging various elements of the vehicle, (4) charging the airborne nose fairing jettison bottles, and (5) supplying the environmental control system.

Logistic constraints on hold capability. The storage and transfer systems at Launch Complex 36 were capable of supporting a 2-hr hold after completion of cryogenic tanking operations. This capability was mainly dependent on the GN_2 supply, since a reserve had to be maintained in the event detanking occurred. The system was further capable of resupply to support a 24-hr turn-around period. There was an 8-day constraint against storage of H_2O_2 aboard the vehicle. At the end of this period the fluid had to be drained and the vehicle system purged and dried.

Spacecraft encapsulation equipment. Encapsulation of the spacecraft for the J-FACT and for launch was accomplished by GD/C in the explosive-safe facility using GD/C-supplied equipment, which included the two halves of the nose fairing, the forward section of the payload adapter, a torus ring for supporting the nose fairing, torus-assembly handling carts, and the necessary slings and work stands. Before encapsulating the spacecraft, the nose fairing halves were installed on nose fairing support ring sections, thoroughly cleaned with solvent, and the fairing assembled by securing the nose fairing latches. The torus ring was placed in the handling cart (Fig. VIII-28) and alignment with the ground transport vehicle was accomplished. The assembled nose fairing was then positioned in the torus ring, the two halves separated, and the thermal bulkhead was installed in each half. The nose fairing was then ready for spacecraft encapsulation. For *Surveyors V, VI,* and *VII*, an alpha scattering instrument purge box, purge line, and nitrogen supply tank were installed on the torus ring. (See Fig. VIII-29 for complete preparation and encapsulation sequence.) To improve cleanness conditions

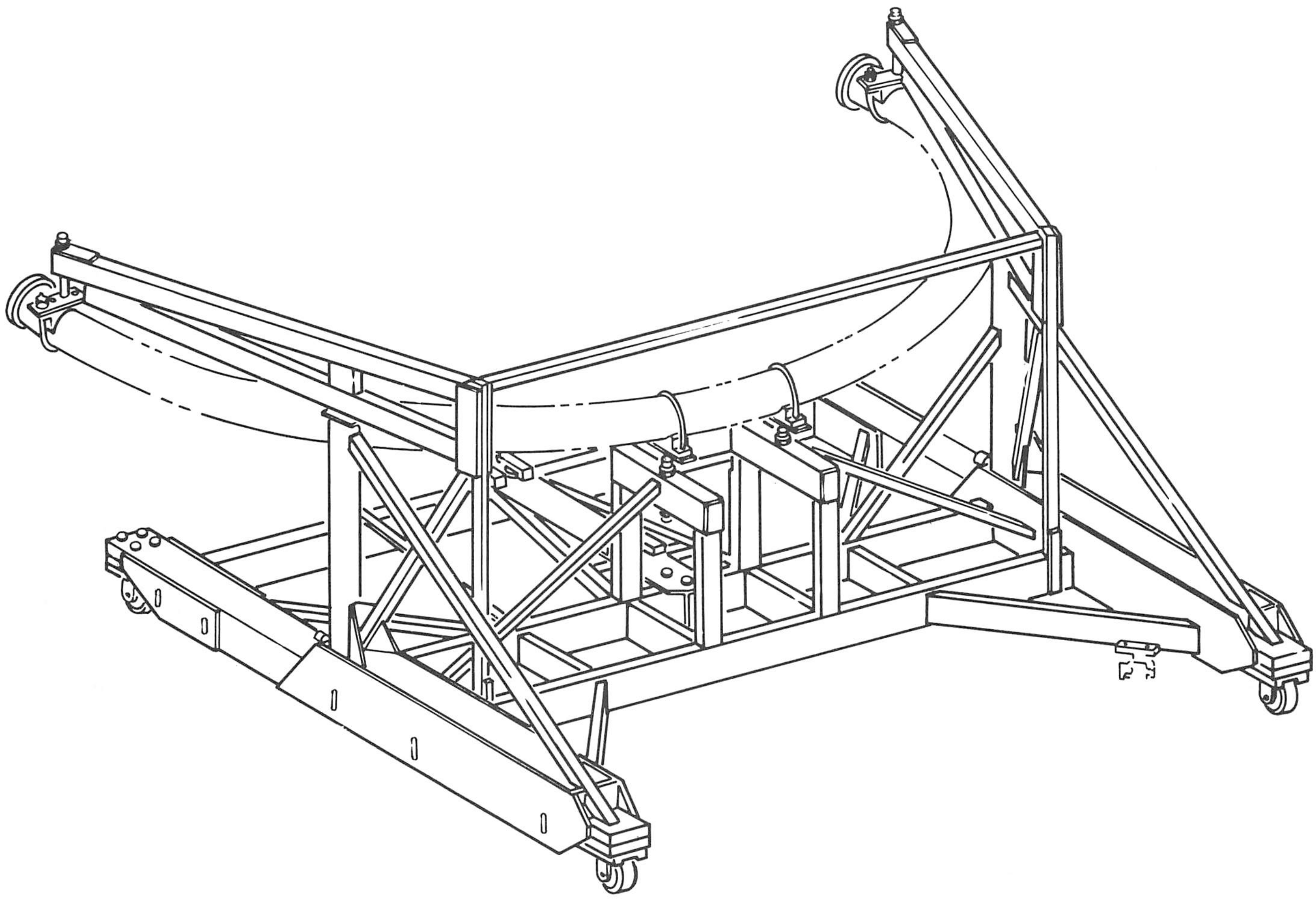


Fig. VIII-28. Nose fairing handling cart

following *Surveyor II* encapsulation operations, all paint was removed from the torus ring, handling carts, and work stands. The adapter ring mounted in the ground transport vehicle (GTV-2) was changed from painted aluminum to anodized aluminum, and the other equipment was coated with zinc chromate primer to preclude paint flaking.

Communications. The blockhouse, Launch Platform 36A, service buildings, service towers, umbilical towers, spacecraft areas, service areas, and the mission operations room were served by a hardline 19-channel operational intercommunication system. Launch Platform 36B facilities had an improved 20-channel transistorized system. During the J-FACT and launch countdown, all communications were switched to channel 1 for direct conductance of the countdown by the GD/C test conductor. In addition to the intercom, a station-to-station telephone system (green phones) operated by key panels at each console, permitted rapid and direct communications between key sites participating in launch operations.

E. Space Vehicle Tests and Operations

The launch vehicle test program for the operational vehicles included acceptance testing of each *Atlas* and *Centaur* stage used for the seven *Surveyor* missions, beginning with factory acceptance testing at the completion of the final assembly. This was followed by system testing at the CSTS in San Diego. The vehicles were then shipped to the AFETR where additional acceptance tests were performed to assure launch readiness.

1. Combined Systems Test

a. Guidance-autopilot and flight acceptance composite test. Following factory acceptance testing, the *Atlas* and *Centaur* vehicles were installed in the CSTS where subsystem verification was performed as the GSE was connected, and where satisfactory operation of the flight control systems in combination with the guidance system was verified by a combined guidance-autopilot test. A launch vehicle FACT without the spacecraft was then performed to demonstrate satisfactory operation of all launch vehicle systems (except propellant loading) during a simulated countdown and flight through *Centaur* retromaneuver. The launch vehicle test configuration was as close to the flight configuration as possible, including simulation of the spacecraft/launch vehicle electrical interface by a GD/C *Surveyor* simulator. First- and second-stage programmers were operated, all electrical umbilicals were either electrically or manually ejected, RF systems were operated via the RF link, and internal electrical power was provided by flight batteries. The launch vehicle FACT was the final acceptance test when the *Surveyor* spacecraft did not participate in the CST. Satisfactory completion of the launch vehicle FACT and acceptance of the test results by NASA was a prerequisite for either proceeding to CST with the spacecraft or shipment of the vehicle to the AFETR.

Before installation of all flight components and completion of factory acceptance testing, a FACT of AC-7/T-21 was conducted at the CSTS on May 21 and 22, 1965, for validation of the CSTS and test procedures (see Subsection C-3-d). Following further testing with T-21 on May 24 and 26, the *Centaur* stage was returned to the factory for updating and to complete acceptance testing. The *Atlas* stage was returned to the factory on June 1 for rework. The two stages were then reinstalled in the CSTS where a third FACT was satisfactorily conducted on July 13, 1965, followed by shipment to AFETR of the *Atlas* on July 27 and the *Centaur* on August 6, 1965.

At AFETR, the vehicle underwent a FACT and a tanking test in preparation for first mate with the *Surveyor I* spacecraft, which it was originally scheduled to launch. However, due to nonavailability of the spacecraft, the AC-7 vehicle was recycled to the factory in San Diego, where it underwent modification, refurbishing, and additional testing. Two major changes were made in the AC-7 vehicle at this time: Due to engine contamination, the original *Atlas* first stage was exchanged with the AC-9 first stage. The *Centaur* guidance system was updated to an improved version as flown on AC-8 and AC-10 (which had been substituted for AC-7 to launch *Surveyor I*).

Additional improvements made to the AC-7 *Centaur* included replacing the old H_2O_2 engines with improved models, and installing a *hi-rel* rotary inverter and a battery voltage monitoring system. After factory acceptance testing of the new AC-7 vehicle, another final FACT (without spacecraft) was satisfactorily conducted in the CSTS on May 25, 1966. The vehicle was then shipped to the AFETR as the launch vehicle for *Surveyor II*.

The *Atlas/Centaur* combinations for *Surveyors I* and *III* (AC-10 and AC-12, respectively) required running of the last 10 min of the FACT countdown at the CSTS due to premature manual disconnect of the interstage adapter electrical connector, which prevented start of the *Centaur* programmer.

b. Joint flight acceptance composite test. It was originally planned to conduct a final J-FACT at the CSTS of each *Atlas/Centaur/Surveyor* space vehicle to demonstrate the operational compatibility of vehicle systems during a simulated countdown and flight through *Centaur* retromaneuver. Because of schedule constraints, only SC-1, SC-4, and SC-5 participated in a J-FACT at the CSTS (which in each case was conducted following the launch vehicle FACT). NASA (LeRC) and JPL acceptance of the test, based on *quick-look* examination of the test data, was a prerequisite to breaking the CST configuration and shipment of the space vehicle to the AFETR. No serious problems developed during any of the J-FACTs which were conducted, and no test reruns were required. The launch vehicle test configuration and minor problems which occurred during the J-FACTs of SC-1 and SC-4 are discussed below. No significant launch vehicle anomalies occurred during the J-FACT of SC-5/AC-13.

The J-FACT was conducted similar to the launch vehicle FACT except that the encapsulated spacecraft was installed on the *Centaur* with special test harnesses and the electrical field joint connected. *Centaur* programmer commands were monitored by telemetry. Receipt by the spacecraft of the commands to extend the omniantennas and landing legs was verified after decapsulation by examination for the firing of squib mufflers. The command to switch to *Surveyor* high-power RF transmission was verified by monitoring the received signal and comparing its relative strength.

Squib simulators were monitored by GD/C to confirm receipt of the spacecraft/*Centaur* electrical-disconnect separation signal and the spacecraft separation signal. Spacecraft/*Centaur* separation was simulated by manual disconnect of the field-joint electrical connector. During SC-4 and SC-5 operations, a power isolation box was installed at this connector to prevent accidental shorting of the connector halves during remating operations. The J-FACT of SC-1/AC-10 included the conductance of a launch vehicle guidance-autopilot test as part of a 480-min countdown. This test preceded the countdown for SC-1 and SC-5, and the countdown was reduced to a duration of 120 min.

Fig. VIII-29. *Surveyor encapsulation operations*

The SC-1/AC-10 J-FACT was conducted on March 7, 1966, with only minor interface problems occurring, among which was a failure to complete the simulated launch ladder at $T-1$ min requiring a hold. The fault was isolated to a recessed pin in the accessory box plug used for EMI monitoring. The circuit was jumpered, the count recycled to $T-5$ min, and the test proceeded without further incident. A quick look of the data revealed no significant problems, the test configuration was broken, and the spacecraft was demated on March 8, 1966 and prepared for shipment to the AFETR.

The J-FACT of SC-4/AC-11 was performed on April 20, 1967. During the test, there was no indication that the squib mufflers (used to simulate release of the spacecraft omniantennas and landing legs) had been activated by *Centaur* programmer command. An additional *Centaur* programmer test was run, and correct operation of the programmer was verified. After the spacecraft was demated and decapsulated on April 20, the problem was traced to incorrect installation and adjustment of the microswitches which indicate firing of the squib mufflers. Consent to proceed was obtained and, after minor modifications and replacements, the launch vehicle was prepared for shipment to the AFETR.

2. Prelaunch Tests at Launch Complex 36

After the launch vehicle and spacecraft arrived at the AFETR, a series of major system tests were performed before the launch countdown for demonstration and assurance of launch readiness. These system tests, which required AFETR range support and KSC/ULO approval of test procedures and acceptance of test results, were conducted by GD/C after the *Atlas, Centaur,* and *Surveyor* systems were individually tested in accordance with their respective AFETR readiness test procedures. The readiness tests of the launch vehicle included a guidance-autopilot integrated test, with the *Atlas* and *Centaur* mated on the launch platform, followed by an integrated launch control test, the last test being performed before mating the spacecraft.

Premate spacecraft readiness tests culminated in a system readiness test of the encapsulated spacecraft in the explosive-safe facility before transport to the launch complex. After the spacecraft was mated to the *Centaur* and the environmental control system was connected and operating, the vernier engine purge system was activated (*Surveyor III* and subsequent missions) and the alpha scattering instrument purge was operating (*Surveyors V, VI,* and *VII*). An RF link optimization test and another system readiness test were performed in preparation for the first of the CSTs. An additional test performed at this time was a compatibility test between DSS 71 and the spacecraft. For AC-10/SC-1 and AC-7/SC-2, the first CST was the launch vehicle flight control and propellant tanking integrated test (described below). The purpose in conducting the tanking test at this time (with the *Surveyor* aboard, the service tower removed, and all RF systems radiating) was to check for any RF interference or adverse effects on the spacecraft from cryogenic tanking. Since no adverse effects on the spacecraft were noted during either the AC-10 or AC-7 tanking operations and elimination of spacecraft participation would reduce the joint simulated-launch test period, the launch vehicle tanking test was conducted after the J-FACT and without the spacecraft for the remaining *Surveyor* missions.

For *Surveyors III* through *VII*, a simulated launch vehicle propellant loading was performed between $T-120$ and $T-55$ min in the J-FACT countdown, wherein the launch vehicle propellant tanking valves and pumping systems were exercised to check for RF interference effects on the spacecraft. No interference was noted during any of these tests, although a prelaunch GSE problem occurred during the test of AC-13/SC-5 (see Table VIII-8).

a. Joint Atlas/Centaur/Surveyor flight acceptance composite test. This test was the second major CST performed at the launch complex for *Surveyors I* and *II* and the first CST for *Surveyor III* and subsequent units. It was performed to verify, on an integrated basis, that all *Atlas, Centaur,* and *Surveyor* ground and airborne electrical systems were compatible and were capable of proper combined-system operation throughout a simulated launch countdown and flight sequence through *Centaur* retromaneuver.

Atlas/Centaur systems were run during the simulated flight sequence on internal battery power with minimum control by GSE. Guidance was in the flight mode with programmers and range-safety-command systems armed. After simulated *Atlas* engine start, a launch release sequence was electrically performed followed by ejection of the ground umbilicals. *Atlas/Centaur* separation was electrically simulated by disconnect of the appropriate plugs with simulation of vehicle electroexplosive devices by squib simulators. All RF systems were operated open-loop in the simulated launch environment. Telemetry and the gantry test rack were used to monitor all *Atlas/Centaur* electrically powered systems; the latter also provided for interruption of battery power to stop the test when required.

Table VIII-8. Launch vehicle and ground-support equipment anomalies during J-FACT

Vehicle	Launch complex	Anomaly	Correction
AC-10/SC-1	36A	*Atlas* programmer failure when switched from internal to external power during *hold-fire* test resulted in loss of programmer reference time and scrambling of pitch program switch positions	Design modification and replacement of *Atlas* programmer. Also replacement of *Centaur* programmer. Correct operation verified during subsequent FACT
		No indication on one of two *Atlas* in-flight pressurization monitors. Loose connection discovered in test box subsequent to test	Connector secured
AC-7/SC-2	36A	No *Atlas* range safety command (RSC) AGC signal. Traced to missing wire in harness subsequent to test	This anomaly did not affect the J-FACT and test proceeded without this function. Missing wire was replaced after J-FACT
AC-12/SC-3	36B	Start of EMI test portion of J-FACT delayed due to inadvertent manual ejection of the P409 umbilical	Umbilical reinstalled and handler cautioned
		Frequency shift of spacecraft shock absorber strain gage on leg 2. After final mate, cause was traced to *Centaur* telemetry interference	Replaced *Centaur* telemetry components
		Field joint connector main power bus shorted to ground during manual reconnection after the J-FACT caused shutdown of spacecraft transmitters	Demated and decapsulated the spacecraft, no damage found. Uncompleted spacecraft RF calibration test rescheduled after final mate. A power isolation box was designed and used during all remaining J-FACT at CSTS and AFETR
AC-11/SC-4	36A	Unprogrammed 12-sec hold at T−2 min in countdown resulting from failure to receive *Surveyor* arm enable activation signal; caused by communication headset failure	Replaced headset and continued with test
		Premature launch release signal prevented fast speed *event* records, resulting in loss of measurement accuracy. Caused by incorrect logic on second-stage engine control panel	Corrected logic. Verified logic and accurate event recorder data obtained during subsequent tests
AC-13/SC-5	36B	During EMI portion of the J-FACT, missile DC power was lost for a few seconds, though AC power remained on. Test constraint for the *Atlas* programmer violated by possible overstress of transistors in high-power circuit	Anomaly could not be repeated and no cause of power failure was found. Programmer left on board for J-FACT and later replaced
		Spacecraft air-conditioning temperature change from 70 ±5°F to 85 ±5°F not accomplished at desired time due to failure to transfer system from manual to automatic on time	Closer monitoring of procedure
		Short in umbilical 407 during plus count after manual ejection due to contact between connector shell and pin in plug	No damage and test continued. Umbilical handlers cautioned on safety
		Momentary loss of guidance *ready* signal due to noisy limit cycle monitor loop	Loop bypassed for J-FACT and subsequently modified
		Range timing lost at the gantry test rack at $L + 1200$ sec caused by faulty relay socket	Test continued without this function. Socket replaced after test
AC-14/SC-6	36B	75-sec hold at T−60 sec due to procedural error in operation of the range safety officer's console	Proceeded with test after clarifying operator instructions. Procedure was revised
AC-15/SC-7	36A	No significant anomalies	

Surveyor subsystems were turned on early in the countdown for a system readiness test between $T-680$ and $T-260$ min. After a period of RF silence, the system was prepared for simulated launch by a spacecraft countdown (concurrent with the *Atlas/Centaur* count). This countdown started during the planned 60-min hold at $T-90$ min. The spacecraft was switched to internal power at $T-4$ min and was operated in this mode with ground umbilical cables ejected. The actuation signals for nose fairing jettison and spacecraft separation were generated and monitored with spacecraft separation being simulated by manual disconnect of the field joint electrical connector. No physical separation or actuation of the nose fairing thruster bottles, nose fairing latches, spacecraft latches, or spacecraft electrical disconnect was performed. *Surveyor* system interrogation and performance monitoring was provided through an RF link, using the combined operation of the STEA, the command and data handling console, and the programmed automatic telemetry evaluator (PATE). A summary of significant problems encountered during J-FACT is given in Table VIII-8.

After SC-1 and SC-2 were first mated to the *Centaur* for the tanking test, the spacecraft receivers were calibrated with the launch platform service tower in place. For SC-3, SC-4, and SC-5, it was planned to conduct this receiver calibration following the J-FACT countdown immediately preceding the demating of the spacecraft. The test was accomplished at this time only for SC-4. The SC-3 calibration was performed after final spacecraft/launch vehicle mating due to accidental shorting of the field joint electrical connector, and the tower removal portion of the SC-5 test was postponed until after final mate due to a thunderstorm.

b. Flight control and propellant tanking integrated test. This quad tanking test verified that the launch vehicle ground and airborne systems were capable of: (1) tanking and maintaining proper propellant levels in all *Atlas/Centaur* tanks, (2) pressurization and chilldown of the *Atlas* helium bottles, and (3) performing ground chilldown of the *Centaur* engines. This test also checked the ability of all launch vehicle subsystems to function properly under cryogenic conditions and in a total launch-representative RF environment, which included operation of the applicable AFETR transmitters and removal of the service tower. During AC-10 and AC-7 tanking tests, SC-1 and SC-2 were mated to the *Centaur* and used in the tests.

For SC-3 and subsequent spacecraft, the flight nose fairing conical section, associated spacecraft forward adapter, and thermal bulkhead assembly were not mated to the *Centaur* for this test. In lieu of these flight assemblies, a geometrically similar test fairing and spacecraft forward adapter assembly were installed to maintain a flight-representative configuration.

A summary of launch vehicle problems occurring during these tests is given in Table VIII-9.

c. Flight acceptance composite test. Although this was an optional, launch-vehicle-only test, it was performed for all seven *Surveyor* missions. The FACT was conducted to provide a reverification of all *Atlas/Centaur* ground and airborne electrical systems near the end of the relatively long period which accrues between *Surveyor* post-J-FACT demate and final remate for launch. The test also provided operational verification of *Atlas/Centaur* corrections required as a result of the previous J-FACT and tanking tests. The test configuration and procedure used generally duplicated the launch vehicle portion of the previous plugs-out J-FACT.

Only minor anomalies were observed during the FACTs of the seven *Surveyor* launch vehicles. These included: faulty operation of the AC-7 *Centaur* inverter temperature recorder; discrepancies in telemetry measurements, guidance gimbal torquing voltage, and performance of the *Atlas* PU unulling operation on AC-13; and loss of guidance optical acquisition at $T-10$ sec in the AC-15 FACT countdown. The loss of guidance optical acquisition, probably due to a severe storm, required a hold of more than 2½ min to reacquire and would have necessitated *Centaur* propellant topping had it occurred during a launch attempt.

d. Composite readiness test. The last CST preceding final spacecraft mating and launch countdown was the composite readiness test with its simulated flight sequence to verify launch-readiness verification of all *Atlas/Centaur* electrical and RF systems after reconnection of the ground umbilicals. The *Surveyor* was not mated to the *Centaur* during these tests.

This test was performed with guidance in the integrated-test mode and programmers and range safety command systems in the armed mode. Pyrotechnics were simulated. The RF systems were operated open-loop and flight pressures were electrically simulated. Vehicle power was provided by a battery simulator. Landline instrumentation, launch control GSE, telemetry, and the gantry test rack were used for event monitoring.

The only significant problems encountered in these tests occurred during the test of AC-12 on April 10, 1967. At

Table VIII-9. Launch vehicle and ground-support equipment anomalies during quad tanking tests

Vehicle	Launch complex	Date conducted	Anomaly	Correction[b]
AC-10/SC-1[a]	36A	April 20, 1966	Failure of *Centaur* LH_2 boost-pump bearing temperature transducer	Replaced
			Optical acquisition difficult to maintain after *Atlas* LO_2 tanking	Acquisition system adjusted
			Insulation panel seal leak	Repaired
AC-7/SC-2[a]	36A	August 11, 1966	LO_2 leak in umbilical boom relief valve in transfer line	Valve repaired and retested
			Failure of regulator pressure transducer	Replaced and revalidated
AC-12 (SC-3)	36B	March 21, 1967	This test was unsatisfactory due to the failure of a pressure rupture disk in the helium Dewar flask which prevented transfer of the liquid. Other anomalies contributing to the decision to rerun the test were: faulty temperature transducer in the hydraulic pump unit, faulty pressure vent valve, failure of the *Atlas* LO_2 tank pressure-indicating circuitry and an inoperative *Atlas* engine abort purge system	Repair or replacement of equipment, except the liquid-helium Dewar flask and the hydraulic pump unit transducer. A substitute Dewar flask was used for the second test; the transducer anomaly was not corrected until after second test
AC-12 (SC-3)	36B	March 24, 1967	Small leak in RP-1 fuel tank pressure line	Repaired
			Low environmental temperature caused by clogged control system	Cleaned, and test continued
AC-11 (SC-4)	36A	June 21, 1967	Inoperative LN_2 rapid fill solenoid valve	Valve bypassed for test and later corrected by repair of a wiring splice
			Improper operation of the *Centaur* hydrogen vent valve prevented boiloff rate test	Valve replaced and correct operation verified during second tanking test
			Erratic outputs from the *Atlas* booster engine pitch channels	Corrected loose terminal on input test signal line
			Inconsistent radar readouts of the C-bank beacon	Beacon replaced but still erratic during second tanking test; problem traced to radar van location and antenna pattern
			Improper *Centaur* propellant utilization system operation	*Centaur* propellant utilization valve electronics package replaced
			Leaking *Atlas* LO_2 tank relief valve	Valve repaired and replaced
			Nonlinear hydrogen tank ullage pressure transducer	Transducer replaced
			Voltage leak between P461 umbilical pins caused by moisture	Umbilical replaced; all other umbilicals examined and dried where necessary
AC-11 (SC-4)	36A	June 27, 1967	No significant anomalies Boiloff rate test satisfactory	
AC-13 (SC-5)	36B	August 18, 1967	Loss of optical acquisition during alignment checks caused by misalignment of theodolite	Theodolite adjusted (during test)
			Pressure oscillation in *Atlas* guidance pod cooling system	Adjusted cooling compressor head pressure (during test)
			Erratic operation of *Atlas* pneumatic and guidance pod cooling duct pressure transducers	Replaced transducers
			Loose plastic material in *Atlas* start tank relief valve	Cleaned valve

[a]SC-1 and SC-2 were aboard for these tests to determine RF effects of cryogenic tanking on the spacecraft.

[b]Unless otherwise noted, corrective actions were taken after termination of tests.

Table VIII-9 (contd)

Vehicle	Launch complex	Date conducted	Anomaly	Correction[b]
AC-13 (SC-5)	36B	August 18, 1967	Leaking liftoff disconnect on *Atlas* helium bottle	Replaced O-ring seal
			Ruptured *Centaur* forward seal	Replaced affected sections
AC-14 (SC-6)	36B	October 19, 1967	Miswired *Centaur* LH_2 probe sensing elements caused improper operation of the probe and prevented tests of the boiloff rate and LH_2 vent lockup	A second tanking test confirmed miswired condition. Control panel and procedure revised to conform to miswired condition
			Loss of one *Centaur* insulation panel purge pressure measurement	Replaced transducer
			Erroneous *Centaur* LO_2 regulator pressure	Corrected miswired instrumentation (during test)
			Ruptured *Centaur* forward seal	Replaced affected sections
AC-14 (SC-6)	36B	October 21, 1967	No significant anomalies; LH_2 boiloff rate and vent valve lockup tests were satisfactory	
AC-15 (SC-7)	36A	December 14, 1967	Excessive LH_2 bulkhead pressure indication	Bulkhead leak repaired and verified by second tanking test
			Inoperative LO_2 transfer system dump valve	Replaced valve. Correct operation verified during second tanking test
		December 15, 1967	No significant anomalies. Satisfactory LH_2 bulkhead pressure permitted boiloff rate and vent valve lockup tests	

[b]Unless otherwise noted, corrective actions were taken after termination of tests.

approximately $T-30$ sec in the countdown, the *Atlas* inverter frequency abruptly changed to a redline condition when switched to internal power, which required replacement of the inverter and restart of the test. Other minor problems encountered included: (1) attenuation of the range safety command signal (partially due to antenna obscuration by personnel in front of the service tower), (2) failure of an autopilot GSE demodulator for *Atlas* vernier engine 1, and (3) a procedural error in sending the RF disable command.

e. Final launch preparations. After performance of the launch vehicle composite readiness test (five days before the scheduled opening of the launch window), a systematic procedure was followed to establish and verify a launch-ready configuration of the vehicle. Final mating of the spacecraft and the *Centaur* was scheduled for $F-4$ days (four days before opening of the launch window). (The actual time varied from three days to seven days due to workday scheduling and late decisions to extend the window opening.) During the final launch preparation period and before the launch countdown, a launch vehicle review was conducted to ascertain the status of all launch complex and vehicle systems as part of a consent-to-launch meeting (see Section V-B-8). Typical tasks which were performed between $F-5$ days and start of the launch countdown included:

(1) Spacecraft encapsulation and performance of system readiness test ($F-5$ days).

(2) *Atlas* and *Centaur* propulsion readiness and leak tests.

(3) Propellant utilization system checks.

(4) Spacecraft television target light checks.

(5) Propellant and gas storage topping.

(6) Erection and mating of spacecraft and performance of SRT ($F-4$ days).

(7) Launch complex and vehicle electrical circuits readiness checks.

(8) Propellant sampling.

(9) Mechanical and hydraulic systems readiness checks.

(10) Test of launch complex emergency power automatic transfer.

(11) Installation of launch vehicle ordnance, including retrorocket destruct shaped charge ($F-4$ days).

(12) Service tower removal for spacecraft RF tests.

(13) Spacecraft retrorocket safe–arm checks.

(14) *Atlas* tanking with RP-1 fuel ($F-2$ days).

(15) Pressurization and venting of start tanks.

(16) Range safety command system tests.

(17) *Atlas* and *Centaur* autopilot tests.

(18) *Centaur* tanking with H_2O_2 ($F-1$ day).

(19) Telemetry system tests.

(20) LH_2 transfer line purge ($F-1$ day).

A summary of anomalies which occurred during the final launch preparation periods is given in Table VIII-10, with two of the more significant ones described in more detail as follows.

Table VIII-10. Final launch preparation anomalies

Vehicle	Launch complex	Anomaly or contributing condition	Correction or improvement
AC-10/SC-1	36A	Leak in RP-1 dual-pump transfer system limited the tanking operation to one pump	Faulty instrumentation connector replaced
AC-7/SC-2	36A	Abnormal *Centaur* engine feedback traces, which were isolated to demodulator in instrumentation circuit	*Atlas* telemetry package and its accessory changed
		Improper telemetry-package commutator operation	
AC-12/SC-3	36B	Generic shorting in sustainer actuator	Replaced *Atlas* sustainer actuators. (Required fuel detanking, hydraulic refill and checks, engine alignment, autopilot recheck, and engine valve sequencing)
		Excessive pressure on shaped charge	Modified nose fairing
		Leak developed in boom system shutoff valve	Replaced valve and lines
		Bent pin in *Centaur* thruster pyrotechnic-relay box	Straightened pin
			Toxic-gas detector sensing lines rerouted for easy access and checking
			Atlas sustainer pneumatic-regulator transducer replaced and calibrated
			Removed *Atlas* displacement gyro in order to replace mounting screw in roll torque amplifier circuit board. Gyro tested and reinstalled.
AC-11/SC-4	36A	Manometer leaked	Replaced RP-1 fuel manometer on *Atlas*.
		Broken squib lead	Replaced *Atlas* in-flight pressurization valve
		Crack suspected in shaped-charge component	X rays proved unit acceptable
		Loose *Centaur* PU connector	Spacecraft demated and suspended in service tower to permit tightening
			Replaced insulation-panel relay box
			Replaced *Surveyor* safe–arm device to support composite readiness tests
			Replaced seal in *Atlas* LO_2 turbopump
			Replaced C-band beacon pressurization valve
AC-13/SC-5	36B	Faulty *Centaur* telemetry transmitter	Replaced
		Inaccurate *Atlas* PU system	Replaced with unit from AC-14
AC-14/SC-6	36B	Leaky *Centaur* attitude-control thruster	Replaced
		Damaged *Atlas* pneumatic control flex hose	Replaced
AC-15/SC-7	36A	Wet mercury in *Atlas* PU system	Replaced
		Inaccurate *Centaur* LO_2 burp orifice	Replaced

After normal tanking ($F-2$ days) of the RP-1 fuel aboard the *Atlas* stage of AC-12/SC-3, a decision was made on $F-1$ day to replace the *Atlas* sustainer actuators because of a generic shorting problem discovered during acceptance testing of another unit. Occurring so close to launch, this was a difficult component change, requiring several major operations, including detanking of the *Atlas* and the retesting of many vehicle systems. However, all required activities were accomplished without affecting the launch schedule and the *Atlas* was retanked before the start of the launch countdown.

Following a normal fuel tanking of the *Atlas* stage of AC-11/SC-4 on July 11, 1967, a manometer leak was discovered and the vehicle was detanked to permit installation of a spare manometer system. Retanking was accomplished on July 12 (opening day of the launch period). However, the discovery and correction of a loose PU system connector at the top of the *Centaur* hydrogen tank prevented a launch attempt on the first day of the launch period. The launch countdown was rescheduled to begin on the evening of July 13.

The spacecraft was suspended in the service tower while the corrective action was accomplished; it was then remated to the *Centaur* and the interface was revalidated.

F. Countdown and Launch Vehicle Flight Performance

1. Launch Countdown

The countdown enabled the space vehicle to be prepared for flight in a logical manner by a controlled sequence of tasks under the direction of the GD/C test conductor. The countdown sequence and events are illustrated in Fig. VIII-30.

Manual action could not be relied upon to verify within the close timing tolerances the many functions which occur during the terminal seconds of an *Atlas/Centaur* launch. For this reason, the final 10 sec of the terminal countdown sequence was automated. This automatic countdown sequence began when the test conductor activated the engine start switch and continued through *Atlas* ignition, thrust buildup, vehicle release, and liftoff. There was a provision for manually halting the countdown at any time up to $T-1$ sec, at which point the vehicle was committed to be released. Preplanned holds were incorporated into the countdown at $T-90$ and $T-5$ min for the purpose of increasing the launch-on-time capability. Any unscheduled hold time was subtracted from the preplanned hold time. The launch-on-time history of *Atlas/Centaur*, including research and development vehicles, is illustrated in Fig. VIII-31.

The hold at $T-90$ min was preplanned for 60 min duration for each vehicle except AC-7/SC-2, which was 70 min. For AC-15/SC-7, an option to extend this hold an additional 35 min was exercised to improve spacecraft telemetry coverage by downrange stations. There were no other extensions of the 60 min hold for the *Surveyor* missions.

The built-in hold at $T-5$ min was planned for 21 min duration for AC-10/SC-1, 15 min for AC-7/SC-2 and AC-11/SC-4, and 10 min for all other vehicles. Extensions of the $T-5$ min hold, with resultant unplanned penetrations of the launch windows, occurred as follows on the *Surveyor* missions:

(1) AC-7/SC-2: Extended 7 min to investigate an indication of a low-temperature reading within the *Centaur* H_2O_2 system. The investigation disclosed an erroneous indication and the countdown was resumed. At $T-115$ sec, the countdown was held and recycled to $T-5$ min to investigate the failure of the *Atlas* LO_2 boiloff valve to close at the start of the final flight pressure sequence, which prevented attainment of proper pressure. The valve was cycled several times with satisfactory closure; however, the automatic LO_2 topping system then failed and manual operation was employed to complete the LO_2 topping. After a total delay of almost 36 min, the countdown was resumed, though the LO_2 topping was still incomplete, in order for launch to occur before closure of the launch window. The vehicle was launched with only 0.176 sec of the available window remaining.

(2) AC-12/SC-3: Extended 51 min awaiting resolution of an indicated anomaly in the position transducer of the spacecraft vernier engine roll actuator. It was a notable achievement that the launch vehicle developed no problems during the countdown and was able to proceed smoothly to liftoff after holding an extra 51 min with all cryogenic chemicals aboard.

(3) AC-13/SC-5: Extended 18 min to investigate an apparent loss of launcher hydraulic stabilization pressure, which could cause vehicle instability under excessive ground-wind loads. The stabilization system was determined to be operational and the countdown was resumed. The problem was later traced to a loss of instrument air pressure (see Subsection D-2-b-3).

As established earlier in the countdown, the $T-5$ min hold for AC-14/SC-6 was extended 17 min to improve downrange tracking. There were no extensions of the $T-5$ min hold for AC-10/SC-1, AC-11/SC-4, or AC-15/SC-7.

A summary of anomalies, which were corrected with no impact on the countdown, are shown in Table VIII-11.

2. Launch Vehicle Flight Sequence of Events

The flight sequence of events for the *Surveyor* parking-orbit mode used on *Surveyors III, V, VI,* and *VII* was basically the same as the direct-ascent mode used for *Surveyors I, II,* and *IV* through *Centaur* first main engine start (MES 1). The parking-orbit mission employed a two-burn powered ascent from launch to injection of the *Centaur/Surveyor* into the required lunar transfer orbit. The direct-ascent mode employed a single, continuous, powered ascent. In each case, following the powered phase there was a brief coast period for damping *Centaur* engine shutdown transients and for executing certain commands to the spacecraft before the spacecraft was separated from the *Centaur*.

After spacecraft separation, the *Centaur* was programmed through a retromaneuver, which placed the vehicle in a completely different trajectory from the spacecraft to increase the separation distance and prevent interference between the *Centaur* and the spacecraft. The retromaneuver also prevented impact of the *Centaur* on the moon. Typical flight profiles for parking-orbit and direct-ascent modes are illustrated in Figs. VIII-32 and VIII-33. Examples of predicted and actual flight-sequence event times for the two types of ascent modes are shown in Table VIII-12.

Table VIII-11. Minor countdown anomalies

Vehicle	Launch complex	Anomaly	Correction
AC-7/SC-2	36A	Low voltage reading from telemetry at $T-260$ min	Telemetry battery replaced
		Commercial power failure at $L+60$ sec	Power returned
AC-11/SC-4	36A	Disconnect cap in *Centaur* H_2O_2 system leaked	Disconnect cap replaced
		Evidence of malfunction in *Centaur* GSE	Replaced overheated decoder for *Centaur* PU system
		Delay in *Centaur* guidance computer's acceptance of pitch and yaw steering program	
AC-14/SC-6	36B	A decay in upper-boom umbilical hydraulic pressure noted at $T-230$ min	Fault was traced to internal system leakage; a pump was used to maintain operating pressure for the remainder of the countdown

Hypergolic ignition of the *Atlas* sustainer and vernier engines occurred simultaneously about 2 sec before liftoff. Providing all prerelease functions were complete and the proper engine thrust was reached, an automatic release signal permitted retraction of the launcher hold-down mechanism and liftoff occurred. Following liftoff, the *Atlas* began a programmed roll sequence from fixed launcher azimuth setting to the desired launch azimuth. At $L+15$ sec the roll program was completed and the vehicle began a programmed pitch, bending the flight path over to maximize booster performance. Beginning with AC-7/SC-2, the pitch program included supplemental pitch and yaw programs for windshear relief. The vehicle continued under the control of the *Atlas* autopilot, with the booster engines being gimbaled for pitch, yaw, and roll control, the vernier engines active for roll control only and the sustainer engine centered. The vehicle reached Mach 1 in 64 sec; maximum aerodynamic pressure occurred at approximately 82 sec (*Atlas* SLV-3C).

Atlas BECO was initiated by an accelerometer signal (backed up by a programmer signal) when the vehicle acceleration equaled approximately 5.7 g at approximately 153 sec. *Atlas* SLV-3C events beginning with BECO occurred later than *Atlas* LV-3C events because of the increased propellant tank capacity of the *Atlas* SLV-3C (see Table VIII-12). Times given in this description are for SLV-3C. Approximately 3 sec after BECO, with the sustainer engine locked in a null position, the booster section was jettisoned by release of pneumatically operated latches. At BECO+8 sec the *Centaur* guidance system was enabled to provide steering commands to the *Atlas* autopilot and the sustainer engine gimbals were unlocked. This allowed the sustainer thrust vector to control the vehicle attitude in pitch and yaw while the two *Atlas* vernier engines continued to provide roll control, as commanded by the *Atlas* autopilot.

About 40 sec after booster jettison, the *Atlas* autopilot issued an insulation panel jettison signal to a remote relay installation in the interstage adapter. The relays

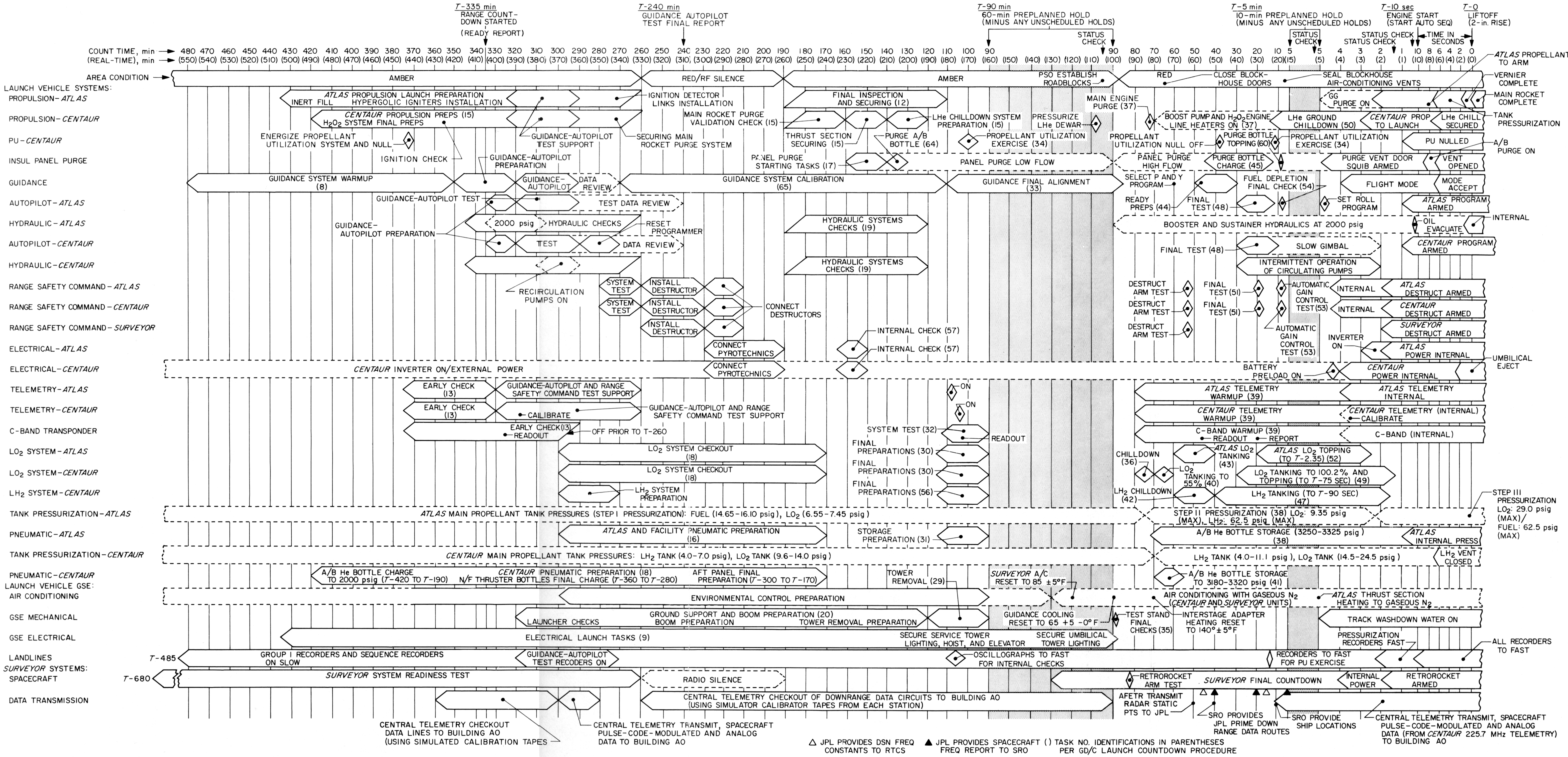


Fig. VIII-30. Typical launch countdown events (AC-13/SC-5)

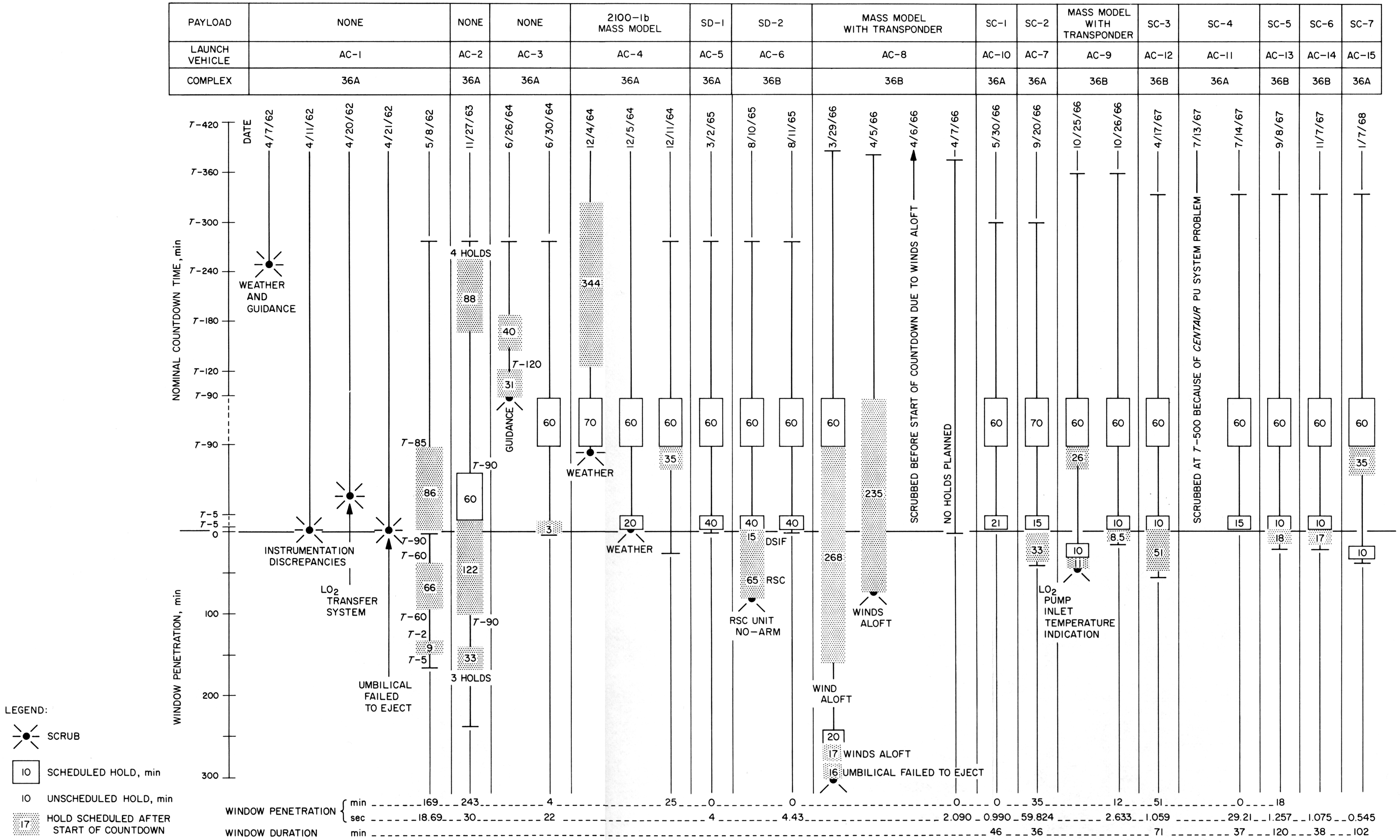


Fig. VIII-31. ***Atlas/Centaur*** **launch-on-time history**

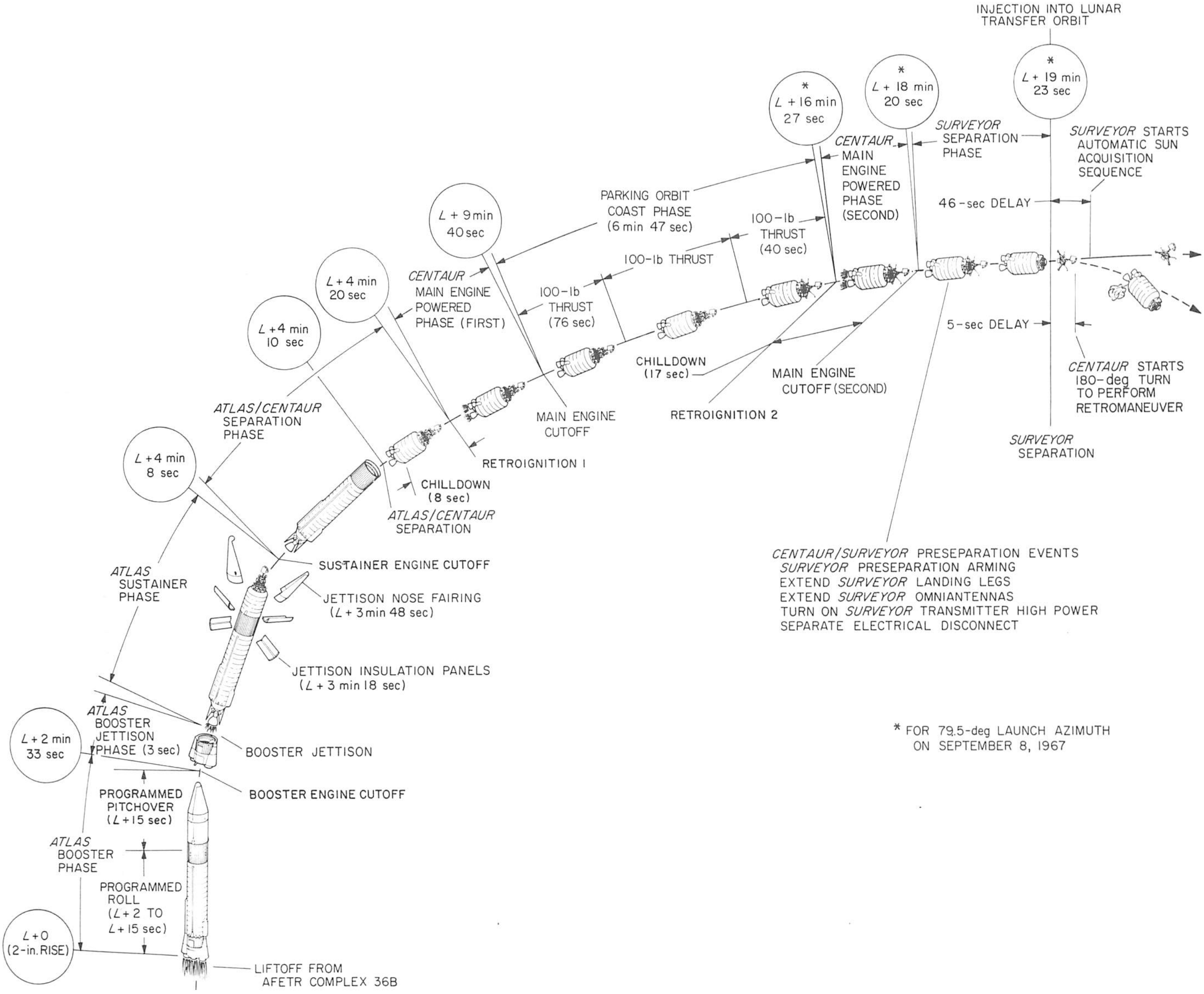


Fig. VIII-32. *Surveyor V* postlaunch events (parking orbit)

closed, sending voltage from the *Atlas* main battery to pyrotechnic detonators. The detonators fired to start detonation of a chain of linear shaped charges, cutting the insulation panels surrounding the *Centaur* tank into four sections. The aft end of each section was hinged at two places to the interstage adapter. As the sections rotated outward under the effects of vehicle acceleration, the hinge fittings released at a rotation of approximately 46 deg, allowing the panels to clear the vehicle. Approximately 30 sec later, the *Atlas* programmer issued a nose-fairing jettison signal which closed two remote relays, sending power from two batteries to the pyrotechnically operated nose fairing latches and, 1 sec later, the nose fairing thruster-bottle valves. The latches were released; GN_2 at 2400 psig forced the nose fairing halves apart. After the fairing valves rotated outward, their hinges released, allowing the halves to clear the vehicle.

Other programmed events which occurred during the sustainer phase of flight were:

(1) Unlocking of the *Centaur* hydrogen-tank vent valve to permit venting as required to relieve hydrogen boiloff pressure.

(2) Starting of the *Centaur* boost pumps about 43 sec before *Centaur* MES 1.

(3) Locking of the *Centaur* oxidizer tank vent valve followed by burp pressurization of the tank.

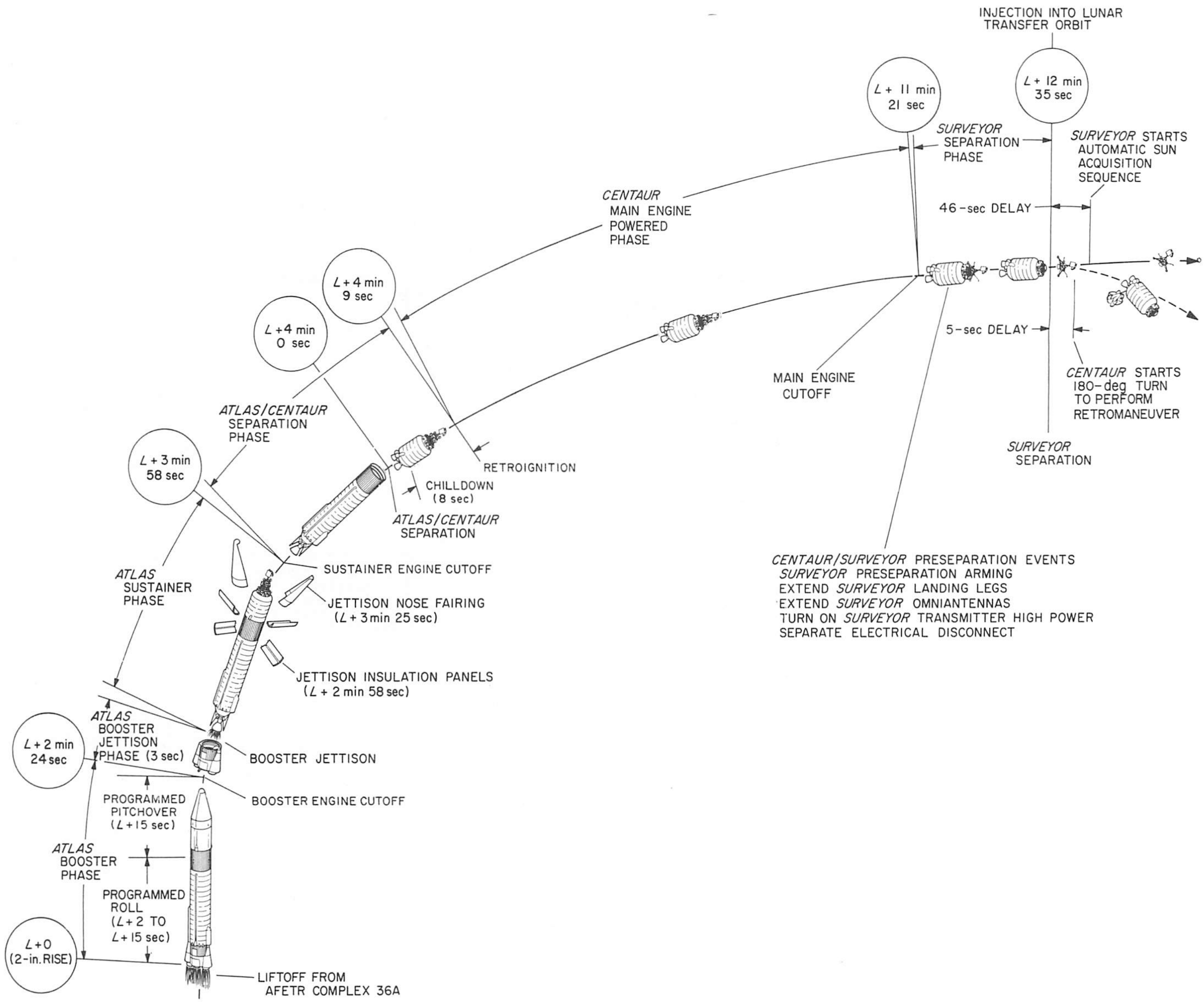


Fig. VIII-33. *Surveyor IV* postlaunch events (direct ascent)

The vehicle continued its ascent with the sustainer and vernier engines thrusting until propellant depletion. A signal from a propellant depletion probe near the bottom of the *Atlas* fuel tank (or in the oxidizer feed line) caused sustainer and vernier engine cutoff. Backup SECO signals were also issued by an accelerometer and the *Atlas* programmer. At SECO, the *Centaur* hydrogen tank vent valve was locked and burp pressurization of the tank was started.

Approximately 2 sec after SECO, the interstage adapter was cut by a pyrotechnic train immediately aft of the *Centaur* aft mating ring. Retrorockets mounted on the aft end of the *Atlas* sustainer tank were fired to back the *Atlas*, together with the interstage adapter, away from the *Centaur*. Both of these events were commanded by the *Atlas* autopilot and powered by the *Atlas* electrical system.

Approximately 10 sec after separation, the *Centaur* main engines were started. The time between separation and MES 1 was used for liquid-helium chilldown of the propulsion system. During this period, attitude control was maintained by the H_2O_2 attitude control engines. All of these events were controlled by the *Centaur* autopilot, which assumed control of the vehicle at separation. During *Centaur*-powered flight, the vehicle attitude was controlled about all three axes by gimbaling of the main

Table VIII-12. Typical boost-phase flight events

Event	Direct-ascent mode (AC-11/SC-4)[a]			Parking-orbit mode (AC-13/SC-5)[b]		
	Mission time (predicted), hr:min:sec	Mission time (actual), hr:min:sec	Mark	Mission time (predicted), hr:min:sec	Mission time (actual), hr:min:sec	Mark
Liftoff (2-in. rise), GMT reference	—	11:53:29.215		—	07:05:01.059	
Liftoff (2-in. rise)	L+00:00:00.00	L+00:00:00.00		L+00:00:00.00	L+00:00:00.00	
Atlas booster engine cutoff (BECO)	L+00:02:23.66	L+00:02:21.9	1	L+00:02:33.49	L+00:02:33.38	1
Jettison booster package	L+00:02:26.76	L+00:02:25.0	2	L+00:02:36.59	L+00:02:36.33	2
Jettison *Centaur* insulation panels	L+00:02:57.66	L+00:02:55.8	3	L+00:03:18.49	L+00:03:17.97	3
Jettison nose fairing	L+00:03:24.66	L+00:03:22.6	4	L+00:03:48.49	L+00:03:47.75	4
Atlas sustainer engine cutoff (SECO)	L+00:03:57.73	L+00:03:59.35	5	L+00:04:08.53	L+00:04:06.40	5
Atlas/Centaur separation	L+00:03:59.63	L+00:04:01.24	6	L+00:04:10.53	L+00:04:08.33	6
Centaur first main-engine start (MES 1)	L+00:04:09.23	L+00:04:10.8	7	L+00:04:20.03	L+00:04:17.80	7
Centaur first main-engine cutoff (MECO 1)	L+00:11:20.99	L+00:11:27.8	8	L+00:09:39.10	L+00:09:47.06	8
100-lb thrust on				L+00:09:39.10	L+00:09:47.06	9
100-lb thrust off				L+00:10:55.10	L+00:11:02.84	10
6-lb thrust on				L+00:10:55.10	L+00:11:02.9	11
100-lb thrust on				L+00:15:46.68	L+00:15:49.6	12
Centaur C1 MES 2				L+00:16:26.68	L+00:16:29.6	13
Centaur C2 MES 2				L+00:16:26.68	L+00:16:29.6	14
Centaur MECO 2				L+00:18:19.49	L+00:18:23.8	15
Extend *Surveyor* landing legs command	L+00:11:53.2	L+00:11:55	9	L+00:18:40.68	L+00:18:43.8	16
Extend *Surveyor* omnidirectional antenna command	L+00:12:03.7	L+00:12:05	10	L+00:18:51.68	L+00:18:54.0	17
Surveyor transmitter "high power on" command	L+00:12:24.2	L+00:12:26	11	L+00:19:11.68	L+00:19:14.1	18
Surveyor/Centaur electrical disconnect	L+00:12:29.7	L+00:12:31.35	12	L+00:19:17.68	L+00:19:20.5	19
Surveyor/Centaur separation	L+00:12:35.2	L+00:12:36.86	13	L+00:19:22.68	L+00:19:25.6	20
Start *Centaur* 180-deg turn	L+00:12:40.2	L+00:12:42.0	14	L+00:19:27.68	L+00:19:31.6	21
Start *Centaur* lateral thrust				L+00:20:07.68	L+00:20:10.6	22
Cut off *Centaur* lateral thrust				L+00:20:27.68	L+00:20:30.6	23
Start *Centaur* tank blowdown (retro)	L+00:16:35.2	L+00:16:36.9	15	L+00:23:22.68	L+00:23:25.6	24
Cut off *Centaur* blowdown, 100-lb thrust on				L+00:27:32.68	L+00:27:35.6	25
Cut off *Centaur* electrical power, 100-lb thrust off				L+00:29:12.68	L+00:29:15.6	26
Cut off *Centaur* electrical power, and end blowdown	L+00:20:45.2	L+00:20:46.9	16			
Energize power changeover switch	L+00:20:45.2	L+00:20:46.9	17			

[a]Utilized *Atlas* LV-3C.
[b]Utilized *Atlas* SLV-3C.

engines. The autopilot controlled the vehicle using steering commands issued by the guidance system. On parking-orbit missions the *Centaur* first burn lasted about 5.5 min, and on direct-ascent missions the *Centaur* burned continuously for about 7.3 min. Upon attainment of required vehicle velocity, the guidance system issued a MECO 1 command, shutting down the main engines.

After MECO 1, for parking-orbit flights, the *Centaur* and spacecraft entered the parking-orbit coast phase of flight. Two of the 50-lb H_2O_2 engines were turned on for 76 sec after MECO 1, after which two of the 3-lb axial engines were turned on to retain the propellants in the proper location in the *Centaur* tanks. All of the H_2O_2 engines were enabled for attitude control as required during the coast phase. The *Centaur* stage could support a parking-orbit coast period of from 116 sec to 25 min for *Surveyor* missions with the actual duration dependent on the liftoff time.

About 40 sec before the end of the coast period, the 3-lb engines were turned off and two of the 50-lb engines were turned on again until MES 2 to ensure propellant control during the events preceding the second ignition. These events include burp pressurization of the propellant tanks, starting of the boost pumps 28 sec before MES 2, and initiation of the *Centaur* engine prestart chilldown sequence 17 sec before MES 2. The *Centaur* second burn on parking-orbit missions was about 1.9 min long.

At MECO 2 (MECO 1 on direct-ascent missions) the H_2O_2 engines were enabled for attitude stabilization prior to spacecraft separation. During the approximate 1-min period between MECO and spacecraft separation, the following signals were transmitted to the spacecraft from the *Centaur* programmer: extend spacecraft landing gear, unlock spacecraft omnidirectional antennas, and turn on spacecraft transmitter high power. An arming signal was also provided by the *Centaur* during this period to enable the spacecraft to act on the preseparation commands. Spacecraft electrical disconnect was commanded about 5.5 sec before spacecraft separation. Upon firing of the squibs for spacecraft separation, three compressed springs at the mechanical interface provided impulse to separate the spacecraft with a velocity of about 1 ft/sec. The *Centaur* attitude control engines were disabled for about 5 sec during spacecraft separation to minimize vehicle turning moments.

At 5 sec after spacecraft separation the *Centaur* began a 180-deg turnaround maneuver under the control of the autopilot using guidance system steering signals and using the attitude control engines to point the aft end of the stage in the direction of the flight path. About 40 sec after beginning the turn, two of the 50-lb engines were operated for 20 sec, while the *Centaur* continued its turn, to provide lateral separation of the *Centaur* from the spacecraft. About 240 sec after spacecraft separation, when the distance between the *Centaur* and spacecraft had increased beyond possible exhaust plume contamination, the *Centaur* hydraulic recirculation pump was turned on and the prestart valves were opened. Residual propellants were expelled for 250 sec. The tank valves were then opened to prevent any subsequent tank pressure buildup. On the AC-12, -13, -14, -15 flights, an H_2O_2 propellant depletion check was conducted after the retromaneuver, in which two of the 50-lb engines were operated for 100 sec. The final *Centaur* event was actuation of the power changeover switch to remove all vehicle power except for telemetry and the C-band beacon. The *Centaur* tank continued in a highly elliptical earth orbit.

3. Performance Summary

The performance of each of the seven *Atlas/Centaur* vehicles used on the *Surveyor* missions was normal, providing a very satisfactory powered-flight phase and accurate injection of each spacecraft into the prescribed lunar transfer trajectory. There were no known failures or functional anomalies of any launch vehicle/spacecraft interface system except for the loss of three telemetry measurements during the launch of *Surveyor II* (see Subsection 3-d). The performance of the various *Atlas/Centaur* systems is discussed below.

a. Guidance and flight control. A measure of the performance of the guidance system was the accuracy with which the spacecraft were injected into the prescribed lunar transfer trajectories. The injection accuracy was expressed in terms of the spacecraft midcourse velocity change which would have been required 20 hr after injection to correct the flight path for a lunar impact at the prelaunch target point. The magnitude of the midcourse velocity change was required to be less than 16.7 m/sec to correct for miss-plus-time-of-flight errors using the square root of the sum of the squared error values, or root sum squared (rss), assuming 1 σ injection errors. The excellence of the guidance system performance was evident by the small midcourse velocity changes which would have been required for each spacecraft, as shown in Table VIII-13. These computations were made using the latest spacecraft premidcourse orbit determinations.

The guidance system discrete commands (BECO, SECO backup, and MECO) were generated as planned during each flight. When guidance steering was enabled from BECO+8 sec until SECO and again between MES and MECO, initial vehicle attitude errors were quickly nulled and close alignment with the commanded steering vector was maintained. A near-circular parking orbit

Table VIII-13. Accuracy of spacecraft injection

Parameter	AC-10/SC-1	AC-7/SC-2	AC-12/SC-3	AC-11/SC-4	AC-13/SC-5	AC-14/SC-6	AC-15/SC-7
Miss distance (uncorrected impact point from prelaunch target point), km[a]	460	142	466	176	46	126	77
Velocity change for miss only, m/sec[b]	3.60	1.02	4.00	1.51	0.55	1.25	0.46
Velocity change for miss plus time of flight, m/sec[b]	5.90	4.44	6.63	1.94	0.74	1.32	1.18

[a]These are the distances by which the *Surveyor* would have missed the prelaunch-targeted lunar coordinates if no midcourse correction had been applied.
[b]These theoretical corrections would apply 20 hr after injection to correct the spacecraft to the prelaunch-targeted lunar point. The actual corrections executed to land at the final aim point were different.

was achieved close to the desired 90 nmi radius for the parking-orbit missions as shown in Table VIII-14.

Autopilot performance was satisfactory throughout each flight with proper initiation of programmed events and control of vehicle stability.

The *Centaur* attitude control system performed properly during each flight, maintaining the required vehicle attitude following the *Centaur* powered phases of flight and providing the necessary low-level axial thrust for propellant control during retromaneuver and during parking-orbit coast periods. The attitude control system also provided the necessary low-level thrust for initial lateral separation from the spacecraft on all flights. At the start of the *Centaur* retromaneuver on each flight, a brief loss of thrust of the attitude control engines was indicated. This anomaly was attributed to gas bubbles in the H_2O_2 system.

During the *Centaur* phases of flight, the vehicle was rate-stabilized in roll rather than roll-position-stabilized.

b. Propulsion and propellant utilization. The *Atlas* propulsion and PU systems performed satisfactorily during each flight. A new sustainer engine LO_2 dome-to-elbow seal was used on AC-14 and AC-15 to minimize the possibility of LO_2 leakage in this area. The PU system satisfactorily controlled propellant levels. The predicted and actual *Atlas* residuals (propellant above pump) are compared in Table VIII-15.

The *Centaur* propulsion system performed well during each flight including both burn periods on parking-orbit missions. Burn times were longer than predicted in all cases; however, no detrimental effects on missions were observed. *Centaur* main engine burn times are shown in Table VIII-16. Engine thrust and specific impulse I_{sp} data are tabulated in Table VIII-17.

Table VIII-14. Parking-orbit parameters

Parameter[a]	Values			
	AC-12/ SC-3	AC-13/ SC-5	AC-14/ SC-6	AC-15/ SC-7
Apogee, nmi	95.3	88.7	89.7	90.7
Perigee, nmi	88.2	82.8	85.1	87.5
Period, min	87.8	87.7	87.8	87.8

[a]Goal was a circular 90-nmi orbit.

The turbine inlet pressure and speed data of the fuel and oxidizer boost pumps exhibited unexpected trends after MES 2 on the parking-orbit flights. These trends were indicative of gas flow through the hydrogen peroxide catalyst beds and had no apparent effects on main engine performance.

The *Centaur* PU system also performed well. The predicted and actual *Centaur* usable residuals after MECO 1 and MECO 2 together with the equivalent additional burn times are shown in Table VIII-18.

c. Pneumatic, hydraulic, and electrical power systems. Operation of all pneumatic, hydraulic, and electrical systems for both *Atlas* and *Centaur* was normal for all flights. Both *Centaur* propellant tanks were maintained at satisfactory levels during all phases of flight, with normal occurrences of the burp pressurization sequences and hydrogen tank venting. There were no unexpected voltage demands or transients.

d. Telemetry, tracking, and range safety command. The *Atlas* and *Centaur* instrumentation and telemetry systems functioned well with only a few measurement discrepancies. During the *Surveyor* II (AC-7) launch phase of flight, three of the five accelerometers located on the spacecraft/adapter interface failed to provide vibration data. Specific cause of this failure is unknown, but faulty accelerometer/amplifier connections are suspected.

The *Centaur* C-band radar apparently operated normally although a weak signal-strength period and some loss of data by downrange receivers occurred on some flights. It is believed that *Centaur* roll contributed to this problem. An evaluation of the system can only be made on the basis of received tracking data and station operator logs, because the airborne system was not instrumented.

The *Atlas* and *Centaur* range safety command systems performed satisfactorily. About 14–16 sec after MECO 1, the range safety command to disable the destruct systems was sent and properly executed.

e. Vehicle loads and environment. Vehicle loads and thermal environment were within expected ranges on all flights. Maximum axial acceleration during the booster phases ranged from 5.69 to 5.74 *g* and occurred at BECO. During the sustainer phases, maximum axial accelerations varied from 1.73 to 1.81 *g* just before SECO. The range of maximum longitudinal oscillations was 0.54 *g*

Table VIII-15. *Atlas* propellant residuals at SECO[a]

Residual	AC-10	AC-7	AC-12	AC-11	AC-13	AC-14	AC-15
Oxidizer (LO_2)							
Actual	386	393	404	401	334	277	453
Predicted	221	504	56	229	138	256	246
Fuel (RP-1)							
Actual	137	133	407	353	193	197	247
Predicted	141	315	197	215	194	199	223

[a]All values given in pounds.

Table VIII-16. *Centaur* main engine burn time

Burn	AC-10/SC-1	AC-7/SC-2	AC-12/SC-3	AC-11/SC-4	AC-13/SC-5	AC-14/SC-6	AC-15/SC-7
First:[a]							
Actual	438.35	439.66	340.49	437.0	327.8	323.9	332.9
Predicted	435.16	436.84	327.24	430.1	319.1	317.3	320.7
Difference	3.19	2.82	13.25	6.9	8.7	6.6	12.2
Second:[a]							
Actual	—	—	111.29	—	114.3	115.7	115.6
Predicted	—	—	109.28	—	112.8	113.5	113.6
Difference	—	—	+2.01	—	1.5	2.2	2.0

[a]All values given in seconds.

Table VIII-17. *Centaur* main engine operating levels

Vehicle	First burn[a]				Second burn[a]			
	Thrust,[b] lbf		Specific impulse,[c] sec		Thrust,[b] lbf		Specific impulse,[c] sec	
	C-1 engine	C-2 engine	C-1 engine	C-2 engine	C-1 engine	C-2 engine	C-1 engine	C-2 engine
AC-10/SC-1	14,955	15,010	432.8	434.2	—	—	—	—
AC-7/SC-2	15,000	14,950	433.0	434.0	—	—	—	—
AC-12/SC-3	14,717	14,506	442.7	442.7	14,885	14,663	443.1	442.9
AC-11/SC-4	15,030	15,050	434.0	434.0	—	—	—	—
AC-13/SC-5	14,900	14,800	444.0	443.0	15,100	15,000	444.0	442.0
AC-14/SC-6	14,700	14,800	443.0	442.0	14,800	14,900	443.0	442.0
AC-15/SC-7	14,694	14,669	443.0	443.0	15,056	14,865	443.0	443.0

[a]Before unnulling of the PU system at MES+90 sec.
[b]Nominal thrust, all vehicles: 15,000 lb.
[c]Nominal I_{sp}, AC-7, AC-10, AC-11: 433 sec; AC-12, AC-13, AC-14, AC-15: 444 sec.

Table VIII-18. *Centaur* usable propellant residuals

Item	AC-10	AC-7	AC-12 MECO 1	AC-12 MECO 2	AC-11	AC-13 MECO 1	AC-13 MECO 2	AC-14 MECO 1	AC-14 MECO 2	AC-15 MECO 1	AC-15 MECO 2
Oxidizer (LO_2), lb											
Actual	99 ±39	131 ±40	6733	367	308	7137	576	7412	647	7133	1564
Predicted	146	208	—	317	295	7180	577	7250	548	7230	1446
Fuel (LH_2), lb											
Actual	46 ±8	56.8 ±9	1447	112	78.9	1539	140.2	1593	164	509	147
Predicted	49	56.6	—	83	72.2	1436	143.6	1450	130	795	139
Equivalent additional burn time, sec	1.6 ±0.7	2.3	6.5		5.3	10.2		11.4		9.0	

Table VIII-19. Maximum measured peak-to-peak acceleration, *g*, *Surveyors I* through *IV*

Event	Surveyor	Accelerometer CY 520	CY 530	CY 540	CY 770	CY 780
Liftoff	*I*	3	4	4	12	2
	II	[a]	[a]	3	[a]	3
	III	4.8	[a]	4.1	16.6	3.1
	IV	4	4	5	7	2
Maximum aerodynamic loading	*I*	[b]	[b]	[b]	[b]	[b]
	II	[b]	[b]	[b]	[b]	[b]
	III	2.4	[a]	2.2	5.7	1.64
	IV	2.7	2.1	2.8	4.7	0.8
Booster engine cutoff	*I*	3	[c]	[c]	[c]	[c]
	II	[a]	[a]	[c]	[a]	3
	III	3.3	[a]	[c]	3.7	1.3
	IV	3	2	[c]	[c]	[c]
Insulation panel jettison	*I*	>20[d]	[c]	[c]	>20[d]	[c]
	II	[a]	[a]	[c]	[a]	[c]
	III	>20[d]	[a]	[c]	[c]	[c]
	IV	>20[d]	[c]	[c]	>20[d]	[c]
Shroud separation	*I*	3	[c]	[c]	[c]	[c]
	II	[a]	[a]	[c]	[a]	2.6
	III	4.3	[a]	[c]	[c]	[c]
	IV	4	[c]	[c]	8	[c]
Sustainer engine cutoff	*I*	1	[c]	1	[c]	[c]
	II	[a]	[a]	[c]	[a]	[c]
	III	1.2	[a]	1.3	[c]	[c]
	IV	2	2	[c]	[c]	[c]
Atlas/Centaur separation	*I*	>20[d]	>20[d]	[c]	[c]	[c]
	II	[a]	[a]	[c]	[c]	[c]
	III	>20[d]	[a]	[c]	[a]	[c]
	IV	>20[d]	[c]	[c]	[c]	[c]

Event	Surveyor	Accelerometer CY 520	CY 530	CY 540	CY 770	CY 780
MES 1	*I*	1	1	[c]	[c]	[c]
	II	[a]	[a]	[c]	[a]	[c]
	III	1.4	[a]	[c]	[c]	[c]
	IV	1	[c]	[c]	[c]	[c]
MECO 1	*I*	4.5	[c]	[c]	4.5	[c]
	II	[a]	[a]	[c]	[a]	[c]
	III	3.3	[a]	[c]	3.3	[c]
	IV	3.3	2	[c]	[c]	[c]
MES 2	*I*	[e]	[e]	[e]	[e]	[e]
	II	[e]	[e]	[e]	[e]	[e]
	III	2.0	[a]	[c]	[c]	[c]
	IV	[e]	[e]	[e]	[e]	[e]
MECO 2	*I*	[e]	[e]	[c]	[e]	[e]
	II	[e]	[e]	[e]	[e]	[e]
	III	0.45	[a]	[c]	[c]	[c]
	IV	[e]	[e]	[e]	[e]	[e]
Legs extend	*I*	20	[c]	10	[c]	[c]
	II	[a]	[a]	[c]	[a]	[c]
	III	18	[a]	[c]	13	[c]
	IV	18	12	[c]	[e]	[c]
Antenna deployment	*I*	20	[c]	12	[c]	[c]
	II	[a]	[a]	[c]	[a]	[c]
	III	16	[a]	[c]	[c]	[c]
	IV	18	[c]	[c]	[c]	[c]

[a]These transducers were inoperative.

[b]Data not available.

[c]Commutated channel. This transducer was not being monitored at this event.

[d]Exceeded bandwidth of channel.

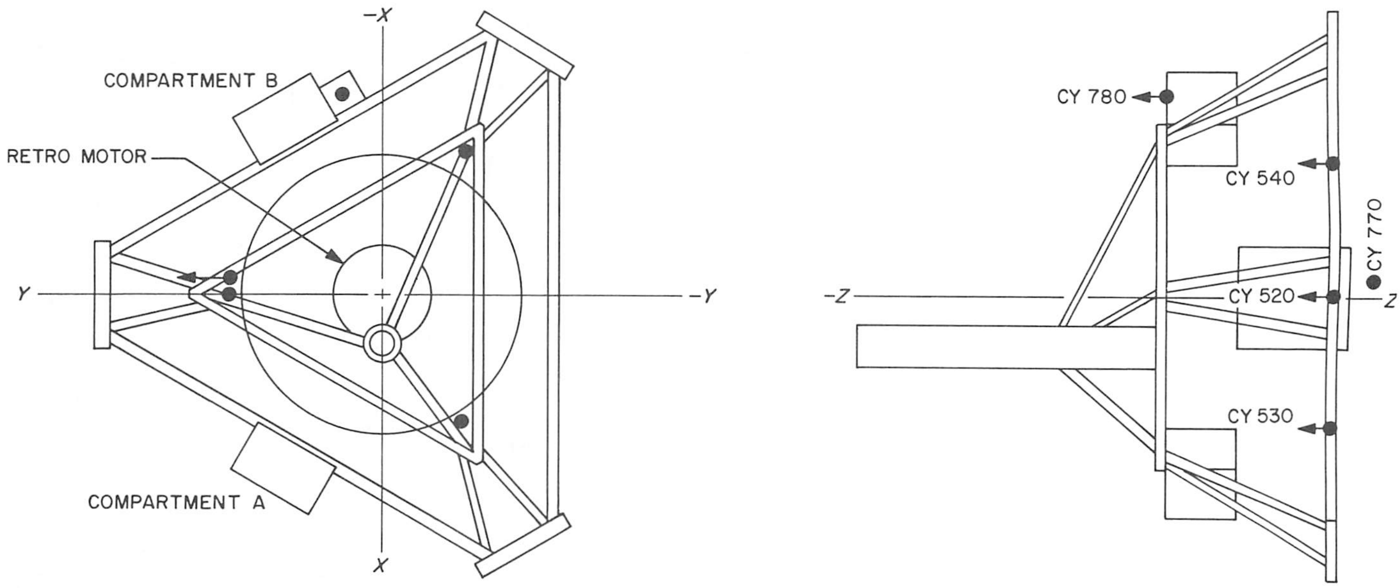
[e]Not applicable.

(peak-to-peak) at 6.0 Hz to 0.72 *g* (peak-to-peak) at 6.1 Hz. These maximum oscillations occurred just after liftoff (0.4–0.7 sec) and were damped out within 15–20 sec.

Five accelerometers were installed in the vicinity of the *Centaur/Surveyor* interface on *Surveyors I–IV* as depicted in Fig. VIII-34. The maximum peak-to-peak accelerations recorded by these accelerometers for each flight are compared in Table VIII-19. The five accelerometers indicated steady-state vibration levels of less than 1 *g* (rms).

Two high-frequency accelerometers were used to obtain vibration measurements on *Surveyors V* and *VI*. A third high-frequency accelerometer was added on *Surveyor VII*. The locations and recorded measurements of these accelerometers are shown in Tables VIII-20 and VIII-21. The radially oriented accelerometer indicated a maximum steady-state vibration level of 2.1 *g* (rms) within a 10–660 Hz bandwidth during the *Surveyor V* and *VI* flights and 2.0 *g* (rms) during the *Surveyor VII* flight. The maximum longitudinal steady-state level indications during launch were 1.6 *g* (rms) for *Surveyor V*, and 1.3 *g* (rms) for *Surveyors VI* and *VII*—all within a 10–2100 Hz bandwidth. Data from the third accelerometer added on *Surveyor VII* was transmitted through the *Atlas* telemetry system.

Low-level, short-duration transients were recorded throughout the launch phase of each flight. Shocks of 600–700 Hz for a duration of less than 5 msec were observed in response to shaped-charge firings at insulation panel jettison and *Atlas/Centaur* separation, with magnitudes exceeding the 20-*g* (peak-to-peak) range of the telemetry channel. Two of the more severe transients not associated with a discrete flight event occurred at



TRANSDUCER	LOCATION	RANGE, *g*	FREQUENCY RANGE, Hz	REMARKS
CY 520	SPACECRAFT, NEAR ADAPTER ATTACH POINT 1	±10	2-1150	CONTINUOUS
CY 530	SPACECRAFT, NEAR ADAPTER ATTACH POINT 2	±10	2-970	COMMUTATED
CY 540	SPACECRAFT, NEAR ADAPTER ATTACH POINT 3	±10	2-970	COMMUTATED
CY 770	ADAPTER, NEAR SPACECRAFT ATTACH POINT 1	±10	2-970	COMMUTATED
CY 780	SPACECRAFT, IN FLIGHT CONTROL SENSOR GROUP	±10	2-970	COMMUTATED

Fig. VIII-34. Launch phase accelerometer location (*Surveyor I–IV*)

Table VIII-20. Maximum measured acceleration during launch phase[a]

Event	Radial accelerometer			Longitudinal accelerometer		
	Surveyor V	*Surveyor VI*	*Surveyor VII*	*Surveyor V*	*Surveyor VI*	*Surveyor VII*
Liftoff	1.95 rms	2.26 rms	1.80 rms	1.26 rms	1.21 rms	1.04 rms
Maximum aerodynamic loading	0.61 rms	0.71 rms	0.83 rms	0.50 rms	0.55 rms	0.62 rms
Booster engine cutoff	2.0	1.4	<1.0	3.0	4.5	3.6
Insulation panel jettison	>10.0	>10.0	>10.0	>10.0	>10.0	>10.0
Nose fairing separation	2.0	2.5	5.1	3.0	4.1	>10.0
Sustainer engine cutoff	1.0	1.5	<1.0	1.0	3.0	<1.0
Atlas/Centaur separation	>10.0	>10.0	>10.0	>10.0	>10.0	>10.0
MES 1	<1.0	<1.0	<1.0	<1.0	<1.0	<1.0
MECO 1	1.8	1.0	1.2	3.8	2.8	3.1
MES 2	<1.0	<1.0	>1.0	<1.0	<1.0	<1.0
MECO 2	3.2	2.7	2.3	2.8	4.5	4.1
Landing gear extension	4.5	7.0	—	7.5	>10	—
Landing gear lock	4.0	3.5	—	6.4	5.6	—
Antenna deployment	2.6	2.9	2.9	7.2	7.2	>10.0

[a]All readings are in earth gravity units (g), zero-to-peak unless otherwise noted.

Table VIII-21. Accelerometer data, launch phase

Accelerometer	Transducer	Location	Range, g	Frequency range, Hz	Remarks
Radial (≈ Y–Y axis)	CA 7730	Forward adapter near spacecraft leg 1 attach point	±10	2–970	Continuous
Longitudinal (Z–Z axis)	CA 7720	Forward adapter near spacecraft leg 3 attach point	±10	2–2300	Continuous
Tangential (Y–Y axis)[a]	CA 2130	Forward adapter just below spacecraft attachment ring, 90 deg counterclockwise from radial accelerometer looking aft			Continuous until *Atlas* separation

[a]Data for the tangential accelerometer not available.

Table VIII-22. Spacecraft compartment temperatures

Parameter	AC-10/SC-1	AC-7/SC-2	AC-12/SC-3	AC-11/SC-4	AC-13/SC-5	AC-14/SC-6	AC-15/SC-7
Ambient temperature at liftoff, °F	88	86	86	84	82	81	84
Minimum temperature, °F	73	70	74	74	70	60	72
Time from launch, sec	85	92	85	86	87	88	76
Temperature at nose fairing jettison, °F	82	81	80	80	78	75	79

532.6 and 683.5 sec after liftoff of *Surveyor IV*. The magnitudes were approximately 8.0 and 4.5 *g* (peak-to-peak), respectively, at a frequency of 1400–1600 Hz. Since these levels were well below the vibration levels to which the spacecraft was acceptance-tested before launch, it was concluded that they were not a contributing factor to the in-flight failure of *Surveyor IV*. The exact cause of the low-level transients, which were not associated with any discrete spacecraft or booster event, are unknown but are believed to be due to a combination of dynamic and thermal loads in the area of the *Centaur* forward bulkhead and adapter structure.

The *Surveyor* compartment thermal and pressure environments were normal throughout the flight. Temperature histories are shown in Table VIII-22. A gradual decrease to the minimum temperatures noted in the table resulted from expansion during ascent. The ambient pressure at liftoff was 14.7 psia for each launch and decayed characteristically to essentially zero prior to nose fairing jettison.

f. ***Separation systems.*** All vehicle separation systems functioned normally. Booster-package jettison occurred as planned, with resulting vehicle rates and high-frequency accelerometer data consistent between flights. Satisfactory insulation-panel jettison was confirmed by normal transient effects on vehicle rates, axial acceleration, vibration, etc. The times of 35-deg rotation of the four insulation panels during jettison was provided by a breakwire at one hinge arm of each panel. Average panel rotational rates to the 35-deg position, derived from the breakwire instrumentation, were approximately 81 deg/sec.

Normal separation of the nose fairing was verified by indications of 3-deg rotation from disconnect wires which were incorporated in the pullaway electrical connectors of each nose fairing half. The spacecraft compartment pressure remained at zero throughout each nose fairing jettison and no pressure surges were noted at thruster bottle pressure discharge.

Atlas/Centaur separation occurred as planned. *Atlas* deceleration during retrorocket firing, as determined from axial acceleration telemetry measurements, was as expected and verified simultaneous ignition of all eight retrorockets. The individual vehicle residual rates after separation in pitch, yaw, and roll were small, indicating no interference between the interstage adapter and the *Centaur* engine compartment components. Examples of separation parameters (for AC-13) are: (1) time required to separate *Atlas* and *Centaur* a distance of 110 in. was 1.03 sec, and (2) average *Atlas* deceleration during retrorocket firing was 0.61 *g* for 0.70 sec.

At spacecraft separation, data from the linear potentiometers incorporated in the spacecraft separation spring assemblies indicated that the variance between the first motion of three separation springs was 0–2 msec. Spring extension to the full 1-in. position was normal and nearly identical, producing a spacecraft separation rate of approximately 1 ft/sec on each flight. Angular rates of the *Centaur* just prior to separation and calculated *Surveyor* rates after separation are listed in Table VIII-23.

g. ***Centaur retromaneuver.*** All phases of the *Centaur* retromaneuver were executed as planned on each flight. The computed *Centaur/Surveyor* separation distance 5 hr after separation was far in excess of the required 336 km at that time. A tabulation of the separation distance achieved on each flight, together with other *Centaur* orbital data, is shown in Table VIII-24.

Table VIII-23. Angular rates at spacecraft separation[a]

Vehicle	*Centaur* rates prior to separation				Calculated *Surveyor* rates after separation[b]			
	Pitch	Yaw[c]	Roll	Vector sum	Pitch	Yaw[c]	Roll[d]	Vector sum
AC-10/SC-1	−0.05	−0.15	0.01	0.16	−0.298	0.163	0.01	0.34
AC-7/SC-2	0.117	−0.151	0.033	0.194	0.104	0.378	0.033	0.393
AC-12/SC-3	0.22	−0.19	−0.03	0.29	0.30	0.47	−0.03	−0.56
AC-11/SC-4	0.09	−0.18	−0.05	0.21	0.16	−0.36	0	0.39
AC-13/SC-5	0.06	−0.14	−0.08	0.17	1.42	−0.04	−0.07	1.43
AC-14/SC-6	0.0	−0.18	0.04	0.19	1.34	−0.09	0.05	1.37
AC-15/SC-7	0.20	−0.12	0.10	0.25	[e]	[e]	[e]	[e]

[a]All values given in degrees per second.
[b]Calculated from potentiometer data (AC-11/SC-4 determined from spacecraft gyro data).
[c]For spacecraft coordinate system; sign is opposite for *Centaur* coordinate system.
[d]Spacecraft roll rate after separation assumed equal to *Centaur* prior to separation.
[e]Data noisy.

Table VIII-24. Relative spacecraft/*Centaur* separation/orbital data

Parameter	AC-10/SC-1	AC-7/SC-2	AC-12/SC-3	AC-11/SC-4	AC-13/SC-5	AC-14/SC-6	AC-15/SC-7
Distance 5 hr after separation, km	1,054	730	1,436	1,167	1,751	1,640	1,423
Closest approach of *Centaur* tank to moon, km	17,291	5,675	34,492	22,469	40,311	28,072	19,743
Orbital period of *Centaur* tank, days	12.360	10.050	8.960	10.470	8.730	10.800	12.630
Apogee altitude, nmi	236,940	321,822	189,854	215,434	186,484	215,957	240,379
Perigee altitude, nmi	91.200	94.370	92.950	91.380	90.660	91.290	90.240

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IX. Spacecraft Aerospace Ground Equipment

The *Surveyor* spacecraft aerospace ground equipment (AGE) included that portion of the ground support equipment that interfaced with, and was associated with, the spacecraft handling, checkout, and launch operations.

For the purposes of equipment development, the spacecraft AGE was broadly categorized as follows:

(1) Operational ground equipment (OGE).

(2) Ground handling equipment (GHE).

(3) Operational support equipment (OSE).

(4) Spacecraft auxiliary equipment.

(5) Shipping and storage containers.

The first category, operational ground equipment, included the system test equipment assemblies (STEA) and other portions of AGE devoted to the electronic testing of the spacecraft.

There were six STEAs constructed during the *Surveyor* Project. Originally, five complete STEAs and a portion of a sixth were fabricated. Later, the sixth STEA was completed as described later in this section. The third was extensively modified (also described in this section).

All five of the categorized items were included in AGE sets 1 through 6 as described in Section IX-C. Each STEA, plus the required elements of the other four categories, represented a complete AGE set. The existence of six AGE sets was the direct result of the existence of six constituent STEAs.

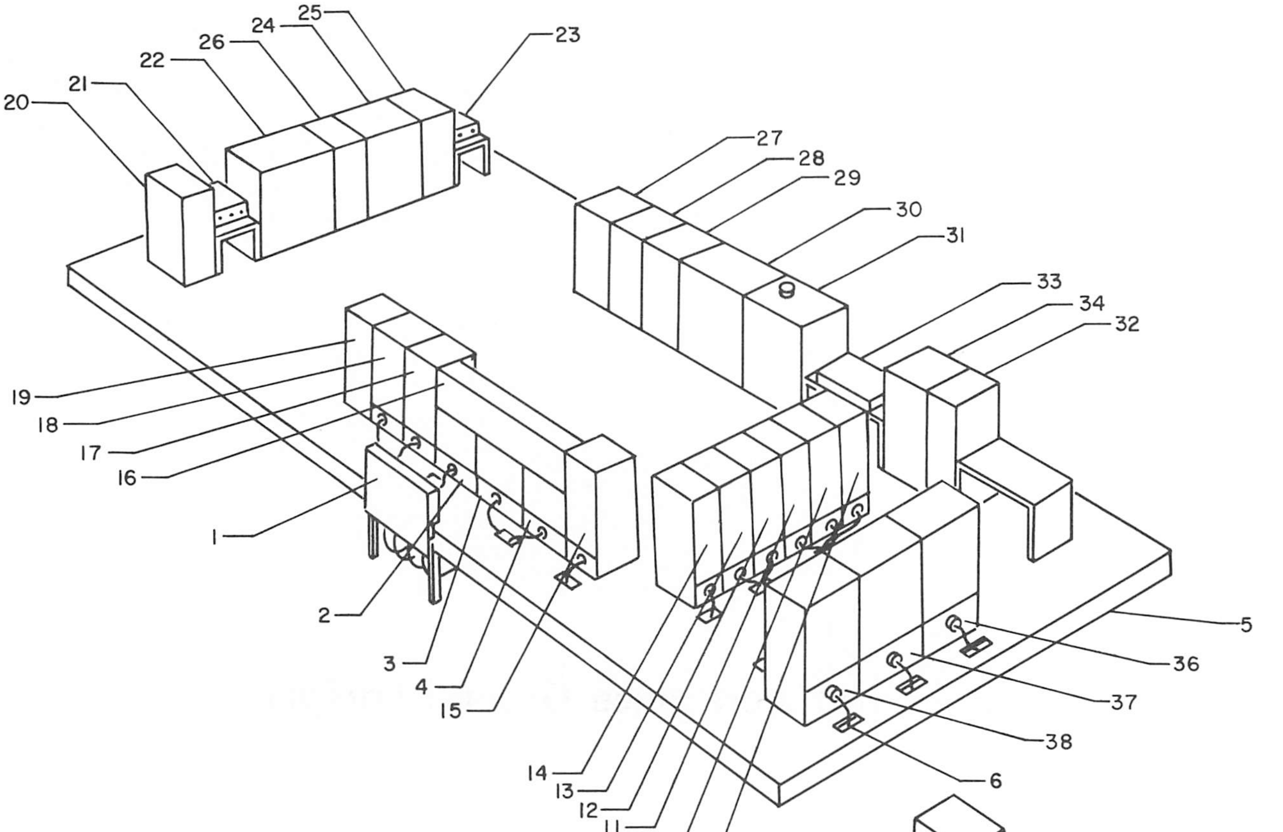
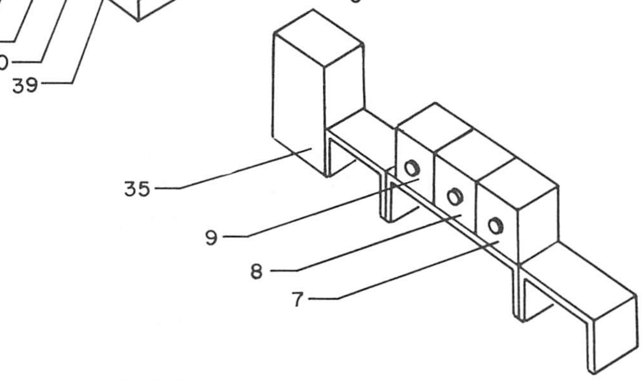
A. Aerospace Ground Equipment Design Objectives and Development Plan

Overall design objectives for the development of adequate spacecraft AGE were established early in the *Surveyor* Project and included:

(1) Provision for the means to test the design integrity and to provide a verification of spacecraft flight readiness via the complete functional checkout of the spacecraft system.

(2) Capability of the AGE and associated launch procedures to accommodate built-in or automatic holds during the countdown sequence

(3) Safeguards were to be provided, as necessary, for personnel who were to work near the spacecraft and the spacecraft operational support equipment.

1 POWER DISTRIBUTION AND GROUNDING ASSEMBLY
2 CDC TELEVISION CONSOLE, BAY 3
3 CDC COMMAND CONSOLE, BAY 2
4 CDC TELEMETRY CONSOLE, BAY 1
5 FALSE FLOOR ASSEMBLY
6 STEA INTERCABLING
7 VEHICLE AND MECHANISMS CHARACTER PRINTER *
8 THERMAL CHARACTER PRINTER *
9 SPECIAL MESSAGE CHARACTER PRINTER *
10 CDC COMMAND SYSTEM, BAY 9
11 CDC BUFFER AND PATCH, BAY 7
12 CDC DECOMMUTATOR RACK, BAY 6
13 CDC DISCRIMINATOR AND HIGH FREQUENCY OSCILLOGRAPH, BAY 5
14 CDC LOW FREQUENCY OSCILLOGRAPH DEMODULATOR, BAY 4
15 PATE RACK
16 CDC ELEVATED CONSOLE, BAY 12
17 TELEVISION TEST RACK
18 CDC TELEVISION PHOTOGRAPHIC RECORDER, BAY 10
19 STV CONTROL RACK
20 TAPE RECORDER REPRODUCER (AMPEX FR 1200)
21 RF CHARACTER PRINTER *
22 RF TEST RACK
23 FLIGHT CONTROL CHARACTER PRINTER *
24 CLOSED-LOOP SIMULATION RACK
25 FLIGHT CONTROL TEST RACK
26 PORTABLE BRUSH RECORDER
27 RADVS RECORDER RACK, UNIT 3
28 RADVS RANGE DOPPLER SIMULATOR RACK, UNIT 1
29 RADVS ANALYZER RACK, UNIT 2
30 ALTITUDE MARKING RADAR TEST RACK
31 POWER SUPPLY TEST RACK
32 PRELAUNCH POWER SUPPLY TEST RACK
33 POWER CHARACTER PRINTER *
34 PORTABLE BRUSH RECORDER
35 LINE PRINTER *
36 DATA INPUT SYSTEM, No. 3 *
37 DATA INPUT SYSTEM, No. 2 *
38 DATA INPUT SYSTEM, No. 1 *
39 SCIENTIFIC INSTRUMENT TEST RACK

* JPL-FURNISHED EQUIPMENT (NOT STEA)

Fig. IX-1. Typical STEA layout

(4) The service-life requirement of the spacecraft AGE was to be 5 yr, including storage and operation.

The spacecraft AGE design provided a safe and convenient means of handling and transporting the spacecraft and provided the capability to perform a complete system level electrical and mechanical spacecraft checkout prior to launch.

B. Aerospace Ground Equipment Design, Development and Operational Experience

1. Operational Ground Equipment

Operational ground equipment included the STEAs, launch complex equipment, and safety console. The equipment described on the following pages is presented by AGE element.

a. System Test Equipment Assemblies. The STEA, with appropriate auxiliaries, contained the electronic equipment for all system testing configured to allow major subsystem testing in a single proximity. Equipment was included for testing the spacecraft power, receiver/transmitter, flight control, altitude marking radar, radar altimeter and doppler velocity sensor, television, command and data handling, scientific instruments, and propulsion (Fig. IX-1).

There were six STEAs; all were functionally identical except that the equipment peculiar to certain tests, such as solar-thermal-vacuum, electromagnetic interference, and vibration, was provided only at the particular testing site.

Interconnections between the STEAs and the spacecraft was by cables and junction boxes and by RF link. The cables connecting the STEA racks to the junction boxes (lower cables) were essentially permanent and remained fixed during all phases of testing. The cables connecting the junction boxes to the spacecraft (upper cables) were selected to provide the proper hardline access for each particular test phase, permitting a logical progression from maximum access to a configuration of minimum access throughout the system test program. Test adapters were used to interface the upper cables and the spacecraft to minimize the number of connections with flight hardware.

Late in 1966, an AGE upgrade program under change orders 126 and 127 was undertaken to improve equipment operability, reliability, and maintainability. This increased the reliability of the intercabling and junction boxes and incorporated other improvements in the flight control and closed-loop test racks, power rack, radar altimeter and doppler velocity sensor (RADVS) range and velocity panel, primary sun simulator, and the intercom system in the manufacturing and test facilities at HAC, El Segundo, Calif. The new cable design included the use of potting boots and controlled cable lay patterns for increased mechanical reliability. The electrical design for the new cables and new junction boxes included strict adherence to correct grounding, power, and signal return practices. The new junction box design permitted greater access for troubleshooting and maintenance. AGE set 4 was the first to receive the upgrade modification, having been changed just before SC-4 testing. AGE set 1 was modified before SC-5 solar-thermal-vacuum test and AGE set 6 was advanced to the upgraded configuration before the arrival of SC-5 at AFETR.

The individual equipment units utilized within the STEAs are briefly described below with reference to location in typical STEA layout, as called out in Fig. IX-1.

RF test rack (item 22 in Fig. IX-1) was used to test the spacecraft receiver and transmitter. The test rack contained a UHF transmitter panel and a UHF receiver panel to test the operating parameters of the spacecraft receivers. Simulated spacecraft RF commands were generated by the subsystem decoder output simulator panel. The running times and on/off cycles generated by these commands were monitored by the timer panel.

Portable RF test set was used in conjunction with the RF test rack to test the performance of the spacecraft RF data link subsystem. It was a miscellaneous collection of RF test equipment used for in-depth evaluation.

Power supply test rack (item 31), except for the rack at STEA 5 which was not upgraded by change orders 126 and 127, functioned as follows: simulated spacecraft +22-V unregulated power was generated by the battery simulator panel and was applied through a control relay in the junction box to the spacecraft 22-V bus via the umbilical connector.

Auxiliary power supply panel generated simulated spacecraft +29-V power for +29-V essential, +29-V nonessential and +29-V flight control buses. This power was supplied through the boost regulator harness connector and would have been used only if the boost regulator were not available in the spacecraft. It was never used for the intended purpose.

Solar simulator control panel simulated spacecraft solar panel impedance characteristics. A constant current signal was generated by the modified solar simulator panel and applied to the solar simulator control panel. The simulated solar panel output was applied through the junction box to the spacecraft umbilical connector or directly to the connector on the mast harness. Spacecraft signals were monitored by the integrating digital voltmeter panel. Certain signals were monitored by meters in the solar simulator control panel, and by an 8-channel strip chart recorder located in the rack.

Prelaunch power supply test rack (item 32) furnished gyro preheat power during system test. It contained a +28-V power supply used to actuate the spacecraft main power switch and retro safe and arm switch. The *Centaur* Program simulator panel provided pulses to the extend landing gear, unlock omniantenna, and high-power transmitter on switches. Signals were also applied to spacecraft engineering mechanism auxiliaries. The engineering mechanism auxiliaries converted these signals to power commands and applied them to spacecraft pinpullers.

Power control panel contained a spacecraft main power switch, which provided on/off control of the spacecraft +22-V bus switch. The spacecraft main power switch control unit contained a relay switching circuit, which turned the spacecraft main power switch off during a STEA power failure.

Prelaunch monitor rack (STEAs 2–5) functioned similarly to the prelaunch power supply test rack.

Altitude marking radar (AMR) test rack (item 30) was used to test the subsystem on the spacecraft. This rack contained four fixed DC power supplies; +22 V, +40 V, −18 V, and −27 V, and one variable voltage supply. This power was used for cooling and power switching, AMR heater, AMR junction box isolation amplifiers, voltage for some rack circuits, and signal for the AMR electronic conversion unit (ECU). The AMR test rack generates three command signals. They were applied through the AMR junction box to the spacecraft; AMR power on, AMR power off, and the AMR enable signal to the video processor. An indication of AMR enabled condition was provided in the display switching panel. Either a manual range sweep or an automatic range sweep was generated from the range and frequency panel. The automatic range sweep continued until reset by panel controls, but the slope had to be adjusted manually during sweep time. The generation of the delay ramp to simulate spacecraft closing velocity was controlled by the arrival of a trigger pulse from the spacecraft AMR synchronizer or RF energy from the spacecraft AMR antenna, which had been detected and amplified in the RF panel. When the coincidence detector had established that there was the desired time relationship between the range sweep and the delay ramp, the output was applied to a panel pulse shaping network. This pulse shaping network generated the modulation pulse and a delay ramp generator reset pulse. During coincidence, the spacecraft AMR video processor generated the AMR mark signal. The spacecraft AMR video processor mark circuit was reset either automatically or manually in the range and frequency panel. Automatic reset was accomplished when the AMR mark signal energized a panel relay. The threshold level of the spacecraft AMR processor was adjusted by a switch and control in the display switching panel.

Flight control test rack (item 25) was used with the closed-loop rack to test the spacecraft flight control sensor group (FCSG). After the redesign by change orders 126 and 127, the following description applied: The flight control test rack contained two power supplies, +24 VDC and +105 VDC; an optical simulator control panel; analog signal display panel; gas-jet display, switching and squib monitor panel; timer–counter panel; digital-voltmeter input selector panel; and digital voltmeter. The output of the 24-VDC power supply was converted to constant current at about 300 mA by circuitry in the optical simulator control panel and supplied power to the primary sun simulator. The analog signal display panel contained meters for simultaneous monitoring of hardline telemetry signals from the FCSG. Each hardline telemetry signal was also routed to the digital voltmeter via the digital-voltmeter input selector panel for more accurate measurements. The gas-jet display, switching and squib monitor panel performed the following functions: (1) Logic switch access switches caged the spacecraft gyros and vernier engine integrators, and simulated the three spacecraft legs-down and separation-of-spacecraft-from-*Centaur* signals; (2) phase detector input switches shorted the error-signal inputs into the spacecraft gas-jet amplifiers; (3) six lamps displayed which gas jets were in operation; and (4) the panel contained three vernier-engine solenoid valves and indicator lamps for loading the solenoid valve circuitry in the FCSG. The six gas-jet signals were also routed to the timer–counter panel where the number of on/off cycles for each gas jet were totaled. In addition to the six gas jet counters, the timer–counter panel contained two running time meters and their associated events counters. Their function was to keep a record of the operating time and on/off cycles of the spacecraft FCSG for coast phase power-on and thrust phase power-on modes of operation.

The digital voltmeter input selector panel contained twenty switches and provided a convenient means of applying signals to the digital voltmeter.

Flight control auxiliary equipment was an equipment group used with the flight control test rack to functionally test the spacecraft celestial sensors and nitrogen system. It was composesd of hoods, light sources, etc., used directly with the flight sensors.

Closed-loop simulation rack (item 24) was used to test the radar altimeter and doppler velocity sensor (RADVS) and the flight control subsystems on the spacecraft in a simulated dynamic real-time mission. After the redesign by change orders 126 and 127, the following description applied: The closed-loop simulation rack was composed of 12 major subassemblies:

(1) Simulator unit.

(2) Simulator control panel.

(3) Interface electronics panel.

(4) RADVS coupling unit.

(5) Multiplex coupling panel.

(6) Computer data system (CDS) signal conditioning panel.

(7) Digital voltmeter.

(8) Digital voltmeter switching panel.

(9) *X/Y/Y* plotter.

(10) Module checker panel.

(11) Function generator.

(12) Frequency counter.

The simulator unit was an analog computer that generated the kinematics of the test by solving the differential equations involved in the descent to the moon. The simulator control panel allowed the test operator to control the integrator initial conditions and to monitor various computed terms and power supply voltages associated with the simulator. The interface electronics panel provided the buffering between the simulator and the spacecraft; simulated the vernier engine loading, time constant and limiting; simulated the main retro burn by generating the mass and inertia as a function of time; generated the angular acceleration terms and total Z axis thrust; and controlled the events of the simulator scale factors to the correct FCSG input scale factors. All signals that passed from the FCSG to the simulator were routed via this panel. The RADVS coupling unit was used to convert the computed velocities and range into proper frequencies needed to modulate the RF at the RADVS STEA test equipment. This unit simulated the RADVS beam geometry so that the frequency shift on each beam would have the correct proportions. The RADVS detected, tracked, and arithmetically added or subtracted the simulated doppler shifts to provide the original computed terms. The multiplex coupling panel provided test access to various parameters in the closed-loop rack primarily for troubleshooting purposes. The CDS signal conditioning panel accepted all parameters of interest in the closed-loop rack and processed them for transmission to the CDS data recording equipment. The digital voltmeter switching panel provided a convenient means of selecting the particular panel from which parameters were to be monitored.

The digital voltmeter was used for precision voltage measurements of all rack parameters. The *X/Y/Y* plotter was used during the terminal descent phases of closed-loop testing only. It presented a real-time indication of spacecraft descent profile. Computed total thrust was also plotted along the ordinate to give an indication of spacecraft acceleration. The module checker panel proved a convenient means for troubleshooting the various circuit modules of the closed-loop rack. The panel accepted any of the three standard sized modules and provided access to all module pins. The function generator was a standard commercial low frequency function generator and was used with the stimulus amplifier in the interface panel for performing phase margin tests on the spacecraft FCSG. The frequency counter was a standard commercial electronic frequency counter with the additional capability of period and time interval measurement.

Closed-loop rack received the three vernier engine thrust commands from the flight control electronics and converted these signals to three DC voltages which were an analog of thrust for each of the vernier engines. In the conversion the engine transfer function and limiting characteristics were simulated. These thrust terms were then passed through circuitry which summed them in the correct proportion based on engine location with respect to the spacecraft axes to produce the torques about the *X*- and *Y*-axes. The three terms were also simply summed to develop an analog of total engine thrust.

These computed torques were then passed through the mass and inertia simulation portion of the interface panel where they were divided by a term representing spacecraft inertia resulting in an analog of angular acceleration about each of the spacecraft axes. The total thrust term was divided by a term representing spacecraft mass; this resulted in a signal analogous to spacecraft linear acceleration along its Z-axis in this same portion of the equipment. This linear acceleration term was then appropriately scaled and inserted back into the flight control electronics to simulate the output of the midcourse accelerometer. The three angular acceleration terms were integrated to develop the angular velocity terms and inserted into the spacecraft gyros to simulate rotation of the spacecraft. The inertial attitude control loops were thereby closed.

The mass and inertia simulation portion of the interface panel consisted of several motor-driven potentiometers plus electronics and simulated a decreasing spacecraft mass and inertia as a function of time during simulated retro burning.

Close-loop auxiliary equipment (items 27–29) was an equipment group used with the closed-loop simulation rack to functionally test the spacecraft flight control and RADVS circuits.

Radar altimeter and doppler velocity sensor (RADVS) racks were used to test the spacecraft RADVS subsystem. The spacecraft RADVS subsystem transmitted four beams of RF energy. During the lunar encounter the differences between the transmitted and the reflected signals were used by the spacecraft to compute altitude, vertical velocity, and lateral velocity. The RADVS racks simulated the lunar surface and evaluated the spacecraft RADVS performance. Four test equipment antenna feed couplers were used to feed the test racks from the spacecraft. The energy was introduced into four electrically identical circuits in the microwave unit panel where each of the four channels of microwave energy was split and processed and then coupled back to the spacecraft antennas. The racks were able to simulate all normal approach positions and rates of approach to the lunar surface that were required to test the RADVS subsystem.

RADVS auxiliary equipment was an equipment group used with the RADVS racks to ensure the specified alignment of the spacecraft RADVS units.

System test stand assembly was used to support the spacecraft and maneuver it in attitude. The stand supported the spacecraft during system test. It was a modified military gun mount which provided azimuth positions through 360 deg and elevation through 180 deg.

System test stand control rack (item 25) provided power for the system test stand azimuth and elevation drive motors. The indications of the positions were provided in the system test stand position indicator panel.

Optical alignment equipment was an equipment group used to optically align the system test stand.

Scientific instrument test rack (item 39) processed data from the flight scientific experiments for monitoring and printout.

Television test rack (items 17 and 18) was used to test the television subsystem on the spacecraft. Coded television group commands, with the appropriate earth-to-spacecraft transit delay, were applied through the spacecraft central command decoders to the television camera. Video signals from the spacecraft were monitored on the signal simulator and monitor panel.

Television auxiliary equipment was an equipment group used with the television test rack to test and calibrate the spacecraft television system.

Junction boxes were located on the system test stand assembly and in the Sector II J-box. They contained relays for signal selection and buffer amplifiers which provided gain and impedance matches and shunts which facilitated current monitoring.

RFI simulation racks provided the simulated electromagnetic environment for the mission sequence/electromagnetic interference (MS/EMI) test phase. The nearby RF radiating sources, the tracking radars at AFETR, and the RF interference generated by the spacecraft onto signal and power lines were simulated.

Portable RFI test set was an equipment group used with the RFI simulation racks, signal generators, and noise meters to simulate and monitor electromagnetic fields.

General-purpose STEA auxiliary equipment was an equipment group used to simulate, monitor, and load STEA and spacecraft signals during acceptance, calibration, and troubleshooting. It was composed of oscilloscopes, vacuum-tube voltmeters, volt–ohm meters, cameras, panoramic adapters, etc.

Breakout boxes were used to provide access to STEA and spacecraft signals for troubleshooting and spacecraft harness acceptance checks.

Squib circuit test set consisted of the squib circuit test adapter, the squib simulator, the stray energy monitor, and a number of squib adapter cables. Squib simulators were used rather than live squibs for safety, economy, and convenience. The purpose of the stray energy monitor was to give assurance that the level of stray electromagnetic energy was not sufficient to fire a squib.

Missile operations paging system intercom rack and auxiliary equipment provided communication between test team personnel while they were performing the system tests. It provided 19 channels of communications.

Power distribution and grounding assembly cabinet (item 1) controlled the 60-Hz power supplied to the STEA assemblies. The input to the power distribution cabinet was 3-phase, 60-Hz power. This power was controlled by phase-balanced branch circuit breakers and applied to each of the power modules. Each module then applied the power to assigned STEA assemblies. The uninterrupted neutral from the power distribution cabinet 3-phase power was connected to a ground bus to chemical ground. All STEA racks featured series-connected emergency off switches except the prelaunch racks, the RF test rack, and the closed-loop simulation rack, which were series-connected in a separate emergency off circuit.

Command and data (handling) console (CDC) was designed to create a complement of ground support equipment which interfaced with the Deep Space Network (DSN) mission-independent equipment to process and display all spacecraft telemetry and video to the extent required to analyze spacecraft performance and generate the commands necessary for control of the spacecraft as indicated by the telemetry data (see Section VII-A-3). CDCs basically identical to those installed at the DSN facilities were required by the *Surveyor* test program because the major portion of testing had to be accomplished via RF link in much the same manner that was utilized during the mission.

The CDC was divided into four basic subsystems consisting of command, telemetry, television, and system test.

(1) Command messages were generated by the command generator from either a manual keyboard, punched tape, emergency command button, or ten individual toggle switches. There was an auxiliary encoder with a separate clock internal to the unit which could be utilized should a failure occur at a critical time. The command generator console displayed GMT, DSIF receiver AGE voltage, receiver lock, and interfaced with the telemetry subsystem for real-time display of certain telemetry parameters. A timer on the front indicated elapsed time since the last transmitted command, and a counter indicated the total number of commands transmitted. Unit spares were provided on site for all portions of the command subsystem to ensure command transmission capability in the event of an on-line failure.

(2) The telemetry subsystem demodulated, discriminated, decommutated, displayed, and recorded data. Two demodulators were rack mounted with one being an on-line spare. The output of the demodulator was routed to a bank of ten phase lock FM subcarrier discriminators which were connected in parallel to the incoming signal. The display capability of the telemetry subsystem consisted of the telemetry display console, low-frequency oscillograph, and the various displays on the command generator console, as previously discussed.

(3) The television subsystem consisted of a television video processor, monitor, and photo recorder. Video data entered the CDC as a frequency-modulated signal. This was demodulated and went to the video processor, which in turn drove the monitor for real-time display and analysis. A Polaroid camera, mounted on a special hood converting the monitor, was used to obtain a permanent record of the picture for real-time analysis. In addition to the real-time display capability, the photo recorder assembly provided a permanent record of each television picture on 35-mm film. The video processor, which drove the television monitor, also drove a low persistence cathode-ray tube (CRT) which was mounted in the photo recorder assembly. The face of the CRT was photographed by a camera mounted directly below it. The camera was triggered each time a picture was processed by the video processor and automatically stepped to the next frame. In addition to the video display and recording capability, the telemetry associated with each in coming frame of data was processed by the telemetry decommutator and displayed on a matrix of lights mounted in the photo recorder assembly. This matrix was photographed along with the picture displayed on the CRT, thus ensuring correlation between each picture and its associated telemetry. In addition, a clock was photographed with each picture.

(4) The data processing subsystem, which provided the interfaces between the CDC and computers,

consisted of a command buffer, telemetry converter, reader punch/typewriter switching unit, input/output selection unit, two keyboard printers, tape punch, reader coupler teletype punch coupler, and computer interface assembly. The functions of these units could have been grouped into the following categories; telemetry interface, command interface, command tape preparation, and computer control.

The notable differences were: An on-line backup was not provided in any STEA CDC; the on-site data processing capability was not provided in any STEA CDC; the major portion of the CDC system test equipment was not provided in any STEA CDC; and the *Surveyor* operations center (SOC) console was not provided in any STEA CDC. However, all STEA CDCs were provided with a programmed automatic telemetry evaluation capability and with a capability to send data to the computer data system (CDS).

b. Launch complex equipment. The launch complex equipment was used to monitor, control, and simulate spacecraft power functions during the prelaunch checks when the spacecraft was mated to the boost vehicle during combined system test and before launch. The launch operations console (LOC) and launch control rack (LCR) provided monitor and control capabilities and simulated spacecraft power functions during prelaunch checkouts. The prelaunch power supply contained in the LCR provided two modes of operation: constant current and constant voltage. The prelaunch power supply in the launch complex was controlled remotely from the LOC.

The solar panel simulator in the LCR contained a prelaunch power supply operated in the constant current mode. The solar panel simulator had the capability to simulate the spacecraft solar panel output voltage and current characteristics entirely or partially. All control and indicators for the solar panel simulator were provided in the power panel in the LOC.

The battery simulator in the LCR consisted of two prelaunch power supplies operated in constant voltage mode, with the master supply controlling a slave supply to produce the required power. The battery simulator simulated the spacecraft battery output partially or entirely. All major switching controls, indicators, and monitoring functions were provided in the LOC as well as in the LCR.

The television light function of the launch complex consisted of a resistive dimmer unit which controlled the illumination of the target lights. Operation of the television target lights was primarily from the LOC. Relay circuits and the dimming network were located in the LCR. Most major circuit controls were also provided in the LCR.

c. Safety console. The safety console was used to power the spacecraft, monitor dangerous flight components, and control safe operating conditions in the explosive safe facility (ESF), and during spacecraft transit from the ESF to the launch pad.

A prelaunch power supply provided constant current or constant voltage modes of operation. A gyro preheat power supply provided +28 V for the gyro heaters. Overvoltage protection was provided external to the power supply.

The dump pulse generator was intended to provide high-current pulses to the spacecraft squibs for dumping high pressure helium and nitrogen gases in the event of an abort.

The toxic gas vapor detector was used to detect, indicate, and measure vapor levels of propellants that might have leaked from the spacecraft propulsion system. This function was continuous during spacecraft loading and encapsulation operations in the ESF and subsequent transit to the launch pad. Leaks were indicated in parts of vapor per million parts of air (ppm) for both propellants. A buzzer and warning light signaled vapor leaks exceeding pre-established points. A similar unit was used on the launch pad.

The temperature and humidity of the air delivered to the encapsulated spacecraft were displayed by the temperature and humidity unit in the safety console. Sensors were located in the *Centaur* nose fairing air-conditioning ducts (Section VIII-C-3).

2. Ground Handling Equipment

Surveyor GHE comprised equipment necessary to support spacecraft assembly, handling, test, and transportation. GHE design included provisions for protecting the spacecraft against damage from handling and transport, and from corrosive materials, atmospheric elements, and other contaminants during test operations.

a. Surveyor ground transport vehicle GTV-1. The GTV-1 provided transport capability for the spacecraft in test areas and was used as an environmentally controlled shipping container between test locations with its

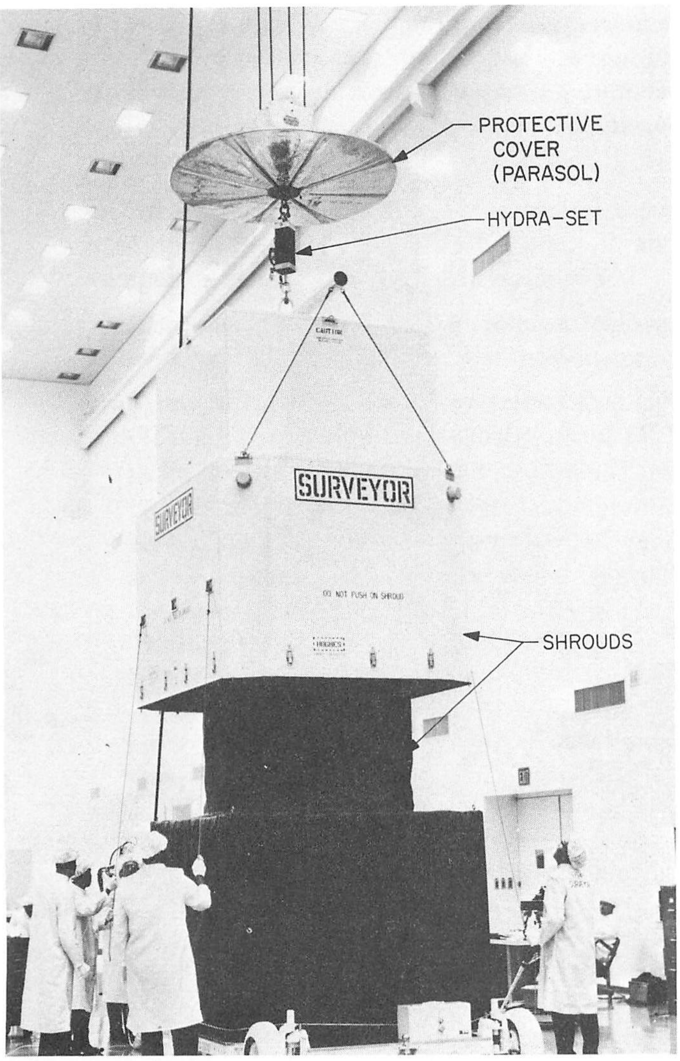


Fig. IX-2. Ground transport vehicle GTV-1

covers installed. Inside controlled areas, wheel dust covers were used to help preserve the controlled-environment area cleanliness. Each GTV-1 inner shroud support frame, outer shroud, and spacecraft handling stand was match-drilled to its trailer and serially numbered accordingly; therefore, inner shroud support frames, outer shrouds, and spacecraft handling stands were not interchangeable among trailers. There were three GTV-1s used (Fig. IX-2).

b. Surveyor ground transport vehicle GTV-2. The GTV-2 served as a mating fixture for HAC and GD/C interface equipment when encapsulating the spacecraft in the *Centaur* nose fairing and provided transport for the encapsulated spacecraft. It was used also at combined system tests (CSTs) to support the spacecraft during test operations. There were two GTV-2s used (see Fig. IX-3).

c. Surveyor handling stand, GTV-1. The handling stands were used as an attaching interface and holding fixture for the spacecraft (less the retro motor) when mounted on the GTV-1.

d. Inner and outer shrouds, GTV-1. The outer shroud was constructed of fiberglass and was designed to protect the spacecraft from dust and rain during transport between clean rooms, between buildings, and between sites. Four large pressure relief valves prevented excess differential pressure buildup between outside and inside, while the shroud was installed on GTV-1. The inner shroud consisted of a tubular frame support and a conductive plastic cover (bag). The inner shroud bag served as a dust cover.

e. Slings. Surveyor slings were used for hoisting, positioning, and holding of spacecraft and spacecraft major components during assembly operations. They were used similarly for other GHE items in spacecraft handling operations. Slings consisted of assemblies of twisted wire cables, load distribution devices, and load attachment hardware. Cables were enclosed in a polymeric sheathing material heat shrunk to the cable. All interconnection and attachment hardware were protected with corrosion-resistant coatings. All slings were designed and built to withstand operational use. Subsequent proof-loading was performed when misuse or requirements indicated. Slings were used in conjunction with:

(1) *Protective cover (parasol).* The protective cover was a parasol-shaped shield. It was installed between the hoist hook and the hydra-set. It served to protect hoisted items from droppings of grease or other contaminants from overhead. The upper and lower shield surfaces were covered with metal foil that provided an electrostatic grounding capability.

(2) *Hydra-set and ballast weight.* The hydra-set, a hydraulically operated device, was installed between the protective cover (parasol) and the ballast weight, to provide precision linear motion for accurate load positioning. Two models of hydra-sets were used. The model O hydra-set was rated for loads of 0–1000 lb; the model C hydra-set was rated for loads of 0–10,000 lb. However, operation of the hydra-set in the lower range of its capacity was very awkward. Because of several incidents of erratic operation, the model C was used only for loads greater than 1000 lb. The ballast weight of 291,627 lb provided additional weight for loads below 1000 lb.

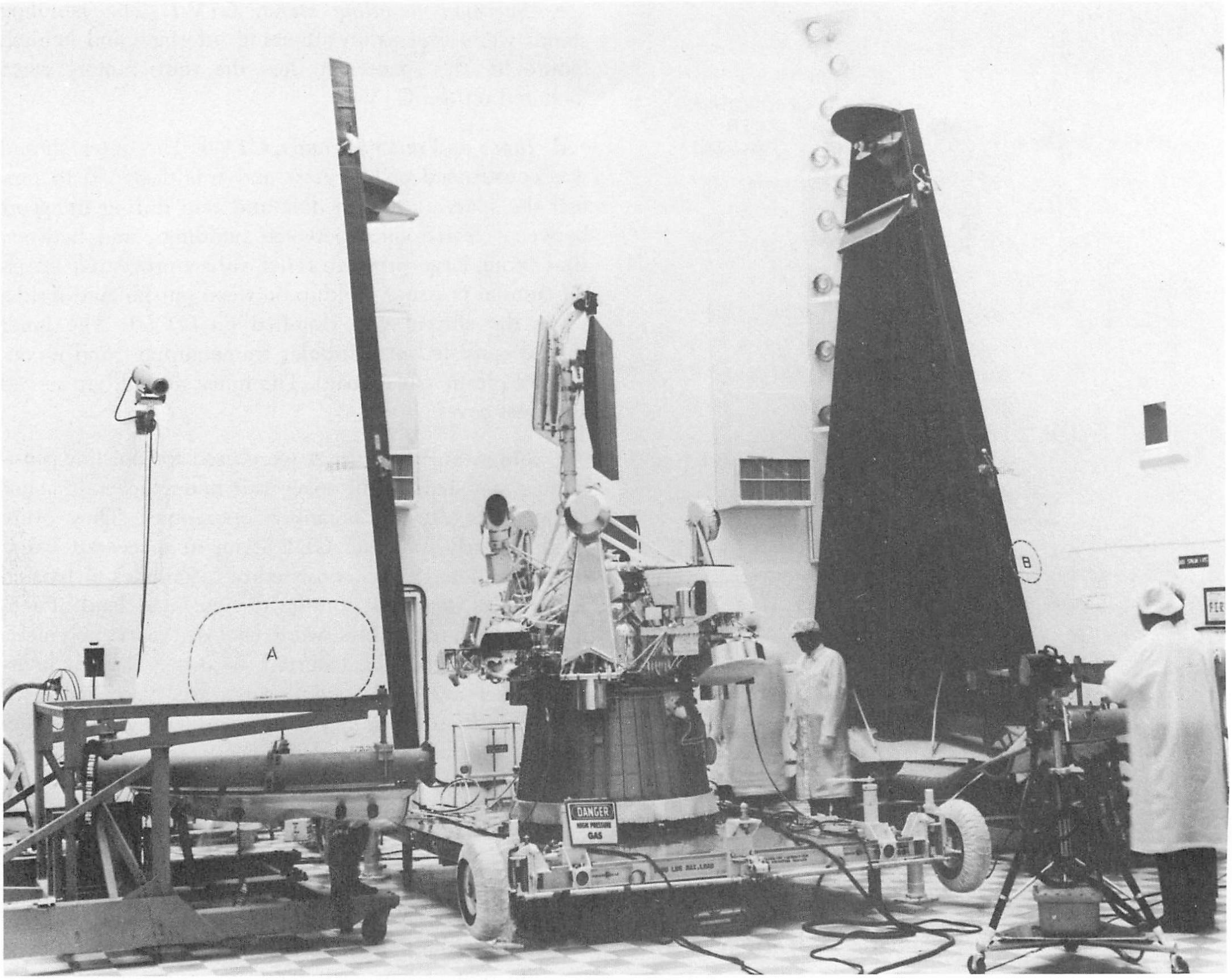


Fig. IX-3. Ground transport vehicle GTV-2

(3) *Handling stands. Surveyor* handling stands provided firm mounting support for spacecraft and spacecraft major components during assembly.

(a) *Spacecraft mobile stand.* This stand was used for holding the spacecraft during installation and removal of major components and protective covers.

(b) *Positioner workstand.* This workstand for the antenna solar panel positioner was used to support the positioner during installation of the antenna and/or solar panel on the positioner.

(c) *Spacecraft assembly stand.* This stand was used to support the spacecraft for removal, redrilling, and reinstallation of the bathtub fittings.

(d) *Spacecraft and retro motor mating stand.* This stand was used to support the retro motor while mating the spacecraft to the retro motor; to support the spacecraft less the retro motor; to provide mobility for the retro motor, the spacecraft, or the mated spacecraft and retro motor, within clean rooms; to support the retro motor while installing the retro motor wrap; and to support the mated motor and spacecraft while installing the AMR.

(4) *Work platforms. Surveyor* work platforms provided test personnel access to hard-to-reach areas of the spacecraft, with provisions for supporting personnel so that they did not damage the spacecraft, especially the finishes on spacecraft thermal surfaces.

The STEA high access stand provided personnel access to the spacecraft upper portions during installation of the antenna solar panel positioner and/or pyrotechnics and during optical alignment procedures (Fig. IX-4).

(5) *Special handling equipment.* These special items of equipment that were used to hold, install, and manipulate various spacecraft components are:

(a) *Retro motor rollover tool.* The rollover tool was used for 180-deg rollover of the retro motor.

(b) *Spacecraft adapter and inner shroud standoff, GTV-1.* These items were used on GTV-1 when transporting the mated spacecraft and inert retro motor for thrust alignment vibration tests. The shroud standoff consisted of pipe sections installed between the trailer bed and the inner frame so that the frame would clear the increased height of the mated spacecraft and adapter.

3. Operational Support Equipment

a. Vernier propulsion system oxidizer servicing cart. The vernier propulsion system (VPS) oxidizer servicing cart was used to store the oxidizer propellant and to evacuate, load and offload the oxidizer tanks of the *Surveyor* vernier propulsion system. The cart operated in a closed fluid-vapor loop with the *Surveyor* spacecraft and contained an integral vapor treatment system for processing toxic propellant vapors to a safe level. The cart was used with nitrogen tetroxide (90% N_2O_4–10% NO) propellant.

The enclosure and trailer consisted of 120-ft^3 stainless steel enclosure mounted on standard, military-type running gear. The trailer utilized two-wheel running gear equipped with pneumatic tires.

Trailer framing included appropriate tie-down rings and forklift channels. The main wheels were equipped with parking brakes. The floor served as a drip pan and was equipped with a drain plug. All operating controls and instrumentation were mounted on a recessed, tilted panel at the rear of the cabinet. The control panel was protected by two stainless steel doors. Panel-mounted components were arranged in accordance with the cart functional schematic. Line markings were used to indicate the various flow paths. Schematic representation of major internal components also were included.

Hose connections were located in this area to provide proximity to the hose storage area and to remove the fittings from the control area for safety purposes. Explosion-proof lighting was provided for the control panel and the cart interior.

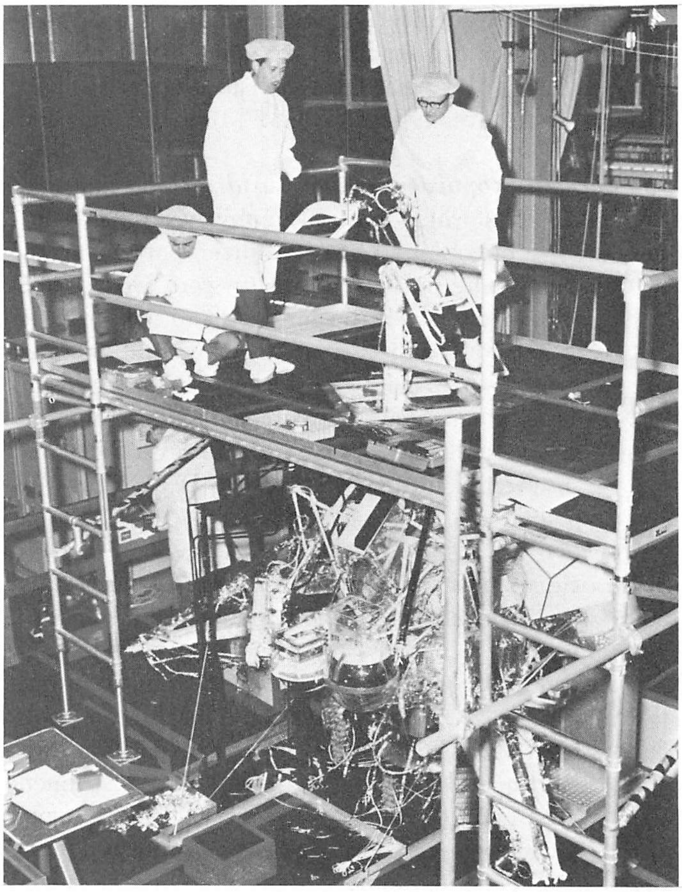


Fig. IX-4. STEA high access stand

The servicing cart was functionally divided into a 0–100-psig gaseous helium circuit, a propellant supply circuit, a propellant bleed and drain circuit, an evacuation circuit, and a vent and vapor disposal system. Helium gas was utilized for pressurizing the gas-side of the spacecraft tank bladders and also for propellant transfer operations within the cart's fluid loops.

A supply tank was provided for storage and supply of 15 gal of propellant to the spacecraft. A second 15-gal tank was provided for spacecraft bleed and offload operations.

The vacuum circuit included a 3-ft^3/min vacuum pump and was used to evacuate the cart drain tank and either or both sides of the tank bladders. The vapor treatment system included a cryogenic nitrogen cold trap for toxic vapor control and protection of the vacuum pump.

Special service facility pneumatic and vapor disposal requirements for the oxidizer servicing cart were: (1) helium gas, grade A, filtered to 10 μm absolute at 800 psig, (2) molecular sieve, type 13K, and (3) liquid nitrogen.

b. Vernier propulsion system oxidizer flushing and purge cart. The flushing and purge unit was for use during servicing operations of the oxidizer portion of the *Surveyor* VPS. The unit was used to store Genesolv D solvent, to provide solvent flushing, and to evacuate the vernier propulsion oxidizer system. The unit operated in a closed fluid loop with the *Surveyor* spacecraft and contained an integral vapor treatment system for processing toxic propellant vapors to a safe level. The unit was also used with nitrogen tetroxide (90% N_2O_4–10% NO) propellant.

The oxidizer flush and purge unit was housed in a 150-ft^3 aluminum enclosure mounted on four standard, 10-in. military-type casters. The enclosure was equipped with an eye bolt for towing, a foot lock, and appropriate tie-down rings, hoist fittings, and forklift channels. The stainless steel floor served as a drip pan and was equipped with a drain plug. Controls and instrumentation were mounted on a tilted and recessed panel on the left side of the unit and were protected by a hinged door. Panel-mounted components were arranged in accordance with the unit functional schematic, engraved on the panel, with line markings indicating the various flow paths.

A stainless steel compartment in the front end provided storage space for five 20-ft hoses used to connect the unit with the spacecraft and the pneumatic control console.

The flush and purge unit was functionally divided into an 0–150-psig gaseous nitrogen circuit, a solvent supply circuit, a solvent purge and flush circuit, an evacuation circuit, and a vent and vapor disposal system. Nitrogen gas was used for solvent transfer operations.

A supply tank was provided for storage and supply of 15 gal of solvent to the spacecraft. A second 15-gal tank was provided for spacecraft solvent flush operations. A 5-gal tank was provided for throttle valve purge operations. This latter tank incorporated a calibrated volume indicator gage and a bottom fitting for draining into the drain tank.

The vacuum circuit includes two 3-ft^3/min vacuum pumps and was used to evacuate either or both sides of the tank bladders. The vapor treatment system includes a cryogenic nitrogen cold trap for toxic vapor control and protection of the vacuum pump.

The special service facility pneumatic and vapor disposal requirements for the oxidizer flushing and purge cart were: (1) nitrogen gas filtered to 10 μm absolute, (2) Genesolv D, (3) molecular sieve, type 13X, and (4) liquid nitrogen.

c. Vernier propulsion system fuel servicing cart. The VPS fuel servicing cart was used to store the fuel propellant and to evacuate, load, and offload the fuel tanks of the *Surveyor* VPS, as described for the oxidizer service cart. This cart also operated in a closed fluid-vapor treatment system. It was designed for use with monomethylhydrazine hydrate (72% MMH–28% H_2O) propellant.

The enclosure and trailer were similar to the oxidizer service cart. The special service facility pneumatic and vapor disposal requirements for the fuel servicing cart were: (1) helium gas, grade A, filtered to 10 μm absolute, (2) Pennsalt 2302C fuel neutralizing solution, and (3) liquid nitrogen.

d. Vernier propulsion system fuel flushing and purge cart. The flushing and purge cart was for use during servicing operations of the fuel portion of the *Surveyor* VPS. The cart was used to store isopropyl alcohol solvent, to provide solvent flushing and monitored system draining, and to evacuate the VPS fuel system. The unit operated in a closed fluid loop with the *Surveyor* spacecraft and contained an integral vapor treatment system for processing toxic propellant vapors to a safe level. The unit was also used with monomethlyhydrazine hydrate (72% MMH–28% H_2O) propellant. The fuel flush and purge unit was housed in a 150-ft^3 aluminum enclosure similar to the oxidizer flush and purge unit.

The special service facility pneumatic and vapor disposal requirements for the fuel flush and purge cart were: (1) nitrogen gas filtered to 10 μm absolute, (2) isopropyl alcohol, (3) Pennsalt 2302C fuel neutralizing solution, and (4) liquid nitrogen.

e. Weight, balance, and alignment equipment. This equipment consisted of weighing and optical components to perform precise optical checks of the alignment of spacecraft components to axes of reference.

A weight and balance fixture in the center of the optical dock employed three electronic load cells for weight determination. It was so designed that the retro motor and spacecraft could be weighed separately (while both were mounted on the fixture) or as an assembled unit. Checkout of the alignment of components to the spacecraft axes was accomplished, by use of precise optical transits and squares that slide on ground ways. Over 35 alignment fixtures were fitted on the spacecraft components. The ways and fixtures were mounted on a vibration-free foundation at both HAC, El Segundo and AFETR locations (Fig. IX-5).

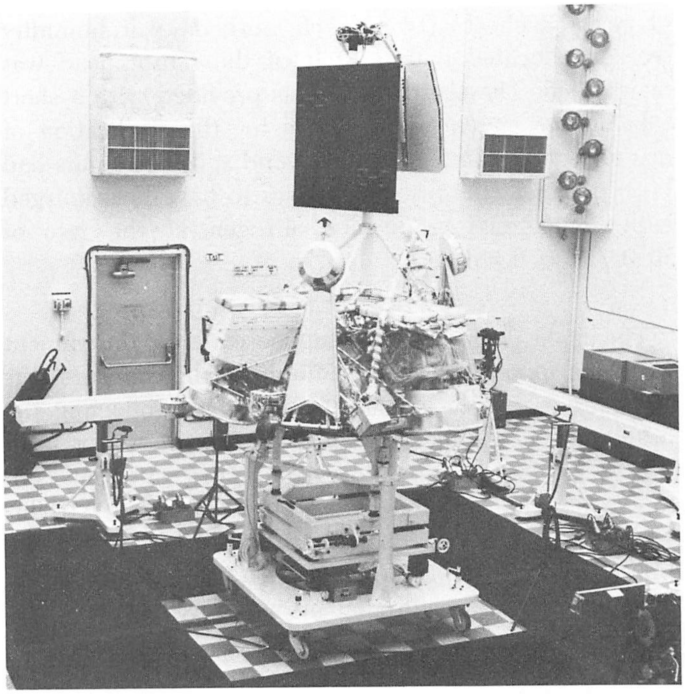


Fig. IX-5. Weight, balance, and alignment dock

f. Mobile temperature control unit. The purpose of the mobile temperature control unit (MTCU) was to supply a predetermined environment to the spacecraft when it was encapsulated in the *Centaur* nose fairing before launch and during transit to the launch pad. An air-cooled, electric motor-driven unit provided filtered air at any selected temperature between 80 and 90°F (within ±1°F) at a relative humidity of 45% or less, whether the unit was parked or being towed at speeds up to 10 mph. The MTCU had noise level, radio frequency interference, and explosion-proofing characteristics consistent with personnel safety and AFETR regulations. It was designed for the extreme environmental conditions in Florida (see Section VIII-C-3 for a description of on-pad air conditioning equipment).

g. Pneumatic control console. The pneumatic control console (PCC) was used to regulate the flow of helium and nitrogen gases to perform functional and leak checks of the VPS, pressurization of the VPS service equipment, functional and leak checks of the gas attitude control system, and pressurization of the spacecraft helium and nitrogen tanks to flight operating pressures.

The PCC contained the necessary valves, regulators, filters and gages to regulate gases from a 6000-psi supply. The PCC was 30-in. wide, 84-in. long, and 54-in. high. Shock-absorbing casters provided maneuverability within the immediate working area. Gages and valve controls were mounted on an inclined surface.

h. Vernier propulsion system loading power supply. The VPS loading power supply unit supplied the electrical power to actuate the vernier engine throttle valve to facilitate loading propellants (or solvents) into the vernier system.

i. Vernier propulsion system servicing equipment. The VPS servicing equipment performed the functions of solvent loading and flushing to meet requirements at El Segundo. This equipment was housed in a sturdy, mobile cart. The design featured easy maintenance and quick removal of equipment to be processed, to meet VPS cleanliness requirements.

j. Support auxiliaries. This category included operational aids as well as items designed to protect the spacecraft and personnel.

1. *Spacecraft protective covers.* Many units on the spacecraft were extremely susceptible to damage in the form of dented or punctured pressure vessels, scratched thermal surfaces, and items knocked out of line. Protective covers, tailored to fit the individual components, were provided to minimize such damage. They were either of Plexiglas or pliable plastic material.

2. *Solar panel assist mechanism.* The solar panel assist mechanism was a mechanism to aid movement of the solar panel in earth-gravity environment.

3. *Alpha scattering instrument purge system.* The alpha scattering instrument nitrogen purge system provided a dry nitrogen environment for the alpha scattering instrument detectors.

The detectors in the sensor head had to be protected from extended exposure to relative humidity of greater than 50%. Before installation of flight sources at AFETR, a continuous monitoring of humidity by means of periodic checks (twice each work day) of humidity recorders located within 50 ft of the sensor head was maintained. The sensor head was provided with a short tube in the viewing port cavity for the application of dry nitrogen. If an exposure beyond specified limits had been detected, the instrument was to have been purged with commercial grade dry nitrogen at the rate of 10 ft^3/hr at 0.1 psig.

The flight source for the alpha scattering instrument was even more sensitive to humidity. The exposure was cumulative and the damage irreversible. Following the installation of flight sources at AFETR, the sensor head was purged with laboratory grade dry nitrogen continuously at a flow rate of approximately 10 ft^3/hr; with a total interruption of less than 3 hr. During the transport to the pad and during *Atlas/Centaur* mating operations, the dry nitrogen purging gas flow rate was reduced to approximately 5 ft^3/hr.

Portable purge equipment (PPE) was used after the spacecraft was encapsulated on GVT-2, during transport to the launch pad, and during hoisting operations and mating to the *Centaur* launch vehicle. The PPE consisted of a single nitrogen bottle and filter package, similar to the on-pad purge equipment (OPPE) below, which was installed on the General Dynamics torus ring fixture.

On-pad purge equipment was used after mating of the encapsulated spacecraft to the *Centaur* until launch. The OPPE consisted of a regulator, a flow meter, a molecular sieve filter and purification chamber, a 0.45 μm millipore filter, and a manifold to connect a multiple bottle gas supply installed in the umbilical towers of pads 36A and 36B. Rigid plumbing was used as much as possible. High pressure flexible hose was used at the flex points of the umbilical boom and at the interface of the boom and launch vehicle.

k. Static charge elimination equipment. Commercial static meters were used for detection and ionized air was employed to neutralize any static charge on spacecraft components, along with conventional grounding methods.

l. Vernier propulsion system shipping and storage pressurization installation. This equipment was used to maintain positive pressure in the VPS during prolonged nonoperational periods when the spacecraft was in storage or shipment.

m. Vibration isolation kit. The vibration isolation kit consisted of a ramp and a system of vibration isolators for use in cargo aircraft to protect the spacecraft from flight loads applied by the aircraft.

The spacecraft ground transport vehicle was rolled up onto the ramp, then jacked up off its wheels so it was supported through vibration isolators.

n. Vernier engine thrust chamber assembly vacuum and purge equipment. The thrust chamber assembly (TCA) vacuum and purge equipment was developed after the *Surveyor II* investigation of possible causes of failure. It was established that small propellant leaks at the TCA could produce salting in the motor throat. The method of prevention, employed from *Surveyor III* on, was the installation of a plug, JPL D143393, into each vernier engine nozzle. The plug was fitted with two tubes to allow flow-through of dry nitrogen as a medium for vapor detection. Between vapor samples, the TCA was evacuated. The plugs were removed by hand at T−175 min, allowing 30 min (until T−145) to clear the tower.

4. Spacecraft Auxiliary Equipment

Spacecraft auxiliary equipment consisted of the squib circuit test set, the special adapter cables, the trailers, the covers, the strain-gage calibration unit, and the accelerometer amplification calibration unit. The squib adapter cables provided connections between the squib simulator and the spacecraft harness. Wear on the spacecraft harness was reduced by reducing the number of connections to the spacecraft harness. Other special adapter cables were used for the same purpose. Strain-gage and accelerometer calibration units were used to calibrate spacecraft equipment.

5. Shipping and Storage Containers

Surveyor long-term storage and reusable containers provided packaged items with adequate protection during shipment and storage, under all environmental conditions. There were three basic shipping container designs: (1) round metal drums (per MIL-D-6054), (2) rectangular boxes (per MIL-C-4150) or of modular construction), and (3) deep drawn aluminum (per MIL-C-4150).

The interior support structures for packaged items were mechanical devices, cushioning materials, or combinations thereof. Each container incorporated a humidity indicator, a pressure relief valve, and provision for desiccant storage.

C. Deployment and Utilization of *Surveyor* System-Level Test Equipment and Facilities

Surveyor test equipment was deployed at three sites during the course of the program. AGE sets 1, 4, and 5 were located at HAC, El Segundo; sets 2 and 6 at the AFETR, Cape Kennedy, Florida; and set 3 installed in a trailer van, was used at HAC El Segundo, California, at Goldstone, and at GD/A (Table IX-1).

1. Equipment Used at HAC

In the Space Environmental Simulation Laboratory, El Segundo, AGE set 1 was used to perform group, system, electromagnetic interference (EMI), solar-thermal-vacuum (STV), vibration, and vernier engine vibration testing. After the installation and activation of AGE sets 4 and 5, AGE set 1 was used exclusively for the 66-hr full-length mission in the simulated space environment provided by the solar-thermal-vacuum chamber and for vibration and ambient mission sequence plugs-out tests on the shaker. A STEA upgrade (change orders 126 and 127) was accomplished after STV testing on SC-4 was completed. At that time also, the floor plan and test equipment arrangement were made essentially the same as AGE set 2 at AFETR. The SC-5, SC-6, and SC-7 spacecraft were tested with the upgraded arrangement including a new system J-box, new cabling, waveguides, shadow shrouds, penetration plates, and mounting hardware.

At the HAC facility in El Segundo, AGE set 4 and 5 were used interchangeably. Primarily, AGE set 4 was used to perform the initial system checkout (ISCO) and mission sequence (MS) testing. The ground support equipment (GSE) included hoisting and handling equipment, weight balance, and alignment equipment (identical to that used at AGE set 2) and fixtures. The GSE and one ground transport vehicle, GTV-1, to transport the spacecraft within the HAC El Segundo location were shared with AGE set 5. The primary use of AGE set 5 was to perform the MS/EMI testing. In addition, this AGE set was used for vibration and ISCO tests (Fig. IX-6).

Table IX-1. Test location and type

AGE set	Type	Location
1	Solar-thermal-vacuum, vibration	HAC, El Segundo
2	Air link during prelaunch countdown and performance verification tests	AFETR
3	Combined system test, vibration, special tests at Goldstone	Mobile-San Diego, El Segundo, and Goldstone
4	Initial system checkout, mission sequence	HAC, El Segundo
5	Initial system checkout, mission sequence, electromagnetic interference, vernier engine vibration	HAC, El Segundo
6	Performance verification tests and air link during prelaunch countdown	AFETR

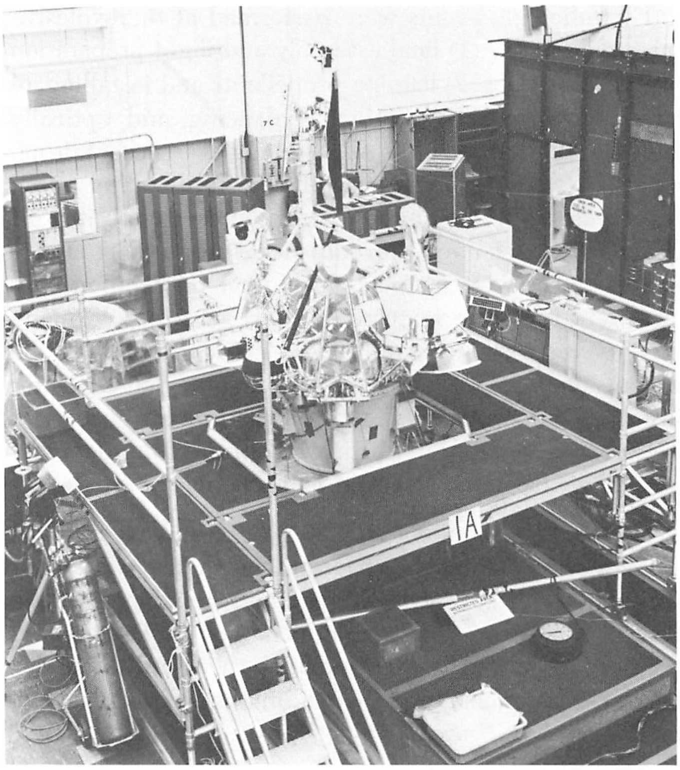


Fig. IX-6. Setup for vibration tests showing workstand

2. Equipment Used at AFETR, Cape Kennedy, Florida

AFETR, Cape Kennedy, Florida used AGE sets 2 and 6. During the *Surveyor* Project, the test plan applicable to AGE set 2 was twice modified from the original use as sole AGE set at the AFETR facility to the AGE set used for hardline testing and finally to the role of air link RF testing only. Originally AGE set 2 included all AGE required to assemble, checkout, and launch the spacecraft at AFETR. This equipment was employed at three distinct geographical areas at AFETR.

Initial inspection and assembly of the spacecraft after arrival and the execution of the performance verification tests (PVT) 1 through 5 took place at the spacecraft checkout facility (SCF) in building AO.

The following events were performed at the explosive safe area (ESA): (1) final assembly and flight preparation of the spacecraft, (2) loading propellants and installing all hazardous items, (3) weighing, balancing, and optically aligning the spacecraft, (4) installing the nose fairing, and (5) system readiness test (SRT) before mating to the flight booster for joint flight acceptance composite test (J-FACT) and launch. Several pieces of support equipment at ESA were employed for final preparation only and were therefore features of AGE set 2 only. Among these were the safety console; the PCC; the MTCU; the propulsion GSE for loading propellants; the GHE for installing hazardous items and transporting the spacecraft between areas; and the RF link for transmitting data and commands between ESF and SCF during the SRTs.

The launch complex (pads 36A, 36B, and blockhouse) was used for mating the spacecraft to the *Atlas/Centaur* booster, performing system readiness tests, performing final flight preparations and launch. The AGE hardware normally used in the area includes the launch complex checkout equipment, and the RF link between the launch pad and the ESF area.

Before the launch of the SC-3 and the arrival of the SC-4 spacecraft, it became evident that one AGE set would not be adequate to support the overlap in spacecraft test operation scheduled at the AFETR facility. Therefore, AGE set 6 was completed and transported to the AFETR facility for the primary purpose of performing all of the air link portions of the spacecraft test operations. AGE set 2 remained in the SC-4 configuration until after the launch, while AGE set 6 was upgraded to operate with SC-5. Equipment from an upgraded HAC, El Segundo AGE set was used to upgrade AGE set 6 at the AFETR when the last spacecraft had passed that point in test. AGE set 2 was used to provide for the possibility of an abort on SC-4 and for air-link RF testing only on SC-5, SC-6, and SC-7.

3. Equipment Used at GD/C, San Diego, California

Originally AGE set 3 had been installed at the GD/C combined system test facility at San Diego, California. However as spacecraft operations progressed through the first launches, the value of STEA 3 as a mobile test facility became apparent. Accordingly, the equipment was installed in a semitrailer and based at HAC, El Segundo.

Testing using the trailer installation was considerably simplified from that previously performed with the permanent installation. In its mobile configuration, STEA 3 was used to perform special tests at Goldstone and spacecraft vibration testing at El Segundo, as well as its prime requirement to perform combined system tests (CST) at San Diego. At San Diego, the launch control rack, operation console and GSE were not in the trailer but were installed inside the CST building. The vibration configuration at HAC, El Segundo was essentially the same as for CST but the power supply rack and the prelaunch monitor rack were near the spacecraft instead of in the trailer.

4. Nonstationary Operational Support Equipment

Spacecraft auxiliary equipment remained with a particular spacecraft from initial testing until launch. Shipping and storage containers were provided for all spacecraft components and flight spares and returned to HAC for reuse. Special test test equipment was provided as required.

5. Statistical Summary of AGE Changes and Problems

Table IX-2 and Figs. IX-7 through IX-13 describe the statistical experience with trouble/failure reports (T/FRs) and engineering change action plans on *Surveyor*. The time period covered is from the inception of the program through September 1967. Peaks in T/FRs reported were usually associated with calibration and preparation periods prior to a launch or test. Peaks in engineering change action plan releases correspond to periods of greatest reported troubles and to periods immediately preceding the first use of AGE sets 1 and 2 and the first use of STEA 6 as a complete test assembly at AFETR.

Table IX-2. Statistical summary of AGE engineering changes

Type	No.	Type	No.
Identification		ECAs[c] released	1365
Master indexes issued	39	Class IIB, record, etc. released	938
Drawings released	7140	Emergency engineering orders processed	997
Control (statistics from third quarter 1965 through 1967)		Individual release transactions	2894
ECRs[a]		Accounting	
		AGE set preparation plans issued	120
Received	2473[b]	Modification kits incorporated	2572
		Configuration indexes released	29
Rejected	1228	Engineering change management reports issued	55

[a]Engineering change requests.
[b]2500 additional estimated from January 1963 to September 1965.
[c]Engineering change authorizations.

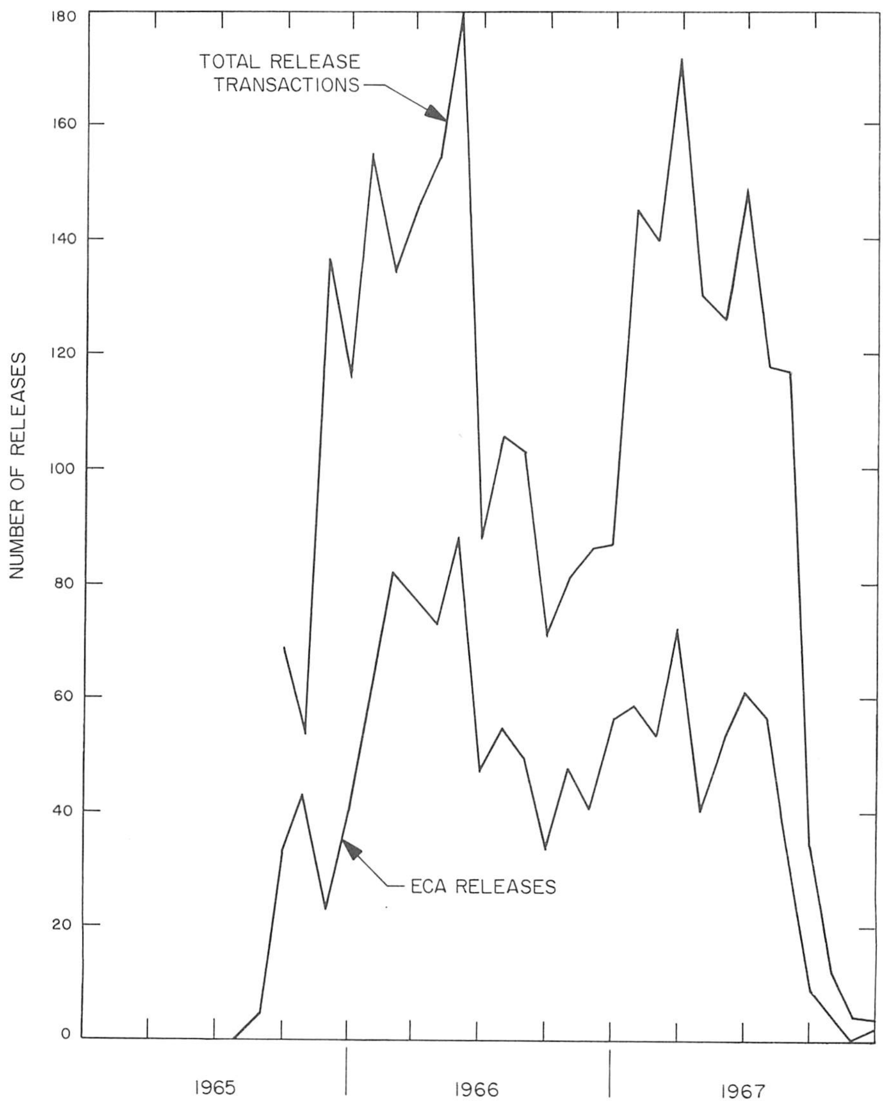


Fig. IX-7. Individual AGE release transactions (September 1965 through September 1967)

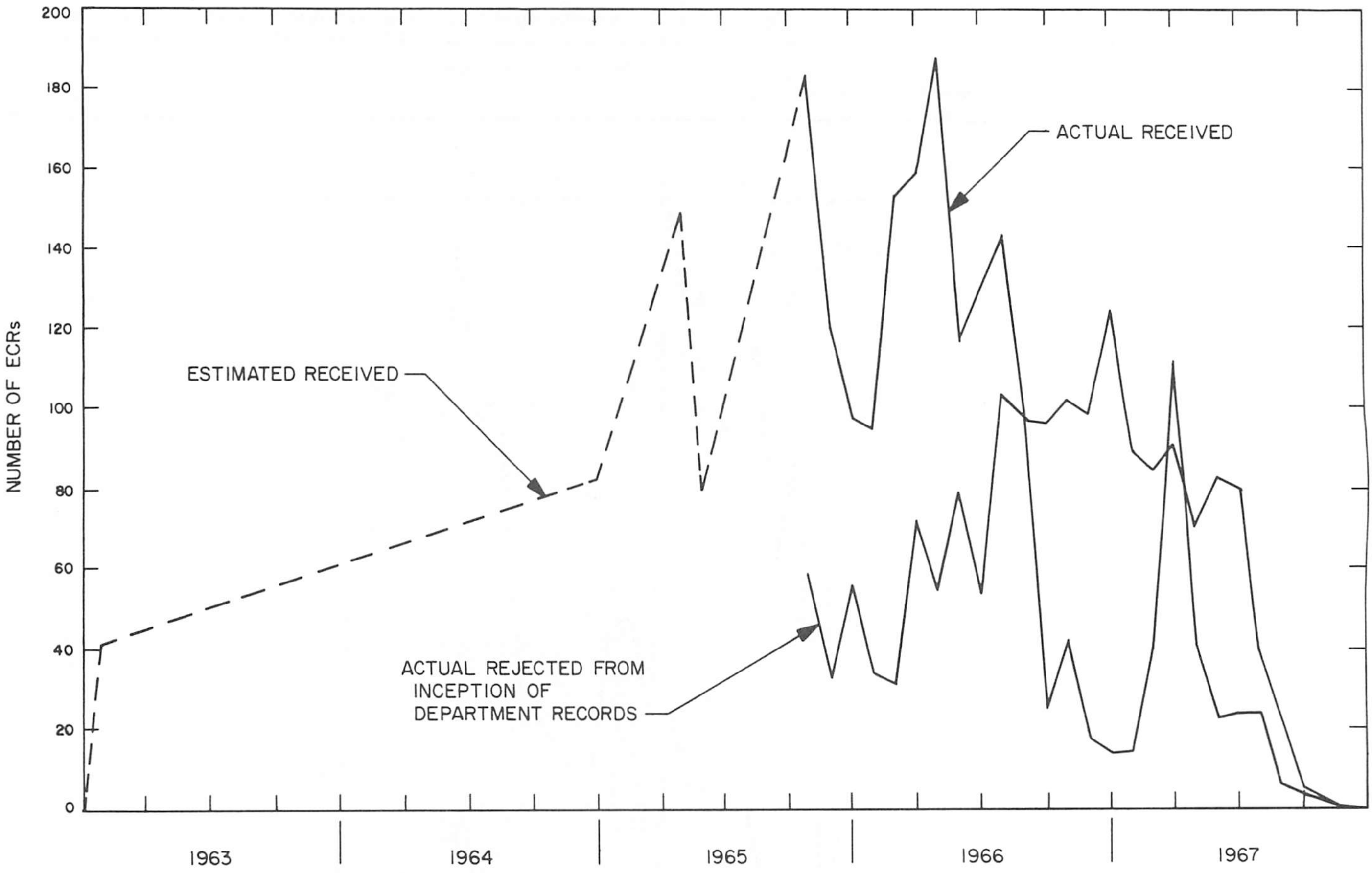


Fig. IX-8. Monthly total of ECRs received (inception through September 1967)

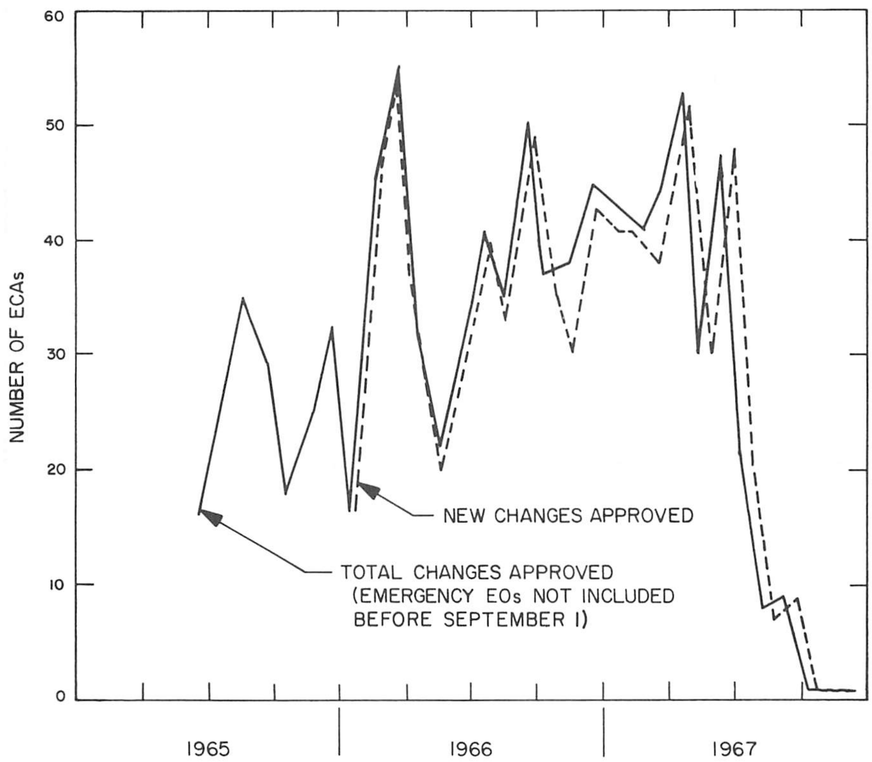


Fig. IX-9. *Surveyor* **AGE ECA approval history (AGE minus CDC)**

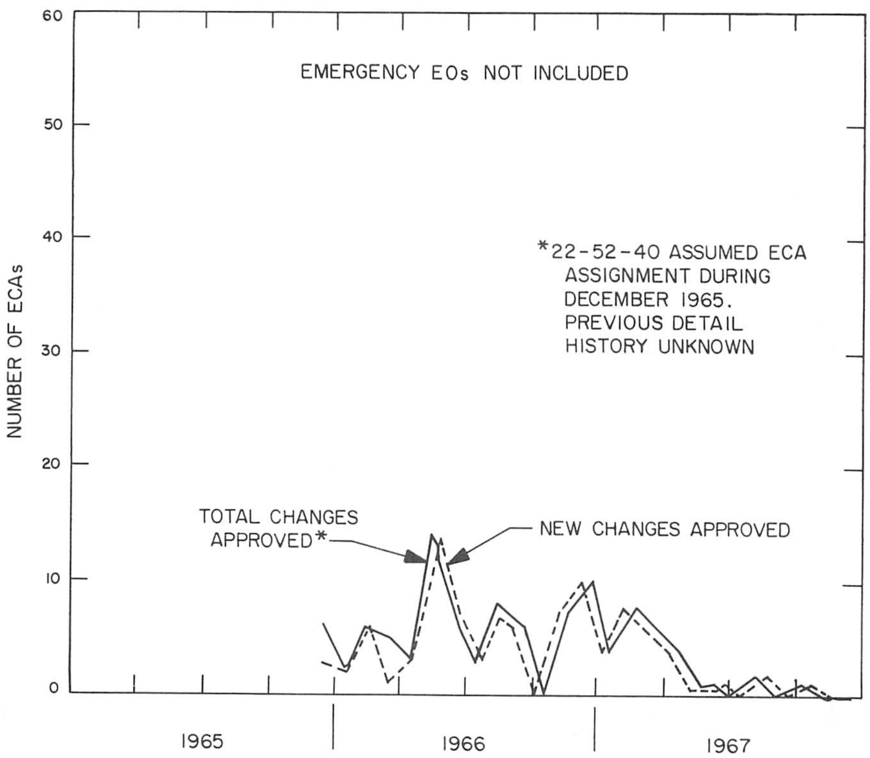


Fig. IX-10. *Surveyor* **AGE ECA approval history (CDC only)**

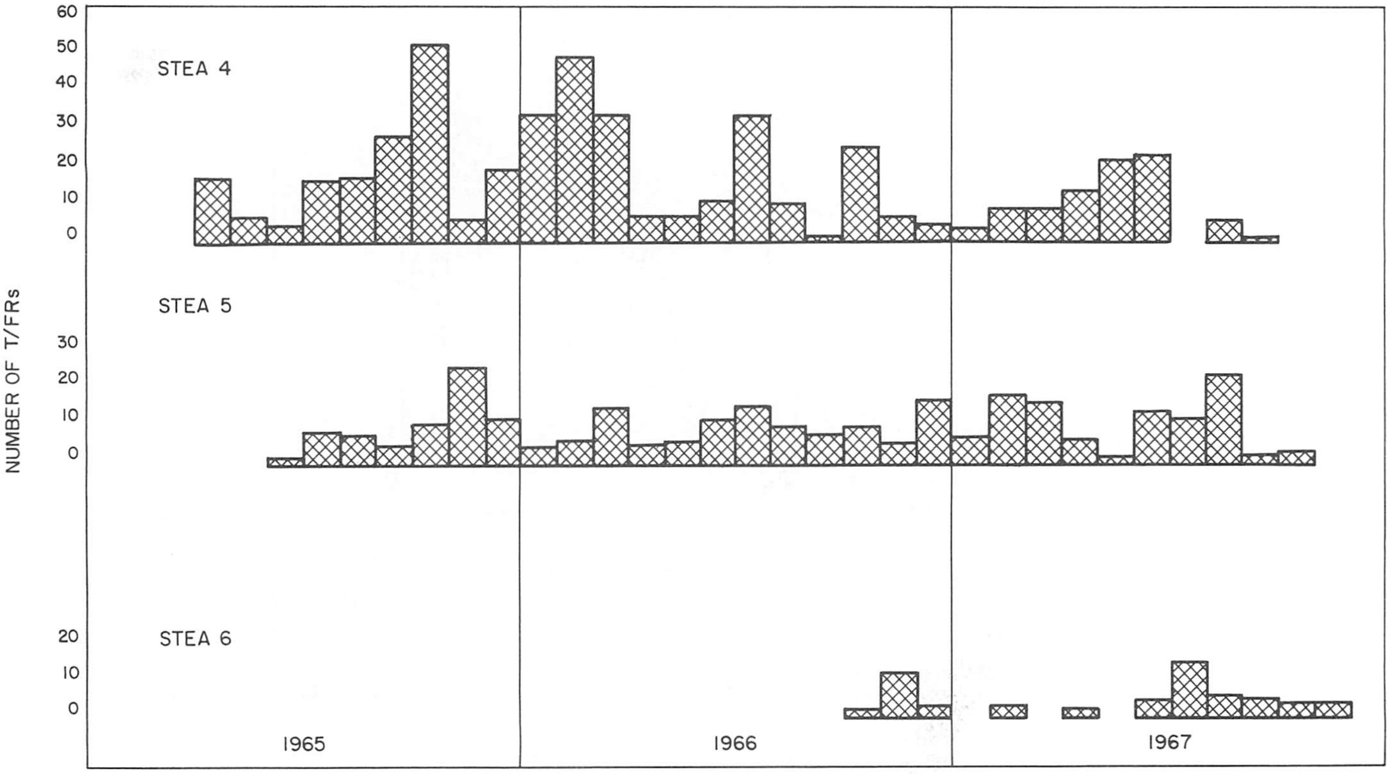


Fig. IX-11. Reported T/FRs plotted against origination date, STEAs 4–6

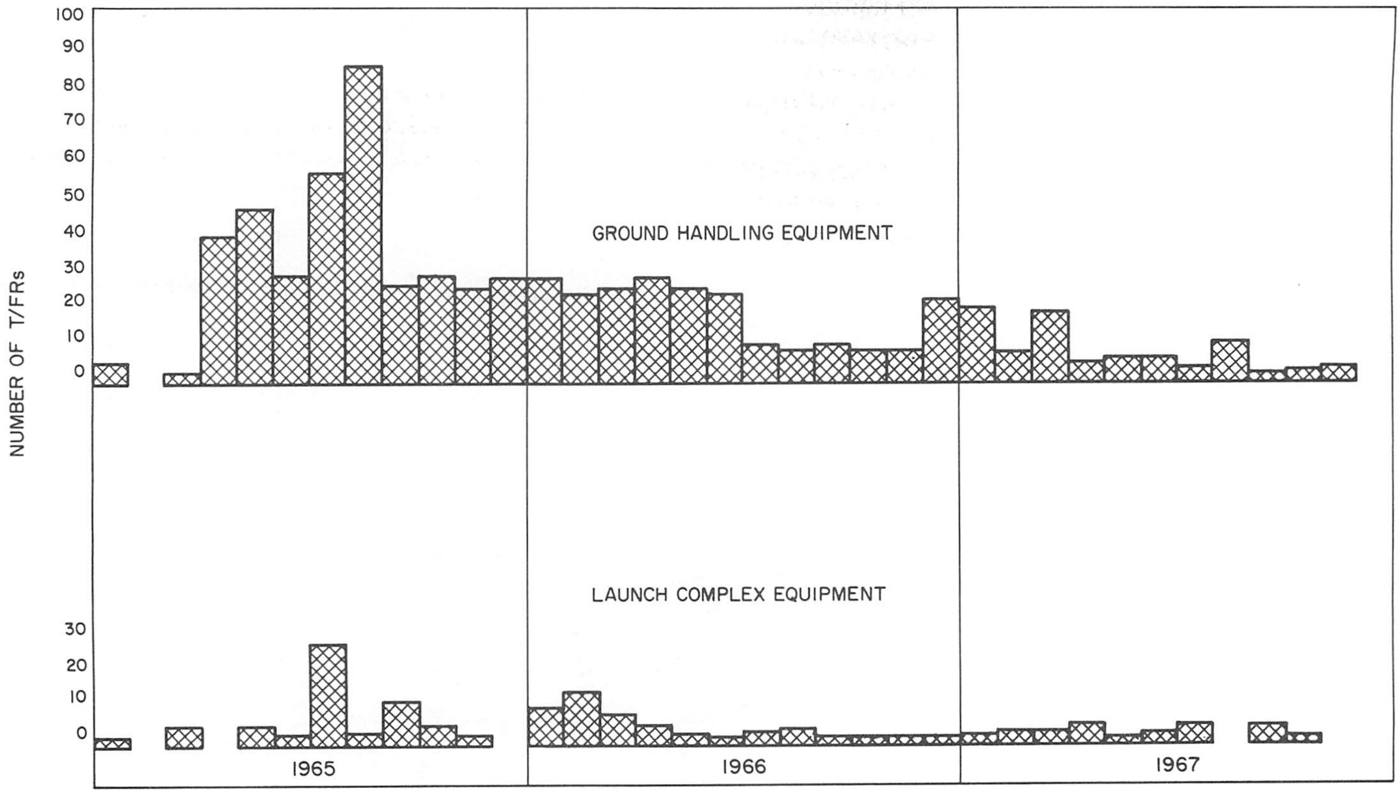


Fig. IX-12. Reported T/FRs plotted against origination date, ground handling and launch complex equipment

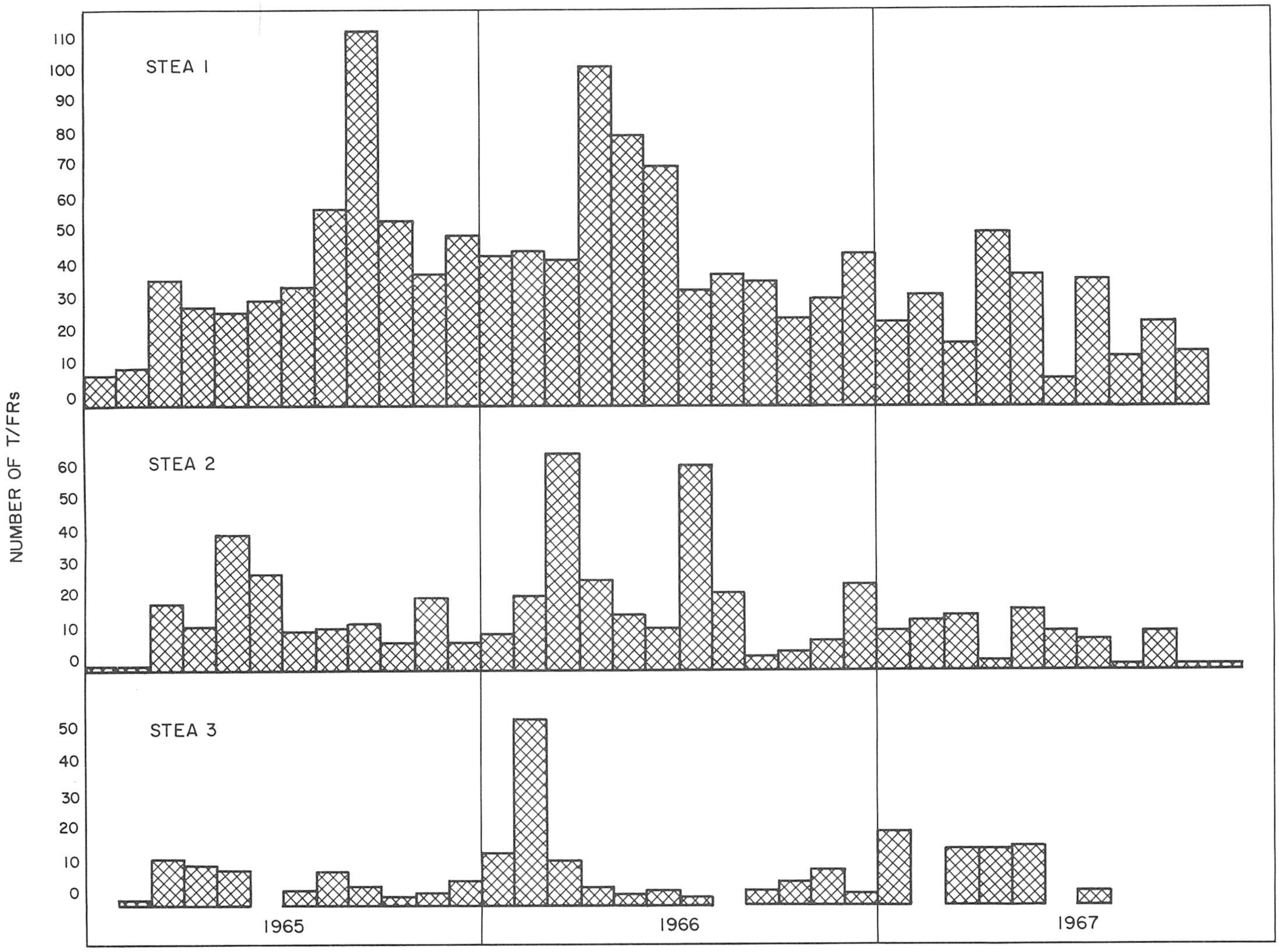


Fig. IX-13. Reported T/FRs plotted against origination date, STEAs 1–3

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X. Spacecraft Environmental Test Requirements

Environmental testing was performed at the subsystem control item level and the system level throughout the *Surveyor* Project. Controlled environmental conditions were imposed on structural and thermal control qualification vehicles, subsystem control items, a prototype spacecraft, and seven flight spacecraft. This section describes the environmental test requirements and discusses the implementation and results of the environmental tests and the environmental data gained from spacecraft flights as each relates to the requirements. See Section XI for a discussion of Spacecraft System test operations and Sections XIV–XXII for specific Spacecraft System test results.

In the role of project manager, JPL was responsible for overall system requirements, and HAC, as contractor, was responsible for the detail requirements necessary to develop a spacecraft capable of meeting the system level requirements. The discussion of the environmental requirements in this section reflects this division of responsibility in that the system level requirements were specified by JPL and the sybsystem control item requirements were specified by HAC with JPL concurrence. Additionally, JPL provided HAC with some control items, designated government-furnished equipment, which were developed under the direct cognizance of JPL. These items are so identified in the following discussion.

A. Derivation of Environmental Test Requirements

1. Vibration Test Requirements

The technical philosophy used to derive the system level and subsystem control item level vibration test requirements is discussed in this section. Detail vibration requirements are given for only the system level requirements, since these can be listed concisely and were the basis for all vibration test requirements. Detail requirements for the control items are given in Subsections E-1 and E-2.

a. Spacecraft System vibration test requirements. The vibration levels originally specified early in the *Surveyor* Project were based on *Ranger* spacecraft flight data. The accelerometers supplying these data were located in the adapter connecting the *Ranger* spacecraft to the *Agena* booster. The random noise level specified for *Surveyor* represented a mean of the *Ranger* random vibration data. The mean level was specified because, at the time *Surveyor* levels were derived, it was felt that the *Surveyor* would be less subject than *Ranger* to acoustic excitation because of the greater distance *Surveyor* would be from the source of the acoustic energy when mounted on top of the *Atlas/Centaur* vehicle. Also, the nose fairing that protected *Surveyor* was assumed to provide more attenuation to acoustic energy.

The sinusoidal environment originally specified for *Surveyor* was higher than the maximum sinusoidal environment derived from the *Ranger* data in the lower frequencies, because it was felt that the *Surveyor/Centaur* combination would have a higher modal density in the lower frequencies than the *Ranger/Agena* combination. It was also assumed that the responses associated with the bending modes of the *Atlas/Centaur* would provide high acceleration levels at the *Surveyor* because of the relatively long length of the vehicle. In the higher frequencies, the original estimate corresponded to the maximum from the *Ranger* data.

Later in the project, it was decided that the original assumptions were questionable and, also, the dynamic data from the *Atlas/Centaur* AC-2 flight indicated that the vibration levels might be lower than anticipated, although the AC-2 vehicle did not have a simulated *Surveyor* payload from which more applicable data could have been extracted. The levels were reduced to those shown in Table X-1, which represent the estimated input vibration levels at the *Centaur/Surveyor* separation plane during flight. These levels were also intended to represent the maximum vibration levels at the center of mass of the main retro motor if the basic resonant frequency of the motor was well above the lower bending frequencies of the *Atlas/Centaur* vehicle.

Table X-1. *Surveyor/Centaur* separation plane input vibration

Frequency, Hz	Level, g	Remarks
Longitudinal (roll) axis		
5–50	1.5 (zero–peak) VFSW[a]	During powered flight
50-1500	1.33 (zero–peak) VFSW	During powered flight
100–1500	2.0 (rms) white gaussian noise	Except during liftoff, maximum q, and/or Mach 1
100–1500	4.5 (rms) white gaussian noise	During liftoff, maximum q, and/or Mach 1
Lateral axes[b]		
1–5	0.275-in. displacement (zero–peak) VFSW	During powered flight
5–50	0.7 (zero–peak) VFSW	During powered flight
50–1500	1.33 (zero–peak) VFSW	During powered flight
100–1500	2.0 (rms) white gaussian noise	Except during liftoff, maximum q, and/or Mach 1
100–1500	4.5 (rms) white gaussian noise	During liftoff, maximum q, and/or Mach 1

[a]Variable frequency sine wave.
[b]Axes perpendicular to the roll axis.

A definition of flight acceptance (FA), type approval (TA), and structures approval levels and the relationship among the three is given in Table X-2. The vibration levels for each type of test can be obtained by multiplying the values in Table X-1 by the appropriate factor of Table X-2. The A level, as given in Table X-2, is the FA level representing an estimate of the 95%[1] flight environment and is the vibration input for the testing of each flight spacecraft. The B level is the TA level and is the vibration input for testing prototype spacecraft to ensure the adequacy of the design and performance of the spacecraft system. A prototype spacecraft is a flight-type spacecraft with all structures and assemblies identical to those of flight spacecraft. The C level is the structures approval level, which is the vibration input for testing the structures test models to ensure the structural integrity of the spacecraft structure and its substructures. A structures test model is a test vehicle with structural members identical to flight spacecraft, but with statically and dynamically simulated mock assemblies.

The ratio between the TA and FA levels generally followed the ratios established for *Ranger* and *Mariner* except in the 1- to 15-Hz range for the lateral axes where the *Surveyor* ratio was 1.25:1 instead of 1.50:1. The ratio was lowered in this range because of critical resonances of the spaceframe, particularly the antenna and solar panel positioner (A/SPP). The ratio of the structures approval to the TA levels was set at 1:15:1 for 1–50 Hz and 1.0:1 above 50 Hz for all axes, which compared with 1.33:1 for *Ranger* and *Mariner*. The ratio of 1.0:1 was

[1]A 95% flight environment is the statistical upper bound of a flight environment as derived from flight data and based on preselected statistical parameters. The term "95%" refers to either the ninety-fifth percentile, 90% confidence level for tolerance limit estimation for the upper bound of statistically nonequivalent data, or the 95% upper confidence level for confidence interval estimation for the mean of statistically equivalent data. Statistically nonequivalent data results from a flight environment that varies randomly from flight to flight, and statistically equivalent data are representative of a repeatable flight environment. For the same sample size and mean value, tolerance limit estimation applied to statistically nonequivalent data will result in a higher 95% flight environment than results from confidence level estimation of statistically equivalent data. Since no *Surveyor* flight data were available at the time the original environmental estimates were made, it was assumed that the environment would vary randomly from flight to flight and the 95% flight environment mentioned in the text was the expected value for the ninety-fifth percentile, 90% confidence level upper bound.

Table X-2. Relationship between vibration test levels

Frequency, Hz	Relative level		
	A	B	C
Longitudinal (roll) axis			
5–50	1.0	1.5	1.73
50–1500	1.0	1.5	1.5
Lateral axes			
1–15	1.0	1.25	1.44
15–50	1.0	1.50	1.73
50–1500	1.0	1.50	1.50

chosen for the high frequencies because the high-frequency vibration was not considered critical for structural loads.

System level testing based on the levels of Table X-1 was accomplished as a sine sweep from 5 to 1500 Hz and 1500 to 100 Hz with random vibration combined with the sine over the 100- to 1500-Hz and 1500- to 100-Hz ranges. The frequency ranges for the testing were dictated by the frequency content of the *Ranger* data and, later in the project, by the frequency content of the *Surveyor* data.

The applicability of the high-frequency combined sine–random testing was subsequently demonstrated. The AC-2 flight acoustic data were analytically transformed into an estimated *Surveyor/Centaur* separation plane ninety-fifth percentile random vibration environment. This environment was then compared with the specified combined sine–random environment by comparing the response, at like frequencies, of a single-degree-of-freedom resonator when subjected to each of the environments. The computed response values are plotted vs frequency in Fig. X-1. A comparison of the two curves shows that the specified environment seems to produce an FA test input lower than the estimated flight environment in the 200- to 1300-Hz region. However, the estimated flight environment was an envelope of data points calculated using the AC-2 flight acoustic data. The methods used to calculate the data points and to envelope the points provided a margin over the actual environment, especially in the 300- to 1100-Hz region, considered greater than the difference between the curves. Therefore, the comparison is considered quite good. Also shown in Fig. X-1 is a plot of the response vs frequency values for the specified TA combined sine–random environment. As can be seen, there was an adequate margin between the TA test levels and the estimated flight environment calculated from the acoustic data.

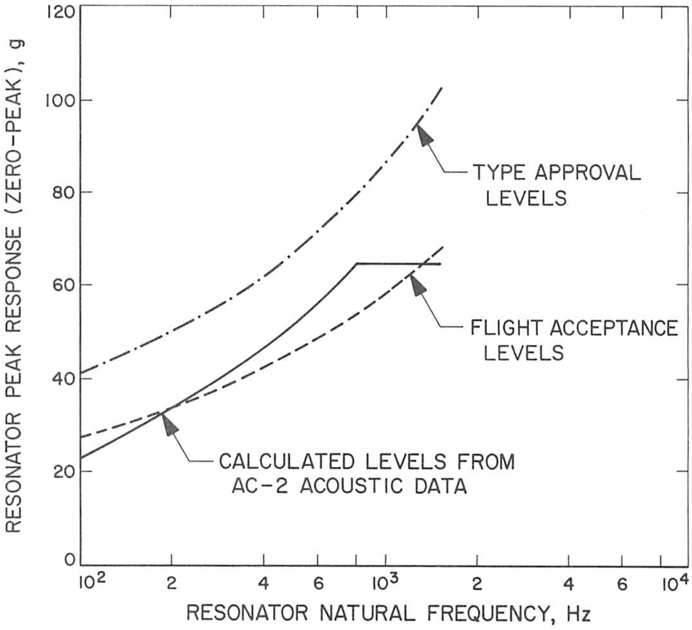


Fig. X-1. Single-degree-of-freedom resonator peak response for excitations consisting of the FA and TA combined sine–random environments

The sweep rates for the sine portion of the various tests (FA, TA, and structures approval), which also determined the duration of the random noise, were specified in HAC requirements documents after agreement had been reached between JPL and HAC on the values. The FA sweep rate of 2 octaves/min was derived by first determining the duration of the AC-2 wide-band acoustic time history at a level with a ratio to the total level that represented the ratio of the response of a single-degree-of-freedom oscillator when excited by the random portion of the specified test levels to the response of the combined random and sine portions. In other words, the portion of the acoustic energy above this time history level was that energy assumed to be represented by the swept sine. This duration was then equated to the duration spent in bandwidth (frequency band between half power points) of a swept sine to compute the sweep rate. The sweep rate, then, provided a duration of excitation in a resonance band simulating the service use duration.

For TA testing, the sweep rate was 1 octave/min, which provided a doubling of the test duration. The HAC unit level TA test durations had been previously established to be twice the FA test duration. The system test durations were made consistent with this ratio.

Sweep rates for structures approval testing varied from 0.5 to 2 octaves/min. For testing conducted late in the program, the sweep rate was specified as 1 octave/min.

Generally, the HAC-established procedures were used to specify structures approval sweep rates.

b. Subsystem control item vibration test requirements. The specified sinusoidal test requirements for the control items were based on the responses of the control items to the predicted *Surveyor/Centaur* separation plane input sinusoidal vibration environment discussed above. The control item responses were first estimated analytically and later empirically established with structural test model vibration tests. These tests and vehicles are described in Subsection B and in detail in Section XIV-B.

The original empirical study consisting of low-frequency vibration tests conducted between 5 and 200 Hz in the lateral axes of the S-1 structural test vehicle provided limited control item response data. The input acceleration amplitude was only 0.25 g rms; this required that response levels corresponding to the much higher predicted environment had to be extrapolated. Control item sinusoidal vibration test requirements were established by enveloping these extrapolated response levels. As more sophisticated and more representative test vehicles became available, the low-frequency longitudinal axis control item sinusoidal vibration test requirements and the high-frequency sinusoidal vibration requirements for all axes were further evaluated and refined from the original analytical and empirical estimates.

These original sinusoidal requirements were considered to be TA requirements because, at the time the levels were derived, the system level sinusoidal requirements, which were the appropriate input vibration levels for the determination of control item responses, were interpreted to be TA levels. The revised requirements established a 1.25:1 or 1.5:1 relationship, depending on frequency, between TA and FA levels with the FA equal to predicted flight levels. Also, the flight level predictions were revised to levels approximately two-thirds of the previous values. The original TA sinusoidal requirements generally were compatible with the revised requirements.

The FA sinusoidal vibration test requirements that evolved from the TA levels generally provided a TA-to-FA ratio of 2:1 at the lower frequencies and 4:3 at the higher frequencies instead of the required 1.5:1 or 1.25:1 ratio. However, the specification of random vibration for TA testing only, as discussed below, resulted in an effective ratio more compatible with 1.5:1 for the overall test environment at the higher frequencies.

The specified random vibration test requirements were based on the assumption that the high-frequency response to this type of environment consists only of local modes on the individual control items. Ideally then, the FA and TA random vibration requirements for the individual control items would have been the same as the system level FA and TA requirements. The actual specified values, however, reflected the philosophy that previously existed in that the TA levels were the same as the predicted flight levels originally specified. For FA testing, a random vibration environment was specified for only a small fraction of the items tested (Subsection E).

Control items developed under the direct cognizance of JPL and supplied to HAC as government-furnished equipment had specified vibration test requirements derived as discussed except that a random vibration environment was specified for FA testing as well as for TA testing. Additionally, the revised ratios specified for TA to FA levels were adhered to for these control items.

Control items required to operate during the descent phase of the mission were subjected to a special vibration test. The specified vibration levels were derived using the expected vibration environment produced by the operation of the retro motor and vernier engines as a basis.

Control item testing using these derived levels was accomplished as a sine sweep covering the frequency range of usually 40–1500 Hz with random vibration combined with the sine during the 100- to 1500-Hz ranges for TA testing. The basis for a combined sine–random test is discussed in the preceding part of this section. The low end of the frequency range was based on a philosophy of testing to one-half of the first fundamental resonant frequency or 50 Hz, whichever was lower. Later, the 50-Hz lower limit was changed to 15 Hz, but almost all control item testing had been accomplished by that time.

In general, sweep rates for control item tests were not linked with the system level sweep rates, which were based on the flight duration of the dynamic environments. However, as desired, a 2:1 ratio of test duration did exist between TA and FA tests and the total duration of the FA control item tests was approximately equal to the total time of the FA system level tests.

2. Thermal Vacuum Test Requirements

a. Spacecraft System thermal vacuum test requirements. Solar-thermal-vacuum (STV) tests were conducted on the Spacecraft System in a simulated solar radiation and vacuum environment.

The Spacecraft System was to be tested at a simulated solar radiation of one solar constant intensity. Flight temperature data from previous *Ranger* and *Mariner* missions, however, revealed variations between the simulated one solar constant intensity and the actual flight radiation intensity. These variations gave evidence of a need for plus and minus margin testing in the STV chamber. Test requirements of 1.12 and 0.87 solar constant intensities, based on black plate measurements, were established.

Maximum operating temperatures were established for all control items. An analysis was made to determine the most severe solar radiation intensity that could be imposed on a *Surveyor* spacecraft without any operating temperature exceeding its maximum limit by spacecraft touchdown on the moon. The analytical study indicated that with a 1.12 solar constant intensity an operating temperature associated with compartment B reached its upper limit at spacecraft touchdown.

Each flight spacecraft was required to be able to function in a 30-min solar eclipse. After spacecraft temperature stabilization in solar radiation, no spacecraft equipment temperature was to drop below the minimum operating temperature limit within a 30-min solar eclipse or within 30 min of termination of the simulated solar radiation in a test chamber. It was determined analytically that if the spacecraft temperature were stabilized in a solar radiation intensity of 87% solar constant and the radiation was then terminated for 30 min, the klystron power supply and modulator (KPSM) subsystem temperature would reach minimum operating temperature. All other spacecraft equipment temperatures would still be in safe regions under these conditions, however. The KPSM, therefore, set the 87% solar constant radiation as the lowest intensity that could be imposed on the spacecraft without violating the specified 30-min eclipse requirement. The 87% solar constant was based on black plate measurements. Figure X-2 is intended to clarify the explanation of the 87% solar constant derivation.

A pressure of 5×10^{-6} torr or less was specified for the vacuum requirement. The pressure requirement was dictated by the test facility capabilities. This level of vacuum provided an adequate test of the spacecraft ability to function as required in a vacuum environment, and was the practical working limit of the vacuum chamber available.

b. Subsystem control item thermal vacuum test requirements. The temperature requirements for control item testing were based on analytical studies made early in the program. Critical upper and lower temperature limits for the operation of any control item were established by the subsystem engineer for each unit. These temperatures formed the basis for the high- and low-temperature thermal–vacuum tests with a vacuum requirement dictated by the capability of the test chambers.

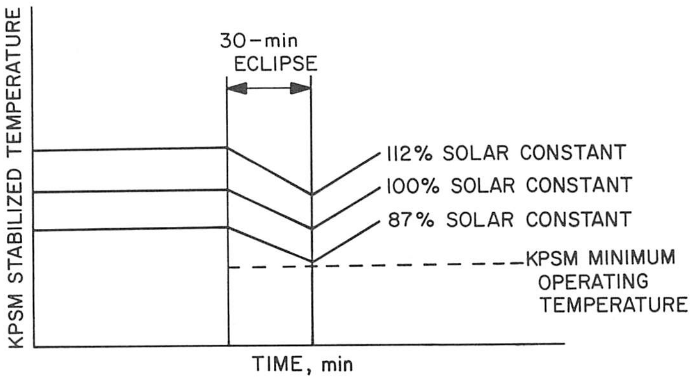


Fig. X-2. Determination of the 87% solar constant test level

The specified TA and FA test temperatures were generally the same, but the durations for the TA tests were much longer than those for the FA tests. The TA test durations were established as the mission duration of a unit or to be sufficiently long to ensure steady-state operating conditions of a unit and to establish degradation properties of that unit. The FA test durations were established to be of sufficient length to insure proper operation of the unit in a thermal vacuum environment, but not of sufficient length to seriously degrade the unit.

3. Electromagnetic Interference Test Requirements

The electromagnetic interference (EMI) test levels were based on calculated incident power densities due to RF sources known to be, or expected to be, operational during *Surveyor* operations at the Air Force Eastern Test Range (AFETR). In addition to these discreet sources, an arbitrary test level derived from military specifications on RF radiation susceptibility testing was defined over the frequency band from 150 kHz to 1000 MHz. In a very few cases, the test levels were limited by the power capabilities of available signal generators.

B. Special Structural Vehicle Qualification Tests

Early in the *Surveyor* Project, the need for special structural test vehicles was established, which resulted in the S-1, S-2, S-2A, and S-9 series of vehicles, and the SD-1 and SD-2 flight test dynamic models. These vehicles evolved both in response to need for test vehicles and to changes in design configuration. The S-1 represented the earliest configuration and was used for the first vibration and static load tests; the S-9 represented the final configuration and was used for the latest structural tests. These vehicles were composed of representative structural members to which were attached either ballast units that simulated CG and mass; dummy units which simulated CG, mass, and inertia; or qualification units which simulated CG, mass, inertia, and function. The earliest vehicles (S-1, S-2) generally had only ballast and dummy units installed, whereas the later vehicles (S-2A, S-9) had predominantly dummy and qualification units installed. A summary description of each vehicle and the dynamic tests conducted on each follows. A more detailed description of the vehicle configuration, test objectives, and results for the S-1, S-2, S-2A, and S-9 can be found in Section XIV of this report.

1. Tests Using the S-1 Structural Test Vehicle

The S-1 structural test model represented an early *Surveyor* configuration and was used for initial vibration testing. In these tests, the S-1 vehicle was tested with lateral sinusoidal vibration inputs parallel to an axis which was parallel to the solar panel/planar array antenna planes in their stowed position. The configurations tested were: (1) vehicle mounted rigidly to the vibration table, (2) vehicle mounted on the *Centaur* payload adapter, and (3) vehicle mounted on the *Centaur* payload adapter with damping tape applied to the inside of the adapter. The rigid mount configuration simulated the specified input, and the second test configuration was introduced to show flexibility effects of the adapter. The input vibration level at the separation plane was 1/4 *g* rms and the test frequency range was 5–200 Hz. This was an exploratory test input, bearing no similarity to the estimated flight environment.

2. Tests Using the S-2 Structural Test Vehicle

The S-2 structural test vehicle represented the early A-21 spacecraft configuration and was used for early modal, vibration, and drop tests.

The modal test was conducted to obtain the elastic modes of the spacecraft for the launch, flight, and landing configurations for use in analytical studies. All modes up to 70 Hz were investigated.

The vibration testing was conducted to qualify the *Surveyor* spaceframe and unit integrating structures for the dynamic loads expected during launch. A longitudinal axis test and a series of lateral axis tests, with the leg 1 azimuth oriented at 0, 45, 90, and 135 deg with respect to the direction of vibration input, were made with the S-2 rigidly mounted to a test fixture at the spacecraft/adapter attach points. Input vibration levels at the spacecraft/adapter attach points were as listed in Table X-3. These levels represented the early *Surveyor* flight environmental estimates.

The S-2 drop test was conducted to qualify the *Surveyor* spaceframe and unit integrating structures for the dynamic loads expected by the spacecraft during lunar landing. Three drop conditions were specified to maximize loads on different critical components. The drop tests were designed to qualify the spacecraft for the touchdown velocity, surface slope, and surface roughness requirements.

3. Tests Using the S-2A Structural Test Vehicle

The S-2A structural test vehicle was a modification of the S-2 vehicle in which spacecraft design changes were incorporated and component simulation improved. The S-2A vehicle was used for vibration and drop tests, and a modal survey.

The S-2A vibration test was conducted to verify the structural integrity of the spaceframe when subjected to C level vibration (see Subsection A-1 for the definition to levels), and to verify the dynamic envelope restrictions and unit level vibration requirements of the A/SPP when subjected to B level vibration input. The test was run along three axes: (1) longitudinal; (2) lateral parallel to the leg 3 azimuth; and (3) lateral perpendicular to the leg 3 azimuth with the vehicle rigidly mounted to a test fixture at the spacecraft/adapter attach points. The input vibration levels at the spacecraft/adapter attach points

Table X-3. The S-2 structural test vehicle vibration test levels

Axis	Vibration input-variable frequency sine wave		
	Frequency range, Hz	g level (zero–peak)	Minimum sweep time, min
Longitudinal	5–40[a]	2.5[b]	3
	40–200	2.0	5
Lateral	5–40	1.25	3
	40–200	2.0	5

[a]Input level at low frequency is not to exceed 0.7-in. double amplitude.
[b]Input to be controlled to prevent the response at the retro motor CG from exceeding 2.5 *g* (zero–peak).

were as given in Table X-4. These levels are the B and C level extensions of the estimated 95% flight vibration environment. The sweep rate of 2 octaves/min was twice that used for later B and C level testing.

The S-2A modal surveys were conducted to obtain data to be included in mathematical models for touchdown studies. Three possible positions for the A/SPP were employed in the surveys with the test vehicle configured to be dynamically similar to the *Surveyor* spacecraft at touchdown. Thirty-two modes were investigated up to a frequency of 60 Hz.

The S-2A drop test sequence was conducted to verify the structural integrity of the upgraded structure during landing conditions. The drop test sequence was designed to subject certain parts of the spacecraft to maximum loads based on specified landing parameters.

4. Tests Using the S-9 Structural Test Vehicle

Initially the S-9 structural test vehicle represented the final A-21 (SC-1 through SC-4) configuration. Torsional vibration tests and modal tests were conducted on the vehicle in this configuration. Vibration and drop tests were conducted on the vehicle after it had been upgraded to the A-21E (SC-5 through SC-7) configuration.

The S-9 torsional vibration tests were conducted to demonstrate that a vehicle with an A-21 flight spaceframe, substructure, and an engineering payload could withstand the specified torsional environment and, also, to measure the dynamic responses of the thermal compartments and the A/SPP for comparison to design loads. All tests were performed with the vehicle in the stowed configuration and attached to the *Centaur* payload adapter which was rigidly attached to a test fixture. The torsional environment was applied about the vehicle's roll axis and controlled at the spacecraft/adapter attach points.

Three different tests were conducted. The first was a low-level (4.4 rad/sec^2 zero–peak), exploratory sine sweep from 5 to 300 Hz and 300 to 5 Hz at 0.5 octave/min sweep rate. The second test was a structures approval level sine sweep. The input, as given in Table X-5, represented the C level extension of the torsional equivalent to the revised lateral axis vibration levels assumed to be acting at the radius of the auxiliary battery. The sweep rate was 0.5 octave/min. The third test was a torsional pulse test. The inputs consisted of taped acceleration/time histories of four torsional pulses, as shown in Fig. X-3. These pulses resulted from a JPL study of torsional motion of the *Atlas* launch vehicle.

The S-9 modal tests were conducted to identify the lower-frequency modes of the spacecraft in both the launch configuration and the landing configuration. For the launch configuration, the spacecraft was mounted on the *Centaur* payload adapter, which was rigidly anchored to a structural floor. Seventeen modes up to 20 Hz were identified. For the landing configuration, the spacecraft was suspended by a low-frequency isolation system. Also, a part of these tests was a determination of the dynamic transfer function between a vibration input at the vernier engines, at a frequency of approximately 40 Hz and an output at the CG of the flight control sensor group.

The S-9 vibration test was conducted to verify the functional operation of the A/SPP after exposure to simulated launch vibrations, to verify the structural integrity of SC-5 type hardware after exposure to a simulated

Table X-4. The S-2A structural test vehicle vibration test levels

Frequency, Hz	Level, g	
	Longitudinal axis	
	B Level	C Level
5–50	2.25 (zero–peak) VFSW[a]	2.60 (zero–peak) VFSW
50–1500	2.00 (zero–peak) VFSW	2.00 (zero–peak) VFSW
100–1500	6.75 (rms) WGA[b]	6.75 (rms) WGA
	Lateral axes	
5–15	0.88 (zero–peak) VFSW	1.00 (zero–peak) VFSW
15–50	1.05 (zero–peak) VFSW	1.21 (zero–peak) VFSW
50–1500	2.00 (zero–peak) VFSW	2.00 (zero–peak) VFSW
100–1500	6.75 (rms) WGA	6.75 (rms) WGA

[a]Variable frequency sine wave, 2.0 octaves/min sweep rate.
[b]Band-limited white gaussian acceleration.

Table X-5. The S-9 structural test vehicle torsional vibration test input levels

Frequency, Hz	Acceleration level, rad/sec^2 (zero–peak)
5–15	9.3
15–50	11.1
50–300	18.3
300–50	18.3
50–15	11.1
15–5	9.3

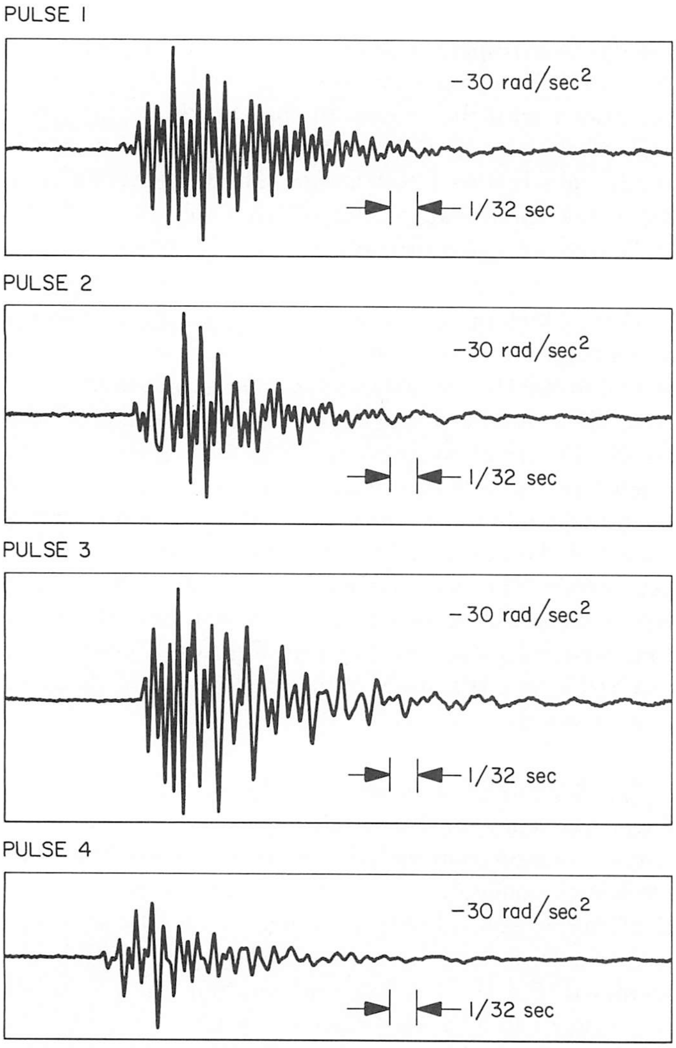


Fig. X-3. S-9 structural test vehicle torsional pulse input at the field joint of the *Centaur* payload adapter

launch vibration environment, and to demonstrate adequate dynamic clearance between the *Centaur* nose fairing and the planar array antenna and the solar panel during launch. The vehicle was rigidly mounted to the test fixture at the spacecraft/adapter attach points for this test. The vibration input at the spacecraft/adapter attach points consisted of two B level and three C level tests, as shown in Table X-6.

The S-9 drop test sequence was conducted to verify the functional operation of the A/SPP after simulated lunar landings and to verify the structural integrity of the SC-5 type hardware following loading, which simulated lunar landings. The drop test sequence was designed to subject the A/SPP to maximum loads based on landing parameters.

5. Flight Tests Using the SD-1 and SD-2 Dynamic Models

The SD-1 and SD-2 vehicles were dynamic models of the operational spacecraft. Each consisted of an operational type spaceframe with dummy components simulating the mass properties of those items they represented. These vehicles were launched aboard the *Atlas/Centaur* AC-5 and AC-6 launch vehicles respectively to check the *Centaur* mission behavior and spacecraft trajectory and to provide vibration data for verification of test requirements. Each spacecraft was equipped with an S-band transponder and a power source to permit tracking after separation from the *Centaur*. The vibration data from these flights was used as a basis for the comparison of flight and test vibration levels in Subsection G.

These vehicles were subjected to TA qualification and FA testing, based on the expected flight vibration requirements.

C. Special Thermal Control Model Qualification Tests

The S-10 thermal control model was a special vehicle used to evaluate and qualify the spacecraft thermal control system. Thermal dissipation of the spacecraft subsystems was simulated during all thermal tests although heat capacity in compartments A and B was not. All thermal finishes were identical to flight spacecraft. The thermal characteristics of the 1250-lbm of main retrorocket propellant were simulated as accurately as possible. The simulated propellant was instrumented with thermocouples at various depths to gain temperature gradient data. The vehicle was instrumented with approximately 800 thermocouples and 50 heaters.

S-10 tests were conducted at both JPL and HAC. The JPL tests included transit and lunar night conditions. The results of the transit were used to compare JPL and HAC facilities. The lunar night tests verified performance of the spacecraft compartment system insulation. The S-10 tests are summarized here, and also in Section XV of this report.

Two transit phase test configurations were tested; one with shadow shrouds simulating the A/SPP and solar panel, and one without shadow shrouds but with the A/SPP and solar panel. The results of the two tests were quite consistent. Each test was conducted at intensities of 80 and 100% of one solar constant (130 W/ft^2 at the level of the preamplifiers on the radar altimeter and doppler velocity sensor antennas). The JPL tests were run

Table X-6. The S-9 vibration test input levels

Test level	Run	Axis of vibration	Sweep frequency range, Hz	Sine level, g (zero–peak)	Random level, g rms	Test time, min	Sweep rate, octaves/min
B	1	Lateral: normal to leg 2	5–15 15–50	0.88 1.02	None	3¼	1
B	2	Lateral: normal to leg 3	5–15	0.88	None	1½	1
C	3A	Lateral: normal to leg 3	5–15 15–50 50–100	1.0 1.2 2.0	None	4½	1
C	3B	Lateral: normal to leg 3	100–1500 1500–100	2.0 2.0	6.75 6.75	3.9 3.9	1 1
C	4A	Lateral: parallel to leg 3	5–15 15–50 50–100	1.0 1.2 2.0	None	4½	1
C	4B	Lateral: parallel to leg 3	100–1500 1500–100	2.0 2.0	6.75 6.75	3.9 3.9	1 1
C	5A	Longitudinal along roll axis	5–50 50–100	2.5 2.0	None	4½	1
C	5B	Longitudinal along roll axis	100–1500 1500–100	2.0 2.0	6.75 6.75	3.9 3.9	1 1

in a 10-ft-diameter chamber requiring the vehicle legs to be removed for clearance. About 400 of the 800 thermocouples were monitored during the transit phase tests.

The compartment system insulation was evaluated during the JPL lunar night tests. The power requirements necessary to hold the compartments at various temperatures between −150 and 50°F were determined to perform the evaluation. The lunar night analytical predictions and test results revealed favorable correlation.

The hot cone test was run on the S-10 at HAC to verify protection from vernier engine nozzle radiation during engine burn. The engine nozzles remained at about 2000°F during a high thrust burn. The test proved that protection measures designed to eliminate any overheating due to thermal radiation from the engine cones were adequate.

Margin testing was also conducted at HAC with the thermal control model in the transit configuration with shadow shrouds. The vehicle legs were on, but the footpads were deleted. The solar intensity, simulated electrical dissipations, and boundary temperatures were varied.

The thermal control model was also used to thermally qualify the pressurized landing gear hydraulic shock absorbers. This was accomplished by placing the vehicle to one side of the chamber and allowing the solar simulation beam to radiate the entire landing leg. The leg plate was equipped with a blanket heater to uniformly heat the leg after solar simulation equilibrium was attained. This uniform heating simulated degradation of the organic thermal control paint on the leg by ultraviolet radiation. Shock-absorber temperature changes were measured concurrently with temperature increases of the landing leg.

D. Special EMI Tests

Two series of special electromagnetic interference tests were conducted. A series of tests was conducted with the T-21 prototype spacecraft to determine the spacecraft's susceptibility to electrical discharge generated by electrification of the launch vehicle during launch.

The second series of special tests was conducted to investigate suspected incompatibility problems at the *Surveyor/Centaur* electrical interface. The second series of interface tests involved T-21, SC-1, and a special interface load simulator.

1. Static Discharge Tests

The objective of the static discharge tests was to determine the susceptibility of the Spacecraft System to possible electrical discharging near or through the spacecraft during launch. The source of the anticipated discharges would be the launch vehicle, through electrification of the vehicle structure by various phenomena. Discharges were considered possible between electrically isolated parts of the vehicle, from the vehicle to the exhaust plume, etc.

A series of discharge tests was included in the T-21 radio frequency simulation test phase. The tests involved discharging through the spacecraft, discharging a charged spacecraft, discharging near the spacecraft, separation of a charged spacecraft, and touchdown of a charged spacecraft. The high voltages associated with the tests were supplied by a 3-kVDC power supply.

It was found that the Spacecraft System was susceptible to several forms of discharge interference. Discharging through the spacecraft involved discharging a charged high voltage capacitor, in parallel with the 30-kVDC power supply, through selected points in the spacecraft circuitry and structure. The energies involved in these discharges were on the order of 0.14 joules. In most cases, the discharge caused only momentary telemetry signal fluctuations. In the case of discharging through the squib return line, several pyrotechnics were detonated and many commutator switches in the engineering signal processor were destroyed. Discharging of the previously charged spacecraft caused one commutator switch failure and consistent detonation of the helium dump squib. All five types of discharging cause telemetry signal fluctuations.

2. *Surveyor/Centaur* Interface Tests

The objective of the *Surveyor/Centaur* interface tests was to investigate predicted incompatibilities between the spacecraft and the launch vehicle due to interference conducted across the electrical interface from the *Centaur* to the spacecraft. The prediction of incompatibilities was based on estimated noise levels supplied by GD/C and analytically determined susceptibility levels provided by HAC.

In response to these predictions, a series of tests was performed to determine the severity of the problem with actual hardware. The test assemblies utilized were the T-21 vs the *Centaur* simulator, the AC-10 launch vehicle vs a special interface load simulator, and finally the AC-10 launch vehicle vs the SC-1 spacecraft. Instrumentation was provided by GD/C and consisted of voltage transient detectors developed by GD/C. Tests were conducted during the pad 36A complex validation, system validation, prototype flight acceptance composite test, and joint flight acceptance composite test phases of initial AFETR operations. The series of tests indicated that the interference levels at the interface were well below those predicted by GD/C. No spacecraft failures were correlated with the interference detected during the tests.

At the same time, GD/C and HAC were directed to reexamine their predictions. The reevaluation performed by GD/C and HAC resulted in less severe predictions by both agencies. Based on these new predicted levels, no interface incompatibilities were expected.

E. Subsystem Control Item Tests

Each spacecraft was an assemblage of approximately 70 different units or control items such as the transmitter, receiver, A/SPP, etc. The control item was the first level assembly that required environmental test before spacecraft installation. Environmental testing at the unit level was performed to environmentally qualify the control items through TA testing and to establish their flight acceptability through FA testing before they were integrated into a Spacecraft System. This minimized the impact of unit failures or deficiencies on the Spacecraft System integrity, program schedules, and cost control.

The TA tests were performed on a representative flight configuration of each control item. These tests were to qualify the unit designs as functionally and/or structurally adequate while exposed to specific environmental conditions. Since the TA test levels were generally more severe than FA levels, TA tested units were classified 4T upon the test completion. This classification prohibited their use as flight equipment.

The FA tests were performed on all control items to verify integrity of workmanship and the ability of the units to function as required in the expected flight environments. Each unit that was TA tested, was first FA tested to establish its workmanship and components as being adequate. All units to be flown or used as flight spares were FA tested before being classified as flight acceptable control items.

Some control items on the spacecraft were developed and fabricated under the direct cognizance of JPL and supplied to HAC as government-furnished equipment. These items included the vernier engine assemblies, the

SC-5, SC-6, and SC-7 solar panels, the alpha scattering instrument, and the Polaroid filters for the television filter wheels. The TA and FA testing of these items was the direct responsibility of JPL.

It was desirable in the environmental test program that the TA and FA test configuration of any unit be consistent, primarily so that TA and FA margins could be established and test results evaluated. If it was not feasible to maintain a consistent configuration for both TA and FA tests of a unit, then the test requirements for both the TA and FA tests generally reflected this configuration difference as well as the specified margins. Almost all control items tested during the course of the *Surveyor* Project had consistent configurations; the few that did not are given in Table X-7.

The various TA and FA test requirements and a summary of test results for each type of test are discussed below. A more detailed description of the tests and test results for any particular control item can be found in Sections XIII–XXII.

1. Type Approval Tests

As discussed in Section X-A, the environmental test levels were originally established to simulate the expected flight conditions. Later in the *Surveyor* Project, this philosophy was changed to require a TA level greater than the maximum expected mission level, although some TA tests were performed with requirements still reflecting the initial criteria.

Table X-7. Environmental test configuration difference

Control item	Environmental test	TA configuration	FA configuration
A/SPP	Vibration	Installed on spacecraft structure	Installed on vibration fixture
Secondary sun sensor	Vibration	Installed on solar panel substrate	Installed on vibration fixture
FCSG[a]	Thermal vacuum	FCSG units tested on control item level	FCSG tested as subsystem
RADVS[b] equipment	Thermal vacuum	Units tested at control item level	RADVS tested at subsystem level
Attitude control, gas supply, plumbing	Vibration	Tested at tank level with mass mockups	Tested as assembly with valves and regulators

[a]FCSG = flight control sensor group.
[b]RADVS = radar altimeter and velocity sensor.

The TA control item testing consisted of vibration tests, thermal vacuum tests, temperature shock tests, pressure shock tests, mechanical shock tests, and static acceleration tests. All control items underwent vibration and thermal vacuum testing. However, only 9% of the units were required to be temperature shock tested, 29% pressure shock tested, and 35% static acceleration tested. Transportation, vibration, and handling shock tests were required for only certain items supplied by JPL. The EMI tests were not required, but could be performed on any unit as optional tests. An acoustic test was required on the solar panel. The surface-sampler control item was not TA tested because of schedule constraints.

a. Vibration test requirements. The TA vibration test requirements were the responsibility of the control item developing organization. Figure X-4 is a typical example of TA vibration test requirements for control items developed by HAC.

Most control item vibration tests covered the frequency range of 40–1500 Hz as a combined sine–random test, although the lower limit of the frequency range was 5 Hz for some control items. The random vibration level ranged from 4.5 to 6.75 g rms, band-limited between 100 and 1500 Hz. A low-level, random vibration input of 2.0 g rms was also generally specified for the control items. A typical test consisted of one 8-min sweep at 6.75 g rms or of one 2-min sweep of 4.5 g rms, and five 2-min sweeps at

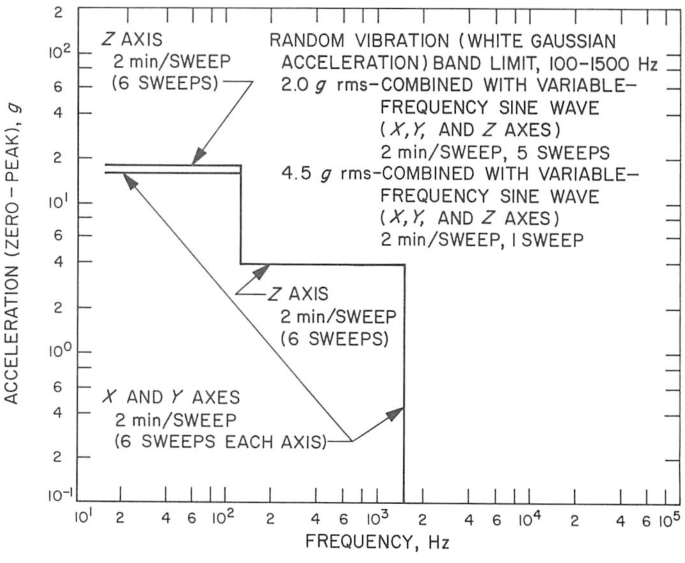


Fig. X-4. Typical TA vibration test requirements for control items

2.0 *g* rms in each axis of test. The random vibration was superimposed on the sinusoidal input from 100 to 1500 Hz in each axis of test. The sinusoidal input ranged from 2.5 to 30 *g* peak at the lower frequencies and from 0.6 to 27 *g* peak at the higher frequencies for the various control items.

The total time duration in each axis of test was generally 12 min. However, if both an upsweep and a downsweep of the sinusoidal input were specified, the duration ranged up to 17 min for each axis. The sweep rates for the sinusoidal inputs were not consistent for all tests since the total time duration was usually divided into multiple sweeps. This resulted in sweep rates ranging from 1.0 to 4.1 octaves/min.

Control items required to operate during the descent phase of the mission were given a special functional vibration test. The flight control electronics unit was subjected to a sinusoidal vibration test in each of three axes.The frequency range was 100–2000 Hz with an acceleration level of 3 *g* peak. The radar altimeter and doppler velocity sensor (RADVS) was subjected to a combined sine–random test. The sinusoidal component had a frequency range of 5–1500 Hz. The random vibration was band-limited 125–1500 Hz. The signal data converter, klystron power supply modulator, and doppler velocity sensor were also tested to these requirements. During this vibration testing the units were required to function in the operational mode they would be in during the descent phase of the mission.

b. Thermal vacuum test requirements. The TA thermal vacuum test requirements were the responsibility of the control item developing agency.

The high-temperature requirements for the compartmentalized units ranged from 125 to 160°F. The low-temperature requirements ranged from 0 to −20°F. The temperature levels for the other control items varied over a much greater range; from +275°F for the high-temperature requirement to −310°F for the low-temperature requirement.

Thermal vacuum temperature levels for the TA and FA tests at the control item level were generally the same. The TA test durations were longer, however. The TA durations were 24 hr for injection and transit, 100 hr for lunar day, and 24 hr for lunar night.

The pressure requirement for the thermal vacuum tests was 5×10^{-5} torr or less. The effects of electrical arc-over were investigated in the transient pressure range of 760 to 1.0×10^{-4} torr for those units required to operate during the initial phases of the mission.

c. Additional test requirements. Additional TA tests such as temperature shock tests, pressure shock tests, mechanical shock tests, static acceleration tests, and others were performed on some of the control items. Several of the tests are discussed in general terms below.

Mechanical shock tests were performed on some of the control items to simulate the spacecraft descent phase environments caused by the retro motor and touchdown dynamics. In the thrust axis four 25 *g*, 5- to 7-msec half sine wave shocks or four 40 *g* 4- to 6-msec terminal peak sawtooth shocks were required. In the lateral axes the *g* levels were 15 *g* and 24 *g*, respectively. The only control item subjected to a mechanical shock based on the response to a pyrotechnic event was the vernier engine assembly. The shock specified for this item was based on the transient produced by the jettisoning of the *Centaur* insulation panels.

Pressure shock tests were performed on some units. It should be noted, however, that these tests were performed with the pressure rapidly increasing rather than decreasing as in the actual case. The control items subjected to this test were required to function during the pressure increase.

Static acceleration tests were usually performed only in the thrust axis. The *g* levels were 10 *g* for 4 min and 5 *g* for 7 min. These levels and durations did vary, however, for some of the units.

d. Summary of TA test results. The results of all control item TA tests were discussed in TA Test Review Committee meetings, where it was the responsibility of the committee to approve the qualification of each control item.

The TA test status of each control item was also discussed in consent-to-ship meetings, which were established for the purpose of reviewing TA and FA test histories at the control item level. Reports prepared for consent-to-ship meeting evaluations, in general, indicated successful passing of the TA tests with a few exceptions. The criteria for evaluating the test results were based on the individual control item test specification. If the requirements of this specification were met, the unit was declared adequately qualified, even if the unit test specification differed from the revised general requirements.

Design changes in the control items were made, as necessary, during the course of the *Surveyor* Project. The TA tests, as required by the cognizant engineers, requalified the modified units. Many design changes were not requalified, however, creating discrepancies between the flight configuration and the TA configuration. These were discussed and approved or disapproved in the unit consent-to-ship meetings where JPL was represented. Typical changes in an electronics unit that was not qualified were component value changes such as a resistor or capacitor value difference. Over 50 such component changes were made in the flight control electronics unit without additional TA tests. The wire gage in the digital equipment was changed after completion of the TA tests. The flight units were fabricated with 24-gage wire, whereas the units TA tested contained 26-gage wire, which weighed less than the 24-gage.

Some configuration changes were qualified with a partial TA retest such as a vibration test performed on a chassis to qualify revised welding techniques.

Examples of complete requalification testing are the A/SPP, television camera, boost regulator, battery charge regulator, transmitter, helium tank, solar panel, and the altitude marking radar. These units were redesigned for the SC-5 through SC-7 spacecraft. (The new solar panel was qualified by JPL.)

Generally, no limitations were placed on the number of environmental tests that could be performed on any unit unless fatigue analyses limited the number of test cycles. Unit rework was flight accepted with partial or complete retests. The retest decisions were dictated by the failure and resulting repairs. Retest decisions were made by the cognizant engineers. Circuit component rework was generally retested with both thermal vacuum and vibration. The vibration testing was often performed in one axis only. This axis was judged to be the most critical for the rework involved.

Some waiving of test requirements occurred when conditions warranted such action. Examples of control item tests waived are the alpha scattering instrument and solar panel test programs.

The transportation vibration and handling shock tests for the alpha scattering instrument were waived with the requirement that the alpha scattering instrument be hand-carried in a special container during in-plant movement and during transit to Kennedy Space Center. The instrument was to be protected from shock such as that produced by rough handling.

Water droplets resulting from a humidity test would have destroyed the solid-state detectors in the alpha scattering instrument. During transit and all prelaunch operations, the alpha scattering instrument was purged with dry nitrogen and was not allowed to be subjected to humid conditions. The humidity test was therefore waived.

Transportation vibration of the solar panels was not performed per specification requirements. The solar panel trailer used as a shipping container was too large for such a test. However, a solar panel shipping trailer with a test panel was traversed over a rough road. Shock and vibration measurements were made with accelerometers and a tape recorder during the test run. The resulting data were analyzed, and the trailer was approved as the shipping container. The trailer satisfactorily isolated the panel from transportation vibration and shock. Each flight panel was transported to the AFETR in a solar panel trailer hard-mounted to the floor of an air ride suspension van.

Test results summary forms were used to document the TA test results of equipment tested under JPL supervision. Figure X-5 contains a typical summary form.

2. Flight Acceptance Tests

The FA testing consisted only of vibration tests and thermal vacuum tests, except for 12 units thermal tested only at ambient pressure. These units were judged to be completely nonsensitive to the vacuum environment.

a. Vibration test requirements. The FA vibration test requirements were the responsibility of the developing organization.

The FA test levels for some control items were greater than the TA test. The main battery, main battery switch, planar array antenna, and shock absorbers were examples of negative TA:FA margins.

Vibration testing of the flight units consisted only of a variable frequency sine wave excitation for all units with the exception of the RADVS, separation sensing and arming device (SS&AD), vernier engine assemblies, alpha scattering instrument, and solar panels. Random vibration was required for these units.

The vibration test requirements for the control items were, in many cases, individually established for each unit. However, most units located in the same spacecraft compartment were tested to the same requirements.

Assembly Polaroid Filters for Surveyor TV Camera Filter Wheel S/N NA Project Surveyor

Operating time ______ Where Performed Bldg. 144 TA ☒ FA ☐

Subassembly Ref. Designation & Serial Nos. Sheet 1 of 2

S/N P-T ______ S/N ______ S/N ______

S/N N-T ______ S/N ______ S/N ______

S/N S-T ______ S/N ______ S/N ______

TA-1 [illegible]

DATE	TEST: DESCRIPTION (VIBRATION, ETC.)	TEST: PARA. No. (FROM TEST SPEC. NO.TS 503679 B)	SIGNATURES: SUBSYSTEM SUPPORT	SIGNATURES: Q.A.	ENV. LAB TEST RPT. NUMBER	PASSED YES/NO	FAILURE REPORT No.	REMARKS
6/19/67	Temp. -Hum.	4. 2. 1	[illegible]	GRN JPL 314	N 370	Yes		
6/22/67	Therm. -Shock	4. 2. 2	[illegible]	GRN JPL 314	N 373	Yes		
7/9/67	Therm. -Vac.	4. 2. 3	[illegible]	GRN JPL 314	N 383	Yes		Delamination occurred along edges. Subsequent evaluation of optical properties indicated only minor effect. Optical changes are within established tolerances.
8/21/67	Vibration	4. 2. 4	[illegible]	GRN JPL 314	360	Yes	007881	Clips were overstressed due to their use more than once. HAC drawing No. 290000 specifies replacement with new clips when filters are replaced. New clips were obtained. This test was to qualify filters not the holding mechanism.
8/18/67	Shock	4. 2. 5	[illegible]	GRN JPL 314	361	Yes	See Remarks	IOM 103.682 reported oscilloscope out of calibration. IOM 103. 695 states calibration performed. No deviation found.
9/1/67	Vibration	4. 2. 4	[illegible]	GRN JPL 314	395	Yes	007990 See Remarks	IOM 2943-482 - Environ. Req. Group requested re-run of Z axis section 2-b of TS 503679 B, Reference PFR 007990.
							007989 See Remarks	During re-run accidental overstress of approximately 175 g occurred. Clips were replaced. One filter was chipped slightly. Re-run Z axis section 2-b of TS 503679 B successfully accomplished with filters O. K.

MAIL TO ENVIRONMENTAL REQUIREMENTS PROJECT REP. AT COMPLETION OF TESTING

Fig. X-5. Typical test results summary form

The start frequencies for the individual sine vibration tests varied from 5 to 40 Hz. The sinusoidal input ranged from 1.5 to 27 g peak in the lower frequencies and from 1.5 to 8.5 g peak at the higher frequencies for the various control items.

There were two types of random vibration tests specified: The vernier engine assemblies, alpha scattering instrument, and solar panel were tested with random vibration combined with the sinusoidal input discussed above. This random level was generally 4.5 g rms bandlimited between 100 and 1500 Hz. The random vibration was superimposed on the sinusoidal input from 100 to 1500 Hz in each axis of test. Additionally, there was a separate random vibration test required for the RADVS, SS&AD, and the vernier engine assemblies consisting of white gaussian noise of 2 g rms. The frequency bandwidth was 50–2000 Hz for the RADVS and 100–1500 Hz for the SS&AD and thrust chamber assemblies. This was an operating test for the RADVS. It was required that each axis be tested for a period of not less than 2 min.

The total time duration in each axis of test was generally 6 min, which was one-half the usual TA duration. The sweep rates varied from 2.0 to 4.1 octaves/min, but were generally the same as the TA sweep rates for any one control item.

b. Thermal vacuum test requirements. Generally, the FA thermal vacuum test requirements differed from the TA requirements in duration only. The duration for the FA tests was typically 2 hr.

c. Summary of FA test results. The FA test history of many control items consisted of a significant number of retests. It was not uncommon for the transmitters to have over 12 axes of vibration testing before their buyoff at the unit consent-to-ship meetings. These same transmitters had often been subjected to five thermal vacuum test cycles of low- and high-temperature extremes. The retests were required primarily by repairs and rework following failures. In some cases, the units were retested following design upgrading.

As for TA testing, HAC placed no limitations on the number of environmental retests that would be performed on a flight unit unless fatigue analysis limited the number of test cycles.

A brief summary of each individual control item test history is available in the unit consent-to-ship documentation for that unit. For those items developed under the cognizance of JPL, test results summary forms were prepared for FA tests. These forms were identical to those for the TA tests (Fig. X-5).

F. Spacecraft System TA Tests (T-21)

The Spacecraft System level TA tests were performed on a prototype spacecraft designated T-21. The T-21 configuration represented the flight spacecraft design electrically and mechanically. All functional requirements of the flight spacecraft were imposed on the T-21.

The TA testing of the spacecraft was performed to qualify the complete Spacecraft System. This was the first test phase to qualify the electronic subsystems integrated into the Spacecraft System. It was, therefore, different from the special vehicle tests discussed previously. (See Section XI of this report for a more detailed discussion of the test description and results.)

1. Vibration Tests

a. Objectives. The primary objectives were to: (1) verify functional operation and structural integrity during and after vibration, (2) establish test criteria level and standards for use in FA testing, and (3) verify adequacy of clearance between spacecraft and *Centaur* shroud during launch. The secondary objective was to qualify *Surveyor* spacecraft design for establishment of test methods, techniques, and procedures used in FA testing.

b. Control procedures. The control method and instrumentation used on T-21 were the same as those used on the SC-1 flight vehicle and described in Section X-G of this report.

c. Test description and summary of results. Table X-8 indicates the test sequence, the tests performed, and test levels and remarks on anomalies encountered in applying the proper test input.

No major failures, which would have demonstrated inherent design weakness, occurred during the T-21 vibration test program. There were, however, minor problems, e.g., loose and/or disconnected coaxial connectors, loose clevises, loose screws, loose damper strut on A/SPP, clearance inadequacies, loose brackets, and cracked spot welds, which developed throughout the test program and were corrected by Engineering Change Request before the first flight spacecraft test.

One of the primary objectives of the T-21 test program was to demonstrate that excursions of the spacecraft parts in all areas were less than a dynamic nose fairing envelope while under the B level vibratory environment. The most critical area in the spacecraft was the solar panel. Four accelerometers were used to determine its dynamic excursion during the shake. Test results showed that the dynamic envelope, as delineated in a JPL engineering drawing, was not violated.

2. Solar-Thermal-Vacuum Tests

The T-21 was functionally tested in STV to qualify the Spacecraft System design in this environment. A secondary objective was to investigate the operational capabilities and limitations of the spacecraft thermal control system while operating under nominal and marginal STV environments.

a. Test requirements. The test requirements imposed on the STV qualification of the spacecraft were the same as those established for the FA tests (Subsection G-2).

b. Test description and results. The STV tests were conducted in three phases. Phase A was aborted during the pumpdown as a result of a flight control malfunction. The solar radiation was to have been 1.0 solar intensity constant. However, a special 12-hr flight control test was later performed.

Phase B consisted of a 66-hr mission sequence. The minimum pressure was 2.5×10^{-6} torr; the average wall temperature was $-305°F$ and the average floor temperature was $-315°F$. The solar radiation was 0.985 solar intensity constant. Problems were encountered during sequence II, but it was completed through posttouchdown operations.

Phase C consisted of three abbreviated mission sequences. The first test was conducted under a 1.0 solar constant intensity level. A pressure of 7×10^{-6} torr was achieved. Although problems were encountered during the various phases of the first test, it was carried through terminal descent.

Table X-8. Vibration levels for the T-21 test sequence

Axis of excitation	Run	Hour: minute	Date 1965	Frequency range, Hz	VFSW g (peak)	WGA, (rms)	Sweep rate, octaves/min	Duration, min	Remarks
Z axis (longitudinal)	1A	16:15	3/26	5–100	0.25	—	1	4.25	
	1B	20:00	3/26	100–1500	0.25	—	2	2.00	
	2A	02:00	3/27	5–50	2.25	—	1	—	TA test
				50–100	2.0	—	1	—	
	2B	20:10	3/27	100–1500	2.0	6.75[a]	1	4.0	TA test
				1500–100	2.0	6.75	1	4.0	
V axis[b] (lateral: along leg 3)	3A	16:28	4/1	5–100	0.25	—	1	4.25	
	3B	18:45	4/1	100–1500	0.25	—	2	2.0	
	4A	23:50	4/1	5–12	0.88[c]	—	1	1.27	TA test aborted[d]
	4AA	20:20	4/2	5–15	0.88	—	1	4.25	TA test
		20:32		15–50	1.05	—	1	—	
		20:56		50–100	2.0	—	1	—	
	4B	12:20	4/3	100–1500	2.0	6.75	1	4.0	TA test
		12:43		1500–100	2.0	6.75	1	4.0	
U axis[b] (lateral: normal to leg 3)	5A	21:15	4/5	5–26	0.25	—	1	2.38	AAL dump[e]
	5AA	22:40	4/5	5–100	0.25	—	1	4.25	
	5B	10:10	4/6	100–1500	0.25	—	2	2.00	
	—	—	4/7	5 and 25	0.25	—	—	15.0	Control system test
	6A	14:30	4/8	5–15	0.88	—	3	0.53	TA test aborted[b]
	6AA	15:15	4/8	5–15	0.88	—	1	4.25	TA test
		15:34		15–50	1.05	—	1	—	
		15:53		50–100	2.0	—	1	—	
	6B	16:50	4/9	100–1500	2.0	6.75	1	4.0	TA test
		17:10		1500–100	2.0	6.75	1	4.0	

[a]Undertest occurred: 4.3 *g* rms.
[b]*V* and *U* axes later redesignated *A* axis and *B* axis, respectively.
[c]Subsequent analysis showed that TA test level was not achieved. Actual level was 0.6/0.88=68% TA test.
[d]A/SPP motion limiter disconnected from solar panel at 13 Hz.
[e]Automatic amplitude limiter (AAL) circuit dumped because of calibration problems associated with footnote d.
[f]Operator error; sweep rate incorrect.

The second test of phase C was conducted under a 1.14 solar constant intensity level. During the midcourse maneuver phase, the transmitter and auxiliary battery temperatures exceeded the red-line, high-temperature limits.

The third test of phase C was performed under a 0.7 solar constant intensity. Problems were encountered with the engineering signal processor and solar panel stepping commands. During the terminal descent phase, the radar altimeter failed to operate, causing simulated spacecraft free fall.

G. Spacecraft System FA Tests

1. Vibration Tests

a. Test requirements. Before launch, each *Surveyor* spacecraft was subjected to a vibration environment to establish flight acceptance. These tests verified functional operation during and following simulated launch, acted as a quality control check on fabrication and spacecraft assembly, and verified that Spacecraft System alignments were not degraded by exposure to vibration.

The test consisted of vibration inputs to the spacecraft in each of three orthogonal directions. These were:

(1) the spacecraft roll axis (Z axis), (2) a lateral axis parallel to leg 3 azimuth (*A* axis), and (3) a lateral axis normal to leg 3 azimuth (*B* axis). These lateral axes were selected because early testing on the project indicated them to be the most sensitive to lateral vibration.

The *A* and *B* axes should not be confused with *X* and *Y* axes used for all spacecraft flight operations and used in some special tests where the *Y* axis is a lateral axis parallel to a plane continuing leg 1 and the Z axis and the *X* axis is an axis completing the orthongonal right-hand system.

The vibration test itself consisted of a low-frequency (5–100 Hz) sine wave portion and a high-frequency (100–1500 Hz) combined sine–random portion in each of the three test axes. Table X-9 indicates the particulars of the test inputs. The low-level sine test indicated for

Table X-9. Spacecraft System flight acceptance vibration test conditions

Test	Axis of vibration	Frequency band, Hz	Sine level, g (peak)	Random level, g rms	Test time, min	Sweep rate, octaves/min	SCAMP[a] inputs	Random inputs
Low	Longitudinal (Z axis)	5–100	0.25	None	4.25	1	FZ 1 RZ 05 RA 06 RB 07	—
FA	Longitudinal (Z axis)	5–50 50–100	1.5 1.33	None None	2.13	2 2	FZ 1 RZ 05 RA 06 RB 07	—
Equalization	Longitudinal (Z axis)	100–1500	None	3.18	As required	NA[b]	—	FZ 1 FZ 2[c] FZ 3[c]
FA	Longitudinal (Z axis)	100–1500 1500–100	1.33 1.33	4.5 4.5	2.0 2.0	2 2	FZ 1 FZ 2 FZ 3	FZ 1 FZ 2[c] FZ 3[c]
FA	Lateral A (along leg 3)	5–50 50–100	0.7 1.33	None None	2.13	2 2	FA 3 RZ 05 RA 06 RB 07	—
Equalization	Lateral A (along leg 3)	100–1500	None	3.18	As required	NA	—	FA 1[c] FA 2[c] FA 3
FA	Lateral A (along leg 3)	100–1500 1500–100	1.33 1.33	4.5 4.5	2.0 2.0	2 2	FA 1 FA 2 FA 3	FA 1[c] FA 2[c] FA 3
FA	Lateral B (normal to leg 3)	5–50 50–100	0.7 1.33	None None	2.13	2 2	FB 1 RZ 05 RA 06 RB 07	—
Equalization	Lateral B (normal to leg 3)	100–1500	None	3.18	As required	NA	—	FB 1 FB 2[c] FB 3[c]
FA	Lateral B (normal to leg 3)	100–1500 1500–100	1.33 1.33	4.5 4.5	2.0 2.0	2 2	FB 1 FB 2 FB 3	FB 1 FB 2[c] FB 3[c]

[a]See Fig. X-6 for accelerometer locations.
[b]Not applicable.
[c]These were added for SC-6 and SC-7. The three channels were then sampled for equal time periods and this signal was sent to the analyzer–equalizer system.

the Z axis was used to establish the integrity of the control circuitry. On SC-1, this low-level run was performed before each axis of vibration test. However, once enough confidence was gained in the system and the operation, it was retained only preceding the Z axis vibration.

b. Control methods and instrumentation. The control method used on the SC-1 through SC-7 sine portions of the vibration test were similar; however, the random control used on SC-6 and SC-7 was an improved technique over earlier tests. Each method with its appropriate instrumentation will be discussed independently.

1. SC-1 through SC-7 sine control technique. The sinusoidal input was controlled by an automatic switching device known as SCAMP (selector control adjustable, multiple point). This device selected the highest of four signals, which it received from four accelerometers, compared this signal to the test reference level and servo controlled the shaker input. For the low-frequency runs (5–100 Hz), three accelerometers were mounted in the three orthogonal axes on the retrorocket (one longitudinal, one lateral along leg 3 azimuth and the other lateral normal to leg 3 azimuth). An additional accelerometer was mounted on the vibration fixture with its sensitive axis in the direction of vibration (Fig. X-6). For the high-frequency combined sine–random, control of noise and sine was accomplished with separate systems. The sine portion was removed from each of three accelerometers mounted on the fixture in the direction of shake. This was done by routing the signals through 50-Hz bandpass filters about the sweep frequency. The outputs of these filters were sent to the SCAMP; the procedure then followed that of the low-frequency runs.

2. SC-1 through SC-5 random control technique. The random vibration signal was obtained from another single accelerometer mounted on the fixture, was sent through a 50-Hz sine rejection filter and then to the random vibration equalizer, which controlled the random vibration to an acceleration spectral density level of 0.0145 g^2/Hz ± 3 db over the range of 100–1500 Hz.

3. SC-6 and SC-7 random control technique. Random signals were obtained from three fixture-mounted accelerometers. These signals were passed through three sine rejection filters with 50-Hz bandwidths and then on to a sampling unit. This unit sampled each of the three input channels for equal time periods and sent this random averaged signal to the analyzer–equalizer, which analyzed the signal with 80 fixed bandwidth filters with center frequencies from 12.5 to 2000 Hz and adjusted the output level to the spectrum shape desired. The time period of the commutated signal was adjustable between 20 and 250 msec, with the actual duration determined by the response of each vehicle. Nominally, a rate of 50–100 msec/scan was found to be optimum.

Test control was maintained through the use of (1) combinations of accelerometers mounted on the fixture with their sensitive axis along the axis of vibration and (2) three accelerometers mounted on the retro motor casing. Figure X-6 indicates the position of the control accelerometers for each axis of vibration and the retro motor positioning, while the final two columns of Table X-9 display the accelerometers used for control during each phase of testing.

The use of the SCAMP peak selector led to some variations in the control profile of each spacecraft. Table X-10 shows the accelerometers controlling the test over the range 5 to 100 Hz for SC-1 through SC-7; Table X-11 indicates the 100- to 1500-Hz control.

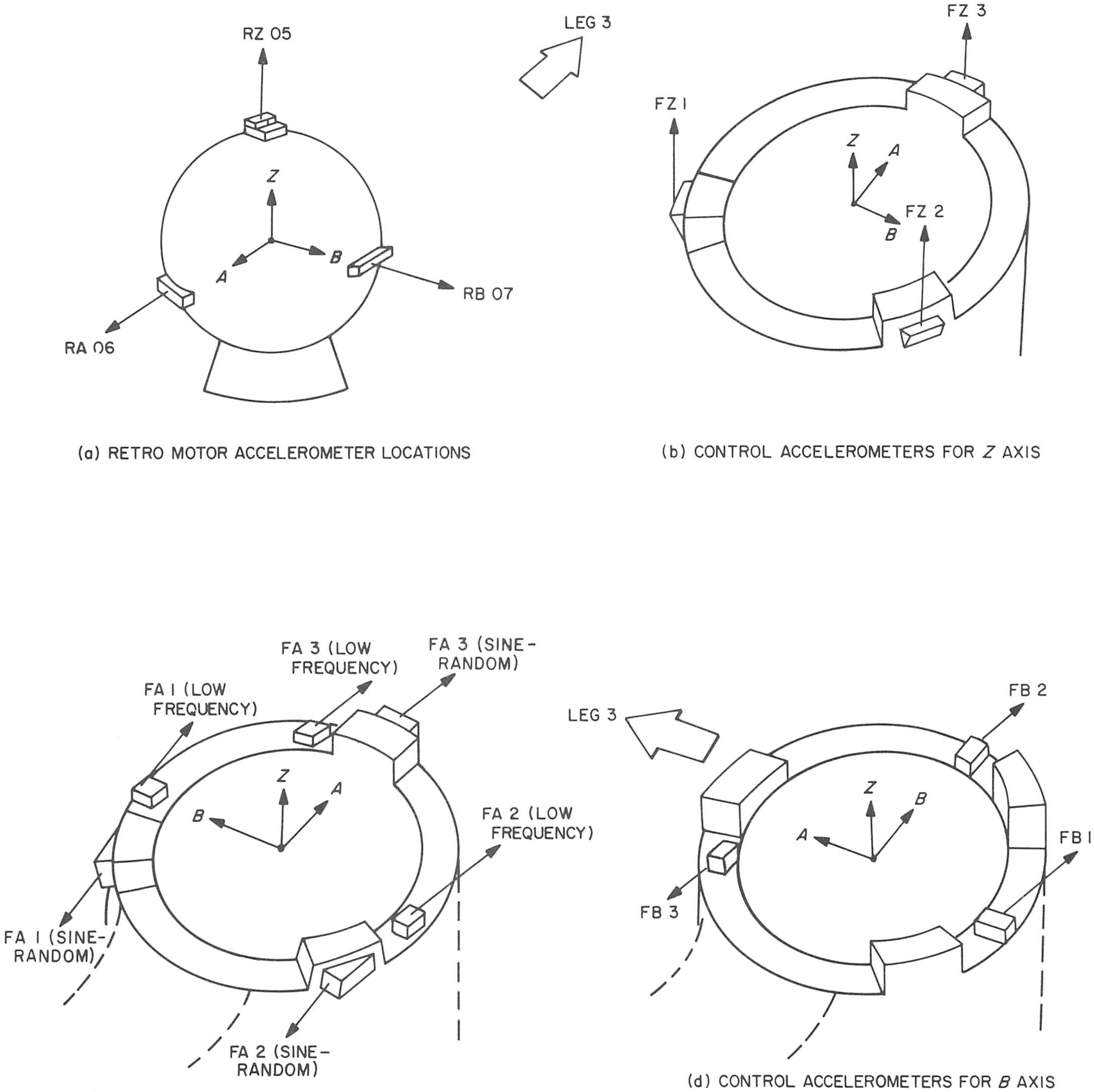
4. Block diagram of control equipment. Figures X-7 through X-9 are block diagrams of the control circuitry for the sinusoidal testing and the combined sine–random testing.

Figure X-7 shows the two-stop system accelerometer channels, 2.5-kHz low-pass filter (fixture) and 40-Hz low-pass filter (retro), that were added after the SC-2 transient; otherwise this is essentially the control system used throughout the project.

Figure X-8 shows the two-stop system accelerometer channels, 2.25-kHz low-pass filter (fixture) and 40-Hz low-pass filter (retro), that were added after the SC-2 transient. The 80-channel automatic equalizer–analyzer was added before SC-1 retest. Otherwise, this was essentially the sine–random control system used for SC-1 through SC-5.

Figure X-9 shows the sine–random control system used for SC-6 and SC-7.

5. System equalization and verification. To apply random vibrations to the *Surveyor* vehicle in the 100- to 1500-Hz range during system level testing, a random noise generator was used. The noise was amplified to drive the exciter to which the *Surveyor* was attached.



(a) RETRO MOTOR ACCELEROMETER LOCATIONS

(b) CONTROL ACCELEROMETERS FOR *Z* AXIS

(c) CONTROL ACCELEROMETERS FOR *A* AXIS

(d) CONTROL ACCELEROMETERS FOR *B* AXIS (LOW FREQUENCY AND SINE-RANDOM)

Fig. X-6. Orientation of system vibration test control accelerometers

Table X-10. Control accelerometer ranges: 5–100 Hz (SC-1 through SC-7)

Spacecraft	Z axis		A axis				B axis		
	RZ 05, Hz	FZ 1, Hz	RZ 05,Hz	RA 06, Hz	RB 07, Hz	FA 3, Hz	RZ 05, Hz	RB 07, Hz	FB 1, Hz
				5–24		24–28		5–23	23–27
SC-1	5–48	48–100	28–30			30–46	27–29		29–43
				46–56		56–100		43–55	55–100
				5–25		25–28		5–23	23–45
SC-2	5–46	46–100	28–30			30–46	—		
				46–52		52–100		45–56	56–100
				5–25		25–48		5–25	25–45
SC-3	5–48	48–100					—		
				48–55		55–100		45–56	56–100
				5--25		25–48		5–24	24–28
SC-4	5–46	46–100					28–32		32–48
				48–58		58–100		48–59	59–100
				5–20	20–21	25–29		5–23	
SC-5	5–45	45–100	29–31	21–25		31–48	23–33		33–48
				48–58		58–100		48–58	58–100
				5–25		25–28		5–24	
SC-6	5–46	46–100	28–31			31–48	24–33		33–48
				48–51		51–100		48–58	58–100
				5–24		24–28		5–24	
SC-7	5–42	42–100	28–30			30–46	24–33		33–47
				46–58		58–100		47–58	58–100

[a]New retro motor mounts installed, better flight simulation.

However, the amplitude response, measured at the attach points, showed peaks and notches caused by mechanical resonances of the *Surveyor*. Thus, the vibration test, as applied to the *Surveyor*, no longer represented the random input to the vibration system. To obtain a flat amplitude spectrum at the attach points, the driving voltage to the exciter was required to vary inversely with the amplitude response of the exciter.

Initially in the project, the only equipment available to accomplish this was a manual analog equalizer. This unit consisted of a chain of shaping devices which, by adjustment of their center frequencies and damping, created the inverse of particular peaks and notches of the exciter's response. The use of the manual analog equalizer was too time consuming for the multiresonant *Surveyor* vehicle and fixture combination, and the equalization itself was a compromise because of the lack of dynamic range and interaction between filters.

To circumvent these limitations, an 80-channel automatic system was obtained and was used from the retest of SC-1 through the end of the system test program on SC-7. This system divided the spectrum into 80 discrete bandwidths, and the amplitude of each segment was adjusted to obtain a close approximation of the desired amplitude curve. This 80-channel automatic equalizer–analyzer saved considerable time and provided more accurate equalization because of its increased dynamic range and additional filtering.

Figures X-8 and X-9 indicate the position of the equalizer–analyzer in the test loop. Although only the automatic equalizer–analyzer is shown, Fig. X-8 would also be correct for the SC-1 test if the manual analog equalizer were substituted.

Verification that proper equalization had been obtained (± 3 db of specified value in the bandwidth) was substantiated by analyzing a 15-sec tape loop with 50-Hz bandwidth filter. After checking the equalization in this manner on the first five vehicles and at the same time monitoring the oscilloscope display of the spectrum shape

Table X-11. Control accelerometer ranges: 100–1500 Hz (SC-1 through SC-7)

Spacecraft	Z axis				A axis			B axis	
	Random	Sine	Sine	Sine	Random	Sine	Sine	Random	Sine
	FZ 1	FZ 1	FZ 2	FZ 3	FA 3	FA 2	FA 3	FB 1	FB 2
SC-1	100–1500	100–200 700–800	200–280 800–1000 1200–1500	280–700 1000–1200	100–1500	100–120 640–800	120–640 800–1500	100–1500	100–1500
SC-2	100–1500		100–280 800–1500	280–800	100–1500	660–900	100–660 900–1500	100–1500	100–1500
SC-3	100–1500	100–200	200–300 440–600 800–1500	300–440 600–800	100–1500	680–760	100–680 760–1500	100–1500	100–1500
SC-4	100–1500		100–130 230–290 440–500 800–1500	130–230 290–440 500–800	100–1500	640–800	100–640 800–1500	100–1500	100–1500
SC-5	100–1500		100–260 460–540 840–1500	260–460 540–840	100–1500	660–740	100–660 740–1500	100–1500	100–1500
SC-6	100–1500[a]	100–200 560–1000	200–300 440–560 1000–1500	300–440	100–1500[a]	620–700	100–620 700–1500	100–1500[a]	100–1500
SC-7	100–1500[a]	260–300	440–540 800–1500	100–260 300–440 540–800	100–1500[a]	1300–1500	100–1300	100–1500[a]	100–1500

[a]All the fixture accelerometers control.

on the equalizer–analyzer, it was felt that sufficient acumen had been gained to buy off the equalization on the basis of scope presentation alone. This was done on SC-6 and SC-7, and a tape loop analysis was performed after completion of testing.

6. *Automatic acceleration limiter.* As a safety backup circuit for the control system, independent signal-sensing accelerometers were placed on the fixture and on the retro motor with their sensitive axes in the direction of vibration. These accelerometers were connected to an emergency tripping circuit, which cut off vibrator armature power when any signal detected indicated local vibration to be 50% greater than the specified level for the test (Table X-12). Also following the SC-2 transient, two additional accelerometers were added to the safety circuitry (Table X-12 and Figs. X-7 through X-9).

7. *High-frequency sine control.* It was found that the sine plots extracted from the combined sine–random test indicated that very little sine input was present, although the level appeared to be high in some cases. The reason was the use of the 50-Hz bandwidth filters. The random level was so great compared to the sine that the sine was overridden by the random coming through the filters. Therefore, the sine plots were actually plots of the average combined random–sine signal passing through the filters.

In some cases there were points that briefly fell below the specified value because of the inability of the servo, which controlled the sine, to respond immediately after being swamped by the random signal. The use of the 50-Hz filter was continued throughout the project, based on the precedent set during T-21 testing.

8. *Crosstalk accelerometers.* Accelerometers were also mounted on the fixture to measure crosstalk. These readouts demonstrated that little crosstalk existed during all phases of testing.

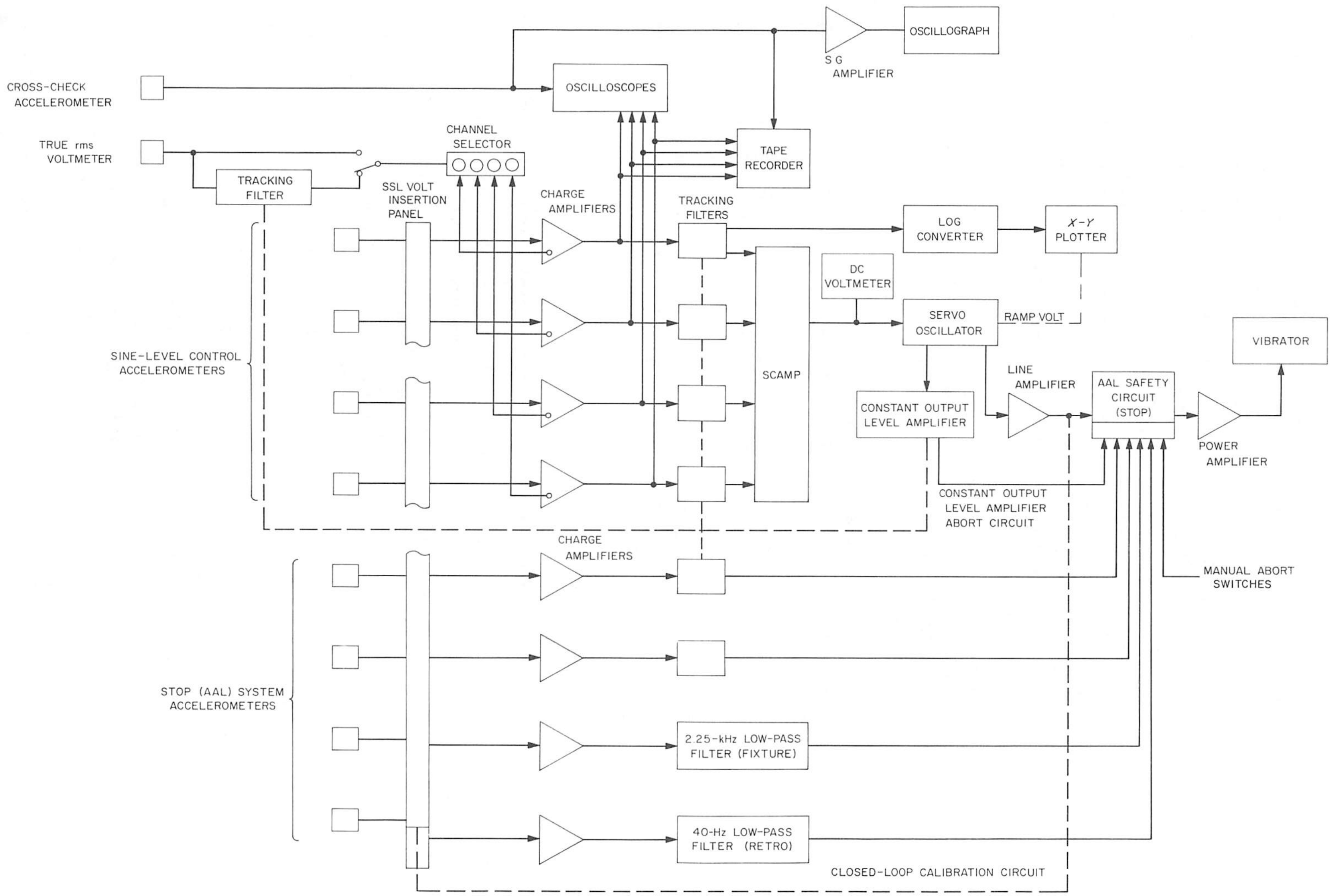


Fig. X-7. Sinusoidal vibration control block diagram

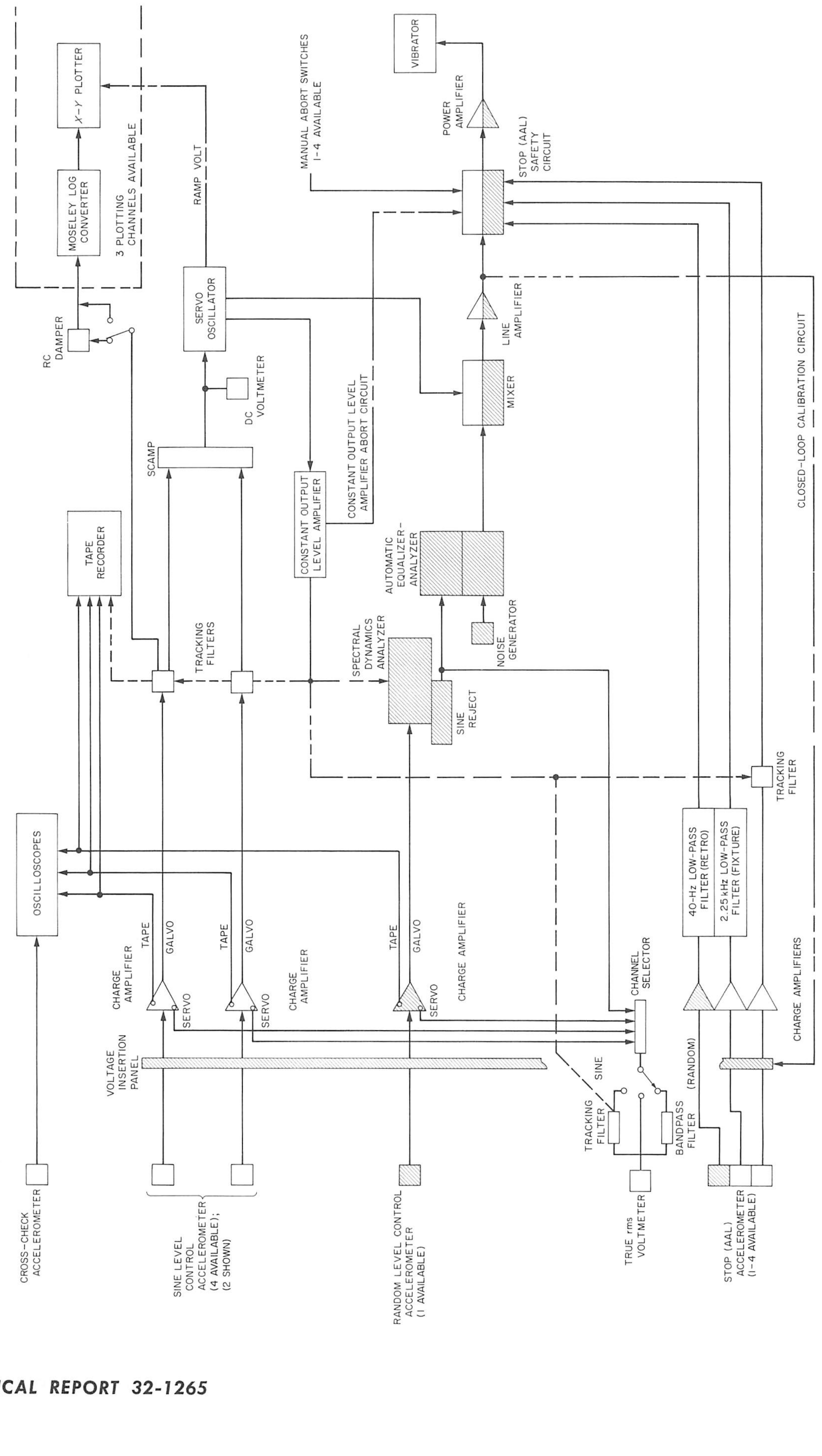


Fig. X-8. Sine–random vibration control block diagram for SC-1 through SC-5

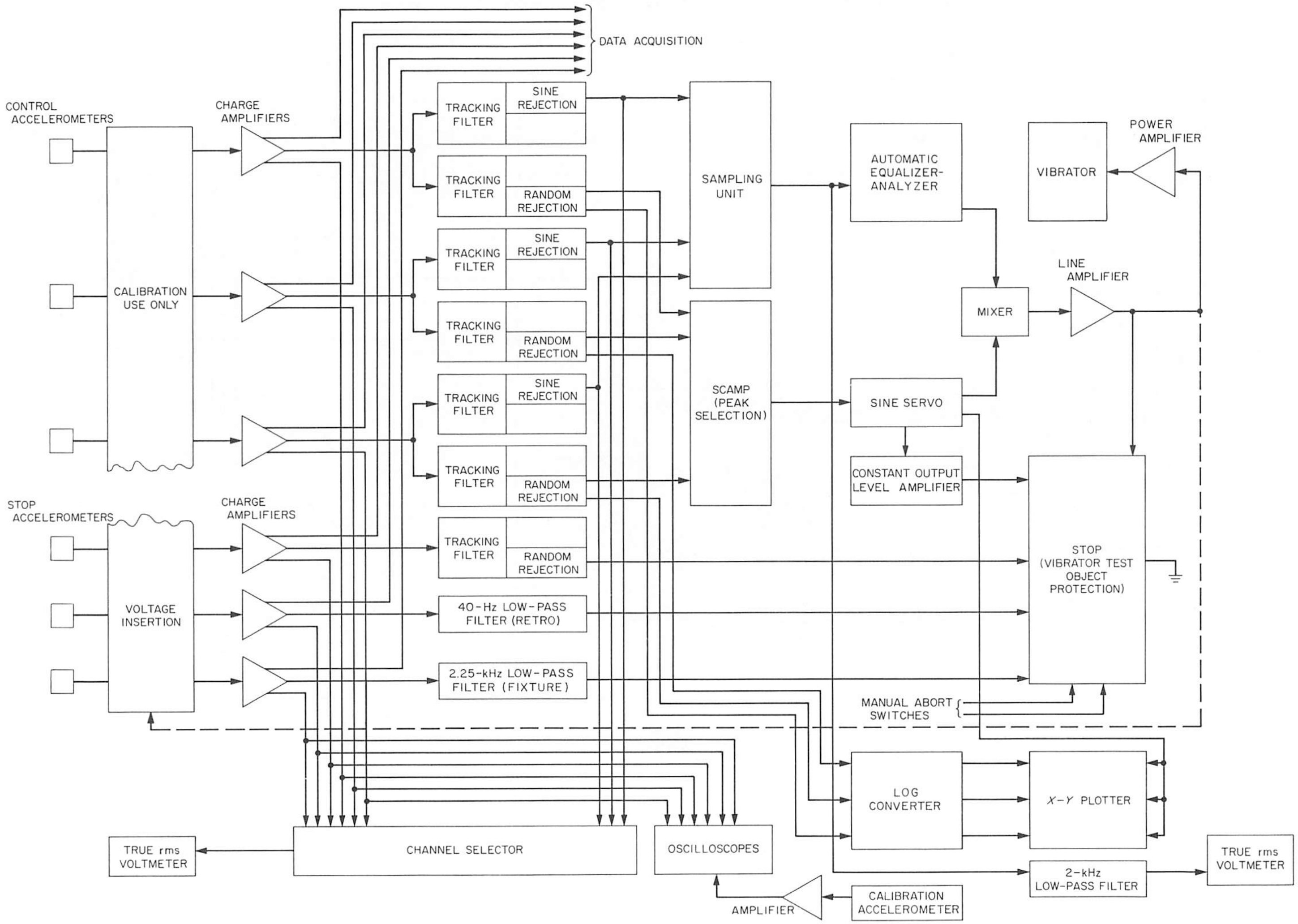


Fig. X-9. Sine–random vibration control block diagram for SC-6 and SC-7

Table X-12. Surveyor acceleration limiting accelerometers

No.	Designation	Location	Filter	Setting	Test	Comment
1	AAL_1–RR	Retro motor in direction of vibration	10-Hz bandwidth tracking	150% of test level	Sine only: 5–100 Hz	All vehicles
2	AAL_2–RR	Retro motor in direction of vibration	40-Hz low pass	2.25 *g*	Z-axis low frequency	SC-2 retest through SC-7
				1.5 *g*	All other tests	
3	AAL_3–F	Fixture near control accelerometers in direction of vibration	10-Hz bandwidth tracking	150% of test level	All sine tests: 5–100 Hz	All vehicles
				3.6 *g* (peak)		
4	AAL_4–F	Fixture near control accelerometers in direction of vibration	2.25-kHz low pass	20 *g* (peak)	All tests	SC-2-retest through SC-7

9. *Flight accelerometers.* Eight flight accelerometers were on board SC-1 though SC-4 during system vibration testing. On SC-1 and SC-2, all accelerometers were recorded during each axis of vibration through spacecraft telemetry. Four of the channels were commutated while the column base accelerometers at each *Centaur* payload adapter attachment point and the one on the flight control sensor group were read continuously. On the SC-3 and SC-4 spacecraft, the testing mode was changed and the commutated channels (RADVS), A/SPP, compartments A and B) could not be read out.

10. *Science payload accelerometers.* On those spacecraft carrying alpha scattering and surface-sampler devices, triaxial accelerometers were mounted and read out during low-frequency tests and during some high-frequency tests. These data were transmitted to the data acquisition area and were of good quality.

c. Test anomalies. On several occasions during the more than two years of flight spacecraft testing, minor anomalies occurred in test procedure, test control, or data analysis. Only one anomaly was of sufficient proportion to require major spacecraft rework and retest: This was a transient input to the SC-2 spacecraft (discussed in the following paragraphs). Table X-13 lists the minor deviations encountered during SC-1 through SC-7 testing.

1. *SC-2 vibrator transient input: cause of anomalous input.* The SC-2 spacecraft was in the final hour of vibration testing at the time of the transient input. The standard procedure preceding the sine–random run of the second lateral axis input had begun. This procedure consisted of (1) increasing the sine level at 100 Hz to the specified value and setting this amplitude into the console, (2) decreasing the sine level to zero, (3) increasing the noise level to -10 db below the specified value as an equalization check, and (4) increasing the noise level to the specified value, superimposing the preset sine amplitude and beginning the sweep. From 20 to 30 sec after the sine signal was reduced and as the noise level was being increased, a transient was introduced to the spacecraft through the vibration system.

Much time and effort was expended by HAC Space Simulation Laboratory personnel to determine the cause of this input, but it was not until 1½ mo later, when a similar transient occurred (with an empty vibrator, fortunately), that the cause was pinpointed.

Additional monitoring equipment in the control circuitry, installed after the SC-2 problem, isolated the problem area to a gassy output tube in the power amplifier. The manufacturer concurred that this tube could have been going gassy during SC-2 testing, during which time a short gas burst yielded an uncontrolled surge of plate current. This surge could have cleaned up the tube enough to prevent recurrence for 1½ mo. The power amplifier tubes were replaced and no further problems were experienced with this equipment since that time (1½ yr).

2. *SC-2 vibrator transient input: response of accelerometers to transient.* The transient input in the direction of vibration resulted in levels greater than 30 *g* zero–peak, and lasted approximately 110 msec. The fixture accelerometers FB 1, FB 2, and FB 3, which essentially measured the transient input to the spacecraft, were calibrated for 15 *g*, 30 *g*, and 30 *g*, respectively; although the shock spectra analysis showed levels up to 55 *g*, no determination could be made of actual levels over 30 *g*.

Table X-13. Minor anomalies occurring during flight spacecraft vibration testing

Spacecraft	Anomalies	Causes	Response of accelerometers	Precaution for future avoidance	Damage to spacecraft
SC-1	1. Improper SCAMP input displayed at vibration console (Z axis)	1. Improper patching of signal through console	1. Control between 60 to 80 Hz was 10% high	1. More careful quality assurance buyoff of patchboard	1. None
	2. Ballantine rms meter behind console gave erroneous readings (Z axis)	2. Bandpass filter was not in meter circuit	2. During equalization rms value of wideband noise rather than 100 to 1500-Hz band was read out	2. Filter installed in meter circuit and checkout incorporated into procedure	2. None
	3. Signal processing failure in compartment B during A axis test. Commutator words lost from engineering signal processor and auxiliary engineering signal processor	3. A piece of mylar dropped from compartment insulation shorting circuitry in the engineering signal processor	3. NA[a]	3. Nylon snood used as inner lining to hold mylar in place; exposed terminals conformally coated	3. None
	4. Excessive helium line leakage under pressure (A axis)	4. Wrong type conical seals installed in lines	4. NA	4. Replaced with proper seals; closer check on hardware	4. None
	5. Six loose bolts discovered in fixture (post A axis)	5. Improper tightening caused bolts to back out	5. Bare fixture vibration before and following retightening yielded essentially the same accelerometer response	5. High-strength bolts to be used in the future. Closer adherence to procedure indicated	5. None
SC-3	Severe phase jitter in transmitter B following equalization for first lateral axis	Microdot connector to the A-4 module loose under compound	NA	Connector was loose before potting. More care in potting operation indicated	None
SC-4	A screw fell from the RADVS antenna area	No screws missing; probably left lying loose from weight, balance, and alignment phase	NA	Closer visual inspection of spacecraft	None
SC-5	Two low-level test runs were made in Z axis (revised specification)	Phase jitter problem in transmitter A	NA	NA	None
SC-6	1. Analysis of average input during lateral axis yields erroneous plot	1. Large variations in levels at individual inputs and sampling rate exceed response time of analyzer system	1. NA	1. Problem cannot be avoided; averaging of individual inputs manually overcame the problem	1. None
	2. Pinpuller on A/SPP backed out under vibration	2. Pin tolerance problem	2. NA	2. Fix made and retest of one low-frequency lateral axis made	2. None

[a]Not applicable.

The accelerometers on the retro motor (RZ 05, RA 06, and RB 07) measured the response of the retro motor to longitudinal, perpendicular to vibration, and axis of vibration excitation, respectively. The response of these three accelerometers was greater than 30 *g*, 9 *g*, and 9 *g*, respectively. Again, no accurate determination of level could be made for RZ 05.

Based on retro motor design constraints, the maximum acceleration level permitted for the retro motor during test was 15 *g*. These transient levels then yielded serious damage potential and accounted for a retro motor support fracture at leg 1. The shock spectra of FB 1, FB 2, and FB 3 revealed that the input transient had energy in broad bands from 20 to 2000 Hz, with peaks at 40, 56, 80, 560, 680, and 1400 Hz.

3. *SC-2 vibrator transient input: precautions taken to safeguard spacecraft.* In addition to the safety circuits described (control method and instrumentation), additional protective devices were installed.

An accelerometer oriented in the vibration axis was attached to the retro motor and was fed into the AAL through a 40-Hz low-pass filter. This was to prevent any high-amplitude, low-frequency input from damaging the retro motor and was set (for SC-2 retesting) at 1.5 *g* (zero–peak).

To prevent damage from high-frequency transients, an accelerometer was installed on the fixture by leg 1 and fed into the AAL through a 2.25-kHz low-pass averaging filter set at 20 *g* (zero–peak). Since this filter was not peak detecting, the amount of protection it afforded was questionable.

4. *SC-2 vibrator transient input: major effects on spacecraft.* There were four major effects on the spacecraft:

(1) Retro motor support system at leg 1 failed.

(2) Bathtub fittings at legs 1 and 2 failed.

(3) Column base fitting inspection indicated holes for attachment to adapter out of alignment from 0.0005 to 0.0015 in.

(4) One solar panel support brace disengaged from the solar panel during the transient, and the threads were found to be elongated.

Further, compartment A and B support tubes were visually inspected and X-rayed; no failures were noted. Torque on all accessible bolts was within specifications; the A/SPP was stepped and functioned properly; all other accessible support structures were visually inspected. Additional low-frequency vibration tests (one at low level, the other at full level) were performed; the agreement in response to the same test before and after the transient appeared to demonstrate that structural integrity of the spacecraft had not been compromised.

d. Spacecraft retests. During the course of the *Surveyor* system level vibration program, four spacecraft underwent retesting. Usually this consisted of a vibration test, or a portion thereof, in only one axis.

On SC-1, a complete one-axis test was performed to subject redesigned flight hardware to Z axis vibration. Special tests and troubleshooting sequences were performed to qualify reworked and upgraded equipment for flight readiness.

On SC-2, to demonstrate confidence in the integrity of the spacecraft and to further investigate a possibility of failure following the transient input, two low-frequency tests in the *B* axis were run. They consisted of 2 octave/min sine sweeps from 5 to 100 Hz at 0.25 *g* zero–peak and 1.33 *g* zero–peak, respectively. Agreement among these runs and the one before the transient helped ensure flight-worthiness.

On SC-3 a short circuit in the ground equipment during initial spacecraft mechanical operations at AFETR resulted in failure of the flight control sensor group and suspected electrical overstressing of other units, necessitating considerable rework; bladders on two oxidizer tanks were also inadvertently collapsed during this period. A decision was made to reverify spacecraft integrity with a complete Z axis vibration test.

On SC-6 a low-frequency test between 5 to 100 Hz in the *A* axis was conducted to verify that the pinpuller pins on the A/SPP did not exceed their motion tolerance and would not back out of position.

e. Comparison of test input levels and specification requirements. Figure X-10 presents typical low-frequency sinusoidal input levels vs specification values for the Z, *A*, and *B* axes, respectively. The data were selected from the SC-7 test for convenience only and, with minor deviations (Table X-10), are representative of the low-frequency control of SC-1 through SC-7. The plots are actually composites of the individual SCAMP inputs and demonstrate the frequency ranges over which the four control accelerometers were effective. Superimposed on

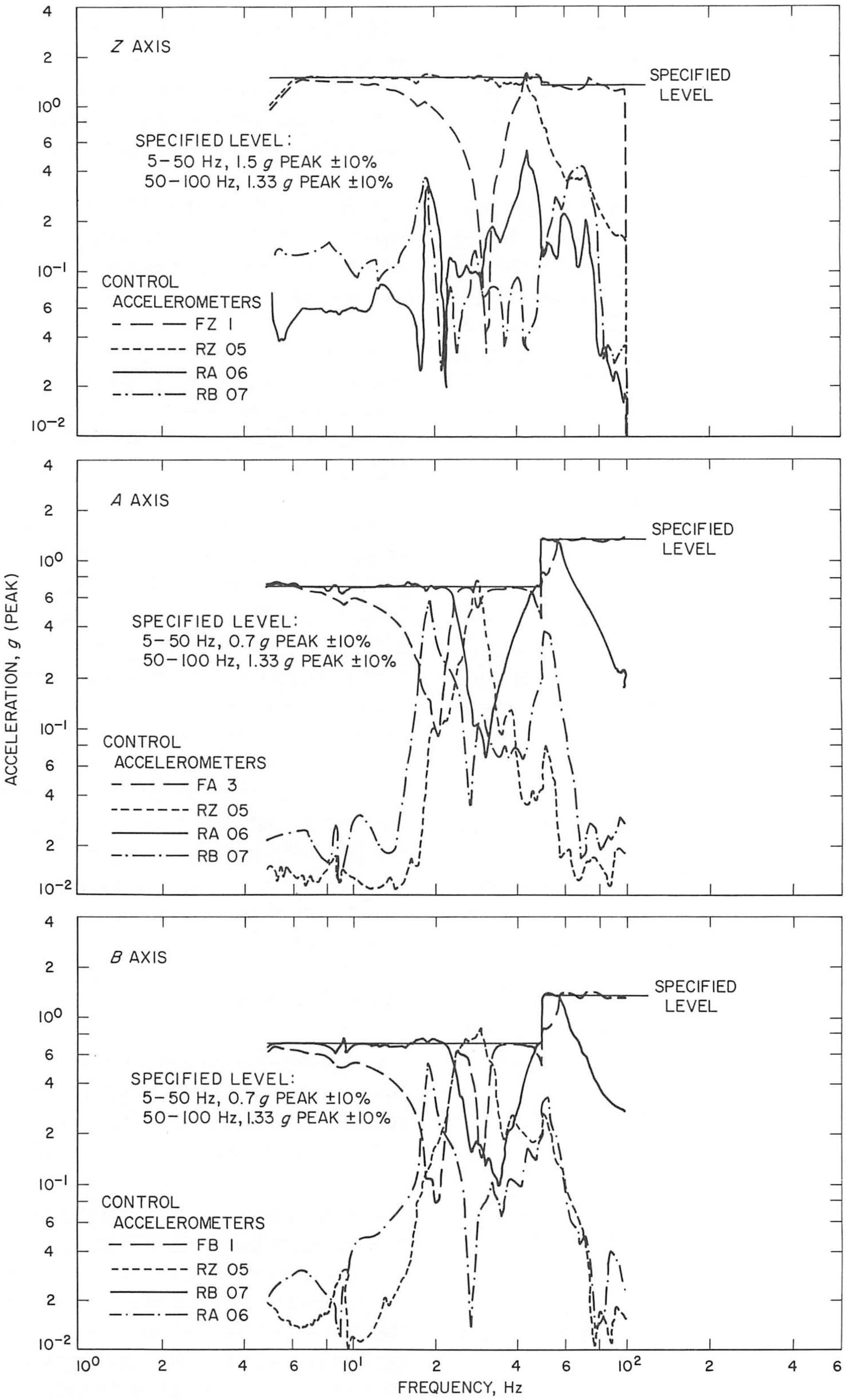


Fig. X-10. SC-7 low-frequency sinusoidal vibration input level vs specification values

each composite is the specified test input level. It should be noted that the minor out-of-specification conditions that did exist occurred at frequencies of changeover from one accelerometer to another and were due to the inability of the SCAMP to respond quickly enough to these changes.

Figure X-11 presents the average and individual random vibration inputs for the Z, *A*, and *B* axes of the SC-7 spacecraft, together with the specified level and tolerances. Because of analysis problems (Table X-13, SC-6, item 1), the average values are not exactly correct, especially in the lateral axes. To obtain true average inputs for the Z axis, for the *A* axis, and for the *B* axis, the individual inputs should be averaged manually.

Typical controlled and uncontrolled random vibration inputs to the SC-1 through SC-5 spacecraft are presented in Fig. X-12. The specified level and tolerance band is included on each. The analyzed outputs of FZ A, FA 3 and FB 1 are typical of the controlled inputs for the Z, *A*, and *B* axes of the SC-1 through SC-5 spacecraft. These three plots could be compared from a control point of view (not from an actual energy input point of view) with the average values, plotted in Fig. X-13, of the final two spacecraft. The response of FB 2 and FB 3 represents a typical input in the lateral axes to the SC-1 through SC-5 spacecraft where the inputs shown were uncontrolled. The same inputs to the lateral axes on SC-7 indicate quite clearly the effect of power averaging on the actual spacecraft input when a comparison is made with the same accelerometer as plotted in Fig. X-11.

Unfortunately, no reduction work on the uncontrolled inputs was performed other than the plots presented here. These inputs were recorded during every test, however, and the raw data are available on tape.

f. Comparison of flight and test vibration levels. The SD-1 and SD-2 dynamic model test flights and the SC-1 through SC-4 flights provided a data base upon which to characterize the 95% vibration environment at the spacecraft/adapter separation plane so that a comparison could be made with the estimated 95% environment, derived from the early *Ranger* flight data, given in Table X-1. The applicable vibration data from these flights were supplied by a series of accelerometers mounted near the spacecraft/adapter separation plane. On SD-1 and SD-2, there were three longitudinal axis accelerometers, one at each column base; a tangential accelerometer on the column base of leg 3; and a tangential and a radial accelerometer on the column base of leg 2. For SC-1 through SC-4, there were the three longitudinal axis accelerometers at the column bases. On SD-1 and SD-2, the accelerometers were monitored continuously; however, on SC-1 through SC-4, the accelerometer near leg 1 was monitored continuously, but the accelerometers near legs 2 and 3 were commutated with two other accelerometers located elsewhere, at the rate of one 0.533-sec reading every 2.7 sec, and telemetered on one channel. The data from the accelerometers were reduced to acceleration spectral densities (ASDs) of the random vibration and shock spectra of the measured transient events.

The 95% random vibration environment derived from this reduced data was the ninety-fifth percentile, 90% confidence level ASD as computed from the 15 separate ASDs of the output of the applicable longitudinal axis accelerometers. (From the six flights, there should have been eighteen separate measurements from the three accelerometers; on SC-2 only one applicable accelerometer was operable, and on SC-3 only two applicable accelerometers were operable.) Because of the small sample size, the 95% sinusoidal environment was obtained by adding 6 db to the steady-state sine equivalent of the maximum envelope of the shock spectra of the measured flight transients for 5% of critical damping. The comparisons of the 95% measured flight separation plane environment to the estimated 95% separation plane environment are shown in Figs. X-13 through X-16.

Figure X-13 contains a comparison between the 95% flight random vibration environment at the spacecraft/adapter separation plane and the estimated 95% random vibration environment of Table X-1. Because of the frequency response characteristics of the data channels, the 95% flight environment was considered valid only up to 800 Hz. As can be seen, the measured environment was less than that predicted, especially above 400 Hz. In specifying a test level based on the measured data, the levels would be increased at least 1.5 db as a test tolerance, making the comparison much closer. Considering the basis of the original estimate, the agreement between the two environments is quite good and, as was desired, test conservatism was obtained in the original estimate. The environment from the flight data is assumed to be for both longitudinal and lateral axes for the following two reasons: (1) the omnidirectional aspect of high-frequency vibration in complex structures remote from the source, i.e., vibration response is more dependent upon stiffness than upon orientation; (2) the three laterally directed accelerometers on SD-2 were frequency response limited to 160, 200, and 330 Hz, respectively, by filtering with Inter-Range Instrument Group bandwidth filters.

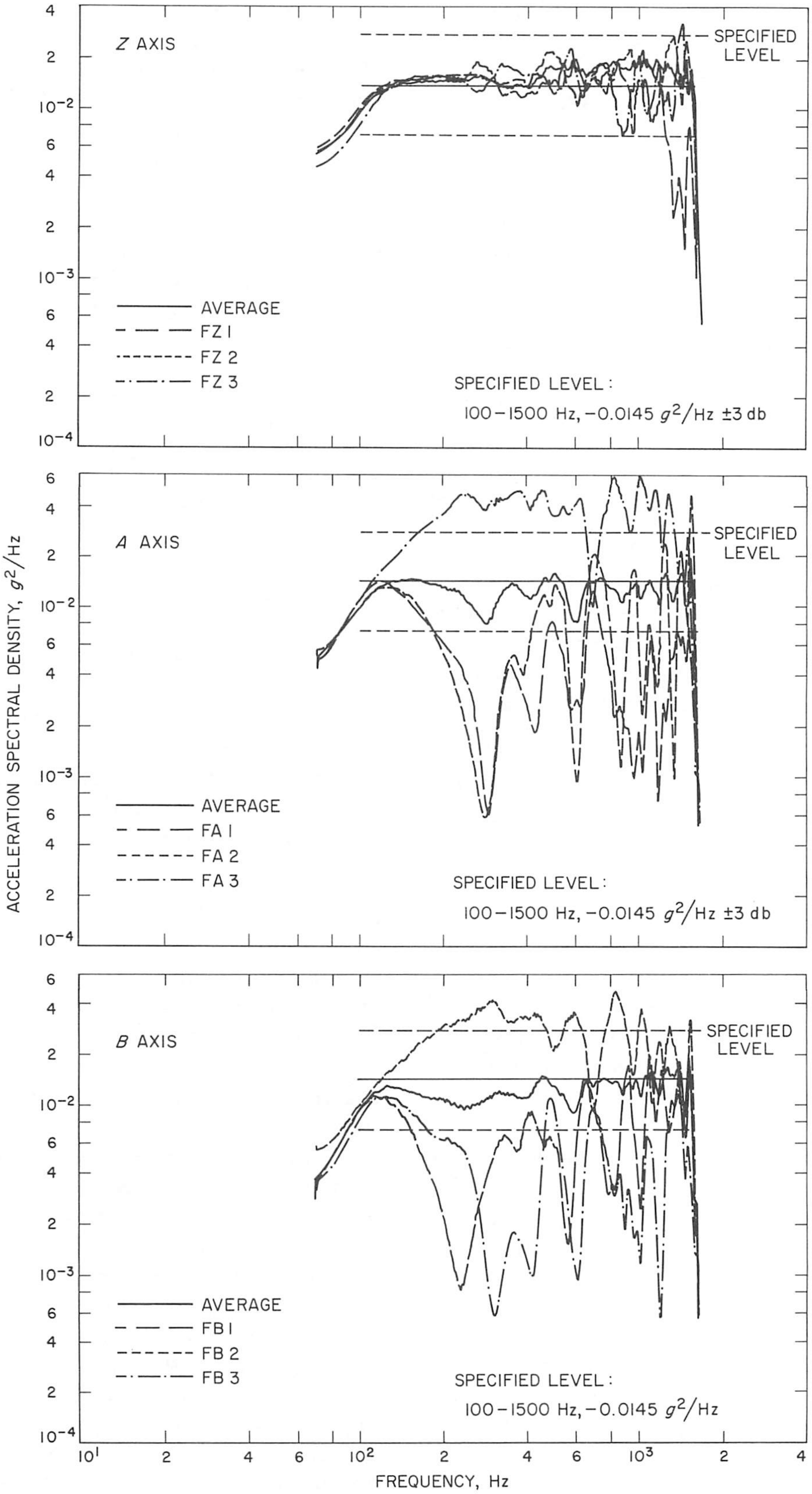


Fig. X-11. SC-7 random vibration test input level vs specification values

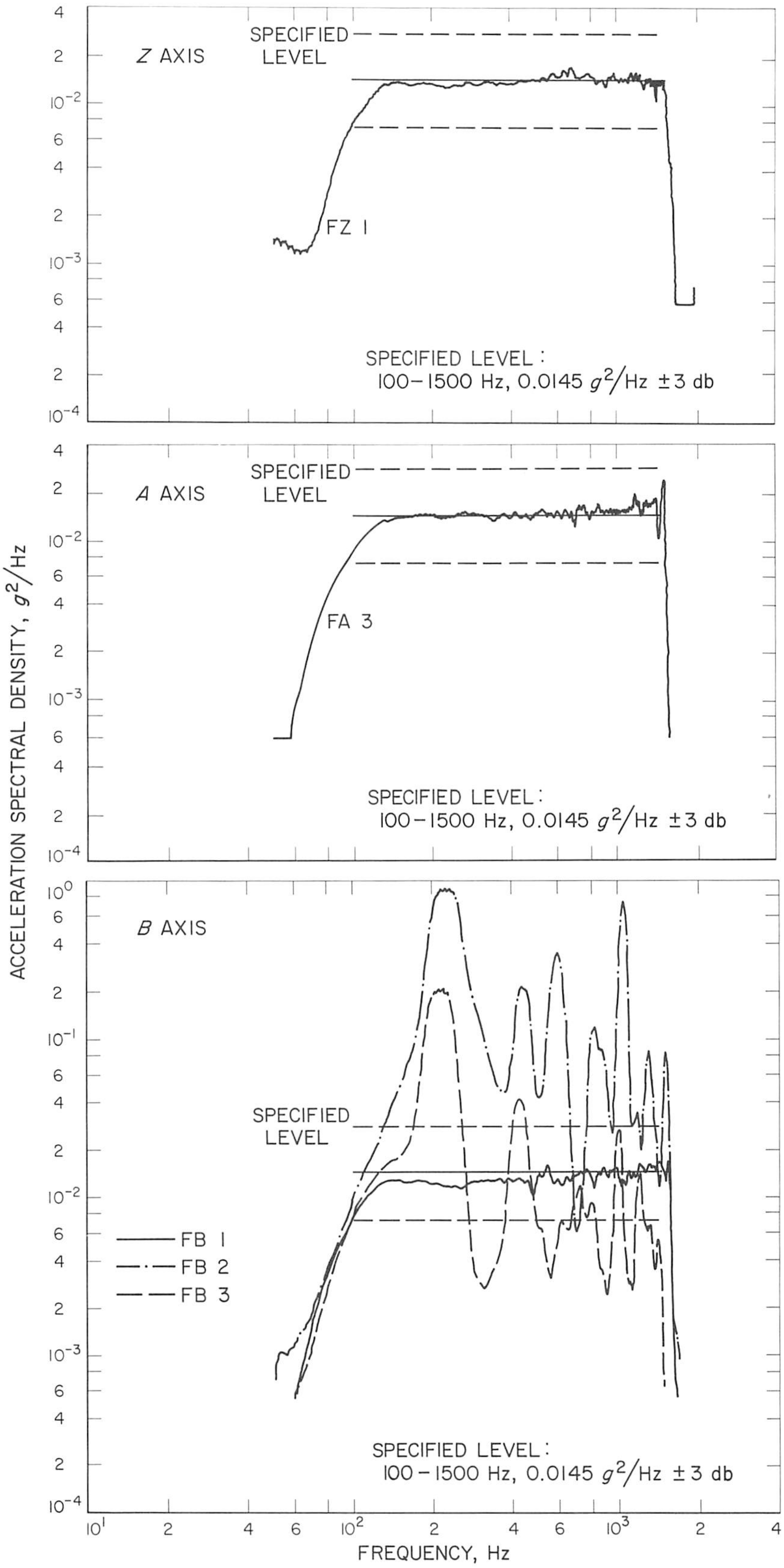


Fig. X-12. SC-3 random vibration input level vs specification values

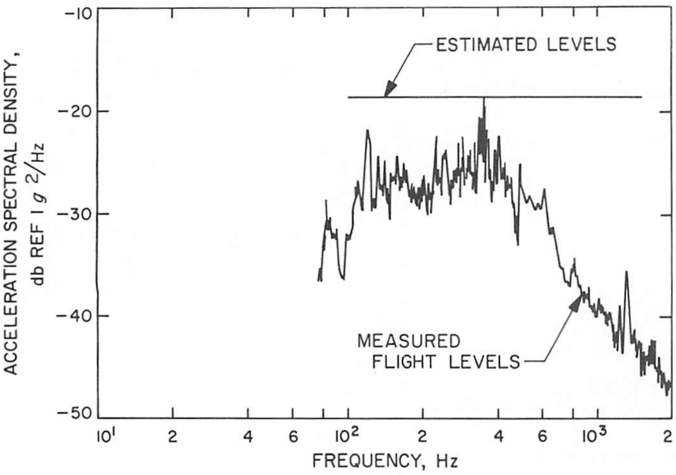


Fig. X-13. Comparison of the 95% flight random vibration environment and the estimated random vibration environment for both longitudinal and lateral axes

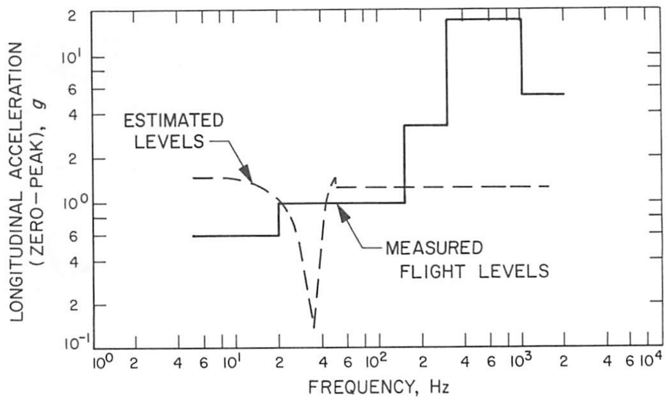


Fig. X-14. Comparison of the 95% flight equivalent sinusoidal vibration environment and the estimated sinusoidal vibration environment for the longitudinal axis

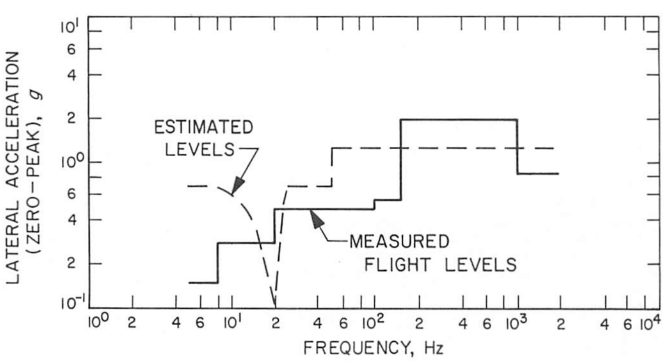


Fig. X-15. Comparison of the 95% flight equivalent sinusoidal vibration environment and the estimated sinusoidal vibration environment for the lateral axes

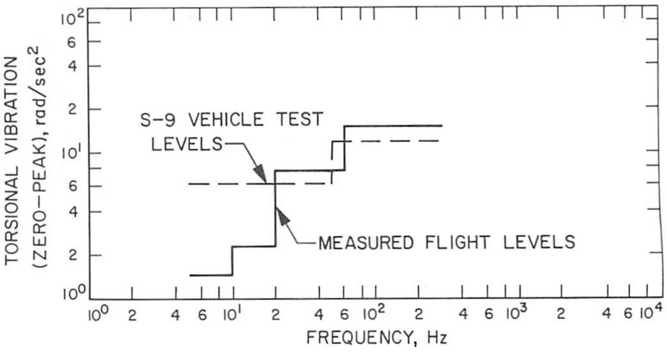


Fig. X-16. Comparison of the 95% flight equivalent torsional sinusoidal vibration environment and the specified torsional sinusoidal vibration environment for the S-9 structural test vehicle

If the random vibration output of these lateral axis accelerometers is examined by comparing ASD plots, it can be seen that the high-frequency random vibration levels increase toward the longitudinal axis levels as the frequency response limit increases, indicating a high-frequency lateral axis random vibration environment greater than indicated by the lateral axis ASDs.

Figure X-14 compares the 95% flight longitudinal axis equivalent sinusoidal vibration environment at the spacecraft/adapter separation plane and the estimated 95% sinusoidal vibration environment of Table X-1. The 95% flight environment was considered valid only up to 800 Hz because of the frequency response characteristics of the data channels. The dip in the separation plane environment in the 10- to 50-Hz range represents the attenuation of the separation plane vibration levels during spacecraft testing when the retro motor acceleration levels reach those specified for the separation plane and control the vibration input (Subsection G-1). The data for this part of the plot were taken from the SC-1 vibration test. The original estimate is seen to have been conservative except in the ranges of 23–40 Hz and above 150 Hz. The high-level equivalent sinusoid in the high frequencies was due to the transient flight events termed insulation panel jettison and *Atlas/Centaur* separation. These were both pyrotechnic events generally characterized by high-level, high-frequency response throughout a structure. Main structural members are generally not adversely affected by high-frequency vibration because they are designed for the high stress levels of low-frequency, large-amplitude response. Spacecraft functional equipment, both mechanical and electrical, are subject to high-frequency vibration failures because the components that make up this equipment have many resonances in the high-frequency region comprised of many different modal configurations. Although the spacecraft was not subjected to any ground vibration testing to ensure its ability to withstand the magnitude of the high-level, high-frequency flight vibration environment or the environment in the region of 23–40 Hz, no failures during flight were attributable to the vibration environment. Also, the multiple axis ground vibration tests, both TA and FA, at the unit and system level, demonstrated a margin for fatigue failures that would not have been exceeded during one flight of a spacecraft, even with the inclusion of the high-level transients and the 23- to 40-Hz environment.

Figure X-15 compares the 95% flight lateral axis equivalent sinusoidal vibration environment at the spacecraft separation plane and the estimated 95% vibration environment of Table X-1. Again, the dip in the separation plane environment in the 7- to 25-Hz range is the attenuation of the separation plane vibration levels during lateral axis spacecraft testing due to retro motor control. Here again, the original estimate showed various degrees of conservatism up to 150 Hz but the estimate was less than the measured values from 17 to 25 Hz and from 150 to 1000 Hz. The comments and discussion included in the comparison of the longitudinal axis measured and estimated 95% environment apply here also.

The only additional comment concerns the derivation of the lateral axis equivalent sinuoid from the flight data. Since the maximum lateral axis vibration is the vector sum of orthogonal components, the X and Y axis data were vectorially combined to form the maximum. However, since the X axis data were determined by the manipulation of data from three accelerometers, erroneous phasing of the three channels could have caused errors in the magnitude of the X axis vibration at higher frequencies. The X axis data were assumed to be valid only to 100 Hz, therefore, above 100 Hz, the vector sum was assumed to be 1.414 times the Y axis value. A factor of 1.414 was chosen because below 100 Hz the X and Y axis vibration magnitudes were approximately equal. However, the 95% flight environment was considered valid only to 300 Hz because of the frequency response characteristics of the data channels.

Figure X-16 compares the 95% flight torsional equivalent sinusoidal vibration environment and the torsional vibration environment specified for the S-9 structural test vehicle torsional test. The torsional environment specified for the S-9 test was the torsional equivalent, at the radius of the auxiliary battery, of the lateral axis sinusoidal vibration environment of Table X-1. Also specified for the S-9 test was a series of four torsional pulses (Subsection B-4). However, these pulses provide a level of excitation considerably below the specified sinusoidal environment and are not included in the comparison. As can be seen in Fig. X-16, the agreement was particularly good above 20 Hz. Below 30 Hz, the original estimate was conservative. Since three accelerometers were also used to determine the flight torsional environment, accuracy above 100 Hz is questionable. The environment was extended to 300 Hz to be consistent with the S-9 levels, but above 100 Hz, the torsional environment was made from one tangential accelerometer. Because this accelerometer would be indicating either torsional or lateral vibration, the environment given for above 100 Hz is not considered reliable.

During flight, the random vibration and the transient events upon which the equivalent sinusoidal vibration levels were based occurred separately except for the *Atlas/Centaur* launcher release transient. The average duration of the random vibration during flight for acceleration greater than 10 db down from the peak level was 40 sec. This compares with a test duration of 3.9 min for the random vibration when combined with a swept sine. The sweep rate for the equivalent sinusoid, based on the combined flight transient durations, was 2 octaves/min for an up and down sweep, which is the same as that used for *Surveyor* testing. The sweep rate computed from the combined durations of the transient events was based on a model that provides excitation in a reasonance band equal to the service use duration.

2. Solar-Thermal-Vacuum Tests

Each flight spacecraft was subjected to STV testing to verify its functional integrity in the solar radiation, cold dark space, vacuum environment. Three phases (A, B, and C) comprised the STV test program.

a. Test requirements. The STV test program required testing to be performed at a pressure of 5×10^{-6} torr or less. It was also required that the chamber wall and floor temperatures be maintained at $-300°F$ or less with thermal shrouds installed. The STV tests were conducted at three levels of solar intensity, identified as test phases A, B, and C. The specified solar radiation intensities were all based on a solar constant defined as 130 W/ft^2 at the test plane.

1. Phase A. The solar intensity for mission sequences performed in this test phase was 1.2 solar constant, as measured with a black plate detector. This test was therefore referred to as the high sun test. The spacecraft mission sequences performed in this test phase included all operations through lunar landing. The launch-to-landing operations period of the test simulated the duration of the flight phase of an actual *Surveyor* mission. A postlanding operations period normally about 4 hr in duration followed the transit portion in the phase A test.

2. Phase B. The second mission sequence was performed with a low sun solar radiation of 0.87 solar constant (phase B) except that for SC-2, the low sun test was run before the high sun test. Spacecraft functional test requirements again included transit operation simulating the actual *Surveyor* mission duration and postlanding operation.

3. Phase C. Following the two full length transit mission sequences of phases A and B, the spacecraft was operated through an abbreviated nominal sun mission sequence in phase C with a solar radiation of 1 solar constant. The total duration of phase C was usually about 32 hr. The cruise mode of the test sequence was shortened, but the launch through midcourse and terminal-descent modes were of normal mission duration. Phase C of the specified STV test requirements was performed on the first three of the seven spacecraft.

Temperature limits were established for all spacecraft flight temperature sensors. This temperature limit information was used by the spacecraft test team throughout the various test operations. Assessments of spacecraft temperatures were performed as part of the test procedures or whenever the test conductor and/or senior thermal engineer thought it necessary. The temperature limits were in the following four ranges:

(1) The operating range lay between the minimum and maximum temperatures the unit could withstand when operating.

(2) The nonoperating range lay between the minimum and maximum temperatures a unit could withstand in the nonoperating mode. However, it was to function within specification limits after returning to the operating range after exposure to the nonoperating range extreme temperatures.

(3) The operating survival range established the minimum and maximum temperatures a unit could survive when operating without damage to itself or other units. The unit may or may not have functioned within specification limits.

(4) The nonoperating survival range defined the minimum and maximum temperatures a unit was required to withstand in the nonoperating mode without damage to itself or other units. When the sensor location temperatures were brought within the operating range, the unit was required to function, but may or may not have been within specification limits.

b. Spacecraft configurations. The spacecraft were in a transit flight configuration except for the following differences existing between the spacecraft STV configuration and the flight configuration:

(1) No live pyrotechnic devices were on the spacecraft during STV tests.

(2) Solar panel and planar array antenna were removed. These units were removed as a result of shadowing problems with imperfectly collimated energy. Shadow shrouds cooled with liquid nitrogen were judiciously placed over units normally shadowed by the solar panel during an actual mission.

(3) Landing gear pads were removed to avoid interference with the chamber wall shrouds when the legs were extended.

(4) Pressurized shock absorbers were replaced with dummy struts for safety reasons.

(5) Live retro motor was replaced with an inert motor.

c. Summary of test results. Control item thermal vacuum test requirements were, in general, compatible with the control item temperatures measured during Spacecraft System STV testing. The STV temperatures were generally conservative when compared to the flight temperatures. A comparison of all unit and system level test temperatures with flight and lunar temperatures was made after the SC-1 mission.

The retest history of the STV FA tests was much more extensive than that of the vibration tests. Approximately 14 phases of the flight spacecraft STV testing were either aborted or rerun after necessary spacecraft rework was performed.

3. Electromagnetic Interference Tests

Electromagnetic interface testing of the flight spacecraft was performed in two phases as part of the flight spacecraft ambient testing at HAC. The first of these two phases was mission sequence III of the mission sequence/

electromagnetic interference test (MS/EMI). The purpose of this test was to verify the compatibility of the Spacecraft System with the radio frequency radiation environment expected during system testing and launch operations at AFETR. The second test phase was the spacecraft transmitter/receiver compatibility test. The purpose of this test was to ensure that neither of the spacecraft transmitters generated spurious emissions at the operating frequency of the spacecraft receivers. Discussed below are the requirements for each of the two test phases and a summary description and results of each test.

a. Mission sequence/electromagnetic interference sequence III. The MS/EMI sequence III test consisted of immersing each flight spacecraft in a radio frequency electromagnetic field simulating that expected at pad 36, AFETR, and performing an air-link controlled mission seqence. The portion of the mission sequence III involving EMI was the initial launch-to-injection phase.

1. Test requirements. The RF radiation environment consisted of radio frequency emissions from discreet sources at AFETR (such as radars and guidance transmitters) as well as discreet sources located on board the *Atlas/Centaur* launch vehicle (such as telemetry transmitters and radar transponders). In addition, an RF susceptibility test level was included over a frequency band of 0.5 MHz to 1050 MHz. Table X-14 contains a summary of the RF simulation sources for the entire test phase.

2. Test description and summary of test results. The simulation equipment consisted of a set of signal generators and antennas arranged to impress the required electromagnetic power densities on the spacecraft. Simulation of three of the AFETR sources (the AN/FPS-8, MOD II, and MOD IV radars) included pulse modulation and were based on peak power. All other sources were simulated at average power levels by unmodulated signal generators. The test levels from 0.5 MHz to 1050 MHz were generated with mechanically swept signal generators. The spacecraft configuration for mission sequence III was as near flight configuration as was practicable except that thermal compartment covers were not installed. Live pyrotechnics were installed.

In general, the flight spacecraft were found to be not susceptible to the RF radiation environment specified, but some problems were encountered:

(1) The SC-3 receiver B failed the tracking range/tracking rate test during a system readiness test in preparation for mission sequence III. The cause

Table X-14. The RF simulation sources for the MS/EMI test phase

Frequency, MHz	Source simulation	Power density, W/m²	Comment
0.5–50.0	HAC specification	0.025 av	Frequency swept
65–500	HAC specification	0.025 av	Frequency swept
260	*Centaur* telemetry 225.7 MHz *Atlas* telemetry 229.9 MHz *Atlas* telemetry 232.4 MHz *Centaur* telemetry 237.8 MHz *Centaur* telemetry 247.8 MHz *Centaur* telemetry 259.7 MHz	0.256 av	Single fixed frequency to simulate all telepacs
450–1050	HAC specification	0.015 av	Frequency swept
1300	AN/FPS-8 radar	2.12 peak 2.86 (10^{-3}) av	Pulsed simulation source
2750	MOD II radar	1.39 peak 2.36 (10^{-3})	Pulsed simulation source
5060	AZUZA Mark II	0.100 av	Continuous wave source
5600	AN/FPQ-6 radar AN/FPS-16 radar AN/TPQ-18 radar C-band radar transponder	0.170 av	Four pulsed sources simulated at average power by two continuous wave sources (see 5900 MHz below)
5900	AN/FPQ-6 radar AN/FPS-16 radar AN/TPQ-18 radar C-band radar transponder	0.011 av	See 5600 MHz above
9050	MOD IV radar	47.8 peak 47.8 (10^{-3}) av	Pulsed simulation source

of this failure was determined to be a self-sustained spurious emission near the receiver passband generated in the traveling wave tube amplifier of the MOD II radar simulation generator. This instrumentation problem was corrected with the addition of a high pass filter at the output of the traveling-wave tube amplifier. The same problem occurred during SC-4 testing when this high-pass filter was deleted from the simulation generator assembly.

(2) Certain low-level current telemetry channels were affected by the EMI environment. This problem was first noted on SC-6 when the 5- to 50-MHz simulation generator was incorrectly calibrated at a high output level. The excessive output of the EMI generator made the anomaly very apparent to the system test equipment assemblies operator. A subsequent check of recorded data from earlier spacecraft indicated that the problem may have occurred before, but at levels normally ascribed to system noise. The problem was found to be associated with sweeping of the generator. No discrete frequency of susceptibility could be determined. Recalibration of the interfering signal generator eliminated the problem.

(3) The alpha scattering instrument showed a definite sensitivity to the specified RF environment. The incident RF was detected and appeared on the instrument sensor head detector guard (AS-5) as a DC bias of about 4.08 V (normal is 20 mV). This alpha scattering instrument anomaly was found to be due to incident RF radiation at frequencies of 204, 260, 1300, and 5600 MHz. Because the alpha scattering instrument would not be active during launch operations, the degradation was considered only a nuisance. The problem was listed on the spacecraft signature list. No further investigation was attempted. The alpha scattering instrument was also found to be susceptible to EMI from the Applications Technology Satellite, which was assembled and tested in the same facility as the *Surveyor.*

(4) Excessive phase jitter was noted on transponders A and B of SC-5. The transponders were found to be sensitive to RF radiation at 47.5, 642, 702, and 732 MHz, when operated at an uplink signal strength of −115 dbm. At an uplink signal strength of −80 dbm, which approximates the normal launch pad signal, the phase jitter dropped to normal levels. The transponder did not lose phase lock because of this anomaly.

b. Transmitter/receiver compatibility test. The transmitter/receiver compatibility test was essentially a low-signal-level spectrum measurement at the spacecraft receiver frequency for each of the two spacecraft transmitters. The JPL EMI group performed this test for HAC because of the unavailability of the required equipment at HAC.

The specified test level was based on a calculation of the worst-case uplink signal strength at the receiver input at lunar encounter distances. The specified maximum for spurious emissions allowable from the transmitter was set at 3 db below the calculated worst-case uplink signal strength.

Instrumentation for the transmitter/receiver compatibility test consisted of an S-band noise meter supplied and operated by the JPL EMI group. This special noise meter had a measuring bandwidth capability of 100 Hz and a sensitivity of −150 dbm per 100 Hz. The measurement was made at the antenna port of the diplexer associated with either omniantenna while each data link transmitter was switched to the chosen omniantenna for the measurement.

Of the 16 data link transmitters tested, only one transmitter was found to generate spurious emissions in excess of the specified level. This transmitter (serial number 13) was originally transmitter B of SC-1.

Both of the original SC-1 transmitters were later replaced. The excessive emission consisted of broadband noise for a period of approximately 10 min immediately following turnon of the high power mode.

During the transmitter testing of SC-2, some desensitization of the noise meter was experienced. The desensitization occurred for both transmitters in both the high and low power modes. At that time the desensitization was ascribed to excessive gain in the noise meter due to recent changes in the meter. Similar difficulties were experienced with the SC-4 test.

During SC-5 testing, desensitization of the noise meter was again experienced. During the test, apparent discreet emissions were detected within the specified frequency range of the test. In subsequent diagnostic testing, it was found that the noise was not generated by the spacecraft transmitters, but by the spacecraft receiver associated with the diplexer being used for test access. These emissions were leakage from the local oscillator multiplier chain through the first mixer of the command receiver. Four receivers were examined at AFETR (two SC-5 receivers and two spares); one SC-5 receiver and one spare were capable of desensitizing the noise meter. A subsequent spectrum analysis of the spare receiver revealed a discreet spurious emission at 2110 MHz, within the RF bandpass of the noise meter. After considerable retesting of the SC-5 data link system it was determined that the data link was not degraded by this anomalous condition. It is probable that a similar situation explains the noise meter desensitization experienced earlier on SC-2 and SC-4.

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XI. Spacecraft System Testing and Launch Operations

A. Introduction

This section summarizes the system level test and operations activities performed on *Surveyor* spacecraft. These include (1) flight spacecraft, (2) special test vehicles (dynamic models and T-2/T-2N aluminum test structure spacecraft), and (3) the type approval spacecraft, T-21.

Included in this section is a summary of the Test and Operations Plans (TOP), which describes the test phases to which the flight spacecraft were exposed, and a description of the techniques, equipment, and organizational structure used to perform the tests and evaluate spacecraft performance during the test program. Refer to Section X for a description of spacecraft environmental test requirements and to Section IX for a description of spacecraft aerospace ground equipment.

1. General Test Objectives and Information

There are three types of system-level testing described in this report:

(1) Design development tests that were performed during the hardware development period to provide data necessary for final design definition. System-level testing of this type was the purpose of the dynamic model testing and the T-2/T-2N program.

(2) Type approval tests that were conducted with prototype hardware to demonstrate that the final design satisfies the functional and environmental performance requirements in excess of those expected in an actual flight. The T-21 was the type approval spacecraft.

(3) Flight acceptance tests that were conducted on flight spacecraft to demonstrate that after assembly spacecraft would perform as intended during the mission, and to obtain preflight calibration data. All seven flight spacecraft were exposed to these tests.

These tests were also conducted on components, modules, units or subassemblies, subsystems, and on special test assemblies. These tests and test results are covered in other sections of the overall report. Only system-level testing on complete spacecraft is described in this section.

The flight spacecraft were tested in accordance with the TOP, described in Subsection B. The testing was conducted at three locations: (1) initially at HAC, El Segundo, for assembly and environmental tests until the spacecraft was ready for shipment, (2) then at GD/C, San Diego, at the combined systems test stand (CSTS) for compatibility tests with the *Atlas* and *Centaur* launch vehicles, and (3) final tests at the Air Force Eastern Test Range (AFETR) for final preflight checks and operations prior to launch. Because of schedule problems, only *Surveyor* spacecraft SC-1, SC-4, and SC-5 were tested at the CSTS in GD/C, San Diego.

The T-21 type approval spacecraft was tested in a similar manner as the flight spacecraft with some additional tests. These included compatibility tests with the Deep Space Station (DSS) at Goldstone, Calif., and a series of propulsion abort exercises at AFETR. These and other type approval tests are described in Subsection D, prototype vehicle tests.

A summary of the system test experience of each flight spacecraft is presented in Subsection D. Figure XI-28 of that subsection shows the actual schedule history of the flight spacecraft as well as T-21. All flight spacecraft were successfully launched after completion of prelaunch tests and operations.

2. Test Management and Organization

The JPL System Test and Launch Operations (ST&LO) section was responsible for monitoring and controlling the tests and operations on the T-21 and flight spacecraft.

A typical organization for each spacecraft is shown in Fig. XI-1. The JPL spacecraft coordinator interfaced with the HAC spacecraft manager, and the JPL spacecraft coordinator interfaced with the HAC test director. The JPL subsystems engineers on the test team provided continuous coverage for all spacecraft test and operations.

All tests were supported by the computer data system (CDS) which was a part of the JPL ST&LO section. A description of the CDS, also known as the Spacecraft Checkout Computer Facility (SCCF), is provided in Subsection A-3.

The test facilities group within the ST&LO section consisting of a space simulator engineer, a vibration facility engineer and a system test equipment assembly (STEA), and property engineer provided support in the maintenance and operation of the various spacecraft test facilities.

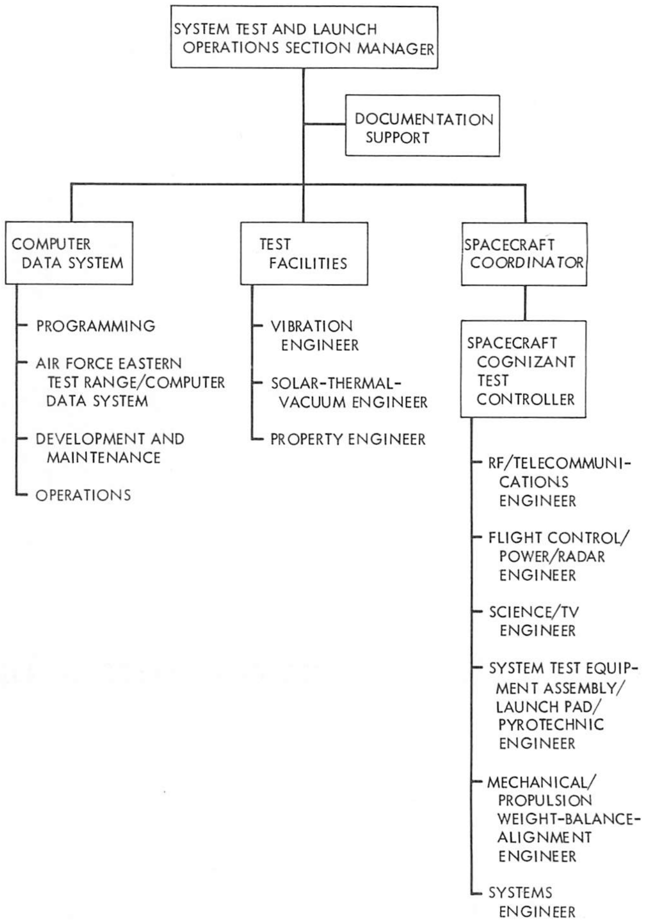


Fig. XI-1. JPL system test and launch operations organization block diagram

At AFETR all operations prior to launch pad activities were conducted in the JPL Spacecraft Checkout Facility (SCF) and the JPL Explosive Safe Facility (ESF). Provision was provided in these facilities for the HAC permanent contingent at AFETR and for the JPL and HAC test team personnel who arrived with the spacecraft. Interface with the AFETR safety organization was also provided by the JPL AFETR organization.

The mission control center, located at JPL Building AO, was the area in which the mission director was located during countdown. Assisting the mission director were the JPL spacecraft manager and the HAC and JPL launch and test team organization. After launch, mission direction as well as control and monitoring of the spacecraft reverted to the flight operations team set up in the Space Flight Operations Facility (SFOF) at JPL Pasadena.

All HAC *Surveyor* functional organizations, as well as HAC supporting activities outside the project, participated in system test and launch operations. Each spacecraft test program, however, was conducted primarily by a separate spacecraft test team, headed by a spacecraft manager, and comprised of individuals assigned essentially full time to that team. Typical team structures during preshipment testing and during AFETR operations are shown in Fig. XI-2.

Three primary functions were represented on each team: spacecraft test operations, facility or base operations, and engineering. Responsibility for conduct of the test resided with the test team headed by the spacecraft test director. Mechanical preparations of the spacecraft were under the direction of the senior vehicle engineer, and the evaluation and acceptability of spacecraft performance were the responsibility of engineering, headed by the senior systems engineer.

3. Spacecraft Checkout Computer Facility

The SCCF was the primary means of monitoring spacecraft performance during all system testing at the El Segundo, CSTS, and AFETR locations. The computer-controlled system received, processed, stored, and displayed spacecraft test information in real-time. It received both hardline direct access and telemetry data from the spacecraft and also provided displays on high-speed line printers and on character printers, incremental plotters, and digital readouts.

One SCF was maintained at HAC, El Segundo, and another at AFETR. The El Segundo facility included two Univac 1219 computers and the AFETR facility utilized one Univac 1218 computer. Each computer was able to support simultaneously two *Surveyor* spacecraft operations.

Input data into the computer data system consisted of the spacecraft pulse-code modulated (PCM) commands, hardline electrical power, and hardline closed-loop flight-control data. The spacecraft PCM and hardline data were suppressed so only significant changes were displayed. Suppression methods consisted of averaging, a suppression limit test, and an alarm limit test. In suppression, data were displayed only if a parameter changed by a prespecified increment. Alarm testing consisted of displaying data only if a fixed upper or lower limit was exceeded.

All data processing results, spacecraft status, and commands were displayed on the line printer, along with time. Specific subsystem data were displayed on six character printers assigned to different subsystems. All engineering data were converted into engineering units for display; identification of data was by nine-character identifiers.

A functional diagram of the SCF is shown in Fig. XI-3. The SCF comprised a central computer complex on local site, and data-input on remote sites. The remote sites received and prepared spacecraft data for subsequent processing, storage, and display at the local site.

Equipment at the local site included a computer, card reader console, input/output (I/O) controller, telemetry and command assemblers, tape transport cabinets, and several data display devices. Remote-site equipment included a data input subsystem, a step number box and line printer, character printers, and a telemetry input subsystem (Fig. XI-4).

B. System Test and Operations Plan Summary

The flight acceptance (FA) tests for the *Surveyor* flight spacecraft were performed in accordance with TOP. This document, written by HAC and approved by JPL, was the top-level document for system-level testing and formed a basis for the detailed procedures and work plans needed to bring the spacecraft subsystems together in an integrated manner. The test phases described in this section and illustrated in the flow diagram in Fig. XI-5 were those necessary to launch the spacecraft with the required level of confidence that the mission objectives would be attained.

1. Initial System Checkout Phase

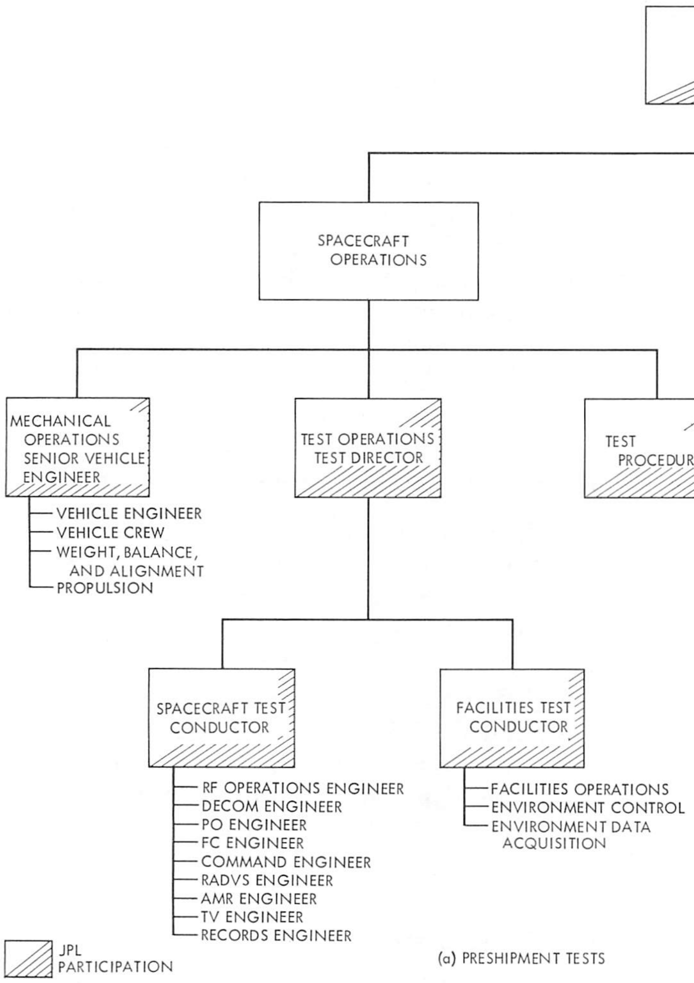
a. Objectives. The integrated system checkout (ISCO) test objectives were as follows:

(1) Perform calibration of engineering and scientific telemetry data channels.

(2) Perform tests to verify compatibility of each subsystem with the spacecraft power subsystem.

(3) Perform subsystem integration tests between all subsystems and the telecommunications (TCM) subsystem.

(4) Perform power and grounding checks.



RF = RADIO FREQUENCY
PO = POWER
FC = FLIGHT CONTROL
RADVS = RADAR ALTIMETER AND DOPPLER VELOCITY SENSOR
AMR = ALTITUDE MARKING RADAR
SP = SIGNAL PROCESSOR
SCF/LP = SPACECRAFT CHECKOUT FACILITY/LAUNCH PLATFORM
ESF = EXPLOSIVE SAFE FACILITY
AFETR = AIR FORCE EASTERN TEST RANGE
EP = ENGINEERING PROCESSOR
DECOM = DECOMMUTATOR

(a) PRESHIPMENT TESTS

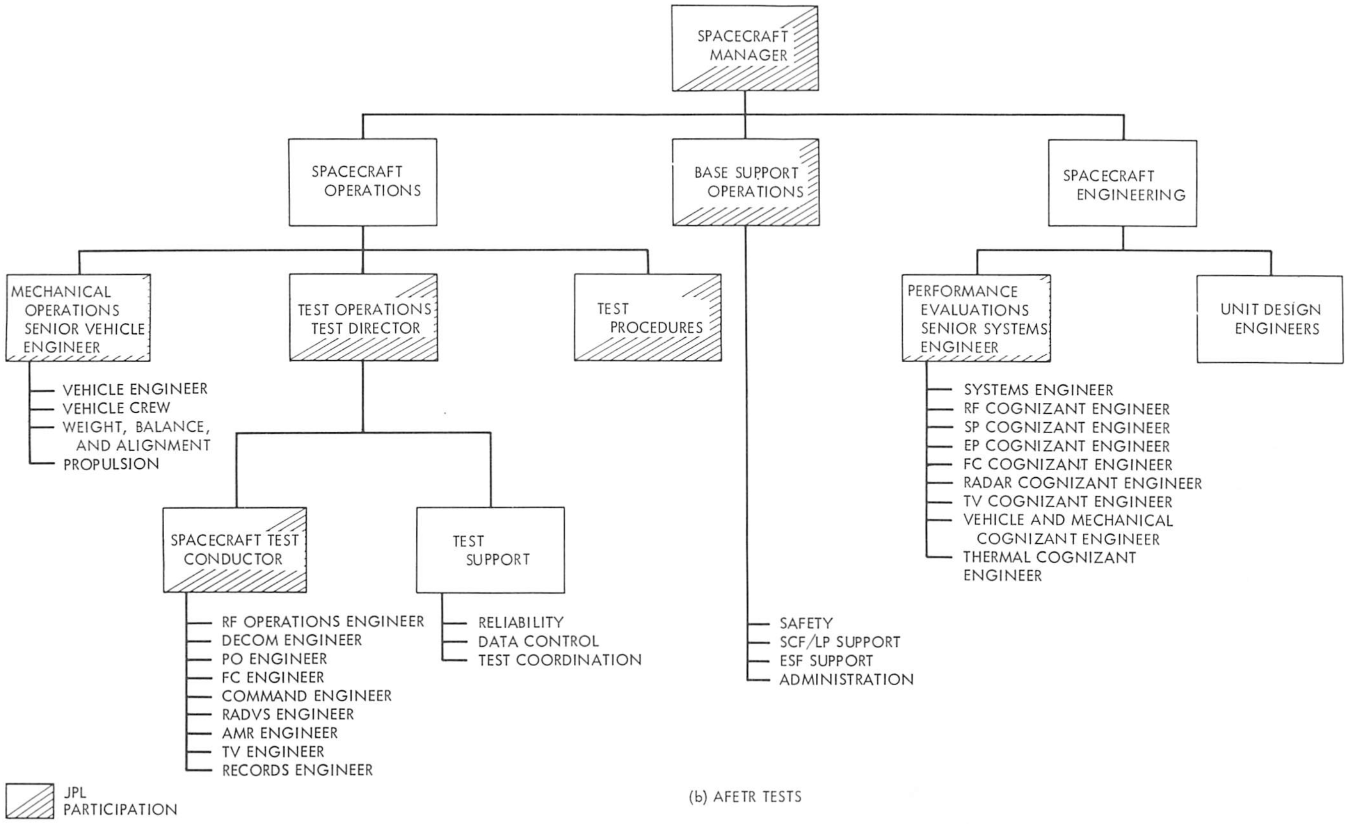


Fig. XI-2. HAC test team organization block diagram

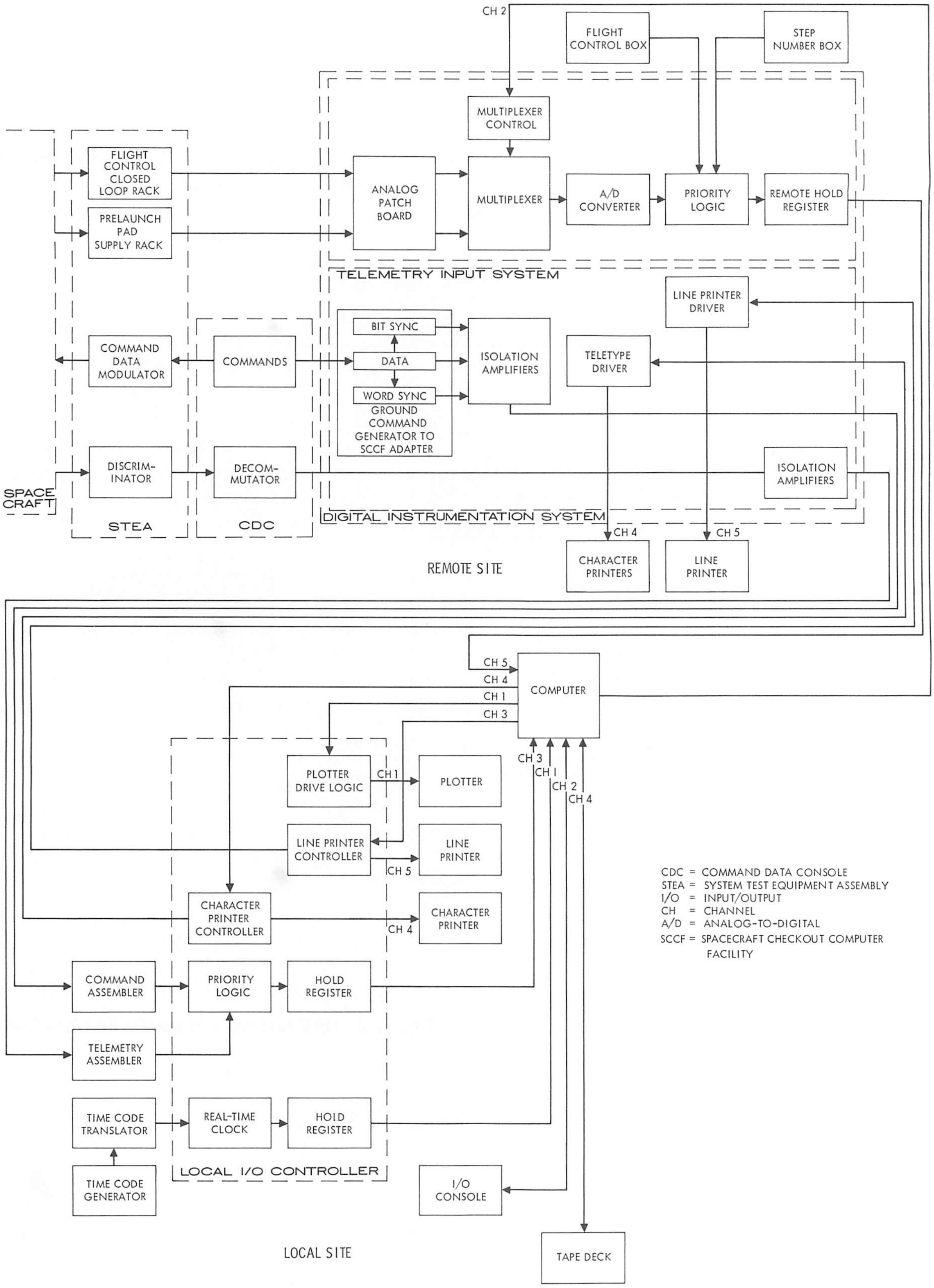
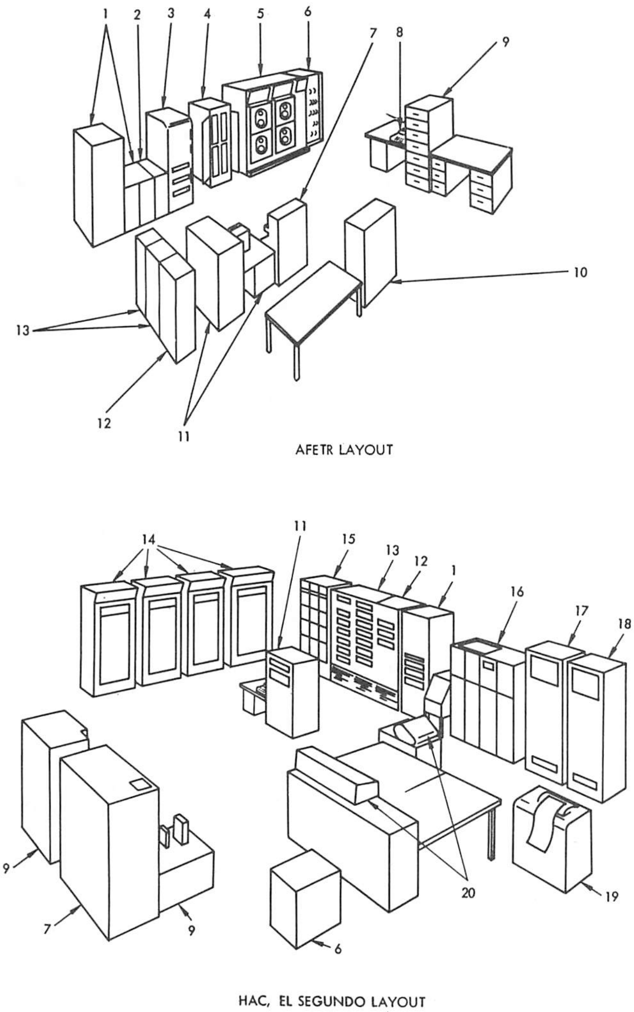


Fig. XI-3. Spacecraft checkout functional block diagram



1. DATA PHONE RACKS
2. LINE VOLTAGE REGULATOR
3. POWER RACK
4. UNIVAC 1218
5. UNIVAC 1240
6. STORAGE RACK
7. CARD READER
8. KEYPUNCH
9. CARD CABINET
10. SPARE PARTS CLOSET
11. INPUT/OUTPUT CONSOLE
12. TELEMETRY AND COMMAND BUFFER UNIT
13. INPUT/OUTPUT CONTROLLER
14. UNIVAC 1541
15. UNIVAC 1219
16. 1510 SYSTEM CONTROLLER
17. TIMING FUNCTIONAL RACK
18. INTERCONNECTING CABLE RACK
19. TELETYPE UNIT
20. X-Y PLOTTER

Fig. XI-4. Computer data system layout

b. Spacecraft configuration. The spacecraft was completely assembled with flight equipment except for the following:

(1) Compartment covers.

(2) Retro motor.

(3) Solar panel.

(4) Live pyrotechnics.

(5) Batteries.

(6) Altitude marking radar (AMR) mounted to test fixture adjacent to spacecraft—connected with test cables.

The spacecraft was mounted on the system test stand and power was provided by an external 22-V supply. Direct electrical access cables were installed for signal injection and hardline monitoring.

c. Test description. Prior to the application of power to the entire spacecraft, an initial power turnon test was performed to verify the proper voltages on each pin and absence of voltage on all other pins. Seven major tests were then performed in the order listed here. A brief description of each test and objective follows (Fig. XI-6).

1. Telecommunication integration test. This integration test was performed to verify the integration of the RF, command decoding, signal processor, and power groups into a system configuration and to demonstrate their ability to operate properly.

2. Television/telecommunication integration test. This test was performed to determine: (1) that survey TV camera mechanisms could be commanded and frame identification information could be received using the telecommunication link, and (2) that TV subsystem video parameters processed through the telecommunication link were acceptable.

3. Flight control/telecommunication integration test. This integration test checked the performance of the spacecraft when the flight control sensor group (FCSG), TCM, power, and gas jet subsystem were integrated.

4. Flight control-altitude marking radar/telecommunication integration test. This integration test checked the performance of the spacecraft with the radar altimeter and doppler velocity sensor (RADVS) and AMR integrated into the system. The most significant tests were the flight control closed-loop tests, which were conducted in ISCO for the first time and which were performed during all succeeding ambient and environmental mission sequences.

Closed-loop tests were made primarily to: (1) check the attitude and thrust-control loop performance during simulated midcourse velocity correction and terminal descent phases of the mission, and (2) verify that the hardware interfaces and interconnections between the flight control subsystem and related elements on the spacecraft were satisfactory.

These tests provided the only preflight closed-loop checks of those portions of the flight control subsystem that were operational during the vernier engine burn phases of the mission. Detailed open-loop tests were performed at the subsystem level during normal FA testing prior to delivery of hardware to the spacecraft.

The closed-loop tests were accomplished by integrating the flight control sensor group, roll actuator, and RADVS with an analog computer which simulated both the spacecraft and vernier engine dynamic characteristics. The closed-loop test equipment was also used to verify gain and phase or stability margins for the various control loops at response frequencies of interest.

5. Mechanism vehicle/telecommunication integration test. This integration test was performed to verify: (1) operation of the antenna/solar panel positioner (A/SPP) mechanisms telemetry and the automatic deployment logic, (2) omniantenna extension telemetry, (3) engineering mechanism auxiliary (EMA) squib ignition energy, and (4) turnon of the EMA thermal switches.

6. Alpha scattering/telecommunication integration test. The alpha scattering/telecommunication integration test was performed to: (1) verify that the alpha scattering subsystem would respond to command and that alpha data could be received using the telecommunication link, (2) verify squib and heater circuits, and (3) calibrate the alpha and proton detectors.

7. Soil mechanics/surface sampler/telecommunication integration test. The soil mechanics/surface sampler/telecommunication (SM/SS/TCM) integration test verified that the mechanism would respond to command and that telemetry was received using the telecommunication link, and checked the operation of the SM/SS subsystem when integrated into the spacecraft.

Figure XI-7 shows the nominal elapsed time for performance of each test during ISCO.

2. Mission Sequence/Electromagnetic Interference Test Phase

a. Objectives. The objectives of the mission sequence/electromagnetic interference (MS/EMI) test phase were:

(1) To verify the capability of the spacecraft to perform MS at minimum voltages simulated battery profile voltage and with flight type battery sources of power.

(2) To verify proper spacecraft performance during exposure to radio frequency interference (RFI) and EMI at signal levels expected at Launch Complex 36 at AFETR.

b. Spacecraft configuration. The spacecraft was fully assembled mechanically and electrically in a flight configuration except for the following:

(1) Mission sequences 1 and 2.

- (a) Dummy solar panel and planar array antenna of one-sixth weight to simulate lunar gravity effects on drive motors.
- (b) Test vernier propulsion system (VPS) thrust chamber assemblies installed.
- (c) Test VPS thrust chamber assemblies installed.
- (d) No compartment A, B, or C covers and no crushable blocks.
- (e) Dummy shock absorbers and footpads installed.
- (f) The AMR mounted on test frame and electrically connected to the Spacecraft System.
- (g) Squib simulators and pinpullers installed.
- (h) No propellants loaded but VPS at pad pressure.
- (i) Hardline test access for simulated power and solar panel inputs, high-voltage alarm circuit, closed-loop flight control testing, RADVS and ARM checkout, and power profile monitoring.
- (j) Spacecraft mounted on the STEA system test stand.

(2) Mission sequence 3. Same as sequences 1 and 2 except for the following:

- (a) Flight type main battery installed.
- (b) Dummy retro motor with flight AMR installed.
- (c) Live squibs and mufflers installed.
- (d) Only test access was umbilical until injection, when umbilical was removed or disconnected through breakout box.
- (e) Spacecraft mounted on a mobile cart in the screen room.

Fig. XI-5. Flight acceptance test plan flow diagram

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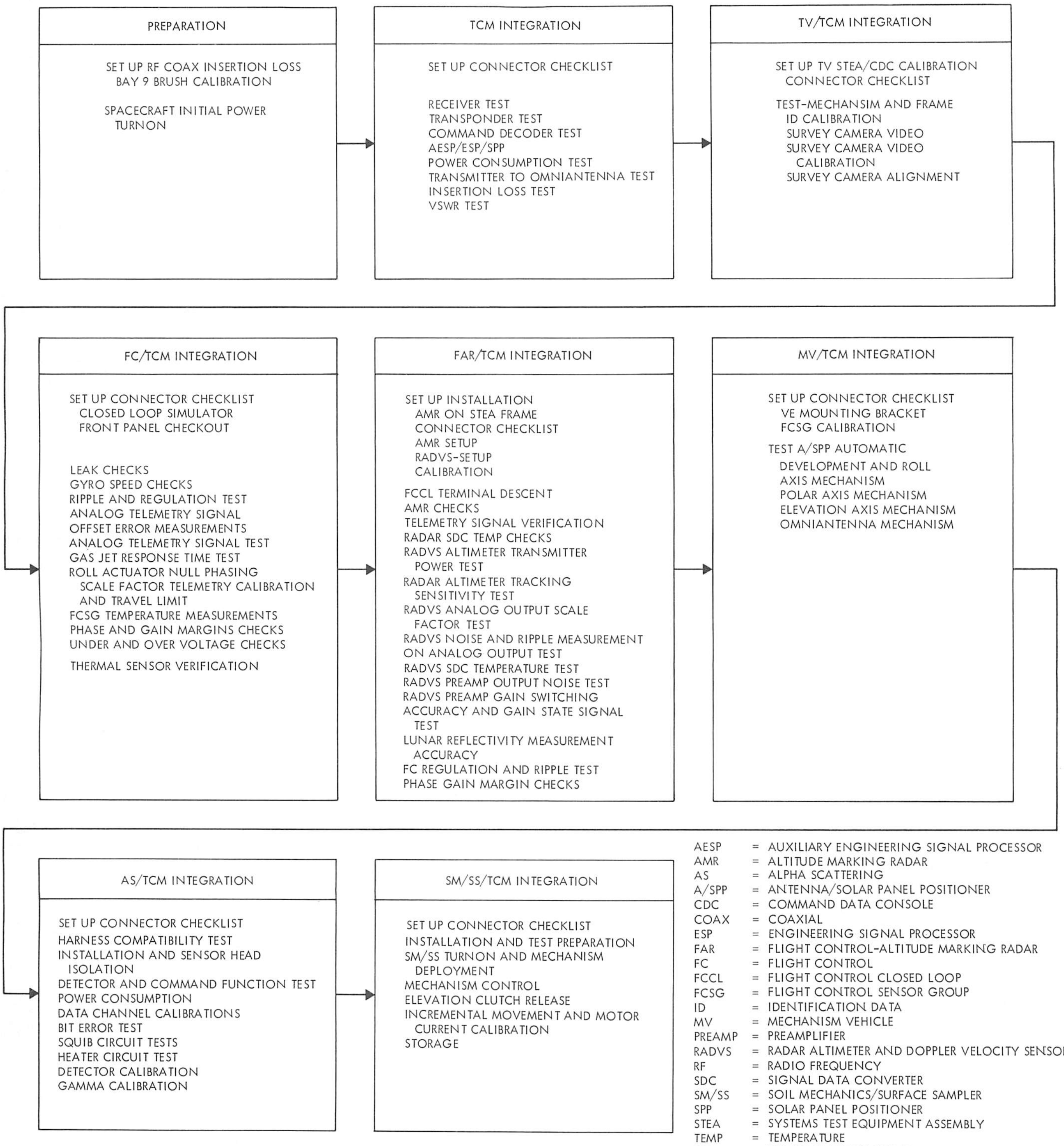
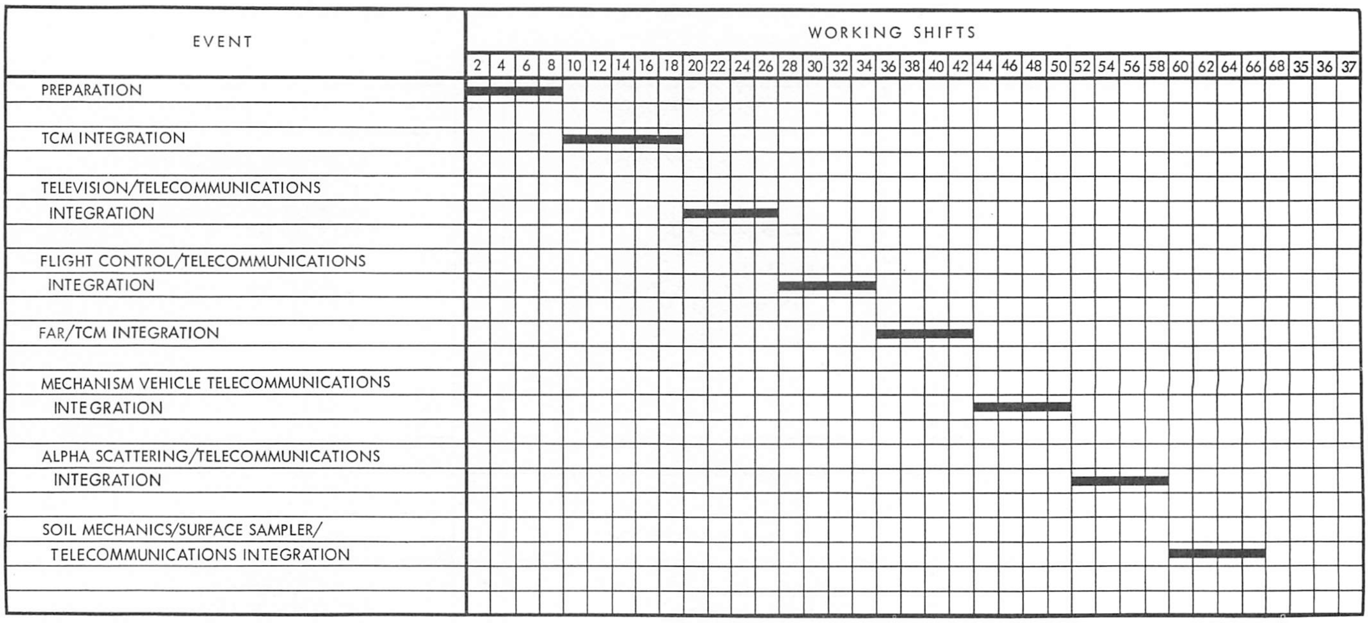


AESP = AUXILIARY ENGINEERING SIGNAL PROCESSOR
AMR = ALTITUDE MARKING RADAR
AS = ALPHA SCATTERING
A/SPP = ANTENNA/SOLAR PANEL POSITIONER
CDC = COMMAND DATA CONSOLE
COAX = COAXIAL
ESP = ENGINEERING SIGNAL PROCESSOR
FAR = FLIGHT CONTROL-ALTITUDE MARKING RADAR
FC = FLIGHT CONTROL
FCCL = FLIGHT CONTROL CLOSED LOOP
FCSG = FLIGHT CONTROL SENSOR GROUP
ID = IDENTIFICATION DATA
MV = MECHANISM VEHICLE
PREAMP = PREAMPLIFIER
RADVS = RADAR ALTIMETER AND DOPPLER VELOCITY SENSOR
RF = RADIO FREQUENCY
SDC = SIGNAL DATA CONVERTER
SM/SS = SOIL MECHANICS/SURFACE SAMPLER
SPP = SOLAR PANEL POSITIONER
STEA = SYSTEMS TEST EQUIPMENT ASSEMBLY
TEMP = TEMPERATURE
TCM = TELECOMMUNICATIONS
TM = TELEMETRY
VE = VERNIER ENGINE
VSWR = VOLTAGE STANDING WAVE RATIO

Fig. XI-6. Initial system checkout test flow diagram



FAR = FLIGHT CONTROL – AMR – RADVS
TCM = TELECOMMUNICATIONS

Fig. XI-7. Initial system checkout nominal test schedule

c. Test description. The MS/EMI test phase was conducted in the system test area in building 350. The spacecraft was commanded through two compressed mission sequence runs without EMI/RFI and then through a complete real-time plugs-out MS with *Atlas/Centaur* and Launch Complex 36 EMI/RFI simulation for the appropriate time periods. The test sequence began with launch pad system readiness checks and terminated with postlanding operations. Figure XI-8 is a flow diagram of MS/EMI tests; Table XI-1 gives the conditions during the three test sequences. The nominal test schedule is shown in Fig. XI-9.

3. Solar-Thermal-Vacuum Tests

a. Objectives. The primary objectives of the solar-thermal-vacuum (STV) tests were as follows:

(1) To verify correct spacecraft functional operation during a real-time MS while exposed to a range of solar conditions in a simulated STV environment.

(2) To verify correct spacecraft thermal performance during simulated STV environments.

b. Spacecraft configuration. The spacecraft was in a transit flight configuration except for the following deviations imposed by test access requirements, facility characteristics, and safety considerations:

(1) Solar panel and planar array antennas were removed because of shadowing problems caused by the solar simulator's decollimation angle; the proper shadow effect was provided by a system of discrete nitrogen-cooled shadow shrouds placed over the units normally shadowed by the solar panel and planar array antennas in flight.

(2) Landing gear pads were not installed because of physical interference with the chamber wall shrouds when the landing gears were in an extended position.

(3) Thermally representative dummy shock absorbers were installed in place of flight shock absorbers; the shock absorbers were outside the solar beam, and the STV test period was beyond their permissible shelf life prior to launch.

(4) Live retro motor was replaced by an inert thermal model.

(5) Propellant tanks were loaded with referee propellants and propellant and pneumatic storage tanks were pressurized to meet the test requirements.

The gases and pressures were chosen to provide the best possible means of measuring gross leakage from the two systems (VPS and gas jet attitude control system) while in the chamber; the measurements were performed with a residual gas analyzer.

The propellant and pneumatic storage tanks were pressurized as follows:

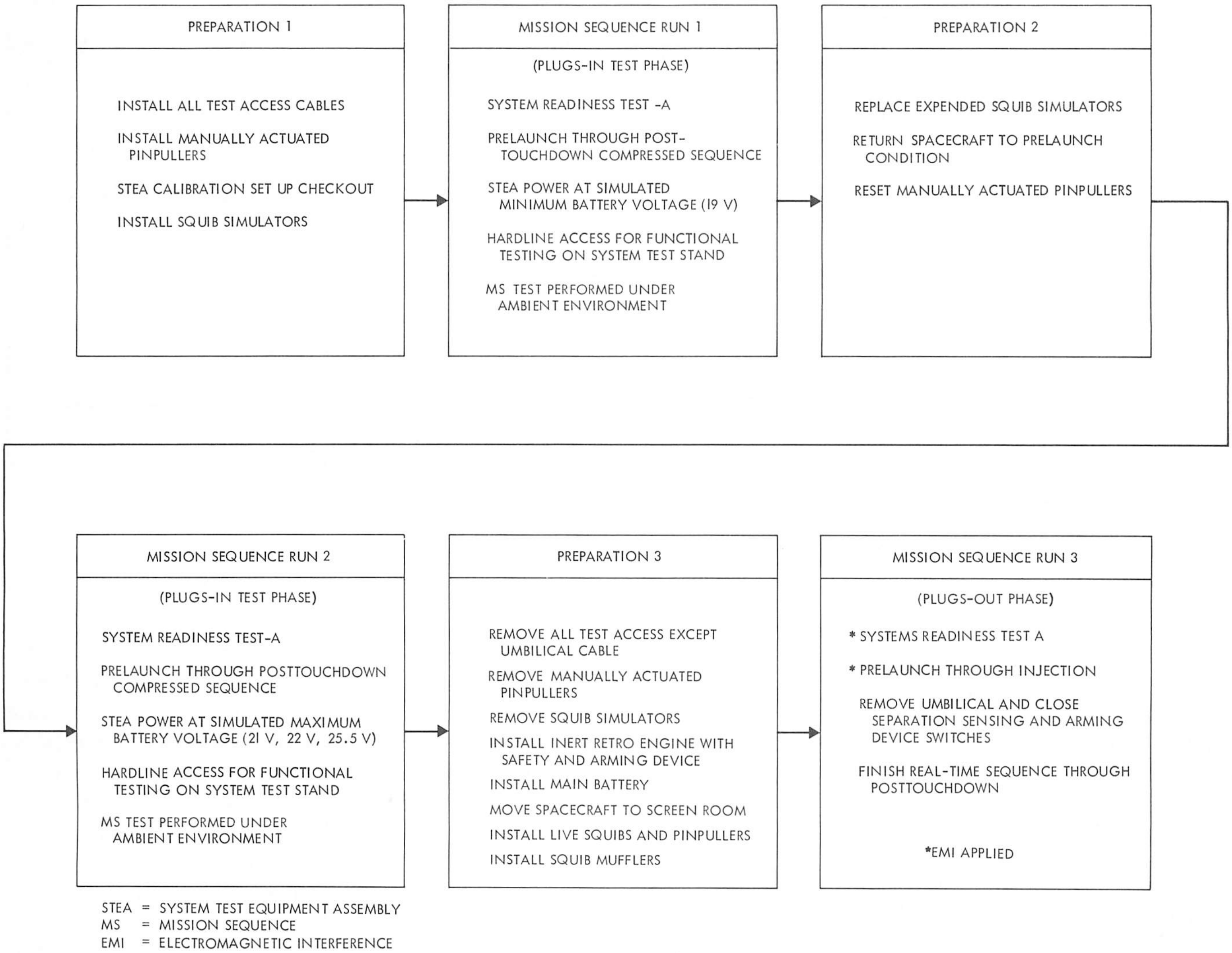


Fig. XI-8. Mission sequence/electromagnetic interference test phase flow diagram

Propellant tank	Pressure, psig	Propel-lant
Vernier propulsion system, helium high-pressure system	2450 ±50	Helium
Vernier propulsion system, helium low-pressure system	350 ±50	Helium
Gas jet attitude control system, nitrogen high- and low-pressure systems	1800 ±200	Argon

(6) Test plumbing provided a means for continuous supply and exhaust of gaseous nitrogen to and from the gas jet attitude control system.

(7) Landing gear and omniantennas were extended prior to the start of the test, but their releasing pinpullers were actuated in a normal sequence during subphase A.

(8) Squibs were fired in special squib mufflers instead of in the flight mechanism during subphase A for the following pyrotechnic actuated devices: (a) helium release and dump valves, (b) main retro-igniter, (c) retro-release devices, and (d) shock absorber locking devices.

(9) Test canisters were installed on the compartments for subphases A and B; the test canisters were essentially identical to flight canisters except that they provided test access to the compartments.

(10) A flight type test battery was installed for both subphases and was used for spacecraft power

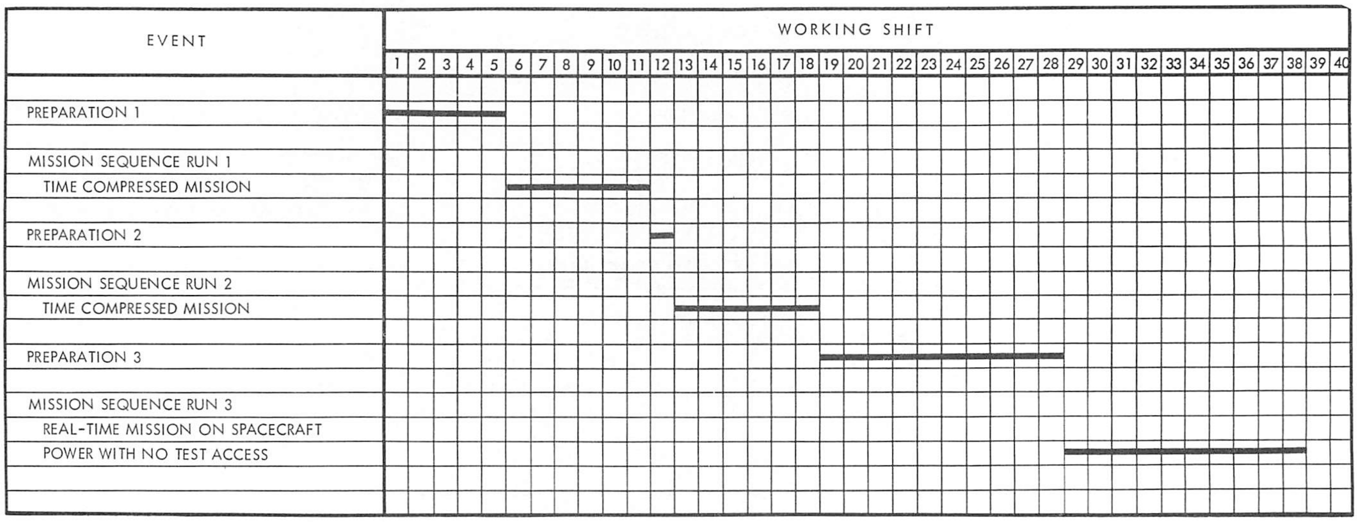


Fig. XI-9. Mission sequence/electromagnetic interference nominal test schedule

during subphase A; the spacecraft was switched to STEA power for subphase B.

(11) Flight VPS thrust chamber assemblies were installed for all STV phases.

c. Test description. The STV test phase was composed of nine preparation or checkout periods (Fig. XI-10) plus the following two major test subphases:

(1) Subphase A (high-temperature) test consisted of a real-time MS (68-hr transit and 4-hr postlanding operations) under a simulated transit environment. During this test, the spacecraft was powered by its own battery. Solar radiation was increased to 112% of nominal sun (130 W/ft^2) to provide high thermal marginal testing.

(2) Subphase B (low-temperature) test consisted of a 72-hr sequence continued from subphase A without interrupting chamber operation, but separated by a temperature stabilization period to reestablish a representative initial temperature condition for the start of the second mission. The spacecraft derived power from STEA. Solar radiation was reduced to 87% of nominal sun (130 W/ft^2) to provide low thermal marginal testing.

Table XI-1. Mission sequence/electromagnetic interference test sequences and conditions

Test condition	Sequence 1	Sequence 2	Sequence 3
Test duration	40-hr compressed mission	40-hr compressed mission	72-hr real-time mission
Environment	Ambient	Ambient	RFI through separation
Spacecraft power	From STEA constant 19 V	From STEA stepped 25.5, 22, 21 V	Spacecraft battery
Access	Complete	Complete	Solar simulator only, spacecraft ungrounded
Command/data link	Hardline	Hardline	RF
Pyrotechnics	Squib simulators	Squib simulators	Live squibs

The test plan for *Surveyors I–IV* included a third planned chamber test (subphase C), which was a compressed time sequence at a nominal solar radiation level (one solar constant). The spacecraft was to be tested in a plugs-out configuration with only sufficient hardline to ensure spacecraft safety.

The STV subphase C was deleted from the test plan after *Surveyor IV*, because the limited additional data did not justify the lengthy schedule time required. Sufficient thermal-vacuum data were acquired in subphases A and B, and adequate plugs-out operation was obtained in MS/EMI sequence 3 and in the ambient mission sequence following the vibration phase.

The STV test phase was conducted in the HAC space simulation laboratory (Building 365) at El Segundo. The 15-ft-diameter vacuum chamber with 8-ft-diameter solar simulator (SERF-C-4S) (Fig. XI-11) provided the simulated environment, and a STEA was used to control the spacecraft and acquire functional data. The simulated

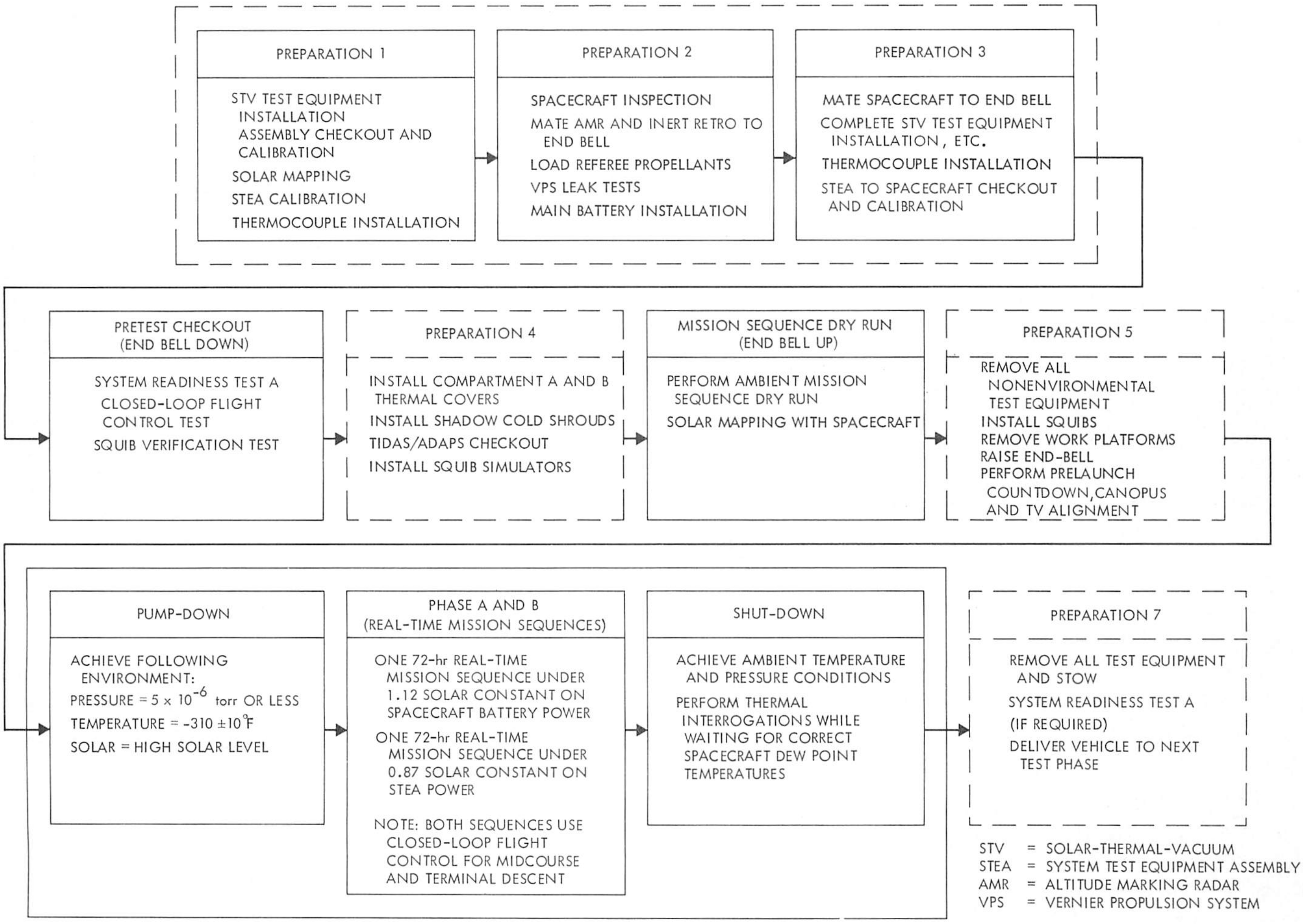


Fig. XI-10. Solar-thermal-vacuum test phase flow diagram

environmental conditions are shown in Table XI-2; Fig. XI-12 shows the nominal elapsed time for performance of the STV test phase.

Table XI-2. Solar-thermal-vacuum environmental limits

Condition	Limit
Temperature	
Range	75 to −320°F or lower
Transit	−300°F or lower
Pressure	
Range	760 to 1 × 10^{-6} torr
Static transit	5 × 10^{-6} torr or lower
Solar radiation (based on black plate radiometer readings)	
Subphase A	112% solar constant
Subphase B	87% solar constant

4. Vibration Test Phase

a. Objectives. The primary objectives of the vibration test phase were as follows:

(1) To verify functional operation during and after simulated launch vibration environments.

(2) To verify proper fabrication and assembly of the A-21 spacecraft.

(3) To verify that spacecraft subsystem alignments are not degraded by exposure to FA test vibration levels.

b. Spacecraft configuration. The spacecraft was fully assembled mechanically and electrically in a launch configuration with the exception of the following:

(1) Fuel and oxidizer tanks loaded with referee fluids and pressurized to 250 psi.

(2) Helium and nitrogen tanks pressurized to 250 psi.

(3) Test main battery installed.

(4) Equivalent structural mockups of live shock struts installed.

(5) Dummy retro motor installed.

(6) Squib simulators used to check the firing circuits of the retroigniter, retro-release bolts, helium release and dump valves, and landing gear locking pins.

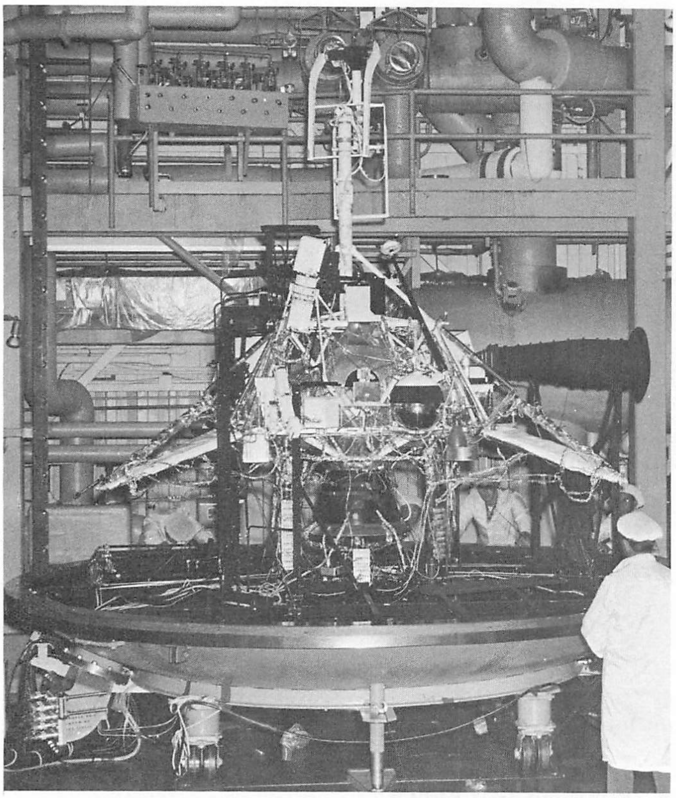


Fig. XI-11. Spacecraft installed on end bell of SERF-C-4S solar simulator

c. Test description. The vibration test was conducted in the HAC space simulation laboratory. The facility incorporated a vibration system which consisted of a power amplifier and a vibrator. Spacecraft control and telemetry data acquisition were accomplished through STEA and CDS. Prior to vibration, alignment was checked to provide a baseline from which to determine changes in critical alignments resulting from vibration.

The vibration test phase plan consisted of nine test subphases, three of which were environmental tests which imposed vibrations in the following three directions: (1) along the spacecraft Z-axis, (2) laterally, in the plane of leg 3, and (3) laterally, normal to the plane of leg 3. The other six subphases included preparation and pre and postvibration system readiness tests (SRT A) to verify system functional operation.

An alignment check was performed after completion of vibration. A flow diagram illustrating the test sequence is shown in Fig. XI-13.

The environmental levels are given in Table XI-3. All vibration tests in the two lateral directions were performed with the vehicle and fixture mounted on the horizontal hydrostatic team units (Fig. XI-14). The Z-axis vibration tests were performed with the fixture mounted directly on the vibrator.

The Z-axis vibration test simulated the vibration frequency/acceleration profile imposed on the spacecraft

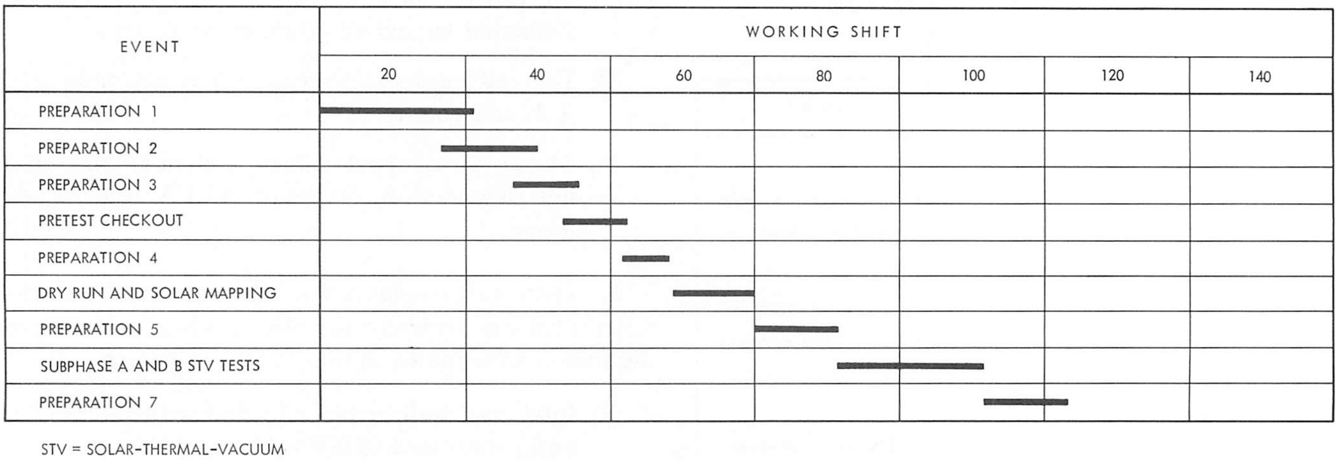
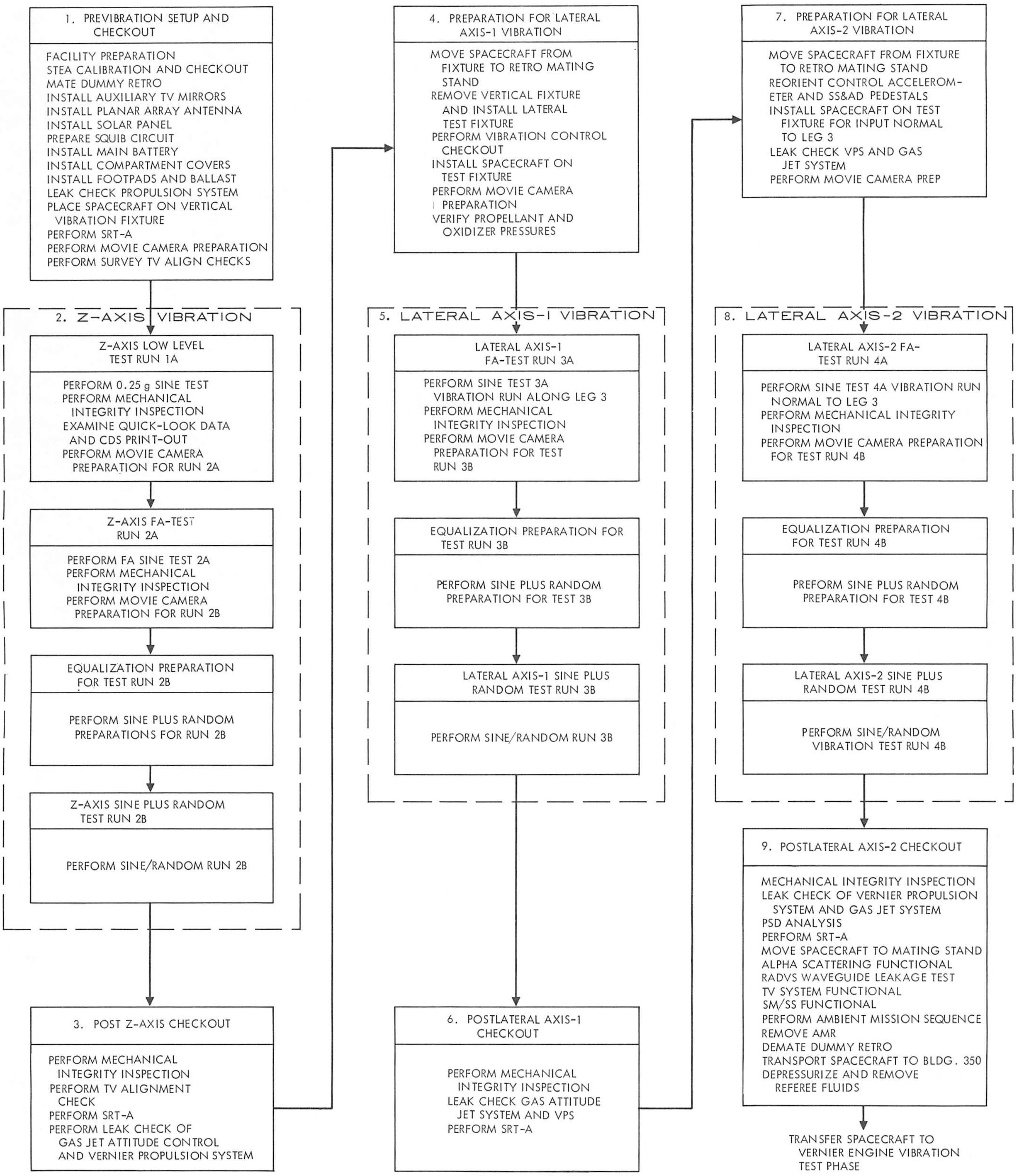


Fig. XI-12. Solar-thermal-vacuum nominal test schedule



SRT-A = SYSTEM READINESS TEST A
VPS = VERNIER PROPULSION SYSTEM
SS&AD = SEPARATION SENSING AND ARMING DEVICE
STEA = SYSTEM TEST EQUIPMENT ASSEMBLY
CDS = COMPUTER DATA SYSTEM
FA = FLIGHT ACCEPTANCE
PSD = POWER SPECTRAL DENSITY
RDVS = RADAR ALTIMETER AND DOPPLER VELOCITY SENSOR
SM/SS = SOIL MECHANICS/SURFACE SAMPLER
AMR = ALTITUDE MARKING RADAR

Fig. XI-13. Vibration test phase flow diagram

Table XI-3. Vibration levels

Force	Rate octave/min	Cycle
Z-axis		
5–50, 1.5 *g* (0–peak) VFSW[a]	2	
50–100, 1.33 *g* (0–peak) VFSW	2	
100–1500, 1.33 *g* (0–peak) VFSW	2	1 sweep up and 1 sweep down
100–1500, 4.5 *g* (rms) white gaussian noise		Combined
Lateral axis		
5–50, 0.7 *g* (0–peak) VFSW	2	
50–100, 1.33 *g* (0–peak) VFSW	2	
100–1500, 1.33 *g* (0–peak) VFSW	2	1 sweep up and 1 sweep down
100–1500, 4.5 *g* (rms) white gaussian noise		Combined
Low level vibration—*Z*-axis only		
5–100, 0.25 *g* (0–peak) VFSW	1	

[a]VFSW = variable frequency sine wave.

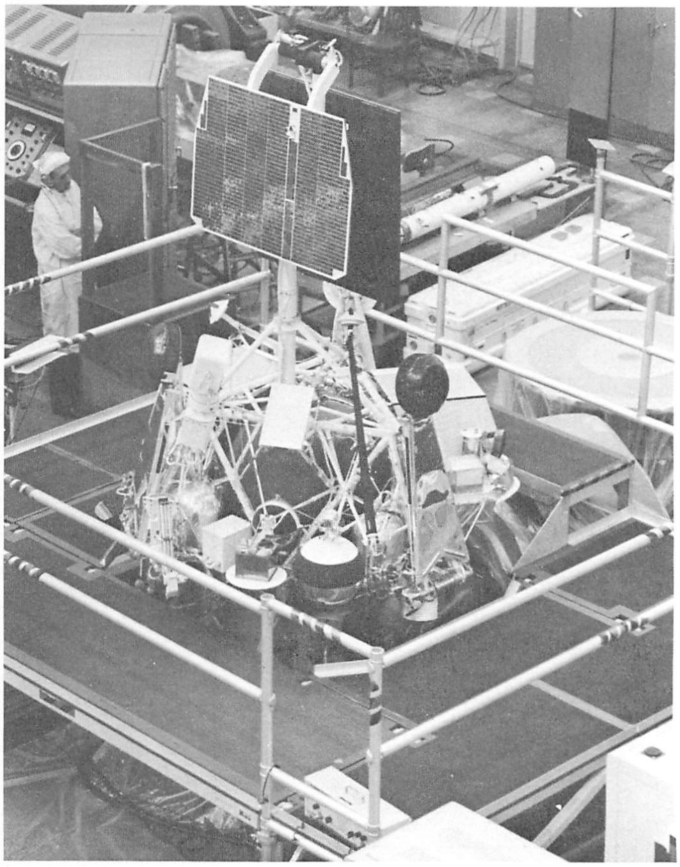


Fig. XI-14. Vibration test setup

Z-axis at launch. The inputs were applied at the separation plane of the *Atlas/Centaur* test vehicle. Inspection for mechanical integrity were performed after each vibration run.

The lateral vibration tests simulated the predicted launch vibrations imposed in the direction parallel to the spacecraft *X-Y* plane. The input directions were parallel to, and normal to, the plane of spacecraft leg 3. After each vibration run, inspections for mechanical integrity were performed and a spacecraft SRT A was conducted. The runs performed are described in Section X, Spacecraft Environmental Test Requirements.

All vibration tests were controlled by the selector control adjustable multiple point electronic device, which incorporated up to four control accelerometers. For the low-frequency runs, three accelerometers installed in three orthogonal axes on the retro motor and one accelerometer on the fixture in the direction of vibration controlled the vibrators input level. In the high-frequency runs, the vibrators input level was controlled by three accelerometers, one at each attachment point on the fixture in the direction of vibration. In the combined sine–random test an additional accelerometer was used for random equalization and control.

The vibrators force was limited by installation of two armature acceleration limiter accelerometers. One was installed on the retro motor and one at a fixture input control accelerometer location in the direction of vibration. The levels of the accelerometers were set at 50% above the input test levels and reset any time a step in level was performed.

Prior to each full-level combined sine and random vibration test, an equalization verification test was performed. The vibration excitation system had to be equalized so that the output from the control accelerometer did not deviate from the specified level by more than ±3.0 db between 100 and 1500 Hz for the noise portion nor more than −10% for the sine portion. An equalization verification curve was made from the random excitation control accelerometer output and combined sine–random rms plots had to conform to the indicated criterion.

Input levels were measured at *Surveyor/Centaur* separation plane and had a tolerance of ±3 db for random power spectral density using a 50-Hz filter and overall rms tolerance of ±10%. Input levels were displacement-limited to 0.8 in. (peak-to-peak) at low frequencies, and response levels at the retro motor CG were not to exceed at any frequency the corresponding input level given in Table XI-3.

Low-level runs were conducted on T-21 on all three axes from 5 to 1500 Hz. Because no information was obtained at high frequencies that was not also available at low frequencies, an appreciable reduction in test duration was accomplished by performing low-level runs on three axes for *Surveyor I* at frequencies only from 5 to 100 Hz. Following *Surveyor I*, low-level runs in the lateral directions were deleted for the same reason.

On T-21 and *Surveyor I* (prior to January 1966) equalization was accomplished manually using peak-notch filters. Because of the excessive time required in the manual system, automatic equalizer/analyzer equipment was procured by the environmental facility and the equalization was performed by the automatic random-vibration system by exciting the spacecraft at 3 db down from full level for 15 sec. A tape loop was cut and a power spectral density analysis performed to verify proper level. The acceleration amplitude of the equalization run was 3.18 *g* rms. On *Surveyors I–V* the analysis was performed with the tape loop; on *Surveyors VI* and *VII*, equalization was verified directly from the oscilloscope on the automatic equipment.

After confirmation that the equalization was proper and within specification, a verification check was performed at 5 Hz. This test verified agreement between control and accelerometers on the vibration test fixture and retro motor. The full-level run was then started.

An ambient MS (non-real-time) followed completion of the postvibration functional checks and was a plugs-out type test similar to that performed in MS/EMI run 3. The initial spacecraft configuration conformed to that at launch. Data links were established using "snood" connections to omniantennas. Squib circuits were configured for testing with squib simulators and stray energy monitors, except at pinpuller positions; for these live squibs were connected and fired in sequence as during a mission. This test was performed in either building 365 or 350. The spacecraft was then prepared for the next phase by removing the main battery (if required), referee fluids, and AMR and checking alignability on the optical dock.

Figure XI-15 shows the nominal elapsed time for the performance of vibration testing.

5. Vernier Engine Vibration Test Phase

a. Objectives. The objective of the vernier engine vibration (VEV) test phase was to verify that the RADVS

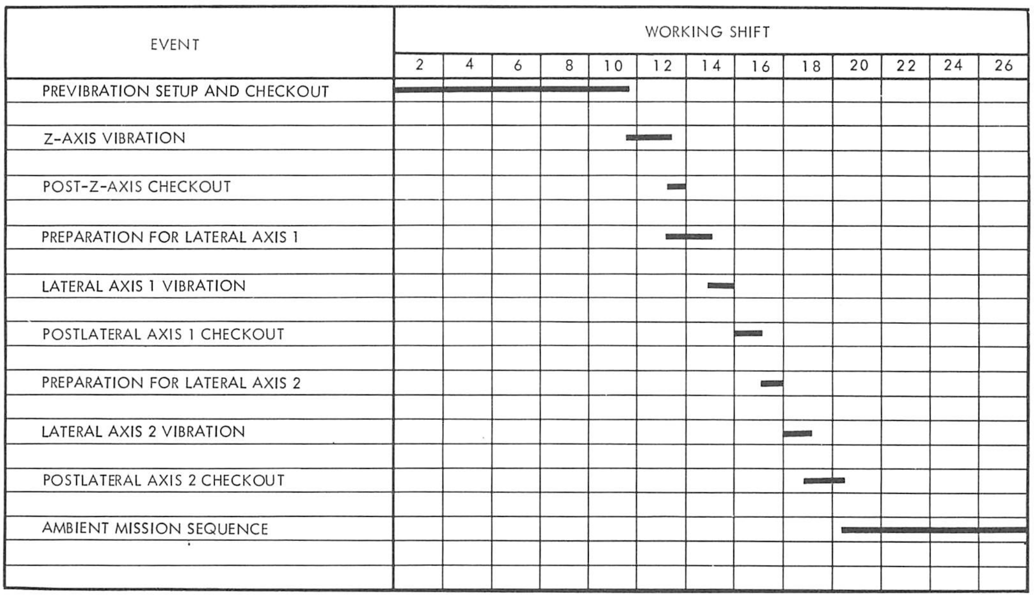


Fig. XI-15. Vibration nominal test schedule

beams did not produce a false lock as a result of vernier engine vibrations.

b. Spacecraft configuration. The spacecraft was in flight configuration with the following exceptions:

(1) No retro motor installed.

(2) No AMR mounted on spacecraft.

(3) Fuel and oxidizer tanks filled with helium gas to 10 psig inside the bladder with 2-psi differential across the bladder; positive pressure inside.

(4) Thrust chamber assemblies replaced with equivalent masses.

(5) Feed horns of RADVS were terminated in microwave loads to simulate a free space environment for the RF transmitters and receivers.

The planar array and solar panel were in the transit position, and the spacecraft legs and omniantennas were extended.

c. Test description. The test was conducted in the building 350 system test area. The test plan consisted of three subphases: (1) previbration spacecraft functional tests, including flight control closed-loop midcourse correction and terminal descent, (2) vibration, applied through dummy vernier engines, while the RADVS was

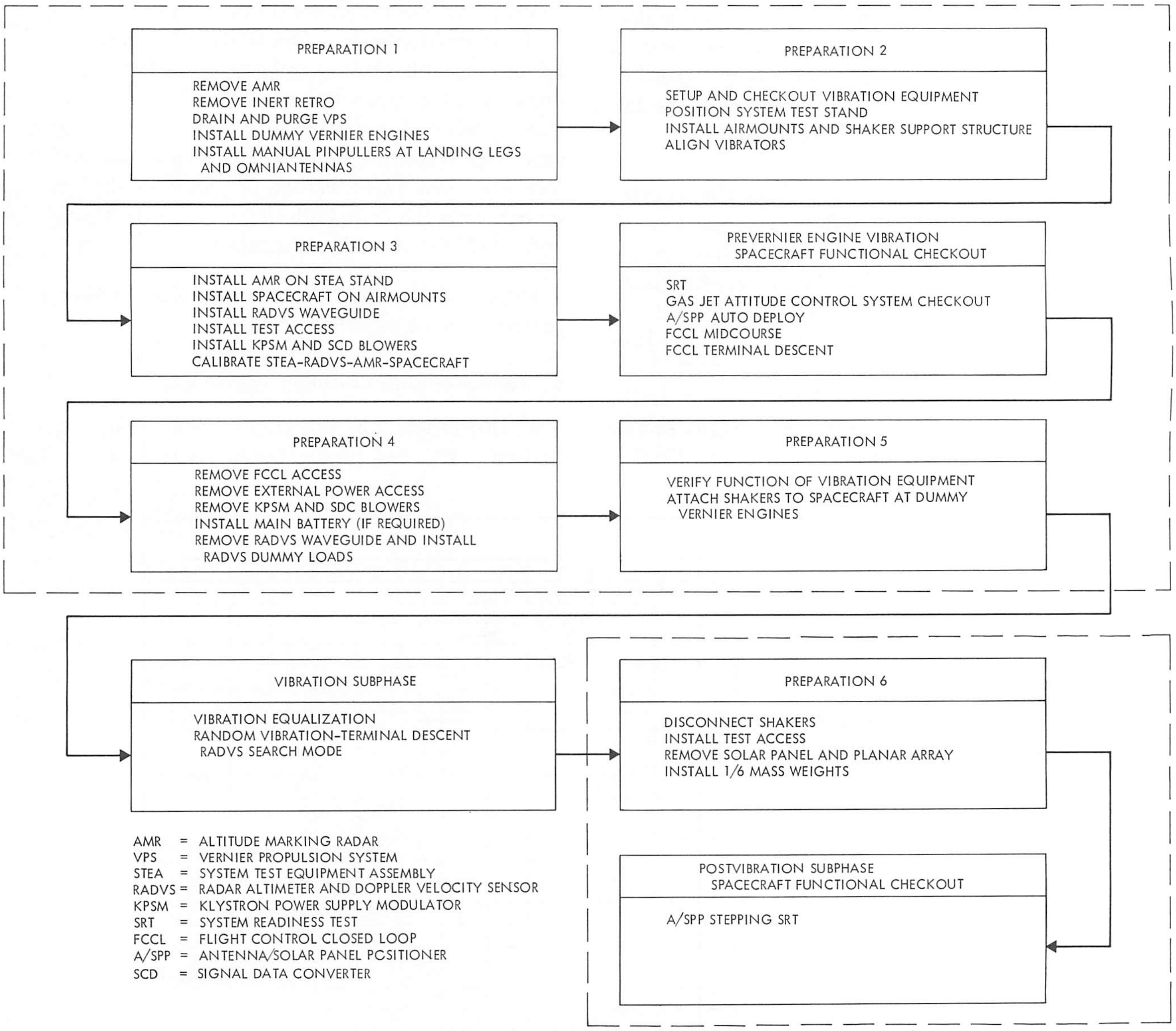


Fig. XI-16. Vernier engine vibration phase flow diagram

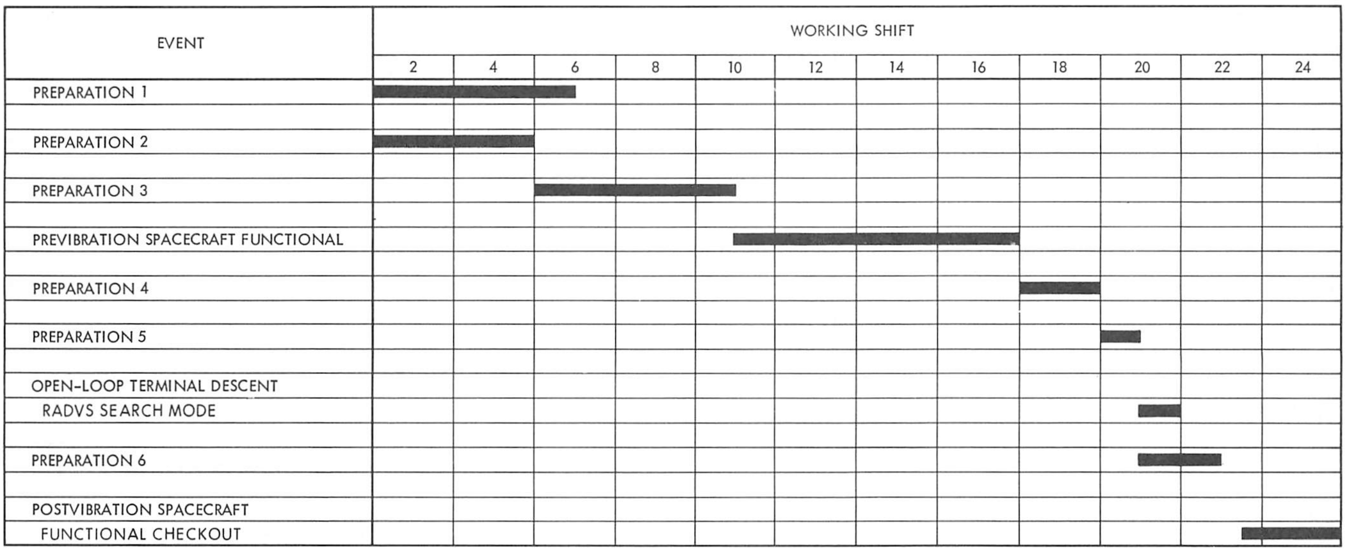


Fig. XI-17. Vernier engine vibration nominal test schedule

functioning in the search mode, and (3) postvibration spacecraft functional tests. Figure XI-16 is a flow diagram of the test phase.

The spacecraft was subjected to vibration excitation in a direction parallel to the spacecraft roll axis, applied simultaneously through dummy vernier engines. Pressure, temperature, and humidity conditions were laboratory ambient. The excitation was random noise with gaussian distribution, band-limited between 84 and 200 Hz, with an average magnitude at each vernier engine of 20 lb rms.

Vibration was produced by three vibrators connected in series to the output of a power amplifier.

Figure XI-17 shows the nominal elapsed time for completing the VEV test phase.

After completion of the VEV test phase, the spacecraft was moved to the alignment dock where critical components were alignment checked to ensure that during vibration no significant alignment shifts had occurred. This also was done to assure that all spacecraft components could be final aligned at AFETR. Figure XI-18 shows an alignment dock.

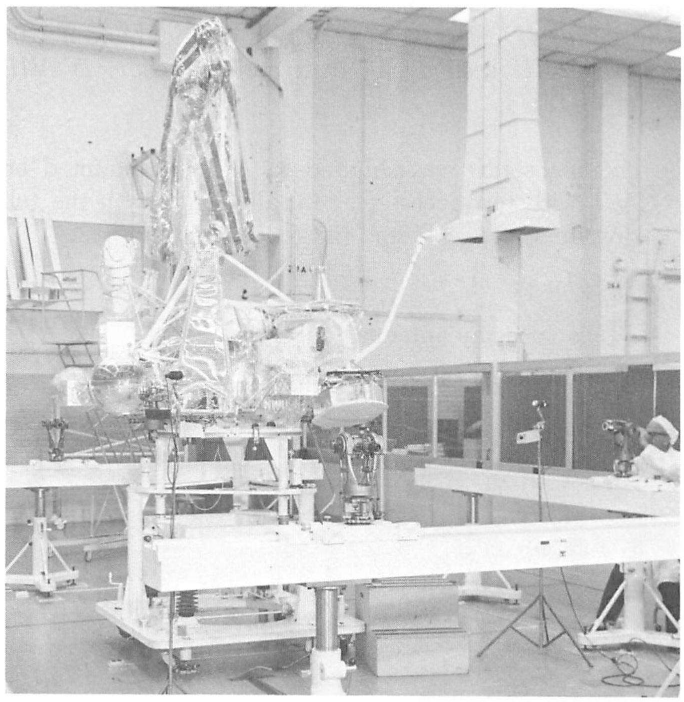


Fig. XI-18. Optical alignment dock

6. Shipment to Combined System Test and Air Force Eastern Test Range

Spacecraft shipment to GD/C, San Diego was via air-ride van. To facilitate transport, various spacecraft components (such as A/SPP, batteries, FCSG, footpads, etc.) were removed and packaged in special shipping containers and the spacecraft was mounted on a ground transport vehicle. Auxiliary cargo, consisting of removed spacecraft components, support equipment, and data files, were included in the shipment.

Originally, it was also planned to ship the spacecraft to AFETR via air-ride van. The transit time was expected to be five days, as verified by van shipment of the T-21 *Pathfinder* spacecraft. However, because of the critical AFETR test schedules and desired launch dates, it was decided to air-transport all subsequent spacecraft to AFETR, from either GD/C, San Diego or HAC, El Segundo.

For air transport of *Surveyor I,* the Douglas C-133 was selected primarily because of its availability and low-level shock characteristics. Due to the limited time available to investigate air transport, the permissible shock conditions were conservative. Subsequent analysis proved that the spacecraft, as secured on ground transport vehicle (GTV 1) for shipment, could safely withstand higher shock conditions, and other cargo planes could be used. For future spacecraft shipments, the governing criteria in the plane selection became availability, cost, and transit time. In succession of use, the four airplanes selected were: Douglas C-133 (*Surveyor I*), Aero Spacelines B377PG (*Surveyor II*), Aero Spacelines B377SG (*Surveyor III* first shipment), and Lockheed L-100 (*Surveyor III* second shipment and *Surveyors IV–VII*) (Fig. XI-19).

The spacecraft was shipped to AFETR mounted on GTV 1 with test access tees installed and with the following items removed and packed separately:

(1) Solar panel.

(2) Planar array antenna.

(3) Antenna/solar panel positioner.

Fig. XI-19. Spacecraft loading for air shipment

(4) Altitude marking radar.

(5) Flight control sensor group.

(6) Retro motor.

(7) Battery.

(8) Pyrotechnics.

(9) Shock absorbers (dummy shock absorbers were installed).

(10) Thrust chamber assemblies.

Humidity was controlled by desiccant bags placed in sealed containers within the inner shroud of GTV 1.

7. Combined Systems Test Phase

a. Objectives. The objectives of the CST phase were as follows:

(1) Demonstrate operational compatibility of the space vehicle (*Surveyor/Centaur/Atlas*) and ground support equipment (GSE) during a simulated flight from launch through simulated *Centaur* retro maneuver.

(2) Demonstrate electrical and mechanical compatibility between the interfaces of the *Surveyor* spacecraft and the *Atlas/Centaur* launch vehicle.

(3) Evaluate space vehicle launch preparations, handling, and countdown procedures.

(4) Provide test team familiarization with an evaluation of equipment and spacecraft handling procedures used at AFETR.

(5) Demonstrate electrical and mechanical compatibility of the spacecraft with aerospace ground equipment (AGE) and GSE.

b. Spacecraft configuration. The spacecraft was in flight configuration except for:

(1) Dummy retro motor.

(2) Dummy altitude marking radar.

(3) Live squibs installed in mufflers in leg pinpuller, omniantenna and helium release positions.

(4) Squib simulation installed in all other locations.

(5) Flight type battery.

(6) Test thrust chamber assemblies.

c. Test description. Combined systems tests were conducted at the NASA/GD/C facility in San Diego (Fig. XI-20) and consisted of six subphases:

(1) Initial preparation.

(2) Premate test.

(3) Encapsulation and mating.

(4) Combined system tests.

(5) Demating and decapsulation.

(6) Disassembly for shipment to AFETR.

Figure XI-21 shows the nominal test schedule for the CST phase.

After arrival at the CST facility in GD/C, San Diego, the spacecraft was received, unpacked, and mounted on the CSTS in the test and assembly area of the facility. The main battery, FCSG and thermal reflector, solar panel and planar array, and compartment A cover were installed. This activity was followed by spacecraft/STEA integration which included the installation of hardline access to the spacecraft and special test items. After this, the crushable blocks and footpads, squib simulators, and auxiliary battery (T-21 and SC-1–SC-4 only) were installed. The final reassembly operation included the installation of the dummy retro motor and AMR.

Spacecraft power was then applied and a commutator assessment was performed. This was to verify proper spacecraft operation after assembly and to establish the data link to the HAC, El Segundo area where the CDS was used to receive the telemetered data and output the data to teletype instruments located at the GD/C, San Diego facility.

The next operation was the mating of the spacecraft to the *Centaur* adapter and verifying that proper mechanical compatibility existed. The *Centaur* nose fairing was then moved to encapsulate the spacecraft. After partial encapsulation, a Spacecraft System readiness test (SRT) was performed with access only through the umbilical connection and TV targets provided by lights inside the nose fairing. After proper spacecraft performance was verified, the encapsulation process was

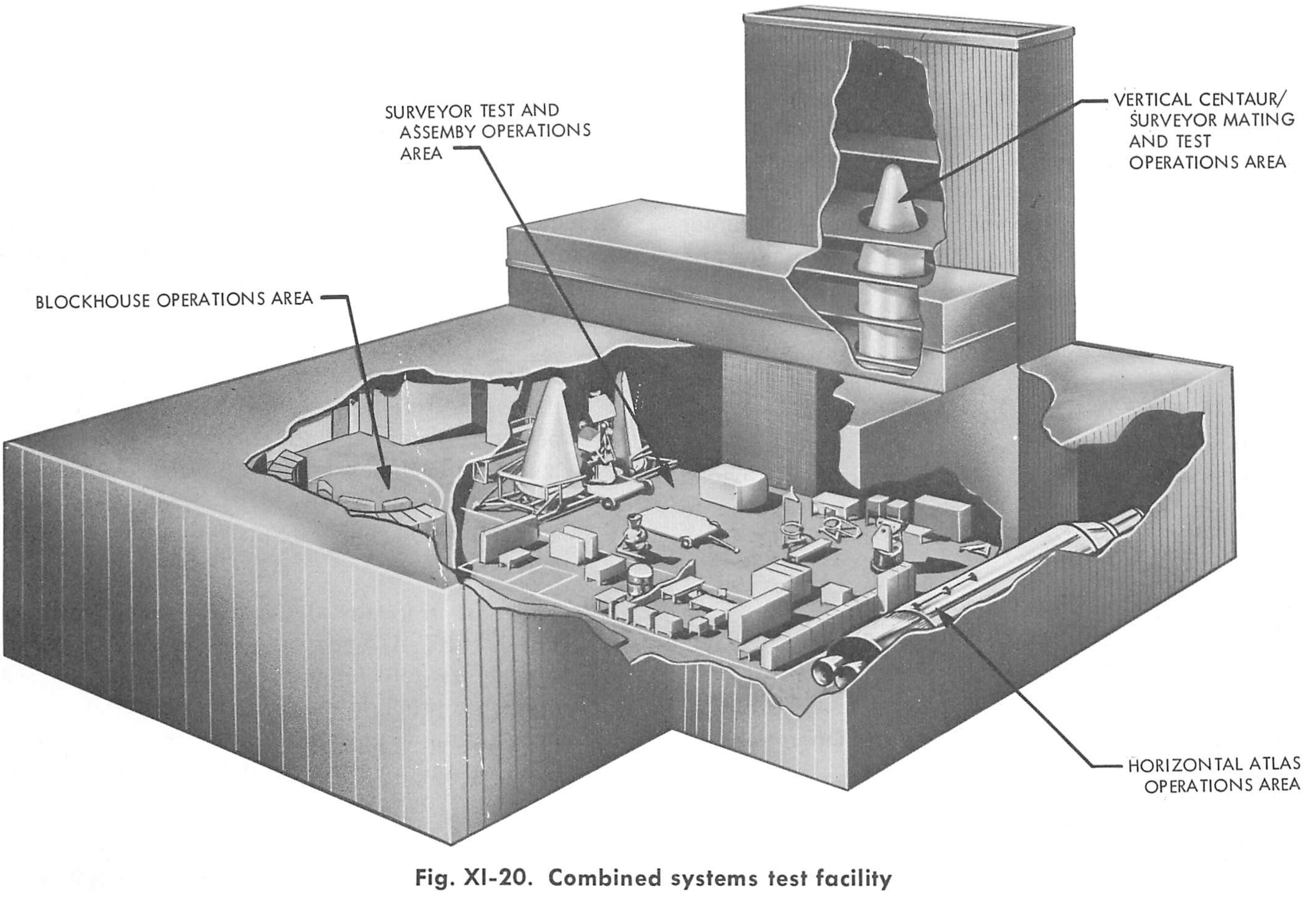


Fig. XI-20. Combined systems test facility

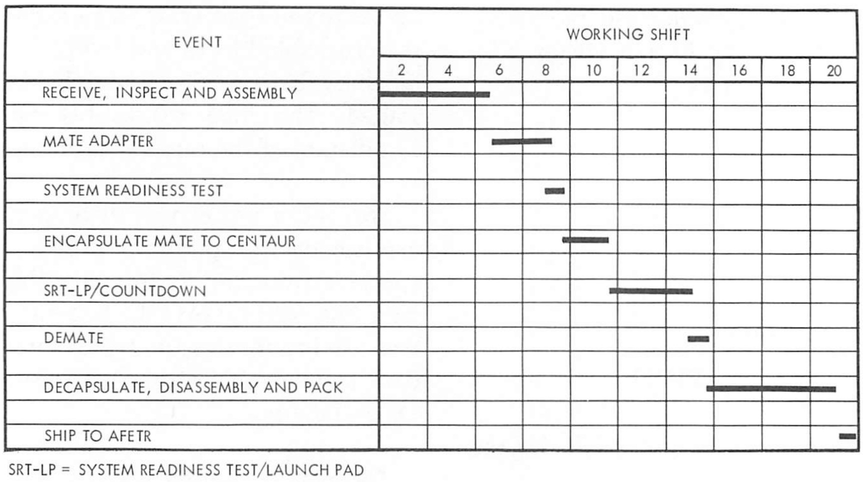


Fig. XI-21. Combined systems test nominal test schedule

completed and the entire assembly lifted to the top of the *Centaur* launch vehicle and mated.

After mechanical mating to the *Centaur* was complete, a Spacecraft System readiness test was performed to verify proper spacecraft performance and to simulate the activities that would be conducted at AFETR.

The CST was a joint test between the *Atlas, Centaur,* and *Surveyor* spacecraft. Each vehicle performed a time coordinated countdown and at simulated liftoff all umbilical plugs were removed. All vehicles operated on internal power and the test was continued until after simulated separation of the spacecraft from the *Centaur* launch vehicle. The umbilical plugs were reinstalled and the test terminated.

After review of the test results and a joint critique of the test, the spacecraft was removed from the *Centaur,* decapsulated and demated from the spacecraft adapter. After disassembly of the spacecraft and preparation for shipment were accomplished, the spacecraft was loaded on an aircraft as described above and transported to AFETR.

The CST facility, launch vehicle only testing at the facility, and joint spacecraft launch vehicle testing are further discussed in Sections VIII-D and -E.

8. Air Force Eastern Test Range Tests and Operations

a. Initial spacecraft preparation and test. After arrival at AFETR (Fig. XI-22) the spacecraft was unloaded at the SCF, located at building AO, for receiving inspection, assembly, and initial tests. Figure XI-23 shows the flow diagram for the AFETR tests and operations and Fig. XI-24 shows a typical schedule of activities from arrival until launch. Additional information on the launch complex facilities is contained in Section VIII-D. Prelaunch testing of the launch vehicle only and joint spacecraft/launch vehicle testing at AFETR is further discussed in Section VIII-E.

During the initial spacecraft reassembly period, the FCSG, AMR, and solar panel were tested in the unit test area to verify flight readiness prior to installation on the spacecraft. After these units were checked and installed, a series of spacecraft performance verification tests were conducted on the spacecraft. The objective of these tests was to verify that the spacecraft had survived the shipment environment and was in flight ready condition. These tests, and a brief description of each, were:

(1) Performance verification test 1 (PVT 1) evaluated the spacecraft power system performance, the power distribution and the power consumption levels for each subsystem. Also checked were the transmitters and receivers for proper performance and the pyrotechnic circuits for proper resistance and operation.

(2) The next test was designated system readiness test A (SRT A) and was used to verify that the spacecraft telecommunications subsystem was operating properly in all modes of operation. The programmed automatic telemetry evaluator (PATE) was used as was the CDS for evaluation of performance.

(3) Performance verification test 3 (PVT 3) was for evaluating seven major spacecraft subsystems in

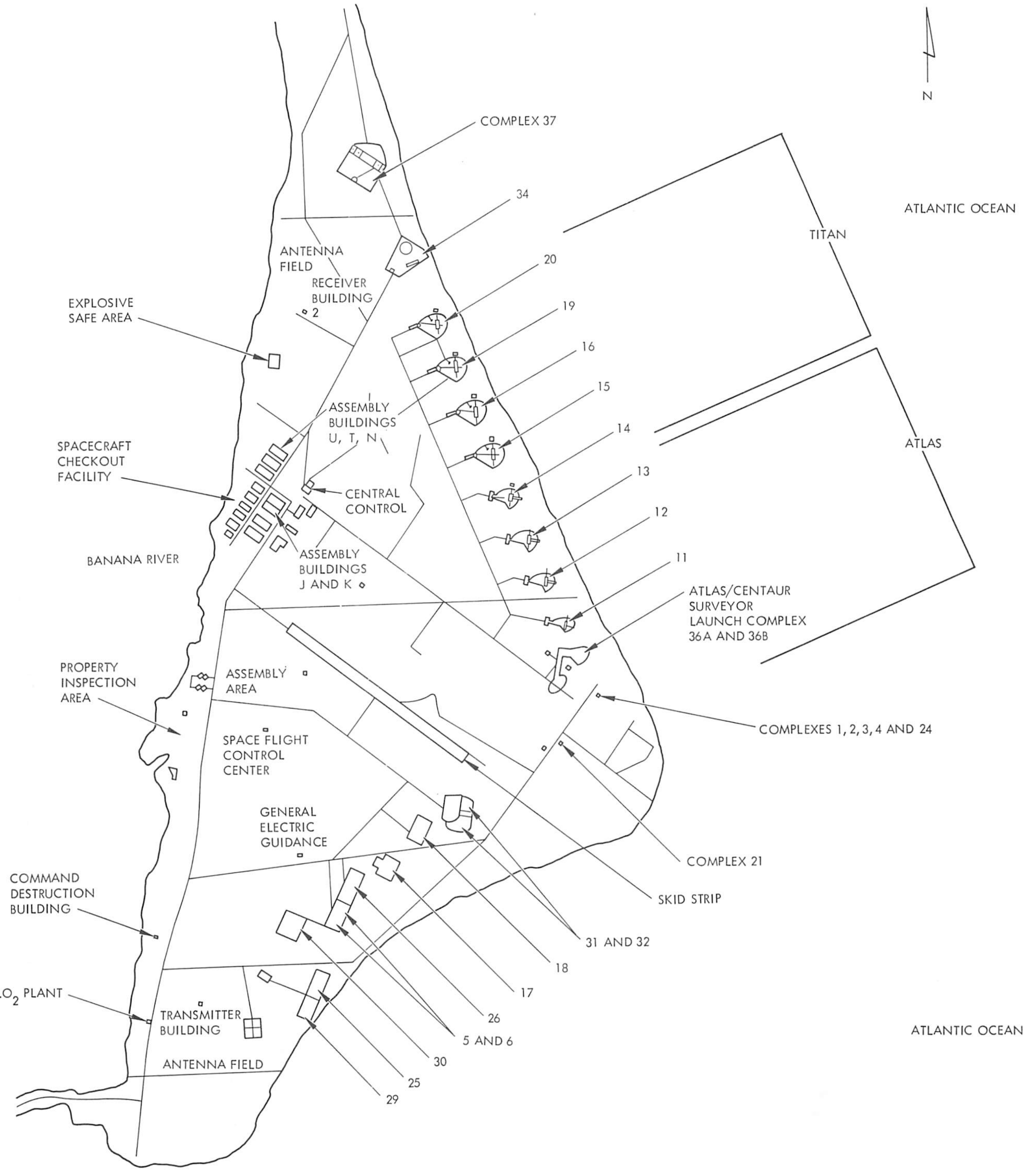
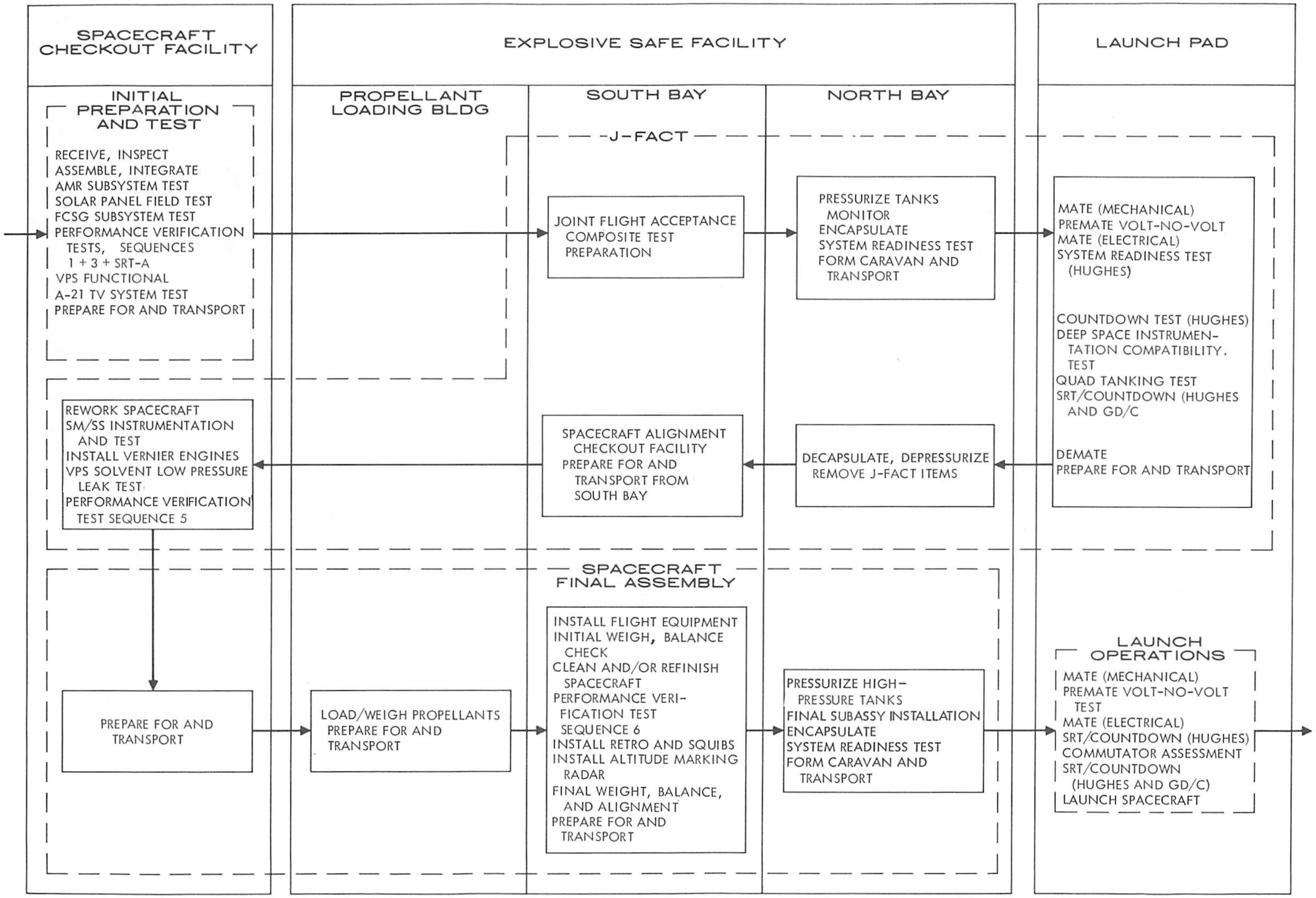


Fig. XI-22. Air Force Eastern Test Range facilities layout

the system environment. These subsystems were flight control, RADVS, AMR, command, signal processing and RF data link. These subsystems were checked during conduction of a simulated mission sequence test.

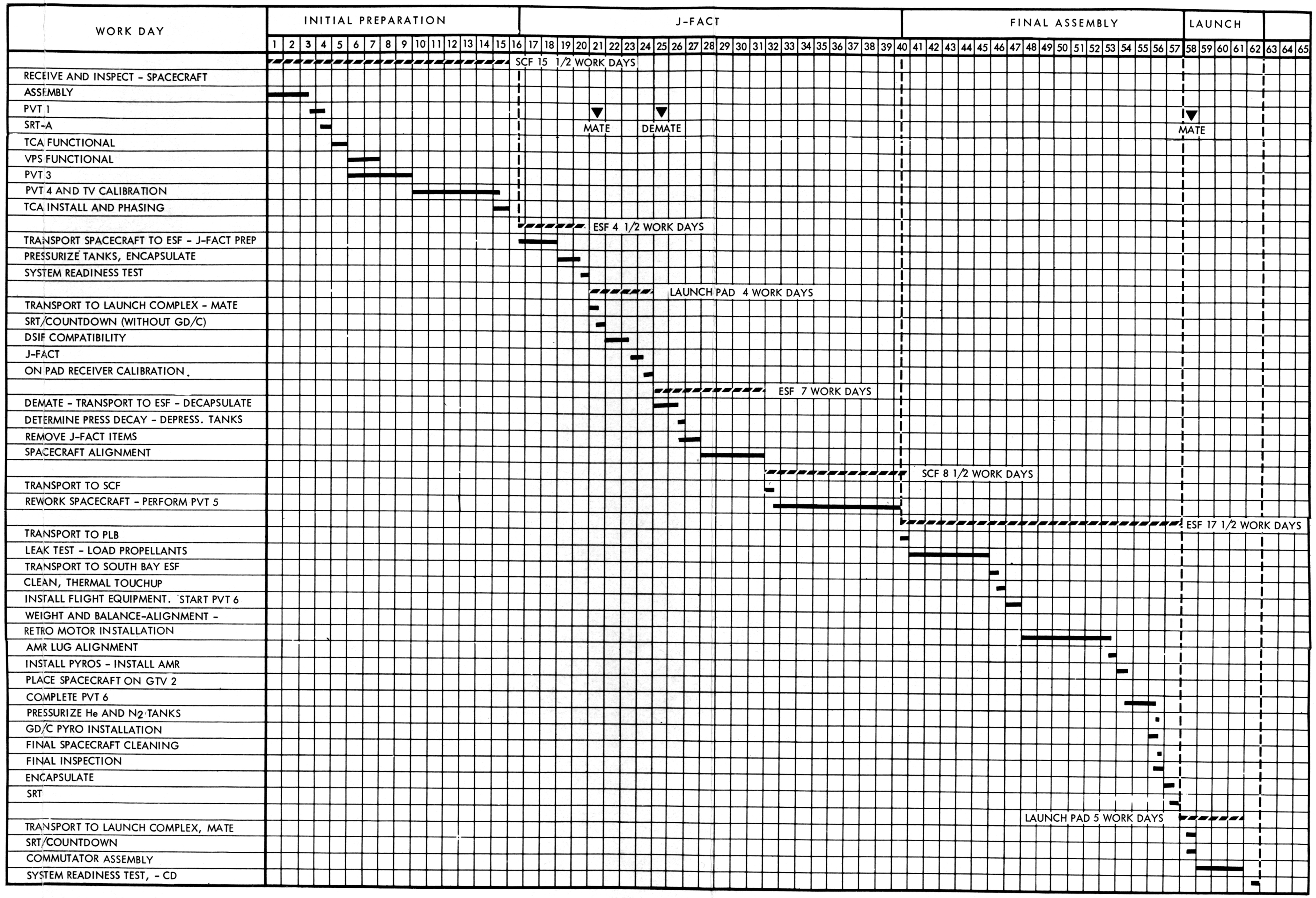
(4) Prior to, in parallel with, and after PVT 3 the following tests were performed:
   (a) Attitude control gas system phasing checks, relief pressures, control pressures and flow rates.



SRT = SYSTEM READINESS TEST
SM/SS = SOIL MECHANICS/SURFACE SAMPLER
VPS = VERNIER PROPULSION SYSTEM
GD/C = GENERAL DYNAMICS/CONVAIR

J-FACT = JOINT FLIGHT ACCEPTANCE COMPOSITE TEST

Fig. XI-23. Air Force Eastern Test Range flow diagram



AMR = ALTITUDE MARKING RADAR
VPS = VERNIER PROPULSION SYSTEM
GD/C = GENERAL DYNAMICS/CONVAIR
SCF = SPACECRAFT CHECKOUT FACILITY
GTV = GROUND TEST VEHICLE
CD = COMMAND DECODING
PLB = PROPELLANT LOADING BUILDING
PVT = PERFORMANCE VERIFICATION TEST
TCA = THRUST CHAMBER ASSEMBLY
ESF = EXPLOSIVE SAFE FACILITY
J-FACT = JOINT FLIGHT ACCEPTANCE COMPOSITE TEST
SRT = SYSTEM READINESS TEST
DSIF = DEEP SPACE INSTRUMENTATION FACILITY

Fig. XI-24. Air Force Eastern Test Range nominal test schedule

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(b) Vernier propulsion system functional tests included oxidizer and fuel system bladder leak checks, helium leak checks, oxidizer and fuel relief and check valve tests, and pressure regulator checks.

(c) Vernier engines-spacecraft functional tests measured fuel and oxidizer flow rates on each engine as each was commanded by spacecraft electronics. This test was conducted with each engine connected electrically to the spacecraft by long lines while the engine was mounted to the solvent flow bench. This test was introduced on SC-3 after the SC-2 flight failure.

(5) Performance verification test 4 (PVT 4) and the TV calibration was next in the test series. This test was a comprehensive and precise calibration of the TV subsystem, the results of which were used after the receipt of lunar pictures to produce more realtistic pictures of the lunar surface.

Following completion of TV testing the vernier engines were installed and the spacecraft was prepared for transport to the ESF for the next test sequence.

b. Joint flight acceptance composite test phase. The joint flight acceptance composite test (J-FACT) involved operations and test in the ESF on the launch complex and a return trip to the ESF before returning to the SCF for final flight preparations. The ESF, which includes a propellant loading facility and two assembly rooms, is shown in Fig. XI-25. Preparations in the ESF included:

(1) Transportation to the ESF, south bay and

(a) Installation of dummy retro motor and dummy AMR to spacecraft.

(b) Main battery installation.

(c) Installation of stray energy monitors, squib mufflers in pyrotechnic circuits.

(d) Footpad installation.

(2) Transportation to the ESF, north bay and

(a) Spacecraft mated to the *Centaur* adapter.

(b) Helium and nitrogen tanks pressurized to flight pressures for a long-term leak test.

(c) Spacecraft encapsulated with the *Centaur* nose fairing.

After these operations, a SRT was performed to verify proper spacecraft operation before transport to the launch complex. During this test, as in all ESF and launch pad tests, the spacecraft performance was evaluated in the SCF, the data were transmitted by RF link to the STEA in the SCF, and processed by the CDS. Following the SRT, the spacecraft was transported to Launch Complex 36A or 36B and mated to the *Centaur* launch vehicle.

After mechanical mating of the spacecraft to the *Centaur*, a volt-no-volt test was performed on the umbilical plugs to assure problem-free operation. Following this, the electrical plugs were connected and a SRT and practice countdown were performed. These were done to ensure proper operation of the spacecraft and to provide training for the spacecraft test and evaluation team. The next test at the launch complex was a spacecraft/DSIF compatibility test. This test, conducted by the spacecraft test team and DSS 71 at AFETR was about 20 hr in duration. All telemetry, command, and RF parameters were examined by the DSS 71 personnel so high confidence existed that no problems would exist between other Deep Space Stations and the spacecraft after liftoff.

Quad tanking of the launch vehicle and exposure of the spacecraft to RF radiation from the AFETR radars was conducted on SC-1 and SC-2. No problems of an RFI or because of the tanking environment were encountered so the test was discontinued on the remaining *Surveyor* spacecraft.

The final on-pad test during this period was the J-FACT. This joint test was to verify proper operation of both launch vehicle and spacecraft during a simulated countdown and simulated flight through spacecraft separation from the *Centaur*. *Atlas*, *Centaur*, and *Surveyor* spacecraft were joined in a simulated countdown and at $T = 0$ the umbilical plugs were removed and all systems operated on internal power as in an actual flight. After simulated spacecraft separation, the plugs were reinstalled and the test terminated. The spacecraft was then demated and returned to the ESF for post-J-FACT operations. The post-J-FACT operations included:

(1) Decapsulation.

(2) Verification that proper squib circuit operation had happened at the launch complex.

(3) Depressurization of the gas systems and calibration of the pressure transducers.

(4) Removal of spacecraft from *Centaur* adapter.

(5) Flight alignment checks on the VPS engines, RADVS, attitude control jets, omniantennas, planar array, secondary sun sensor, and FCSG.

After completion of the alignment, the spacecraft was transported to the SCF for the final preflight preparations and tests.

c. Final preflight preparation phase. The spacecraft was returned to the SCF for the final electrical tests with the STEA. This test sequence was referred to as performance verification test 5 (PVT 5) and consisted of the following tests:

(1) Automatic sun acquisition and star acquisition tests were performed to verify the optical sensors and associated electronics.

(2) Terminal descent tests were performed to verify the spacecraft range/velocity profile descent segments. The spacecraft was placed in the descent configuration and the radars and flight control were operated in an open-loop configuration. Spacecraft performance was verified via spacecraft telemetry.

(3) Flight solar panel and secondary sun sensor operations tests were performed to qualitatively verify the electrical performance of the flight solar panel and secondary sun sensor. The solar panel was illuminated with an exciter and the spacecraft was exercised through the various battery charge regulator modes of operation. The spacecraft performance was evaluated via spacecraft telemetry. The secondary sun sensor was illuminated and its operation verified via spacecraft telemetry.

(4) Propellant valve response and electrical integrity tests were performed to verify proper pull-in/drop-out valve currents. The spacecraft was commanded to exercise the propellant valves and recordings were made of the pull-in, steady-state, and drop-out currents. Following evaluation of the recordings, each valve was connected to the spacecraft harness and resistance measurements were taken to verify their connection.

(5) Motor coil resistance tests were performed on the A/SPP to verify correct resistance of the motor coils while they were cold.

(6) A Canopus pressure switch test was performed to verify proper Canopus sensor pressure.

(7) Roll actuator null, travel limit, phasing, and scale factor tests were performed to verify the performance of the roll actuator. The roll actuator was unpinned and the spacecraft was commanded into a mode to enable the roll actuator. The spacecraft was then rotated using the system test stand and roll actuator performance measured.

(8) Mission sequence tests, consisting of injection, separation, RF acquisition, gyro precession checks, midcourse timing accuracy, and sun acquisition, were conducted to verify that the spacecraft could perform as would be expected in a true mission. The various tests were performed by setting up the spacecraft configuration to simulate the true mission as much as possible and spacecraft performance verified via spacecraft telemetry.

(9) Ranging tests for RADVS were performed to verify radar ranging accuracy. The radar system was allowed to range in free space and into a known delay line and performance was verified via spacecraft telemetry.

After completion of PVT 5, the spacecraft was transported to the propellant loading building in the ESF. Following flight pressurization of the propellant system using solvents, the spacecraft was loaded with a measured amount of oxidizer first and then fuel. The procedure was the same for each propellant. With the spacecraft on the weighing stand, the VPS was evacuated, the system completely filled, and entrapped gas vented from the propellant feed system. Propellant was then offloaded in small increments until the exact amount specified for the particular flight remained on board.

The spacecraft was then transported to the south bay of the ESF assembly laboratory and final flight preparations were continued. The following flight equipment was installed:

(1) Main battery.

(2) Pyrotechnic board (prior to SC-5).

(3) Compartment A cover.

(4) Auxiliary TV mirror (SC-3–SC-7).

(5) Thermal reflector and ballast.

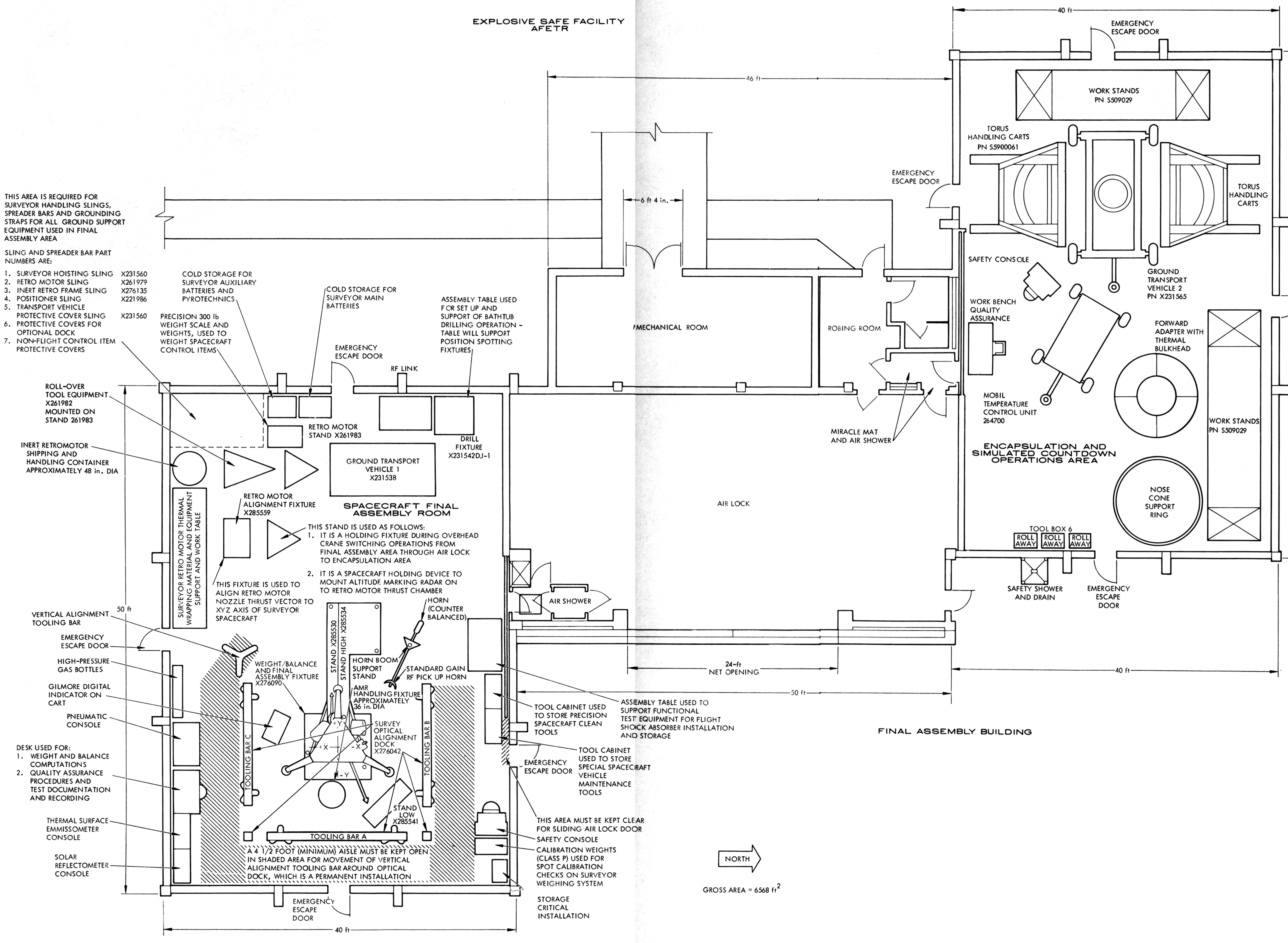
Final thermal finish touchup and cleaning were then performed and PVT 6 was performed which included:

(1) Verification of umbilical, squib circuits, and separation sensing and arming device switch positions.

(2) Calibration of strain gage telemetry channels.

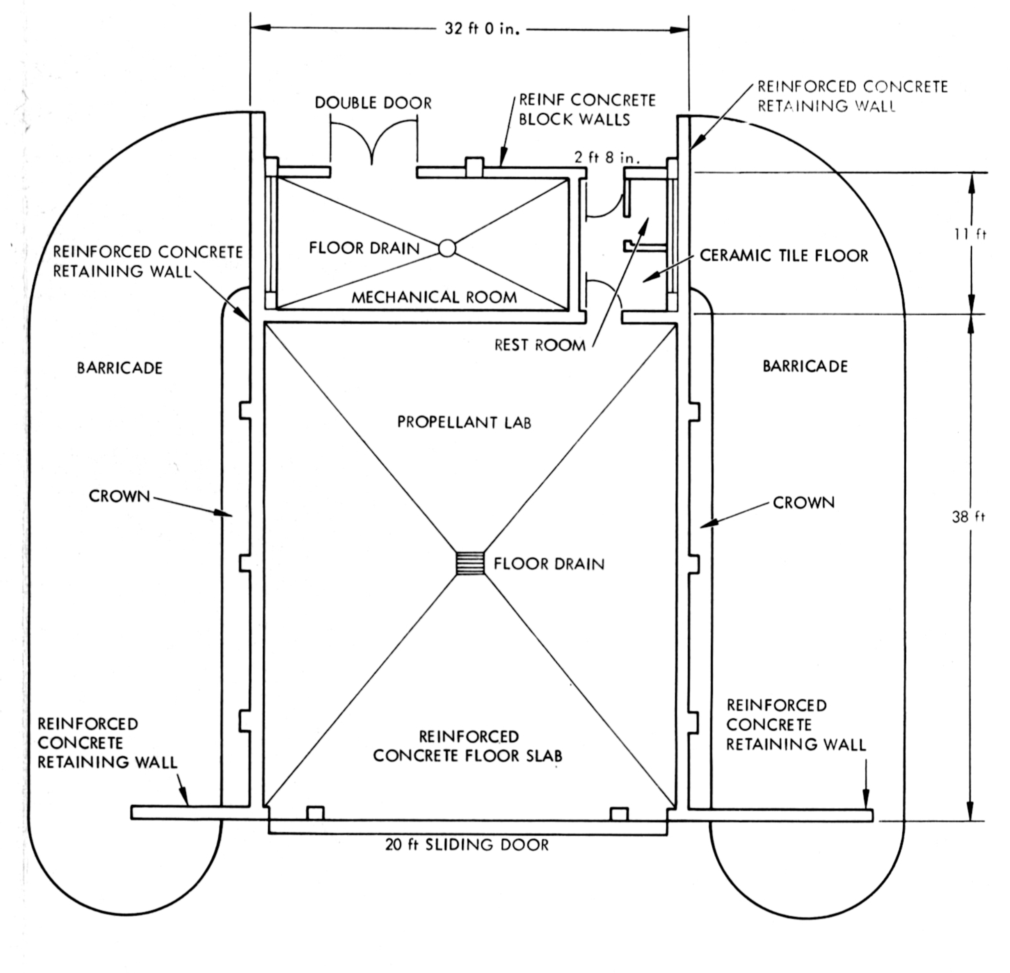
(3) Altitude marking radar open-loop operation.

(4) Radar altimeter and doppler velocity sensor lockup tests.



QTY	GD/A FURNISHED	APPROXIMATE SPACE REQUIREMENTS, ft
1	SLING ASSEMBLY AND SPREADER BAR	4 × 4
1	ERECTOR SLING FOR NOSE FAIRING ASSEMBLY	4 × 6
1	TORUS HANDLING SLING	4 × 5
1	FORWARD ADAPTER SUPPORT RING	5 × 5
1	TORUS RING SHIPPING DEVICE	12 × 12
1	SPACECRAFT SUPPORT ARMS	2 × 6
QTY	HUGHES FURNISHED	SPACE REQUIREMENTS, ft
1 set	CALIBRATION CHECK WEIGHTS (MOUNTED ON CART)	4 × 7
1	GRANITE SURFACE PLATE (QUALITY ASSURANCE)	4 × 5
1	GROUND TRANSPORT VEHICLE (HARD COVER)	5 × 10
1	GROUND TRANSPORT VEHICLE (OUTER HARD COVER)	5 1/2 × 11

COVERS MUST BE REMOVED IN A CLEAN ROOM TO PREVENT INSIDE OF COVER FROM BECOMING CONTAMINATED



PROPELLANT LOADING BUILDING

Fig. XI-25. Explosive safe facility final assembly and propellant loading building

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(5) Thrust chamber assembly vernier engine throttle valve checks.

Following this, the retro motor was installed and a final weight, balance and alignment was performed. During this period omniantenna and roll actuator pinpullers were installed as were helium release and SM-SS squibs and the AMR. The spacecraft was then moved to the north bay where the following was installed:

(1) Crushable blocks.

(2) Shock absorber squibs.

(3) Landing gear pinpullers.

(4) Auxiliary battery.

Encapsulation was performed after the spacecraft was checked to assure flight configuration and removal of all nonflight items. An SRT was performed to demonstrate that the spacecraft was ready for mating to *Centaur*. The SRT was performed by RF data link between ESF and the STEA/CDC in SCF.

d. Launch preparations. This phase of launch operations prepared the spacecraft and *Atlas/Centaur* system for the launch. It verified that the spacecraft was ready to be launched and had remained completely operational until the moment of actual launching.

The launch preparation phase began with the transport of the spacecraft to the launch complex. The spacecraft remained at the complex until launch. The following operations were performed.

After completion of the SRT postencapsulation at ESF the spacecraft was placed on GTV 2 and transported to the launch complex. The spacecraft was connected to the groundwire of *Centaur*, hoisted and mechanically mated to *Centaur*. A volt-no-volt test was made before electrical mating. The *Surveyor* operations console in the blockhouse monitored and controlled the spacecraft through the *Centaur* umbilical connector until launch.

To confirm that the spacecraft was in a launch-ready condition, a SRT launch pad (SRT/LP) was run. This verified that the major subsystems were in a state of readiness.

To verify that the spacecraft could be counted down in a launch configuration, a countdown was performed. The countdown verified or exercised the following major systems:

(1) RF data link.

(2) Vehicle power.

(3) Accelerometer checks.

(4) Commutator assessment.

(5) Spacecraft power system checks.

(6) Commutator switch checks.

(7) Launch preparation.

T-minus-4-day retro arm check was performed.

With the successful completion of the above checks, the spacecraft was ready for launch. The launch operation was a combined operation of GD/C, HAC, and JPL. The spacecraft was verified prior to countdown by a SRT/LP.

The actual launch countdown procedure was then initiated and terminated with the launch of the spacecraft.

C. Special Vehicle Tests

1. Dynamic Models and Mass Models

The primary objective of the *Surveyor* dynamic model flights was to monitor the environment during the *Atlas/Centaur* boost phase (including spacecraft separation) using a vehicle that simulated the dynamic characteristics of the flight spacecraft. The program included two dynamic model flight vehicles (SD-1 and SD-2), two test vehicles (SD-3 and SD-4), and four flight sets of S-band transponder assemblies. Two of the sets were installed on SD-1 and SD-2; the other two were provided to GD/C for installation on the mass models used with A/C-8 and A/C-9 launches. Flight dynamic models were equipped with an S-band transponder and power source to permit tracking after separation from the *Centaur* for at least 20 hr.

The dynamic model launches also provided a basis for an early verification of the test vibration specification for the spacecraft system and the interface adapter. Vibration data during the flights were furnished by dynamic transducers mounted on the spacecraft and adapter, the outputs of which were telemetered by the *Centaur* telecommunication system.

a. Dynamic model description. The dynamic models were designed to simulate the structural response and mass properties of the spacecraft. The basic spaceframe was similar to that of the flight spacecraft, with a difference being the exclusion of certain reinforcing substructure (Fig. XI-26). Dummy masses were mounted on the spaceframe to simulate the mass properties of the components they represented. The planar antenna, solar panel, and mast assembly were similar in appearance to the *Surveyor* counterpart at launch. An omniantenna, which was used with the S-band transponder, was mounted at the top of the vehicle on a short mast.

The vehicle was instrumented with temperature sensors, accelerometers, microphones, and strain gages to monitor reaction during the boost phase.

A dummy retro motor was mounted within the vehicle in approximately the same position as the spacecraft retro motor. The simulated retro motor contained the electronic equipment required to convert the vehicle transducer inputs into the appropriate frequency modulated subcarrier signals for modulation of the telecommunication transmitter. The 1300-lb motor simulator also included an adjustable assembly of removable weights to permit off-loading while retaining the ability to properly locate the overall center of gravity of the dynamic model.

b. Dynamic model qualification, acceptance, and development tests. The qualification, acceptance, and development tests were accomplished as follows.

Flight acceptance vibration testing of the SD-1 dynamic model was completed on May 8, 1964, to demonstrate proper manufacture and assembly of SD-1. The test was performed with the SD-1 model mounted on the vertical vibration test fixture, and consisted of a single sinusoidal vibration sweep of increasing frequency along the Z-axis (thrust).

Modal surveys conducted with SD-1 (completed on August 27, 1964) were used to compute the rigid body modes of the dynamic model.

Flight acceptance vibration testing of the SD-2 model, which was identical in scope to the SD-1 test, was successfully completed on May 28, 1964.

Dynamic model SD-3 was designed and fabricated early in the program to grossly simulate a 2100-lb spacecraft with respect to weight and center of gravity location. The model was used in conjunction with the *Centaur* low-level vibration tests and was delivered to Lewis Research Center in 1963.

Type approval vibration tests were conducted with SD-4 to qualify the dynamic model structural design for launch vibration environment. The unit was subjected to type approval vibration tests along the *Z*, *X*, and *Y*-axes. During the *X*-axis tests, a failure occurred in the mast. Design changes were incorporated and proven satisfactory by repeating some portions of the test. The tests were completed satisfactorily in February 1964.

c. Dynamic model test flights. Launch of the SD-1 dynamic model with the *Atlas/Centaur* A/C-5 boost vehicle was attempted at 16:25:00 GMT on March 2, 1965 from Launch Complex 36A. The launch failed when an engine on the *Atlas* shut down immediately after liftoff and the launch vehicle fell back on the pad and was destroyed.

The SD-2 dynamic model was launched from AFETR Launch Complex 36B on August 11, 1965, using the *Atlas/Centaur* A/C-6 boost vehicle. Launch occurred at 14:31:04 GMT.

The SD-2 dynamic model spacecraft was tracked by the DSIF stations at Johannesburg (DSS 51), Goldstone (DSS 11), and Tidbinbilla (DSS 42). Initial two-way lock by DSS 51 was accomplished 31 min and 12 sec after launch. Spacecraft operation was normal until 16⅓ hr after launch, at which time loss of signal occurred. Since the operation of the dynamic model was predicted to be greater than 30 hr, a failure analysis was conducted. The problem was identified as a thermal problem, internal to the transponder, and corrective action was taken on all remaining transponders.

d. Mass model test flights. Hughes provided only the transponders and power sources. General Dynamics/Convair provided the mass models used for the A/C-8 and A/C-9 test flights. The flights used spare transponder units remaining from the earlier SD-1 and SD-2 launches.

The A/C-8 test flight was the first flight using the two-burn parking orbit ascent mode. The flight, launched on April 7, 1966, was partially successful even though the *Centaur* second burn did not occur as planned. Transponder operation during this mission lasted approximately 26 hr.

The A/C-9 launch vehicle successfully launched its mass model spacecraft via parking orbit and injected it into the desired lunar transfer type trajectory. This was accomplished on October 26, 1966.

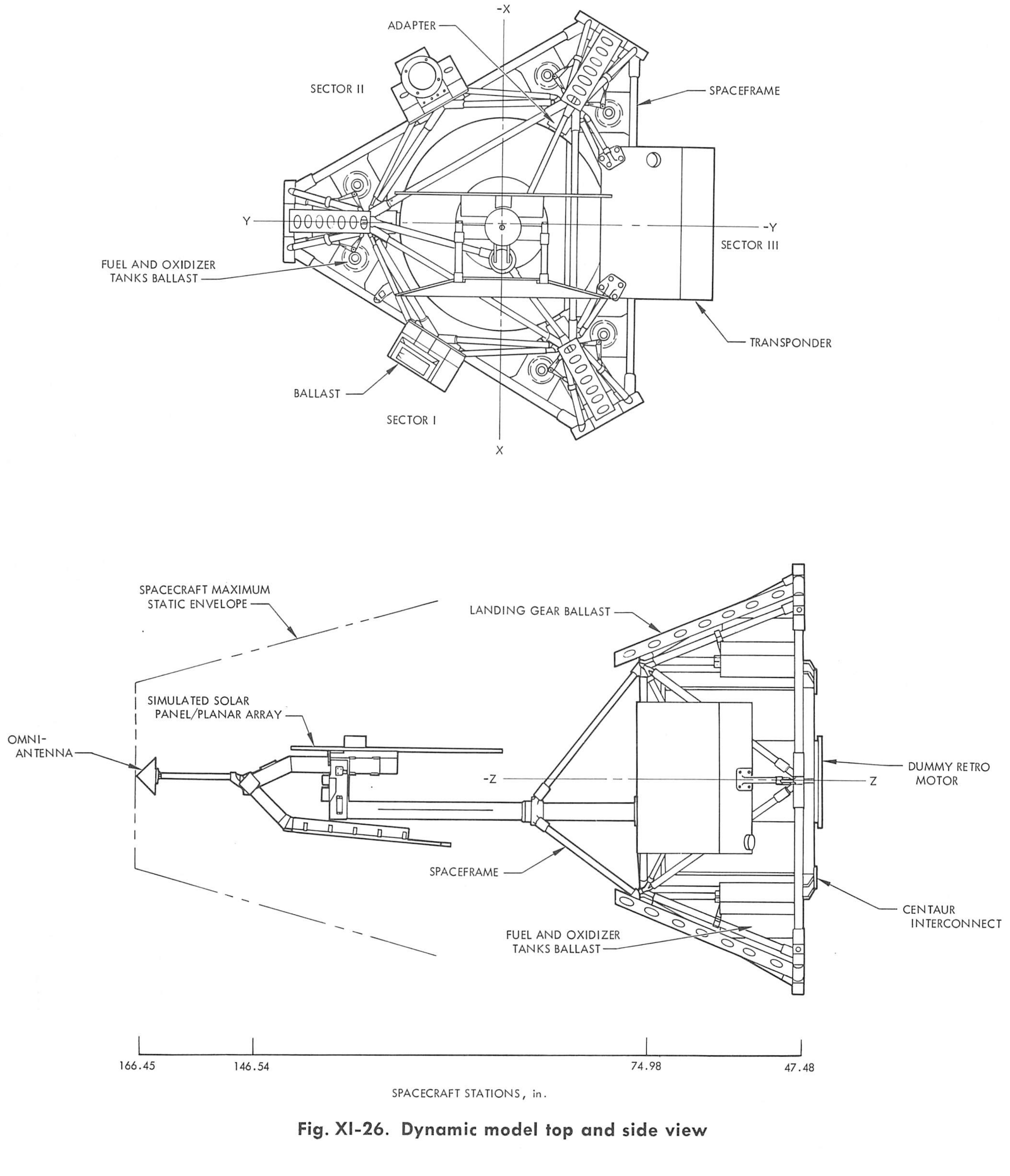


Fig. XI-26. Dynamic model top and side view

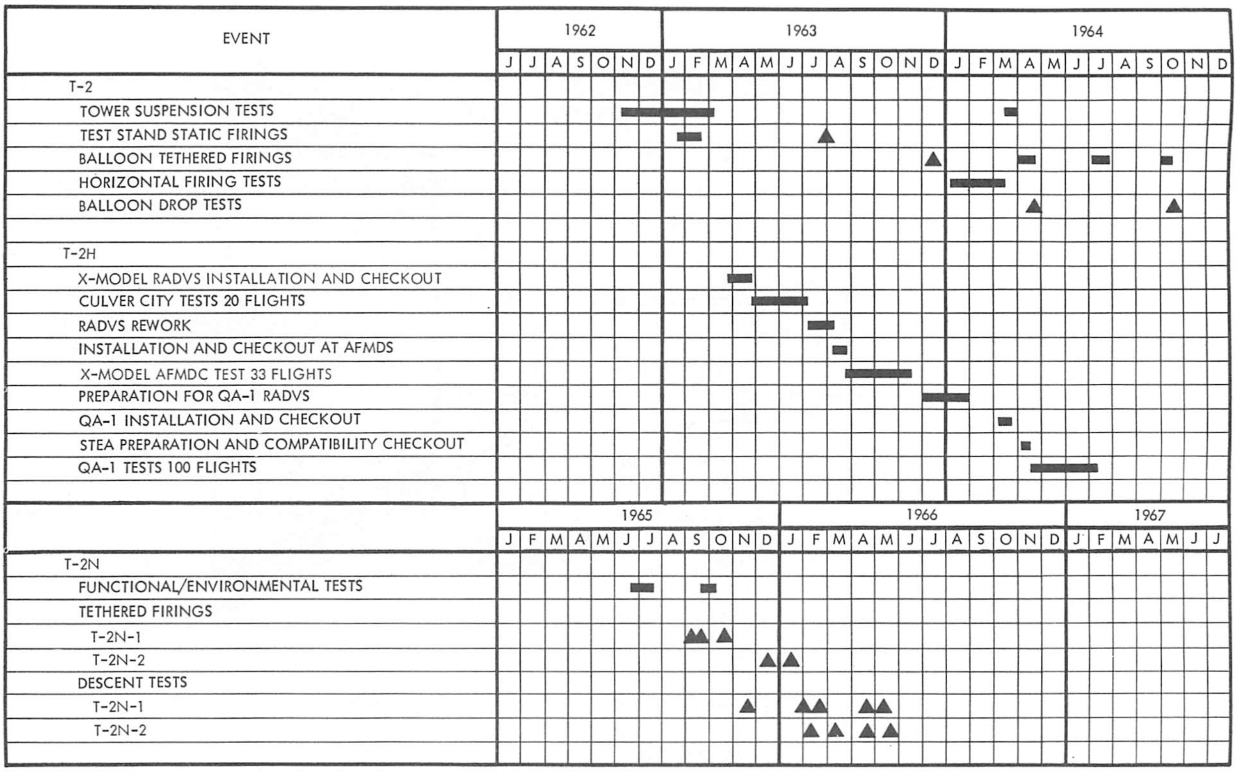
2. Test Program, T-2/T-2N

The primary objective of the T-2/T-2N test program (Fig. XI-27) was to verify the capability of the *Surveyor* RADVS, flight control, and propulsion subsystems to perform the final phase of the lunar soft landing.

a. Test vehicle description. The basic vehicle, designed and fabricated for use on the T-2/T-2N program, was a pyramidal aluminum structure on which were mounted the RADVS, flight control, and vernier engine propulsion subsystems. The vehicle also contained the T-2/T-2N command, telemetry, power, and recovery subsystems required for vehicle control during the missions. The three test flight subsystems were functionally identical to their *Surveyor* equivalents, with minor exceptions required because of the earth environment. The vernier engines provided the controlled thrust for deceleration and stability control during vehicle descent. The pressure feed system was composed of a single helium tank, two propellant tanks (each containing a positive expulsion bladder), and the necessary valves, interconnect lines, quick disconnects, and instrumentation. Three sets of independently actuated throttle valves received thrust control signals from the flight control subsystem. The feed system was pressurized by squib firing a helium release valve. Pressure dump valves were provided for safety purposes.

The flight control subsystem consisted of the inertial reference unit, flight control electronics, and roll actuator. The flight control electronics received and processed inputs from the inertial reference unit, and RADVS, and delivered the thrust control signals to the vernier engine throttle valves. The roll actuator mechanism swiveled engine 3 for vehicle roll attitude control. The inertial reference unit included three gyros that detected vehicle motions about the roll, pitch, and yaw axes, and an accelerometer that detected acceleration along the vehicle Z-axis.

The flight control subsystem operated in three different control modes: inertial attitude control, lateral velocity attitude control, and trajectory control. The inertial attitude control mode was used to control vehicle attitude



RADVS = RADAR ALTIMETER AND DOPPLER VELOCITY SENSOR
AFMDS = AIR FORCE MISSILE DEVELOPMENT SYSTEM
STEA = SYSTEM TEST EQUIPMENT ASSEMBLY
QA = QUALITY ASSURANCE
AFMDC = AIR FORCE MISSILE DEVELOPMENT CENTER

Fig. XI-27. Test program schedule, T-2/T-2N

from the time of engine ignition prior to release until RADVS lockup; it was also used when the vehicle longitudinal velocity V_Z was reduced below 10 ft/sec. Attitude control signals were derived from the gyros. The lateral velocity attitude control mode was operative during the RADVS controlled portion of the flight, when RADVS X- and Y-velocity signals were used to command pitch and yaw attitude changes that tended to null the X- and Y-velocity components. In the trajectory control mode, the vehicle Z-axis velocity versus altitude followed a preprogrammed profile.

Radar altimeter and doppler velocity sensor output signals, proportional to slant range and velocities in the three axes, were fed to the flight control electronics for processing and control of the vehicle during flight. The RADVS had a minimum velocity lockup range. Upon lockup, control signals were developed which activated the switching between RADVS output and flight control input.

The electrical power subsystem furnished the electrical power, switching, and power distribution to operate the electrical equipment during countdown, flight, and recovery. A high-capacity battery in the gondola provided the electrical energy required for operation during the major part of the countdown. Minutes before release, operation was switched to the vehicle main battery (internal power). An independent 28-V battery was supplied in the gondola for operation of the command receiver. A 22-V emergency recovery battery that was not connected to the main 22-V bus, was also provided. In the event of main battery failure, or a short of the main bus, recovery could still be accomplished.

The telemetry subsystem provided 75 channels, and consisted of two commutators, each operating into its own independent transmitter. Signals to be telemetered were converted into a pulse-duration-modulated/frequency modulated/phase modulated (PDM/FM/PM) format for transmission to the operations console. Selected signal processing circuitry was included to provide data for verification of commands, monitoring of selected parameters during countdown and flight, and postflight data evaluation.

The recovery subsystem, activated by an RF command link from the ground operations console, fired squibs to activate the recovery parachute deployment and inflation of the air bag.

The command subsystem consisted of the command receiver and the required logic circuitry necessary to receive, process, and route to the vehicle those command signals required to execute the sequence countdown events and release. The command receiver contained 10 standard Inter-Range Instrument Group (IRIG) channels; the command assembly provided the required circuitry for processing the outputs from the command receiver.

The vehicles were assembled at the HAC, El Segundo facility and transported to the test site at the Air Force Missile Development Center (AFMDC), Holloman Air Force Base, N. Mex., where the test was conducted.

b. Tests (T-2). The T-2 test operations (Fig. XI-27) began with subsystem testing, and then progressed to larger portions of the system, the final test being a drop test which simulated the lunar soft landing. The test plan was divided into four major phases: (1) tethered hover tests, (2) drop tests with the RADVS open-loop, (3) drop tests with RADVS closed-loop control, and (4) soft-landing drop tests. The tethered hover tests were to be performed with the vehicle suspended from a tower. Vernier engine operation in the flight control inertial mode was evaluated. A series of free-fall drop tests was planned in which the vehicle was to be released from a balloon, under attitude control in inertial mode, and recovery made at the required altitude after completion of the tests. The objective of this group of tests was to evaluate radar performance under dynamic conditions, but without the RADVS providing inputs to flight control. Free-fall drop tests evaluating all phases of trajectory control were also planned. Two drop tests simulating lunar landing conditions were planned. The parachute portion of the recovery system was not to be activated and the vehicle was to be released from an appropriately lower altitude. The test consisted of raising the vehicle to 1500 ft by means of a balloon, performing a countdown during which all subsystems (including vernier engines) were turned on, and then releasing the vehicle and allowing it to descend under its own control to perform the programmed descent required for the soft landing.

During the T-2 testing described as follows, several major problems arose which prevented accomplishment of all the major test objectives.

1. Tethered hover test. During the period from November 9, 1962 through March 5, 1963, six T-2 vernier engine test firings were performed at AFMDC. The original objectives of the tests, as stated in the T-2 test plan, were as follows: to determine general vernier engine operation, to evaluate the stability of the inertial mode under varied thrust conditions, to check the effects of engine-induced vibration, to evaluate operation of the auxiliary system, and to obtain a propellant utilization factor. For these tests the vehicle was suspended from an *Aerobee* tower 30 ft above the terrain. Ballast was used in place of the RADVS system.

During these tests, problems were encountered with vehicle stability and as a result a static firing test program was initiated to investigate the engine system.

2. *Static engine firing test.* On January 23, 1963, four static engine firings were performed to determine if hysteresis could be reduced to an acceptable level by varying dither amplitude or frequency. Hysteresis was reduced, but not to an acceptable level. Two of the three restarts were rough (erratic thrust chamber pressure). Defueling procedures were modified to include a liquid solvent flush in an effort to reduce sticky vernier engine throttle valve spools.

A test plan was devised to investigate the cause of the uncommanded thrust transient and the rough restarts. Additional propulsion instrumentation was incorporated and provisions were made to flow propellants through the bleed ports of one engine. On February 20, 1963, a test was conducted with the T-2 mounted on the static test stand. Four liquid flow tests and five hot firings were made with the T-2 electronics either on or off. A rough start occurred only on the first hot firing. Results of the test were inconclusive.

Instrumentation data review indicated additional instrumentation and modification of the testing sequence were required. On March 1, tower test 5 was performed. Instrumentation data indicated that the first rough start may have been due to gas left in the system because of propellant loading procedures and that subsequent rough starts may have been due to gas originating at various points in the system from the thermal environment.

On the following day four firings were conducted using a new loading technique (fill and bleed) that eliminated any gas in the system from fueling. No rough starts occurred on the first or fourth firing; rough starts were experienced on the second and third.

Sufficient data were obtained on the rough start problem, and testing at AFMDC was suspended pending receipt of modified throttle valves.

3. *Static engine firing retest.* Vernier engines with bellows assemblies were delivered to AFMDC and a static firing retest was performed. Ramp and sawtooth inputs were applied and results indicated that hysteresis, deadband, and discrete command response of the redesigned throttle valve were well within specification.

4. *Helicopter testing (T-2H).* After suspension of tether testing because of the vernier engine problem, the T-2 test program plan was modified to include special test phase to evaluate RADVS performance in a dynamic test operation using the helicopter-mounted unit. This testing was particularly important since RADVS operation was carried out at elevations and velocities greater than those used in the T-2 drop test.

The T-2H helicopter testing was divided into two phases, the first used the X model RADVS, and the second used the improved QC-1 model. The RADVS system and associated equipment were mounted on a special test fixture attached to a helicopter. A helicopter was selected as the flight vehicle since the terminal descent trajectories of an actual space vehicle could be simulated and the aerodynamic stress restrictions imposed by the antenna material could be observed. The Z-axis (roll) of the special test fixture was mounted at various angles with respect to the vertical to allow special trajectories to be planned and flown with the RADVS beams oriented at various angles of incidence.

5. *Horizontal static firing tests.* During a balloon tether test in preparation for the descent test phase, the RADVS doppler trackers locked up erroneously when vernier engine ignition occurred. Investigation into this problem of RADVS susceptibility to vernier engine noise led to the implementation of the horizontal static test phase.

An extensive investigation was conducted by operating T-2 in a horizontal position on a static test stand. The horizontal mounting position allowed the radars to be operated so there would be no objects in the paths of the radar beams. In this way, the direct acoustic or mechanical vibration effects on the RADVS receiver could be assessed. Effective acoustic isolation was obtained with the use of Stafoam.

After evaluation of test results, the RADVS was considered ready for the balloon drop test.

6. *Descent tests.* On April 28, 1964, a test of the descent system using the T-2 vehicle was attempted. The vehicle completed ground system checks in which all mission modes, including the emergency recovery provisions,

were checked. Approximately 40 min before the scheduled drop, the vehicle fell free from the gondola. The test director sent the command for emergency recovery, but the main parachute had already separated from the craft, leaving it in free fall. The vehicle was destroyed by impact and fire. A failure review board was appointed to analyze the cause of the failure and to recommend corrective action. The investigations failed to establish positive evidence of a single initiating event. However, laboratory and field tests revealed susceptibility of the auxiliary control circuits to static discharges.

The loss of the T-2 vehicle required that the T-2S (spare) be assembled in order to continue the test program. Protective measures against static discharge were taken in the form of circuit desensitization and addition of a trailing ground wire between the vehicle frame and earth. The vehicle was assembled, integrated, and tested in preparation for the drop test.

On October 22, 1964, the T-2S drop test was accomplished at AFMDC. The vehicle was released after a successful tether. A series of malfunctions in the vehicle and recovery system resulted in a touchdown at approximately 57 ft/sec which caused substantial damage to the T-2S vehicle and subsystem components.

At this point, the T-2 program was suspended, and after a review and planning phase, the T-2N program was initiated.

c. Tests (T-2N). The T-2N program was essentially a continuation of the T-2 program, the objective being to accomplish descent tests in which the vehicle was allowed to perform an unconstrained terminal descent profile under its own control. An upgrade period was initiated to improve the overall quality and reliability aspects of the program. The general physical and functional configuration of the vehicle remained the same as that of T-2, but redundancy and fail-safe features were incorporated into the new design, particularly in the recovery subsystem, and the quality of the hardware was improved.

On completion of the redesign phase, fabrication and assembly of the new vehicles were begun. Two vehicles (T-2N-1 and T-2N-2) were built so a number of tests could be accomplished within a short time span by performing a test with one vehicle while the second was undergoing the refurbishment/retest cycle at HAC, Culver City. Each assembled vehicle was subjected to a functional and environmental test phase at HAC to qualify it for the test operational phase at AFMDC (Fig. XI-27).

The tests were of three types: (1) tether, (2) descent, and (3) descent/touchdown. The tether test was the final phase of system test before the descent test. The vehicle was raised to altitude by a balloon, and a countdown was performed that included turnon of all subsystems, certain subsystem checks, and engine ignition. Proper response to these events was verified, followed by a shutdown, terminating the test. The descent/recovery test was a terminal descent test performed at a sufficiently high starting altitude (approximately 1450 ft) to allow completion of the programmed descent at an altitude of approximately 600 ft, at which time vehicle recovery was commanded. The descent/touchdown test was similar to the descent/recovery test except that the descent was started at an altitude of 850 ft and the descent terminated at earth touchdown. The T-2N test program was successfully carried to completion with no major perturbations.

The system functional and environmental tests were conducted to qualify the T-2N vehicle. Functional testing provided for group testing, or initial checkout, and system level testing. The following five tests constituted the environmental tests:

(1) A thermal test was conducted to verify operating performance within specified tolerances at temperature extremes of 20 and 100°F.

(2) A vernier engine vibration test was performed to verify operating performance within specified tolerances during vernier engine vibration input period. A closed-loop mission sequence was performed during vibration.

(3) A RFI and static discharge test was performed with the vehicle installed in a screen room and subjected to a simulated AFMDC RF interference environment. Operation within specified tolerances was required during the test.

(4) An acoustic test (T-2N-1 only) was performed with the vehicle mounted on an inverted test fixture and the RADVS turned on. An acoustic environment, simulating that of the vernier engines, was created by F-104 engine runs. The RADVS was monitored for lockup on acoustic noise.

(5) A landing shock test was performed by dropping the vehicle from a height of 6 ft with landing air bags inflated. This test duplicated the expected landing shock velocity of 20 ft/sec. This was a nonoperational test. A functional pre- and post-test was performed to verify unchanged vehicle performance.

The complete system functional and environmental test phase was carried out with no major incidences occurring. The T-2N vehicles met their requirements under the specified environmental conditions.

The history of the test operation at AFMDC is presented chronologically by vehicle tether and descent tests. A number of repeat tether tests were required because of problems encountered. These problems were satisfactorily resolved prior to continuing into the descent test phase. All descent tests were successful.

The first of the T-2N tether and descent tests was a T-2N-1 tether test scheduled for September 3, 1965. The test was aborted September 2, 1965 during the closed-loop MS of the countdown, when the klystron power supply/modulator (KPSM) malfunctioned. This first test was rescheduled and conducted on September 10, 1965.

The second T-2N-1 tether test was performed on September 23, 1965.

The third T-2N-1 tether test was conducted on October 26, 1965, and considered successful. Minor deficiencies encountered in tests 1 and 2 were corrected and both the vehicle and ground control equipment were considered ready for a descent test.

Descent test 1 attempted on November 12, 1965, was aborted by telemetry frequency interference. Rescheduled for November 22, the test was successfully completed with no major malfunctions or anomalies. The vehicle was returned to HAC, Culver City, and testing was started on T-2N-2.

The first tether test with T-2N-2 was performed on December 14, 1965, with unsatisfactory performance of both the propulsion and RADVS subsystems.

The second T-2N-2 tether test, conducted on January 13, 1966, was aborted when the gondola main battery failed during the countdown. On January 17, the mission was successfully completed with no malfunctions or major anomalies, and the vehicle was judged ready for descent testing.

The second T-2N-1 descent test was completed on February 1, 1966. Because of findings from this test mission, modifications were made to improve the recovery RF link.

Originally scheduled for February 9, 1966, the first T-2N-2 descent test was never successfully completed. It was cancelled because of high winds on February 9, aborted because of balloon leakage on February 10, and terminated on February 11. The test mission was not rescheduled.

The third T-2N-1 descent test was conducted on February 24, 1966. All vehicle and ground control subsystems functioned satisfactorily during the descent. The test was judged a success and the vehicle returned to HAC, Culver City for incorporation of landing modifications.

The second T-2N-2 descent test was performed on March 7, 1966. Operation of all subsystems was judged satisfactory during the descent, and the vehicle was returned to HAC, Culver City, for incorporation of modifications for the touchdown tests.

The fourth T-2N-1 descent test was scheduled for April 13, 1966, but was cancelled because of high winds. The test was rescheduled for April 14, but was aborted when conditions required total emergency recovery of the vehicle. Minor damage was incurred due to twisting and swinging of the vehicle beneath the balloon, but no damage was incurred on landing. The balloon ruptured after vehicle release and the gondola fell to the ground, incurring severe damage. No attempt was made to recover the gondola with the balloon branch parachute due to insufficient altitude. A slack tether line was also entangled around the parachute. The test was rescheduled for April 21, and successfully completed. Vehicle performance was satisfactory, and touchdown modifications were verified.

The third T-2N-2 descent test was successfully completed on April 26, 1966. This was the first test for this vehicle with the touchdown modifications incorporated.

d. Conclusions. The objectives of the T-2/T-2N program were met. All parameters of the integrated RADVS, flight control, and propulsion subsystems were determined to be within specifications during the test program. A number of significant design and test procedure improvements were incorporated into the *Surveyor* flight equipment as a result of both the T-2 and T-2N test phases.

Performance characteristics of RADVS not evaluated in the descent tests were thoroughly evaluated and found to be satisfactory during model T-2H helicopter testing of the RADVS QA-1 model.

The propulsion subsystem servicing methods and techniques used for flight spacecraft were developed on the

T-2/T-2N program. Included in this category were procedures for functional testing of the upstream pressurization system, vacuum fill and bleeding procedures, and off-loading procedures. The methods of predicting helium and propellant utilization for the spacecraft were the same as those used on the T-2N vehicle. The accuracy demonstrated in the T-2N predictions added confidence to the spacecraft predictions.

Confidence was obtained in the closed-loop MS. The descent profile resulting from this test was found to agree closely with the descent profile of flight spacecraft on *Surveyor* missions.

D. Prototype Spacecraft Tests and Operations

The objectives of the prototype spacecraft, designated T-21, were to qualify the *Surveyor* functional, mechanical, and thermal design through a series of type approval tests in an ambient environment and at environmental levels more severe than those anticipated during the actual mission. The T-21 spacecraft tests also proved the equipment, facilities, techniques, and procedures to be used for flight acceptance tests of the flight spacecraft. Each of the phases of the T-21 test program is summarized. The T-21 test schedule is included in Fig. XI-28.

1. System Functional Tests

Initial spacecraft checks were started on January 2, 1963, and consisted of integration of the spacecraft units with each other and integration with the STEA. The ISCO phase started in December 1963 and continued until July 1964. Included in this initial test period was a match-mate test between the *Surveyor* spacecraft and the *Centaur* adapter and nose fairing (also see Section VIII-C).

2. Mission Sequence Testing

The objectives of MS testing were to verify that the T-21 spacecraft would perform properly in a simulated MS and to demonstrate that procedures, STEA and facilities were adequate to support the flight spacecraft test program. This test sequence was performed during July 1964. Some of the MS testing was performed in a screen room where the spacecraft was exposed to an RF environment more severe than the anticipated RF environment of the launch complex at AFETR.

3. Static Discharge Tests

The objectives of the static discharge tests were to determine the susceptibility of the spacecraft to electrostatic discharges from it to other bodies and the susceptibility to arc discharge in proximity to the spacecraft. These tests were started on July 23, 1964, and completed by mid-August. The tests consisted of charging the spacecraft to voltage levels of 30 kV and discharging the voltage to ground from various points on the spacecraft. The tests did show susceptibility and resulted in corrective measures to desensitize some spacecraft circuits.

4. Solar-Thermal-Vacuum Testing

The objective for STV testing was to verify spacecraft performance in a solar-thermal-vacuum environment that was above (110%) nominal and below (90%) nominal solar intensity.

The first real-time MS attempt in the STV chamber began on January 29, 1965. The test was aborted because of a flight control temperature problem. The test sequence was modified to include one full-duration (66-hr) mission and two abbreviated (32-hr) missions. This sequence was started on February 4, 1965, but was discontinued at the end of the 66-hr sequence due to RADVS malfunctions.

After RADVS repair and other spacecraft problem investigation, sequence 3 was initiated. This sequence consisted of 20-hr abbreviated missions under nominal, 110 and 90% solar intensity levels. While some problems did occur, the test was considered satisfactory.

5. Vibration Tests

Prior to starting the vibration tests, the T-21 spacecraft was used to verify the weight, balance, and optical alignment equipment and procedures. This was accomplished in early May 1965.

The vibration tests consisted of tests in the spacecraft Z-axis and tests in two lateral axes at type approval vibration levels. While some spacecraft problems did occur, they were not considered major and the tests were considered satisfactory. This testing was completed in mid-April 1965.

6. Combined Systems Test

The CST activity began at the CSTS at San Diego, on May 13, 1965.

Prior to shipping the spacecraft from HAC, El Segundo, a series of PVTs were performed to verify spacecraft readiness to support the upcoming tests and to check out the procedures to be used. This was preceded by a spacecraft upgrade period. The activities at CSTS consisted of the following:

(1) Spacecraft receiving inspection and reassembly.

(2) Performing a SRT (PVT 2) performed in the spacecraft assembly area.

(3) Mating the spacecraft to the *Centaur* adapter and encapsulation of the spacecraft with the nose fairing.

(4) Hoisting the spacecraft assembly to the top of the *Centaur* launch vehicle.

(5) Performing a SRT.

(6) Performing the launch countdown simulation in conjunction with the launch vehicle personnel. This CST continued after simulated liftoff and all systems were operated until simulated spacecraft separation from the *Centaur*.

No significant problems were encountered during the tests and the spacecraft was demated and prepared for shipment to AFETR (also see Section VIII-D and -E).

7. Air Force Eastern Test Range Tests

The T-21 spacecraft arrived at AFETR on June 1, 1965. The purpose of T-21 testing at AFETR was to verify the readiness of the SCF at Building AO and the assembly and propellant loading facilities at the ESA. Another objective was to demonstrate alignment and propellant loading procedures and to check out on-pad operations (also Section VIII-C).

After receiving inspection and assembly operations were completed at Building AO, the spacecraft was installed on the CSTS and PVT 1, 2, 3 and 4 were completed. Other spacecraft functional checks were performed on the attitude control and propulsion system and the spacecraft was transported to the propellant loading building in the ESF.

Propellants were loaded into the T-21 spacecraft and the weighing exercise performed. Some fuel leaks were experienced and resolved, but an oxidizer leak in one of the tanks necessitated downloading that tank.

The propellant-loaded spacecraft was then taken to the ESF assembly building and the final assembly operations were continued. These included installation of the dummy retro motor, mating to the adapter, and encapsulation of the spacecraft on July 10, 1968. An on-pad abort exercise was practiced with the aid of a plywood simulated shroud. This was repeated under dark (night) on-pad conditions and no problems were encountered.

The spacecraft was transported to Launch Complex 36A and mated to a *Centaur* simulator. Launch complex SRT, countdowns and EMI tests were conducted. An RF compatibility test with DSS 71 was also conducted with no difficulty. At the completion of these tests, the spacecraft was demated from the *Centaur* simulator and transported to the ESF.

After decapsulation and depressurization in the assembly building, the spacecraft was taken to the propellant building and downloaded.

The T-21 was returned to the SCF and some functional testing was accomplished along with a repeat system readiness test and countdown. After propellant loading procedures were revised, the spacecraft was again returned to the propellant loading building where referee fluids were used to verify the adequacy of the revised procedures. These operations were satisfactorily completed and the spacecraft was returned again to the SCF for shipment preparation. The T-21 spacecraft was shipped to Goldstone, Calif., on July 31, 1965, by air-ride van.

The T-21 AFETR testing was successful since it exposed many areas that conceivably would have hindered AFETR operations on *Surveyor* flight spacecraft. The RF links between the SCF, ESF, and Launch Complex 36A were demonstrated, DSIF compatibility was shown, gas detection techniques were improved, loading and downloading of propellants were performed, joint agency operations were performed on a fueled spacecraft, a fueled spacecraft was transported to and from the launch complex, abort procedures were demonstrated, and compatibility between spacecraft and major support/test items and environments were demonstrated.

8. Goldstone Operations

Activation of the temporary spacecraft facility at Goldstone, Calif., consisting of three trailers and a screen room, was completed prior to arrival of the T-21 spacecraft. The spacecraft, aboard the ground transport

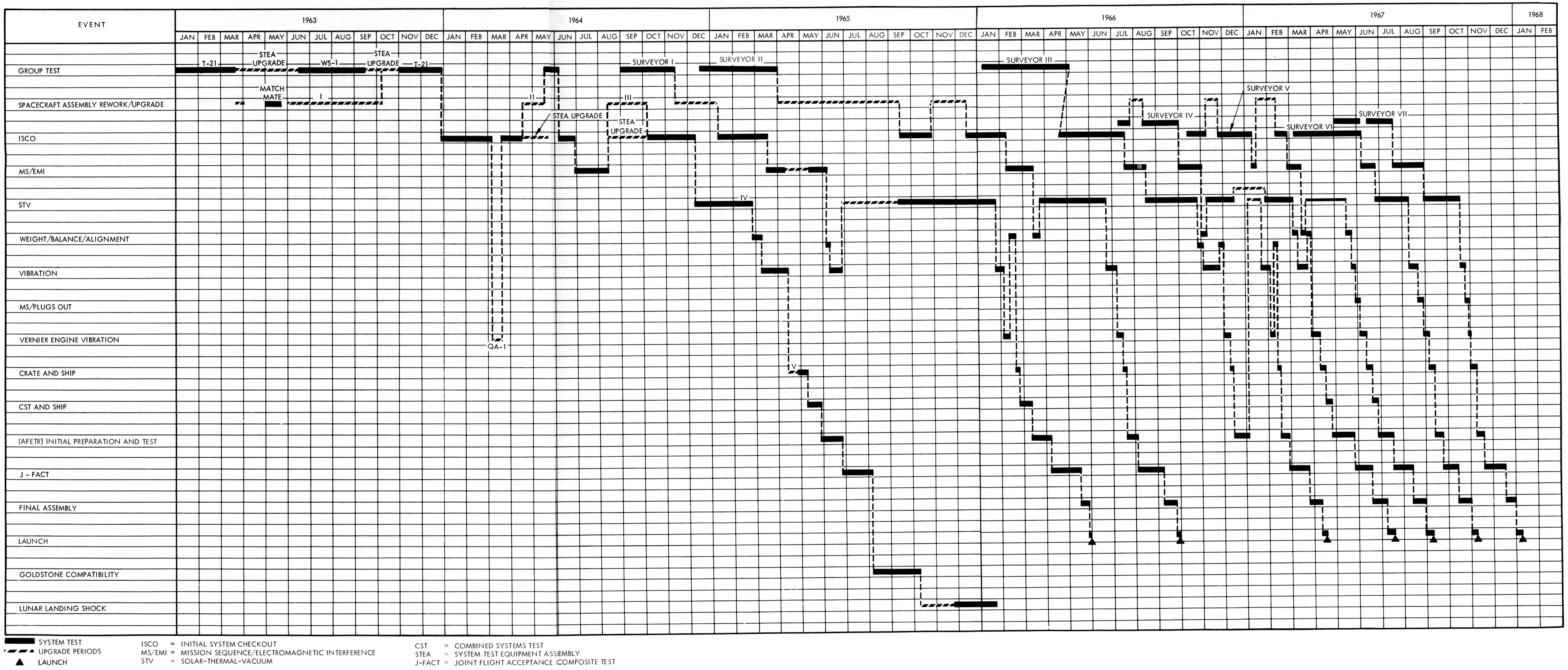


Fig. XI-28. Surveyor system level test history

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vehicle, was placed in the RF screen room and assembled to a flight configuration except for installation of compartment canisters, AMR, retro motor, shock absorbers, solar panel, planar array, and pyrotechnics.

Preparation of the spacecraft for testing consisted of running PVT and the system readiness test. In addition, several operations were performed to make the spacecraft compatible with DSIF test requirements, such as exchange of transmitters for those containing the latest modifications, checkout of TV cameras 3 and 4, and calibration of receivers.

The DSIF/DSN/spacecraft compatibility tests were started on August 14, 1965 and included the following tests:

(1) Radio frequency compatibility tests demonstrated the ability of the DSIF S-band system to communicate with the spacecraft for all spacecraft RF configurations.

(2) Command compatibility tests demonstrated the ability of the DSIF S-band system to command the spacecraft on both automatic phase control and automatic frequency control modes.

(3) Telemetry compatibility tests demonstrated the ability of the DSIF and command data handling console equipment to receive, detect, process, display, and record the pulse code modulated/frequency modulated/phase modulated telemetry from the spacecraft.

(4) Television compatibility tests were performed using special test patterns for obtaining certain quantitative measures of the system performance. Tests were performed by focusing the TV cameras on landscape to test effects of the microwave link transmission characteristics on the quality of the TV pictures.

(5) Space flight operations compatibility tests verified the compatibility and functional operation of the system tied in with space flight operations facilities (SFOF).

(6) Operations training tests provided an opportunity for DSS 11 and HAC station personnel to practice and demonstrate proficiency in performing mission operations procedures, by using the T-21 spacecraft as a signal simulator. The final tests involved the simulation of various spacecraft malfunctions unannounced to the mission operations personnel. The purpose was to exercise these personnel in analysis of spacecraft failure symptoms.

(7) A power group integration test was performed to obtain static and dynamic information when the solar panel, battery charge regulator, and booster regulator were interconnected and loaded in a manner experienced in actual flight. The solar panel was placed on top of the screen room and the solar energy delivered to the spacecraft. Different spacecraft power and load configuration were commanded, thus completing a successful integration.

The T-21 tests at Goldstone, Calif., demonstrated the engineering and functional compatibility between a flight-equivalent spacecraft and the *Surveyor* space flight operations/DSN systems, and provided operational training for their personnel.

The T-21 spacecraft was shipped from Goldstone, Calif., to HAC, El Segundo via air-ride van on October 12, 1965, and upgrading for the lunar landing shock test was immediately started.

9. Lunar Landing Shock Tests

The final phase of the T-21 type approval testing began December 21, 1965. This was the lunar-landing shock test performed on a flight-configured spacecraft in the space simulation laboratory. The test was performed by dropping the spacecraft a prescribed distance onto an inclined wooden platform studded with wooden blocks simulating rocks. Four worst-case lunar-landing conditions were chosen. The first three drops at 15 ft/sec represented the 3 σ expectance of lunar-landing conditions, while the fourth drop at 20 ft/sec represented the 10 σ expectancy.

Selected procedures from the short systems confidence check were used to verify spacecraft readiness before each drop. Phase jitter, shock absorber strain gage channels, and critical power parameters were monitored during the drop. Spacecraft power was supplied by batteries with the flight-control thrust-phase power and RADVS power on. Five seconds after impact, the flight-control thrust-phase power and RADVS were commanded off. Fifteen seconds after impact, the spacecraft was raised clear of the drop platform, and a cycle print of all commutators was performed by utilizing the programmed automatic telemetry evaluator (PATE). Upon conclusion of each drop, functional integrity of the spacecraft was verified by repeating the short systems confidence check procedures. For verification of alignment, the spacecraft was returned to the optical dock.

Due to a conflicting requirement for use of the test equipment, drop test 3 was eliminated. Functional test results for the three drops performed were satisfactory.

These tests verified that the spacecraft could functionally and structurally survive lunar touchdown.

E. Flight Spacecraft Tests and Operations

This section describes the test results of all seven flight spacecraft. Deviations from TOP are listed and significant problems are noted. The test schedules of each of the flight spacecraft as well as T-21 are shown in Fig. XI-28. Launch vehicle and combined spacecraft/launch vehicle prelaunch tests at AFETR are further discussed in Section VIII-E-2. The countdown and launch vehicle flight performance is discussed in Section VIII-F.

1. Surveyor I Spacecraft (SC-1)

The SC-1 system tests were started in early September 1964 and ended with launch on May 30, 1966. The test program included the FA test at HAC, El Segundo, the CST tests at GD/C, San Diego, and the AFETR tests.

a. Group tests and initial systems checkout. The ISCO test phase began on September 1, 1964 with the installation of major control items. Group tests were conducted through November 17, 1964, and from then to December, 30, 1964, the *Surveyor I* spacecraft was in upgrade. The ISCO tests were resumed on December 30 and continued through March 20, 1965.

Although ISCO tests were completed, the test results were difficult to assess due to numerous problems existing in most subsystems.

b. Mission sequence and mission sequence/electromagnetic interference. This test phase consisted of MS tests, MS/EMI tests, and upgrade of the spacecraft. The test phase began March 20, 1965, with MS associated tests which were completed April 5. The spacecraft was moved to building 365 on April 5, where MS/EMI was conducted through April 8.

Upgrade operations began April 8, 1965 and terminated May 14, 1965 and included such major spacecraft configuration changes that the MS and MS/EMI phases were essentially nullified.

Additional testing began on May 14, 1965, with the delivery of the spacecraft to building 365, where it was mounted on the CSTS in the screen room. Test/operations for this retest period were ended on June 2, 1965 with completion of MS/EMI and an SRT.

c. Solar-thermal-vacuum test phase. The STV test phase was preceded by some special tests. These consisted of four RF tests, five power system checks, four signal processing tests, four Spacecraft System tests, one RADVS, and one TV special test performed between June 26 and August 10, 1965.

The STV test phase ran from August 10 through December 27, 1965 and consisted of five aborted mission sequences prior to successful performance of mission sequences A, B, and C.

The STV-1 test phase began with vehicle preparation on August 10. Mission sequence under STV environment began September 9 and was aborted because of spacecraft problems on September 10 after the spacecraft had been subjected to approximately 22 hr of STV environment. Troubleshooting sequences and tests were conducted until the test phase was concluded on September 29, 1965.

The STV-2 test phase began, after incorporation of spacecraft design changes, with vehicle preparation on September 29, 1965. Mission sequence in the STV environment began October 2, and aborted on October 5. Tests, troubleshooting sequences, and subsystem upgrade followed the MS. The test phase was terminated with completion of spacecraft reassembly on October 17, 1965.

The STV-3 test phase began with vehicle preparation on October 17, 1965, and MS in the STV environment began October 22, and aborted October 24. Post-STV tests and troubleshooting steps followed the MS. The test phase was terminated October 30.

The STV-4 test phase began with an MS dry run October 30, 1965. The MS in the STV environment began on November 1 and terminated November 5. Special tests, AGE checks, troubleshooting, and spacecraft unit upgrade followed MS. This test phase was terminated December 3, 1965.

After the STV-4 abort, an extensive series of special tests, troubleshooting sequences, and calibration procedures were performed. The special tests which were performed provided integration of the reworked hardware units into the spacecraft system. System test equipment assembly preparation was extensive and broad in scope.

The STV-5 test phase began December 3, 1965 following subsystem/unit configuration changes. Mission sequence was started December 16, 1965, and aborted December 17 because of a nitrogen leak in the STV chamber. Tests and troubleshooting sequences followed the aborted mission. This test phase was terminated December 25, 1965.

The STV-6 test phase began with the phase A (low-sun) MS on December 27, 1965. Completion of phase A was followed by phase B (high-sun) on December 31, 1965. Tests and troubleshooting sequences preceded phase C (plugs-out) which began January 20, 1966. The test phase was terminated January 25 with STV depreparation operations. All MS events were completed successfully for two 66-hr missions which satisfied the test requirements for STV phases A and B. The plugs-out configuration, phase 6C, required 17 days of preparation. These preparatory tests included SRT (PVT 2) and a plugs-out MS dry run. The STV phase C sequence was successfully completed on a time-compressed scale.

d. Vibration test phase. The vibration test phase ran from June 2, 1965 through February 7, 1966 and consisted of *X*, *Y*, *Z*, and *Z* rerun vibration tests. The vibration test phase began with alignment operations in Building 350. The spacecraft was moved to Building 365 where the balance of the test phase was conducted. Completion of the initial vibration test phase occurred on June 24, 1965, with movement of the spacecraft to Building 350 for weight, balance, and optical alignment. The three-axis vibration test phase was successfully completed with the vehicle in a near-flight configuration.

As a result of problems which occurred after completion of the original flight acceptance vibration test phase in the STV test phase, a decision was made to rerun the Z-axis vibration phase.

Vibration in the Z-axis was accomplished on January 30 and 31, 1966, followed by transfer of the spacecraft to the system test stand in the screen room of Building 365. Further investigation of anomalies was accomplished by special tests and troubleshooting sequences. These tests were concluded on February 7, 1966.

e. Vernier engine vibration. The VEV test phase began with tests on February 8, 1966 in the screen room of Building 365. Special tests and troubleshooting sequences continued until February 12. Following this, alignment operations and special tests were performed in Building 350, where the test phase was completed on February 17, 1966. No significant anomalies were noted during this test.

f. Combined systems test. The spacecraft was transported by air-ride van to the CST facility in GD/C, San Diego, for the CST. This test phase ran from February 18, 1966 to March 7, 1966.

The only serious anomaly during the CST period was the discovery after decapsulation that 10 of 12 retro motor release squibs were fired. After extensive research, it was determined to a high degree of confidence that the squibs were probably expended before installation.

The spacecraft was shipped by C-133 aircraft to AFETR on March 12 1966.

g. Air Force Eastern Test Range tests. The AFETR phase began with the spacecraft arrival on March 12, and the test plan as described in Subsection B was followed except for the following significant problems which required some retest:

(1) A harness short during J-FACT preparation necessitated removal of the battery charge regulator for repair.

(2) The RADVS signal data converter failed during PVT-5 and was replaced with a flight spare.

(3) During final flight preparation in the ESF prior to moving to the launch stand, the A/SPP roll axis-drive motor was replaced with a flight spare.

(4) A short circuit in the main battery monitor cable necessitated battery and cable replacement during final assembly in the AFETR.

Surveyor I was successfully launched on May 30, 1966, at 14:41:00 GMT.

2. *Surveyor II* Spacecraft (SC-2)

Flight acceptance testing for SC-2 began with the start of group tests in December 1964, and ended with launch on September 20, 1966. Major test and operation plan deviations were revised MS/EMI test sequences and omission of combined systems tests at GD/C, San Diego.

a. Group test and initial systems checkout. Group tests were completed in March 1965, and no major problems were encountered. Most of the problems involved the system test equipment assembly/spacecraft interface.

Because SC-2 was in an assembly and upgrade program to incorporate mandatory spacecraft modifications from April through September 1965, no systems test operations were conducted. Following completion of the upgrade program, SC-2 was subjected to new and more extensive power and grounding checks.

The *Surveyor II* spacecraft started ISCO on September 13, 1965 and, after an upgrade period in November, completed ISCO on February 7, 1966.

b. Mission sequence/electromagnetic interference tests. Mission sequence testing began February 7 with MS/EMI testing conducted between February 16 and March 12, 1966. The following significant changes were made to the MS/EMI testing plan for SC-2:

(1) Two ambient (non-RFI) sequences with hardline access instead of one ambient and one RFI.

(2) One of the ambient sequences at high simulated battery voltage and one at low voltage.

(3) Extension of the length of the plugs-out RFI sequence to full mission time.

The spacecraft successfully completed the MS/EMI test phase. Real-time and posttest evaluation indicated that the spacecraft was in excellent technical condition. There were no spacecraft failures which would have caused a mission abort.

c. Solar-thermal-vacuum tests. After a weight, balance, and alignment operation, SC-2 was prepared for transfer to Building 365 on March 23, 1966; the space simulation laboratory and STV tests were completed July 14, 1966.

Original test plan called for a phase A (low-sun) back-to-back, with a phase B (high-sun) STV test, followed by a phase C (nominal-sun) test. Due to test aborts, no back-to-back tests were accomplished. However, a significant improvement over SC-1 was achieved in this test program. While STV testing of SC-1 took 19 weeks and eight mission attempts, the test program for SC-2 was only 10 weeks long and required just six mission attempts.

The first solar-thermal-vacuum sequence STV-1A (low-sun) was conducted between April 22 and April 27, 1966. The MS for STV-1A was performed in real-time, taking 66 hr, with the sequence of events as defined in EPD-180.

The test was terminated prior to starting run B because of a ruptured main battery. After opening the chamber and removing the battery, the transmitter B and the thermal tray assembly were changed because of physical damage from the spilled potassium hydroxide within the compartment. Four compartment A thermal switches were replaced with switches set for a lower closing temperature of 40 $\pm 3°F$ to maintain battery temperature during high-sun test at approximately the same values observed during low-sun test.

Sequence STV-1B (high-sun) of the solar-thermal-vacuum tests on SC-2 were started on May 9, 1966. Thirteen hours after simulated launch the MS was aborted because of improper operation of two thermal switches on compartment A. This caused the battery and compartment A temperatures to be higher than normal.

Sequence STV-2B (high-sun) was conducted from May 12 to May 16, 1966. The STV-2B was the second attempt for the high-sun test (simulating 112% solar intensity), in accordance with EPD-180.

Sequence STV-3B (high-sun) was started on June 1 and ended on June 4, 1966.

Temperatures were higher over most of the spacecraft during this mission than in previous tests. Further, the compartment B temperatures were above the upper temperature limits. There were two major reasons for these high temperatures: (1) a higher ultraviolet content in the STV-3B simulated solar intensity, and (2) two or more compartment B thermal switches were not operating properly.

The dry run for sequence STV-1C (nominal-sun) was performed on June 10; no anomalies were observed during the prelaunch countdown, and launch took place at 15:08:00 GMT. No A/SPP stepping was planned for the dry run, so the solar panel was not deployed after separation. However, the solar panel deploy logic was turned on automatically just before separation. The sequence STV-1C was started on June 12, 1966.

The test proceeded normally until shortly after the start of RADVS-controlled terminal descent. The spacecraft failed to respond to commands as result of boost regulator failure, and all communications with the spacecraft were lost. RADVS power was on at the time of communication loss and remained on for 11 minutes, resulting in excessive thermal stress on the signal data converter and KPSM units of the RADVS. Spacecraft power was turned off through spacecraft main power switch. The test was aborted on June 13, 1966, and the spacecraft was brought out of the chamber.

The boost regulator failure resulted in some design changes and a special STV test was conducted to test the modified boost regulator and the replacement units in the RADVS system. This special mission consisted of 10 hr of a normal phase C MS, followed by a 6½-hr period of thermal preconditioning, a terminal descent using the flight control closed-loop simulator, and a special 6-hr TV test after touchdown. The test was started on June 24 and successfully completed on June 25, 1966.

d. Vibration test phase. The flight acceptance vibration tests for SC-2 were run between June 30, 1966, and July 7, 1966. In addition to the normal tests a special vibration test was performed after the spacecraft received an abnormal shock due to test vibrator malfunctions during one of the vibrator runs.

e. Vernier engine vibration tests. The VEV test phase was performed July 8 and July 13, 1966. On July 11, 1966, the dry run terminal descent was performed. The terminal descent was a failure when RADVS range lock was lost at the 1000-ft mark. This was caused by a defective crystal in the RADVS.

The RADVS system was removed from the spacecraft and shipped to GD/C, San Diego for rework and the VEV tests were continued with RADVS components borrowed from SC-3.

After the RADVS was repaired and returned, a successful midcourse and terminal descent was completed on July 13, 1966 and the spacecraft was prepared for shipment to AFETR.

f. Air Force Eastern Test Range tests. Spacecraft SC-2 arrived at the AFETR skid strip via Super Guppy aircraft on July 19, 1966 and testing was concluded with the launch on September 20, 1966. The spacecraft was unloaded at the skid strip and transported to the spacecraft check-out facility. The GTV-1 and outer covers were cleaned and the spacecraft moved into the high bay where the GTV 1 covers were removed.

No significant problems or test plan deviations were noted in the AFETR test phase for SC-2, and the spacecraft was launched on September 20, 1966, from Launch Complex 36A.

3. *Surveyor III* Spacecraft (SC-3)

The SC-3 system tests were started on April 14, 1966 and concluded with launch on April 17, 1967. The major deviation from the standard test and operations plan was the deletion of the CST at GD/C, San Diego and a recycle for some FA retest at HAC, El Segundo after the initial shipment to AFETR.

a. Group test and initial systems checkout. Group tests were performed during the period from January 10 through May 2, 1966. The ISCO was conducted between April 14 and July 14, 1966 with no unusual difficulties being encountered.

b. Mission sequence/electromagnetic interference test phase. Mission sequence tests began on July 25, 1966 and were satisfactorily completed on August 13, 1966. The only major spacecraft problem was a failed commutator switch in the ESP unit. This was repaired and retested prior to the start of MS run number 3.

c. Solar-thermal-vacuum test phase. SC-3 was delivered to the space environmental laboratory on August 14, 1966, and STV was completed on October 27 after five mission sequences, which included the first successful back-to-back STV-A and -B runs.

On September 3, 1966 sequence STV-A (low-sun) was initiated. This test simulated a real-time MS of approximately 69-hr transit, and 4-hr postlanding operations. The STV-A mission was completed successfully on September 6, 1966.

Sequence STV-B (high-sun) was started on September 7, 1966 after the completion of STV-A and was completed successfully on September 10, 1966.

After sequence STV-B the spacecraft was removed from the chamber and the SM/SS experiment was installed.

During preparations for the STV-C phase, the RADVS inadvertently turned on due to a procedure error, and the temperature limits for the SDC and KPSM units were exceeded. The SDC and KPSM units were replaced and the new units were retested with no problems encountered.

On September 24, 1966 subsequent to *Surveyor II* mission failure, an investigation of the SC-3 VPS disclosed apparent contaminants in the oxidizer tanks. Oxidizer tanks 1 and 3 were changed out and the VPS analyzed. Results of the examination brought about the removal and changeout of all three fuel tanks. Furthermore, based upon recommendations from the *Surveyor II* Failure

Review Board, several modifications were incorporated into SC-3. These changes were commutator modifications to assure accurate data at 4400 bits/sec, filters in the propulsion strain gage data channels, a filter in the regulated current data channel, and a reassignment of certain commutator data channels. Confidence testing and preparations continued, but, due to the problems encountered during STV testing and required spacecraft modifications, the decision was made to slip the launch date one month. On October 6, 1966 the spacecraft was readied for the STV-C test phase.

Sequence STV-1C was conducted from October 6 through October 9, 1966, as a real-time MS with approximately 35 hr of mission time in low-sun and the remainder in high-sun. It was required because of spacecraft configuration changes after STV-B. During the midcourse maneuver, the roll actuator failed because of an open wire in the spacecraft harness. The spacecraft was removed from the chamber and the wire was repaired.

Sequence STV-2C was performed from October 15 through October 16, 1966, as a shortened low-sun MS.

Sequence STV-3C was performed from October 24 through October 27, 1966, as a nominal-sun test in real-time sequence, being required to gain confidence in spacecraft performance. It was completed successfully on October 27.

d. Vibration test phase. Prior to the start of the vibration test phase, the spacecraft was optically aligned and transported to Building 365 for vibration testing. Vibration tests were performed between November 7 and December 1, 1966.

Preparations for the vibration test phase were initiated on November 7 and completed on November 20, 1966. The delay was caused by a modification to the spacecraft harness and a changeout of a portion of the RADVS waveguide. On November 21, 1966, vibration runs 1 and 2 were successfully completed and the spacecraft was removed from the fixture and placed on a work stand where an SRT was performed. During performance of the SRT, transmitter B developed phase jitter. Compartment A was opened and the transmitter was removed, repaired, tested, and reinstalled.

The spacecraft was returned to the vibration fixture and on November 27, 1966, runs 3 and 4 were completed. At the conclusion of the test, input sensitivity for receiver A was 20 db below normal. The unit was repaired and retested satisfactorily both in the unit area and on the spacecraft.

A postvibration test was then successfully performed, and the spacecraft was transported to Building 350 where postvibration optical alignment tests and spacecraft weighing were performed.

e. Vernier engine vibration test. On December 1, 1966, the spacecraft was moved to Building 350 and the spacecraft was prepared for VEV testing.

Pre-VEV/RADVS testing indicated excessive noise in the doppler velocity sensor (DVS) preamps. The KPSM unit of the RADVS was returned to Ryan, San Diego where the problem was corrected. The unit was returned to the spacecraft and retested satisfactorily. Final preparations for VEV were completed and the VEV tests started on December 6, 1966. They were successfully completed on December 7, 1966, and shipment preparations were started.

f. Air Force Eastern Test Range tests (initial). The AFETR tests of SC-3 began after the arrival of the spacecraft at Cape Kennedy via air on December 14, 1966.

During PVT 1–3, a series of spacecraft problems occurred that necessitated hardware replacement. These problems included:

(1) Use of an ungrounded soldering iron damaged a AESP commutator.

(2) Strain gage amplifier assembly replaced for same reason.

(3) The EMA and FCSG were damaged because of a STEA calibration-box short circuit. Both units were returned to HAC, El Segundo for repair.

(4) Oxidizer tank 2 was replaced for thermal standoff problem.

(5) Oxidizer tanks 1 and 3 bladders were partially collapsed because of a procedural problem. Testing indicated no problem.

(6) Antennas and preamplifiers of the RADVS were removed and returned to Ryan for antenna pattern measurements.

Because of the flight hardware problems and subsequent removal and replacement required, a decision was made to return the spacecraft to HAC, El Segundo for repair and retest through a modified environmental test sequence.

After preparation for shipment, the spacecraft was shipped from AFETR to HAC, El Segundo by air-ride van on January 4, 1967.

g. Solar-thermal-vacuum retest. During test preparation and checkout for STV, the camera mirror assembly failed to respond to commands in azimuth. Upon disassembly, it was discovered that the teeth on the azimuth gear were severely worn at the point where the mirror was positioned during the truck shipment from AFETR to HAC, El Segundo. The damage was attributed to a high vibration level during shipment due to a faulty air-ride system on the van. The camera was returned to the unit area for repair and the TV camera from SC-4 was installed on the spacecraft for use throughout STV testing.

The STV retest was started on January 19 and completed on January 22, 1967. There were no significant spacecraft difficulties during this test. Three terminal descents were successfully performed.

h. Vibration retest. Vibration runs 1A, 2A, and 2B were performed on January 29, 1967, and successful operation of all spacecraft systems was realized.

i. Vernier engine vibration retest. On February 1, 1967, the spacecraft was moved to Building 350 for VEV retesting. On February 4, the tests were performed with no significant anomalies. On February 5, the spacecraft was moved to the alignment dock area where weighing and alignment exercises were performed. On February 7, the spacecraft was prepared for shipment to Cape Kennedy to reinitiate launch preparations.

j. Air Force Eastern Test Range tests (final). The spacecraft was air-shipped to AFETR on February 14, 1967, and launch operations testing was started. Testing proceeded in the prescribed sequence with the addition of several propulsion system tests to gain confidence in the flight readiness of the spacecraft. These included a long lines test in Building AO, where the vernier engines were operated in the propulsion laboratory from spacecraft outputs from the test floor, and a test at flight pressures in the propellant laboratory at the ESF after propellant loading.

The only major problem during the testing phase was a momentary short, created between the spacecraft battery and ground when the field joint connector was reconnected after J-FACT.

The spacecraft was successfully launched on April 17, 1967.

4. *Surveyor IV* Spacecraft (SC-4)

The SC-4 systems tests were started on June 11, 1966, and concluded with launch on July 14, 1967. There were no significant deviations from the basic test plan.

a. Initial systems checkout. The SC-4 ISCO test phase was initiated June 11, 1966 but because of some late hardware deliveries, formal ISCO tests were delayed until July 14, 1966.

From August 25 to September 9, 1966, some TCM retesting was performed due to changeout of transmitter A, and a number of troubleshooting sequences were performed.

b. Mission sequence/electromagnetic interference tests. The MS/EMI test phase began September 30 and ended November 3, 1966. From October 7 through October 11, a number of ISCO TCM retests were performed. Mission sequences 1 and 2 were conducted according to plan without significant anomalies. At the completion of MS run 2, a special test was conducted as a result of the *Surveyor II* failure. The purpose of the test was to attempt to determine the solenoid valve currents from current shunt EP-4.

Mission sequence 3 with EMI was performed with excellent results and was completed on October 28, 1966.

From October 28 through November 2, 1966, some troubleshooting sequences, which investigated strain gage amplifier noise and regulated output current problems, were performed. This post-ME/EMI troubleshooting resulted in the engineering signal processor and auxiliary engineering signal processor being returned to the unit area for design changes affecting current calibration channels. Up to this time, SC-4 did not have the SM/SS installed. The mounting brackets for the SM/SS were installed and the spacecraft was delivered to the Building 350 alignment area.

Weight, balance, and alignment of the spacecraft were performed according to plan from November 7 to 10, 1966.

c. Solar-thermal-vacuum test phase. The STV test phase was performed between November 10, 1966, and February 28, 1967, and was comprised of three MS: A, B, and C; no repeats were required.

With the arrival of SC-4 in Building 365, pre-STV preparations were initiated. On November 15, 1966, the

SM/SS was installed on the spacecraft. The pre-STV preparations were continued and on December 9, 1966, STV phase A was started, followed immediately by phase B. They were successfully completed on December 15, 1966. A brief rundown of the four most significant anomalies follows:

(1) Receiver B acted erratically on five occasions. For no apparent reason the automatic gain control (AGC) drifted. Following phase B, the receiver was removed from the spacecraft and sent to the unit area for repair.

(2) Altitude marking radar (AMR) electronics temperature, R-7 gave indications of approximately 60°F, whereas the expected value was 20°F. A troubleshooting sequence, run during the coast phase, revealed a problem in a grounding circuit of the spacecraft. The AMR was removed and returned to the unit area for repair.

(3) The spacecraft transferred from auxiliary battery to main battery mode without having been so commanded. The problem was a defective RF transfer switch. The defective switch was removed and replaced after phase B with a redesigned switch.

(4) Vernier line 2 heater exhibited a higher than expected duty cycle during both phases of STV. While in phase B (low-sun), it was approximately 85% and went as high as 99% during solar eclipse and minimum unregulated bus. The complete assembly of lines, heater, thermostat, and sensor were removed and replaced after phase B with newly designed units. The resistance of the heaters was reduced by nearly one-half.

On December 16, 1966, phase C preparations were started. While preparations progressed, it was learned that *Surveyor* SC-3 was to be returned from Cape Kennedy and an STV retest performed; thus, rescheduling of *Surveyor* SC-4 testing was necessary. A decision to start vibration preparation was made, the vehicle being transported to Building 350 for optical alignment prior to vibration testing. Optical alignment was completed on January 9, 1966, and vibration preparation continued. On January 13, SC-4 was moved to Building 365, where the SDC was removed and returned to Ryan, San Diego for the cross-coupled side-lobe logic modification. The TV camera was removed and used on SC-3 for its STV retest, and then the camera was returned in preparation for the SC-4, phase C, STV testing. By this date, SC-3, having completed its STV retest, was proceeding into vibration retest, so SC-4 vibration plans gave way to preparation for STV phase C testing. Phase C was started on February 24, 1967.

The STV phase C was performed under low-sun conditions with plugs out, and simulated a real-time MS of approximately 63 hr. In addition, two posttouchdown terminal descents, TV testing, and SM/SS testing were performed following the first terminal descent and before the test chamber was returned to ambient conditions. This combined posttouchdown activity lasted approximately 23 hr.

Only two problems of significance were observed during phase C:

(1) Transmitter B exhibited a power decrease of 2 to 4 db after sustained periods of operation. The transmitter was removed after phase C and sent to the unit area where the problem was discovered to be a missing part in the transmitter attenuation which connects to the input of the traveling-wave tube.

(2) Television camera mirror assembly would not respond properly to elevation commands. The camera was later removed from the spacecraft and returned to the unit area for rework and retest.

The engineering signal processor was removed from the spacecraft and returned to the unit area for investigation of yaw gyro error-signal telemetry fluctuations. A transistor was found shorted and cracked. The unit was repaired, subjected to a special thermal-vacuum test, and returned to the spacecraft.

With STV phase C completed on February 28, 1967, the spacecraft was sent to Building 350 for previbration weight, balance, and alignment, and on March 9, was returned to Building 365 for the vibration test phase.

d. Vibration test phase. On March 20, the vibration test phase was completed according to the test plan; no spacecraft hardware anomalies were observed. The spacecraft was moved to Building 350 for postvibration alignment and then into the screen room where special high-pressure propulsion tests were performed. On March 28, SC-4 was moved to STEA 5 for VEV preparation.

e. Vernier engine vibration. Following weight/balance operations and thermal finish touchup, the spacecraft was prepared for shipment by van to GD/C, San Diego for CST.

f. Combined systems test phase. The CST test phase was performed April 13 and 21, 1967.

Prior to shipping the spacecraft to GD/C, San Diego a commutator switch in the engineering signal processor was believed damaged. The unit was returned to the unit area, repaired, and sent to the spacecraft at CST. In addition to the formal tests, seven troubleshooting sequences were performed at CST. Results of all testing clearly showed the spacecraft to be performing correctly and that shipment to AFETR was indicated.

g. Air Force Eastern Test Range tests. The spacecraft was shipped by air and arrived at Cape Kennedy on April 25, 1967. The testing followed the general test plan without significant anomalies and the spacecraft was launched on July 14, 1967.

5. Surveyor V Spacecraft (SC-5)

The SC-5 systems tests were started October 21, 1966, and concluded with launch on September 8, 1967. Configuration differences between SC-5 and preceding spacecraft required procedure changes and some deviations from the standard test plan.

a. Initial system checkout. The SC-5 ISCO tests were planned as a two-stage phase. The initial testing was done with the old SC-1–4 type harness and testing was completed with the SC-5 harness. Initial power turnon and TCM integration tests verified the new spacecraft power system and checkout of the new upgraded STEA 5 configuration. The new SC-5 type wiring harness was then installed and the ISCO phase was reinitiated on November 28, 1966. The ISCO was completed according to plan except for the following deviations:

(1) The redesigned TV camera scheduled for SC-5 (290512 series) could not be delivered to the spacecraft due to problems encountered during the fabrication and unit testing of the camera so a spare camera was used for ISCO testing.

(2) The AS/TCM ISCO test was also conducted during ME/EMI due to late delivery of the unit.

b. Mission sequence/electromagnetic interference test phase. The MS/EMI test phase ran from January 14 through March 15, 1967, and was comprised of three MS.

At successful completion of MS 1, the spacecraft went through an extensive upgrade period. The alpha scattering subsystem, including the new compartment C, was installed and several control items were recycled through the unit area to incorporate latest design changes. This upgrade necessitated two weeks of electrical retests to verify the compatibility of the subsystems. At this time the camera originally intended for SC-5 was installed on the spacecraft and the TV/TCM tests were again conducted with the new camera. The AS/TCM integration test was performed for the first time, thus completing all ISCO requirements.

Sequence 2 was completed with satisfactory results.

The overall performance of the spacecraft during sequence 3 was nominal. A few minor anomalies and one TV noise problem were noted. The TV problem occurred during the TV test sequence of posttouchdown operation; a filter was added to eliminate the noise.

A partial alignment test of the spacecraft was performed during the period March 16 through March 20, 1967. The RADVS antennas, omniantennas, thrust chamber assemblies, jets, and flight control were aligned, and the new solar panel STM model was temporarily installed to prove mechanical compatibility.

c. Solar-thermal-vacuum test phase. The STV test phase was performed between March 20 and May 14, 1967 and was comprised of an anomalous phase A sequence, an aborted phase B sequence, followed by successful back-to-back (continuous) phases A and B sequences.

Sequence STV-1A, (battery, high-sun), conducted during the period from April 21 to April 24, 1967, uncovered excessive helium relief and check valve temperatures, low nitrogen tank temperature, spurs generated by transmitter A, and a leak in the vicinity of vernier engine 3. The spacecraft was removed from the chamber for modification and repair at the completion of phase 1A.

The paint patterns on the helium relief and check valves were modified to decrease the steady-state temperatures of the valves; the shaded portion of the nitrogen tank was covered with aluminum to reduce heat losses, and the sunlit portion of the tank was modified to provide for greater heat absorption; and transmitter A, SN 11, was replaced by SN 15 transmitter that had excessive phase jitter and was subsequently replaced by SN 12. Posttest troubleshooting sequences failed to disclose the source of the leak which was detected in the vicinity of vernier engine 3. The spacecraft was prepared for the next sequence (STV-1B) with full cognizance of this fact.

Sequence STV-1B, (battery, low-sun), was initiated on May 4, 1967. However, the mission was aborted early in coast phase 1 due to the recurrence of an alcohol leak in the area of vernier engine 3. It was noted during a special high sun test that the temperatures of the helium check and relief valves were above the 125°F specification. The spacecraft was subsequently removed from the chamber and vernier engine 3 was replaced. The helium check and relief valve bushings were replaced by a set with a higher thermal conductance. The spacecraft was then prepared for the STV-2B and 2A (modified) sequences.

Sequences STV-2B, (low-sun), and STV-2A, (high-sun), were performed as back-to-back missions between May 8 and May 14, 1967. Both phases consisted of 69 hr each for transit, and 15 and 19 hr, respectively, for post-touchdown operations. Final evaluation of the test data confirmed that the overall test objectives were met and the overall performance of the spacecraft throughout both missions was excellent.

Following STV de-prep operations, a high-pressure test and a bladder integrity check were conducted. Pre-vibration alignment checks were then performed on RADVS, vernier engines, flight control, and A/SPP in the launch position.

d. Vibration test phase. The vibration test phase was conducted from May 22 to May 28, 1967. Evaluation of data verified that all test objectives were met.

The ambient MS test was compressed to 14 hr and was completed on May 31. Evaluation of the test data indicated the performance of the spacecraft was excellent.

The spacecraft was moved to the optical dock area of Building 350, for mechanical work and for postvibration alignment.

e. Vernier engine vibration. The VEV tests were conducted from June 6 to June 9, 1967, without significant anomalies.

f. Combined systems test. The SC-5 underwent the CST according to plan between June 18, 1967 and June 28, 1967. Results were satisfactory and the spacecraft was shipped to AFETR from GD/C, San Diego via air transportation on June 28, 1967.

g. Air Force Eastern Test Range tests. Launch readiness tests were performed between July 5 and September 8, 1967. Testing followed the standard plan except for the following problems and resultant retests.

During AFETR testing, the new TV camera SN 11 was installed due to the questionable ability of the original SC-5 camera vidicon to survive the lunar night; a preliminary check of performance was conducted. The SDC was received from Ryan, San Diego following retrofit action, and installed.

Difficulties arose in conducting the RADVS ranging tests. Considerable troubleshooting led to the conclusion that the problem was due to the relatively wider spectrum of KPSM SN 12. The unit was sent to Ryan, San Diego for further examination.

During VPS functional, fuel tank 1 failed the bladder leak test and was replaced. Fuel tank 3 exhibited a somewhat higher (within specifications) leak rate and was retained on the spacecraft, pending additional testing. In the course of the flow tests of flight vernier engines, excessive shutoff time was observed in engine SN 561, resulting in its replacement with thrust chamber assembly SN 560. The shutoff valve from SN 561, which exhibited superior characteristics, was installed on SN 560.

Additional TV testing was necessary during PVT 4 because the original SC-5 camera had been replaced.

Propulsion operations were conducted in parallel with TV tests. Flow test of spare engines was performed. The oxidizer check and relief valve assembly, which was found to have developed an excessive leak, was replaced. Both check and relief valve assemblies were reoriented on the spacecraft so as to point away from the alpha scattering system.

Pin retention failure on the safe/arm connector of the flight retro resulted in replacement of the safe/arm with a spare. A dent was noted on one of the struts of the spare retromotor, and X-rays revealed a void in the nearby weld joint. Subsequent analysis indicated that these discrepancies had been previously bought off and were acceptable for flight.

Delay in repair and return of original SC-5 AMR SN 11 led to the decision to designate the SP-1 AMR SN 14 for *Surveyor* flight; this AMR was then successfully group-tested.

Results of analysis and of ranging tests led to the decision to replace the KPSM SN 12 with a KPSM SN 11, and the new RADVS system showed excellent performance in the final spacecraft ranging test.

During final flight preparation at the ESF, the propellant tanks were loaded with solvents and pressurized to nominal pad pressure of 300 psi. During the first attempt at conducting a high-pressure leak test, a leak was observed at the bottom of oxidizer tank 2 around 700 psi. Upon replacement of this tank, a similar leak was uncovered at about 500 psi at the oxidizer tank 1 on the second attempt of the high-pressure test. Analysis revealed that O-rings on the oxidizer tanks were found to have come from the same suspect batch, so oxidizer tank 3 was replaced, including the defective oxidizer tank 1. A high-pressure test and pressure decay test, as well as calibration of pressure transducers, were then completed. Subsequently, alcohol was found on the gas side of the fuel system, and it was determined to have been caused by bladder tears which had developed on fuel tanks 2 and 3. Fuel tanks 2 and 3 were then replaced and the entire fuel system was thoroughly cleaned and vacuum purged. A bladder leak test was then conducted on fuel tank 1 on the spacecraft. The fuel system was reloaded with solvents and the high-pressure decay test was conducted on the fuel system. Upon removal of solvents and subsequent purging and vacuum cleaning of both the oxidizer and fuel systems, a composite bladder leak check was performed. Following subsequent loading with fuel and oxidizer propellants, a bladder integrity check and a special high-pressure test were then successfully accomplished.

The AFETR test phase was successfully completed with the launch on September 8, 1967.

6. Surveyor VI Spacecraft (SC-6)

The SC-6 systems tests were performed between March 16 and November 7, 1967. Major deviations to the test plan included omission of the CST phase at GD/C, San Diego.

a. Initial systems checkout. The SC-6 ISCO tests were conducted between March 16 and June 3, 1967. Flight hardware problems and late delivery of some flight units were the reasons for the longer period of time required to complete the ISCO tests.

b. Mission sequence and mission sequence/electromagnetic interference. Mission sequence tests were performed between June 3, 1967 and June 27, 1967. No significant problems were encountered during the MS tests; and all testobjectives were successfully accomplished.

c. Solar-thermal-vacuum test phase. The SC-6 was exposed to the STV test phase between June 27, 1967, and August 10, 1967. Problems were experienced in some of the sequences that required rerunning of some tests. The major problems are described as follows:

The first sequence STV-1A, under high-sun conditions, was marred by difficulty in the terminal descent sequence. A series of four terminal descents were required before success was attained. The problems were later determined to be in the ground test equipment. In addition, problems with inadvertent loss of uplink and a dropout of high power resulted in a changeout, at a later date, of transmitter B and receiver A.

The low-sun sequence (STV-1B) was started without breaking the chamber, as in the original plan. No significant problems were detected in this sequence.

This high-sun sequence (STV-2A) was initiated on July 29, 1967, because of the replacement of transmitter B and receiver A after the completion of STV-1B. Prior to the midcourse correction portion of the test, the test was aborted because of a failure in transmitter A. The transmitter was replaced on July 31, 1967.

Sequence STV-3A was necessary to qualify the two transmitters and receiver A which had not operated for any significant period in system-level STV environment. The test was marred by excessive phase jitter on transmitter B. The transmitter was later removed, checked in the unit area, and replaced on the spacecraft at AFETR.

d. Vibration test phase. Vibration testing on SC-6 was conducted between August 17, 1967, and August 21, 1967. Problems discovered during the vibration phase were:

(1) Antenna/solar panel locking pin retracted during vibration. This resulted in a design change to the pinpuller assembly and subsequent replacement.

(2) Transmitter B power output was noted as steadily decreasing. This transmitter was finally changed-out at AFETR.

(3) The extension force for leg 3 was above specification and the spacecraft A frame was replaced prior to vernier engine vibration.

e. Vernier engine vibration. The VEV testing was performed between August 25 and August 30, 1967. During the test, only one significant anomaly was encountered—it being associated with the RADVS beam 3 preamplifier. The doppler velocity sensor (DVS) antenna was returned for rework to Ryan, San Diego, where a broken solder joint was discovered and repaired. The unit was then subjected to partial FA test and returned to AFETR just before the start of PVT 1.

The SC-6 weight, balance, and alignment was performed on August 31, 1967, without incident, and preparation for shipment commenced the following day. Shipment of SC-6 to AFETR occurred on September 4, 1967, aboard a chartered commercial jet.

f. Air Force Eastern Test Range tests. Launch readiness tests at AFETR were started on September 5, 1967. After reassembly of the spacecraft, the test plan was conducted as outlined in Subsection B. Major problems during the testing were:

(1) During RADVS ranging tests in PVT-3, problems were experienced with KPSM tracker lock. The KPSM was replaced and retested.

(2) Shutoff time of vernier engine 2 was found to be excessive and erratic during engine functional checks. This engine was replaced with a flight spare.

(3) During PVT 5, the alpha scattering instrument power supply indicated a ripple in excess of what was expected. The alpha scattering electronics unit was returned to the University of Chicago for repair and the unit replaced on the spacecraft during final flight preparations in ESF.

After an uneventful countdown, SC-6 was successfully launched on November 7, 1967.

7. Surveyor VII Spacecraft (SC-7)

System testing of the SC-7 began during the latter part of May 1967 with the start of ISCO. Most problems were encountered during the STV tests and the major deviation from the basic plan was the omission of the CST phase.

a. Initial system checkout. The ISCO test phase was completed in the middle of August 1967. Due to the loss of control items to SC-5 and 6 and other problems, the ISCO tests were interspersed in the MS/EMI test phase.

b. Mission sequence and mission sequence/electromagnetic interference. The MS/EMI tests were initiated during the last week of July 1967, interspersed with ISCO tests, and completed by the end of August 1967. The significant problems pertained to the TV camera and the vernier engine strain gage channels.

During MS tests, no video resulted when TV camera SN 14 was operated in open shutter mode. The camera was replaced with SN 13.

The vernier engine strain gage 1 telemetry channel fluctuated randomly. The problem was isolated to the strain gage amplifier which was then replaced with SN 8. During SRT prior to MS/EMI 3, all three vernier engine strain gage channels (P 18, 19, and 20) fluctuated when the new amplifier was turned on. The unit tests later verified that the strain gages were susceptible to electromagnetic interference. As the EMI environment is not encountered during the flight, the strain gage performance was considered acceptable.

c. Solar-thermal-vacuum test phase. Several setbacks were encountered during STV testing and resulted in phases 1A, 2A, 3A, 3B and 4B. The STV test preparation started during the first part of September and the tests were concluded around mid-October 1967.

The only major problem during STV prepartion was the replacement of the three oxidizer tanks. Two tanks were replaced because of a suspect O-ring problem and the third because of a low tank-heater insulation resistance.

The first high-sun sequence (STV-1A) was started on September 20, 1967, and was terminated after about 40 hr because of low power output from transmitter B. After transmitter replacement and retest, one of the spacecraft receivers also developed a problem and was replaced prior to returning to STV testing. Two different vernier engines were installed on the spacecraft because of a questionable lubricant used during manufacture.

High-sun sequence (STV-2A) was initiated on September 25, 1967, but had to be terminated after about 8 hr because of a chamber leak.

After repair of the chamber leak, test (STV-3A) was started again on September 26, 1967. The test was completed and the following problems were encountered.

(1) During the second terminal descent, the AMR marked immediately after the AMR enable command. This later resulted in replacing the AMR with another flight unit.

(2) The 600-line mode of operation of the TV system was abnormal. This resulted in repair and retest of the camera prior to vibration.

Sequence STV-3B, at low-sun conditions, was initiated without repair of the problems noted in run 3A. However, because of a battery failure, resulting in high battery temperature and pressure, the test was aborted at 44 hr. An additional problem was a failure of flight-control thrust-phase power noted during the second terminal descent of the planned test.

Because of the problems noted during the previous missions, it was decided after repair to qualify the replaced and repaired equipment with a 65-hr low-sun MS STV-4B. This mission was started on October 8, 1967, and was considered successful except for minor problems.

d. Vibration test phase test. Initiated on October 18, 1967, vibration tests were completed in approximately three days. All control items (electrical) including the TV camera, were flight units, and the tests met all objectives.

Following the vibration tests, a plugs-out MS test was performed at ambient conditions. The test was performed on October 21, 1967, and proper spacecraft operation was noted.

e. Vernier engine vibration test. The VEV tests were performed on October 27, 1967. Spacecraft hardware consisted principally of flight type items with the exception of SDC SN 3. The test met all objectives with no significant spacecraft problems.

In preparation for VEV test, however, a KPSM (SN 6) instrument ground-shorted to its preregulator output, resulting in excessive current to the spacecraft ground via SDC ground and the flight control return. The excess current did not damage the FCSG, but it burned open a printed circuit line in the SDC. All three control items were removed from the spacecraft, and the KPSM and SDC were repaired.

Following VEV tests, weight, balance, and alignment tests were performed from October 30 to November 1, 1967. There were no spacecraft problems.

The CST at GD/C, San Diego were omitted and the spacecraft was shipped by air from HAC, El Segundo to AFETR on November 5, 1967.

f. Air Force Eastern Test Range tests. Launch readiness tests at AFETR started on November 6, 1967, and the following were the significant problems encountered during this test phase:

(1) During PVT 1, in the initial series of tests, a shift of AGC calibration was noted on receiver SN 21. The receiver was removed, checked and tested in the unit area at HAC, El Segundo, and replaced on the spacecraft for flight.

(2) The retro accelerometer, used for telemetry only, was replaced prior to launch because of unexplained output variations.

(3) The KPSM was replaced during PVT 3 tests with another flight unit because of undesired noise bursts.

(4) During bladder leak tests on November 21, 1967, the fuel tank 1 leakage exceeded the specification for leakage. Fuel tank 3 was exceedingly close to the upper limit and both tanks were changed.

(5) An O-ring was replaced in the helium tank pressure transducer during final flight preparations. This was discovered during the pressure decay tests.

(6) An access cover was accidentally dropped into the TV camera body. The camera had to be removed from the spacecraft, returned to the unit area in HAC, El Segundo, fixed, and retested and then returned to the spacecraft at AFETR. This was accomplished prior to the J-FACT.

(7) A problem was detected in the alpha scattering instrument during on-pad tests during the J-FACT period. The problem was later isolated to defective guard detectors. Several proton-guard detectors were replaced prior to final testing; no further problems were noted.

After an uneventful countdown, SC-7 was launched on January 7, 1968.

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XII. Quality Assurance and Reliability

In direct support of *Surveyor* Project goals and in compliance with quality assurance and reliability (QA&R) provisions of NASA Contract NAS 7-100, Task Order RD-5, the JPL *Surveyor* quality assurance and reliability office was established to manage *Surveyor* QA&R programs. The manager of QA&R was responsible for the overall QA&R programs. Implementation of QA&R functions was the responsibility of the individual *Surveyor* Project system and hardware suppliers: Lewis Research Center (LeRC), Hughes Aircraft Co. (HAC), Deep Space Network (DSN), and Mission Operations System (MOS). Launch vehicle quality and reliability were the responsibilities of NASA, LeRC as directed by NASA Headquarters through its *Centaur* Program office.

The JPL QA&R activities were staffed with appropriate personnel and supported project efforts to achieve successful missions with quality hardware. Particular emphasis was placed on a complete and integrated program of component, subsystem, and system testing to increase mission reliability and assure mission success.

The *Surveyor* QA&R office reported functionally to the project manager's office and administratively to the JPL QA&R office. The *Surveyor* QA&R office assisted in the conduct of mission and major system reliability oriented tasks and recommended measures for improvement of mission success probabilities. The *Surveyor* QA&R office monitored mission operation readiness tests and Spacecraft System and subsystem tests. The *Surveyor* QA&R office was responsible for upgrading and effective reorientation of HAC QA&R program plans midway through the program. The HAC program plans were brought into consonance with applicable tasks of NASA reliability publication NPC 250-1 (Ref. XII-1) and the NASA quality publications NPC 200-1 (Ref. XII-2) and NPC 200-2 (Ref. XII-3).

Specifically, the *Surveyor* QA&R office audited the activity effectiveness of the established program plans and provided assistance to the project office on reliability and quality activities. The QA section monitored contractor receiving, in-process, and final inspection; it also represented the *Surveyor* Project office in accepting and certifying flight hardware, flight spacecraft, operational support equipment (OSE) and government-furnished equipment (GFE). The reliability section performed a control and support function as the JPL interface with the HAC trouble/failure reporting (T/FR) system, conducted a control and technical review for the GFE JPL problem/failure reporting (P/FR) system, and provided a failure reporting interface with the DSN/Deep Space Instrumentation Facility (DSIF) and LeRC activities. Where reliability section activities were in direct support of the Spacecraft System (primarily the HAC/JPL T/FR loop, subcontractor control, and an electronic parts control function), full coordination was maintained with that section of the JPL *Surveyor* Spacecraft System which had the assigned responsibility for system design reliability engineering.

A. Quality Assurance

1. Quality Program Provisions

The nature of the *Surveyor* mission dictated that specific quality approaches be defined and controlled if the paramount overall objective of mission success was to be realized. The importance of the QA&R program comes into proper perspective when full consideration is given to the complexity of the Spacecraft System and its related support equipment, its operating environment, and the potential finality of the mission launch.

*a. **Objectives.*** Quality assurance (QA) programs were designed, therefore, to accomplish the foregoing by assuring that the *Surveyor* spacecraft, scientific experiments, and associated ground support systems would comply with established contractual design, quality, and reliability requirements. This was accomplished by JPL imposing requirements (appendix) on the contractor to establish and implement controls through the various phases of design, development, fabrication, assembly, test, and end usage. Conformance was verified by auditing and monitoring contractor activities, and by performing inspections and tests at various key locations during the program.

*b. **Requirements.*** The *Surveyor* Project development plan specified that quality requirements be compatible with the applicable provisions of NASA publications NPC-200-1A, NPC-200-2, and NPC-200-3 (Ref. XII-4). Quality requirements were defined, however, by the application of MIL-Q-9858 (Ref. XII-5) to the *Surveyor* Spacecraft System contractor and JPL quality control (QC) requirements to the other first-tier subsystem contractors. The application of quality requirements to *Surveyor* activities performed at the laboratory was accomplished through the issuance of *Surveyor* Project quality directives and instructions. Basic *Surveyor* quality requirements are summarized in the appendix.

*c. **Concepts.*** These quality requirement documents were imposed in lieu of specified documents, i.e., the NPC-200 series, because at the time the basic *Surveyor* contract was issued for bid (1960) and subsequently let to HAC (1961), the only major government quality specification available for usage was MIL-Q-9858. Therefore, MIL-Q-9858 was utilized and subsequent review of the HAC Quality Plan was made to determine its conformance to NPC-200-2. From this review, it was decided that the continuous upgrading of HAC quality system and corresponding revision of the *Surveyor* QA plan to reflect this upgrading, the intent of NPC-200-2 had been fulfilled. The contract itself was not changed to cite NPC-200-2, as discussions on the subject indicated that possible contractual implications might occur from its inclusion.

A JPL specification based on earlier *Mariner* and *Ranger* quality requirements was written to identify *Surveyor* quality requirements. This specification provided more rigorous requirements at the subsystem level than NPC-200-3, and its enforcement corresponded to partial application of NPC-200-2. The specified approach and requirements were those deemed necessary to meet project goals and requirements in relation to the technical scope and complexity of individual items of hardware to be procured.

It should be noted that problems were encountered in the process of having requirements established on scientific instruments provided by outside (non-JPL) principal investigators. The basic cause of these problems was attributed to the black-box concept and the university-designed-and-built experimental-model concept.

Final resolution was not obtained due to deletion of the 2500-lb spacecraft configuration and its associated scientific payload. The alpha scattering instrument requirements are discussed under the subsystem portion of this section.

2. Spacecraft Quality System

*a. **Organization.*** The JPL *Surveyor* QA organization originated in May 1963 with the establishment of the *Surveyor* QA&R office. *Surveyor* QA had two roles to play in order to fulfill its project obligations: first, the monitoring, auditing, and evaluating of contractor and subcontractor quality activities; and second (but minor, due to limited quantities), the inspection of JPL-built items (e.g., television filters and magnets) and the monitoring of on-lab testing (e.g., vernier engines and solenoids).

From 1961 to 1963, laboratory QA personnel assisted individual project personnel, as needed, with efforts directed primarily toward negotiations and development of project requirements. In addition and as a result of prior negotiations, Air Force QA personnel provided a HAC plant surveillance service to the *Surveyor* Project from May 1962 to June 1963 and inspection service from June 1963 to July 1965.

In mid-1964 the project sharply increased its staffing requirements and deepened its involvement in monitoring and evaluating contractor performance. In consonance with these changes, the *Surveyor* QA organization began to augment inspection services being provided by Air Force QA, with spacecraft-experienced inspection personnel from the then recently completed *Mariner* and *Ranger* Projects.

By December 1964 the *Surveyor* QA staff included experienced personnel in the fields of subcontractor control, propulsion, manufacturing, and scientific instrument development. A systems and procedures group and a staff engineering function were also activated in support of these disciplines.

Thus, *Surveyor* QA activities were organized functionally in relation to task performance and location of activities. *Surveyor* QA utilized the existing on-lab quality services to support on-lab activity (e.g., mechanical/shipping, inspection, environmental testing, etc.). Later in the program, these services were extended to include coverage of the solar panel at Electro-Optical Systems (EOS) and the alpha scattering experiment at the University of Chicago. However, *Surveyor* QA retained project technical responsibility in all cases. During significant *Surveyor* activities, personnel were assigned to residency at the following activities and subcontractors:

Subcontractor or facility	Activity
Holloman Air Force Base	T-2 program
Reaction Motors Division	Vernier engines
Space Technology Laboratory	Vernier engines
Link	Ground data handling/television (GDH/TV)
Electro-Optical Systems	Solar panel
Air Force Eastern Test Range	Spacecraft testing and launch
Combined Systems Test (San Diego)	Spacecraft testing
HAC, El Segundo, Culver City, Airport facilities	Manufacturing, assembly, test

During 1966, additional changes were made to JPL *Surveyor* QA organization, to support spacecraft activities at Air Force Eastern Test Range (AFETR).

This increase in field activity offset a decline in the personnel required in the manufacturing effort, and personnel no longer needed to support the fabrication cycle were effectively used to support launch preparation activities at AFETR.

b. Responsibilities. *Surveyor* QA was assigned the basic responsibilities of:

(1) Establishing and/or approving quality requirements.

(2) Assuring implementation of requirements.

(3) Assisting project in quality related matters.

(4) Evaluating and/or reporting quality activities.

In discharging these responsibilities, *Surveyor* QA activities were divided into two functional areas: one basically responsible for QA requirements, evaluations, project interface; the other primarily responsible for implementation control and/or inspection activities.

Surveyor quality assurance directives (SQADs) were released to define *Surveyor* QA responsibilities. Air Force QA activities and responsibilities were established through letters of delegation and the JPL-approved Air Force Quality Plan.

The original HAC Spacecraft System contractual concepts considered quality costs as part of the cost of each individual engineering discipline. However, as a result of contract modifications negotiated in early 1964, SQA was charged with the additional responsibility of negotiating and assessing the total contractor QA costs and efforts for the *Surveyor* Project.

A detailed breakdown of total JPL *Surveyor* QA responsibilities is contained in Table XII-1.

c. Reporting. Reporting, particularly in this program, played an important role in gauging contractor performance in establishing and maintaining a timely and effective QA system.

The QA reporting system that evolved, provided written communication with project management, engineering, and the contractor on matters pertaining to results

Table XII-1. Jet Propulsion Laboratory *Surveyor* quality assurance responsibilities

Task	Definition
Establishing/approving requirements	Establish project QA requirements in consonance with applicable tasks of NPC 200-1A, NPC 200-2, NPC 200-3 Prepare and develop SPQD s Review and approve HAC, and JPL subcontractor QA plans and procedures Coordinate QA requirements with *Surveyor* Project activities Participate in review of project documents
Project assistance	Assess, evaluate, and negotiate contractor quality costs and level of effort Provide quality assessment in hardware design reviews and failure review boards (FRB) Provide quality considerations as a member of *Surveyor* Design Team Assist in evaluating spacecraft flight worthiness by participation in consent-to-ship/consent-to-launch (CTS/CTL) evaluations Participate in determining acceptability of flight units and major support equipment as a member of unit consent-to-ship (UCTS) boards Assist in coordinating and resolving launch vehicle QA problems Provide project interface to contractor/subcontractor QA activities Resolve QA problems
Implementation assurance	Perform surveys of JPL subcontractors to determine adequacy of QA system Perform in-process inspection of JPL fabricated items
	Perform receiving and shipping inspections of all *Surveyor* items processed through JPL Monitor handling, testing, environmental controls of *Surveyor* items processed through JPL Participate in accepting flight items to be delivered as government-furnished equipment (GFE) Audit QA system implementation for compliance to QA procedures Perform random inspections of hardware to ensure flight workmanship Establish mandatory inspection points at critical and/or problem activities to ensure flight workmanship Assess adequacy of controls and implementation at HAC subcontractors Monitor and assess adequacy of tool and gage, calibration, handling, packaging, shipping, configuration control, and environmental controls Monitor and verify adequacy of manufacturing and special process controls, and manufacturing test planning Perform final inspection of spacecraft before launch Represent JPL as customer representative on MRB review
Evaluation/reporting	Identify all system and hardware discrepancies to cognizant activity and obtain resolutions and corrective action Evaluate contractor QA performance Provide a reporting system to identify problems, performance, and status

of inspections performed, trends of contractor performance, identification of QA problems, and material review actions. A description of the reporting methods utilized is outlined below:

Hardware and system discrepancies were identified and corrective action was achieved as described below:

Hardware discrepancies were recorded on appropriate contractor QA documentation. Customer complaints identified in this manner required customer buyoff of corrective action taken before the item was cleared.

1. *Discrepancy Notice.* Repetitive hardware or quality system problems were listed on a discrepancy notice (DN) form. The DN, which identified a problem and requested corrective action, was utilized to notify the cognizant contractor QA supervisor of action required. Contractor response was reviewed for adequacy by JPL *Surveyor* QA. Followup action was initiated where necessary to ensure that adequate corrective measures were taken and maintained.

2. *Corrective action request.* The corrective action request (CAR) served the same purpose as the DN, except that it was reserved for more critical problems. The CAR was issued by the JPL *Surveyor* QA manager through contractual channels to the contractor QA manager, to focus management attention on particular problems.

3. *Inspection summary sheets.* This report was a score sheet of inspections performed by JPL QA representatives at mandatory product control (MPC) points in the

assembly, test, and acceptance cycle of flight hardware. Analysis of these reports identified areas of weakness or strength in the contractor's fabrication control, and further served as a prime source of information for computing statistical data.

4. *Unit history logs.* Logbooks were maintained by QA representatives as a means of tracking the history of each spacecraft through assembly, system test, and launch operations.

5. *Activity reports.* Prepared weekly by JPL QA supervisors, activity reports summarized quality activities for areas covered and apprised management and engineering of the day-to-day activities and problems encountered. Secondary function was to provide cross-pollination of knowledge and information among various elements of the QA organization.

6. *Quality status report.* Issued monthly by *Surveyor* QA office, this report provided visibility of current hardware quality and trends and included:

(1) Statistical data comprising percent defect trends for hardware presented to the customer.

(2) Defect characteristics by distinct hardware categories.

(3) Significant problems requiring attention by project organization.

(4) Summary of outstanding DNs.

(5) Material review board (MRB) summaries by hardware and type of defect, including recurrence rates.

d. Audits. To ensure flight hardware integrity, it was necessary to perform audits of the contractor's quality activities to ascertain compliance with requirements.

The audit activity can be summarized in four major categories: survey and audit of JPL subcontractors, major QA system program audits of HAC, continuous area audits of HAC, and audits of HAC subcontractors.

1. *Jet Propulsion Laboratory subcontract audits and surveys.* Quality assurance performed survey type audits of all major JPL subcontractors to evaluate effectiveness of quality system and validity of plans prior to approval of the quality plan.

2. *Major audits of Hughes Aircraft Co.* There were two separate major audits of HAC by JPL and NASA. The first of these audits, performed by NASA–Western Operations Office (WOO) from April through June 1963, was not unique to the *Surveyor* project. Major problems found and subsequently corrected were:

(1) Inadequate QC organization and relationship to management.

(2) All MRB membership composed of HAC personnel only.

(3) Inadequately trained and certified soldering personnel.

(4) Lack of adequate inspection procedures.

As *Surveyor* QA was reaching peak manloading and hardware deficiencies were being observed in significant numbers, JPL decided to perform a second major audit of all HAC QA activities to evaluate the effectiveness of its QA system. This audit took place in February and March 1965. Major problems found and subsequent corrections were:

(1) Inadequate subcontractor control and requirements (revision of supplier QA plan and improved source inspection).

(2) Lack of control of environmental and contamination requirements.

(3) Release of various quality documents and ultimate establishment of particulate evaluation facilities at AFETR.

(4) Undefined (class II) support equipment quality requirements: revision and release of 37 series SQADs.

(5) Deficient application of workmanship standards (implementation of a training program).

(6) Multiplicity of quality documents and conflicting requirements: revision of QA plan, JPL SQA review/approval of SQADs, HAC QA project review of all QA documents, and ultimate (late 1966) revision of divisional QA document system.

(7) Minimal project QA control (increased staffing and performance of hardware reinspections).

3. *Continuous area audits.* The need to evaluate and monitor the effectuality of QA system was recognized and implemented by requiring that work areas without a JPL MPC activity were to be reviewed periodically to ensure that hardware was being adequately controlled and properly built. In those areas in which MPC was performed, the QC system effectiveness was reviewed concurrently with detailed hardware inspections. Problems encountered were those normally associated with hardware production.

4. *Hughes Aircraft Co. subcontractor audits.* To establish an initial level of confidence, JPL QA conducted audits of each critical supplier. Analysis of the 28 audits performed indicated there were three major common deficiencies:

(1) Quality assurance contractual requirements.

(2) Insufficient application of existing quality requirements.

(3) Ineffective contractor source personnel.

Subsequent actions resulted in a complete rewrite of the Spacecraft System contractor's controlling supplier QA plan, and the initiation of a personnel training program to ensure uniform enforcement of supplier control requirements. Followup audits showed definite improvement in the performance of most suppliers and in the response of source personnel. There were exceptions, however, and JPL handled these by imposing MPC at supplier level to ensure compliance.

e. Material review board. As in any large and complex project, much hardware failed to conform to engineering criteria during manufacture, assembly, and test.

Therefore, a program was implemented to review discrepant hardware for determination of its usability, nature of repair and rework, and to initiate corrective action to prevent recurrences. This material review system at HAC was supplemented by the T/FR system for resolution of functional discrepancies which occurred during subsystem flight acceptance (FA) testing and Spacecraft System Testing.

Surveyor QA personnel represented the JPL *Surveyor* Project as the customer representative on the HAC MRB. To assure proper technical evaluation and awareness of problems, the cognizant JPL engineering representative was contacted for his concurrence with MRB evaluations and decisions.

Beginning in June 1966, a summary (Table XII-2) of MRB actions by subsystem was included in the monthly JPL-SQA status report. Information in this summary was continually updated to reflect hardware completion, problem solution, and corrective action taken. In addition, charts depicting overall trend, defect characteristic trend, and corrective action cause trends were maintained in the project control room.

Results of tracking and disseminating this information were as follows:

(1) Increased attention to problem areas.

(2) Less reliance on customer for determining disposition.

(3) Increased emphasis on corrective action.

(4) Improved corrective action.

(5) Provided a tool for evaluating contractor performance changes for specific periods.

f. Inspection. The most important single action that can be taken in measuring and/or ensuring the quality level of the product and system is that of performing in-process and final inspection of the hardware. The performance of this inspection, being a project requirement, was accomplished primarily by the responsible contractor's QC organization. However, to ensure contractor performance and awareness of the expected flight quality, source inspection including key and critical in-process operations and final spacecraft inspections were also performed by JPL inspectors.

Resident JPL inspection personnel were assigned to HAC, EOS, RMD, STL, and Link to perform necessary inspections. Itinerant JPL inspection personnel visited other JPL subcontractors. Hardware fabricated or tested at JPL was inspected by JPL inspectors to ensure conformance to engineering criteria.

The JPL-SQA inspection activities at HAC were basically in the three areas of launch support, spacecraft assembly, and control item fabrication, as described below.

1. *Launch support.* The JPL-SQA inspection activities at combined systems test (CST) and AFETR were performed as a parallel function to all HAC inspection. Due to the complexity of the mission and the nature of the spacecraft, all spacecraft operations (except electrical test) including assembly and disassembly, rework and repair, modifications, adjustments, alignments, etc., were witnessed 100% by JPL inspection personnel.

2. *Spacecraft assembly.* Inspection activities at spacecraft assembly were primarily oriented toward a monitoring function except that all control items (items or units whose next higher assembly is the spacecraft itself), when installed on the spacecraft, received a mandatory visual inspection and documentation review by SQA. All activities, except functional electrical test, were inspected 100% by HAC QC.

Figures XII-1 and XII-2 show spacecraft and control item flow with typical inspection points.

*	=	JPL CONCURRENCE/ACCEPTANCE
ES	=	EL SEGUNDO (CALIFORNIA)
ESA	=	EXPLOSIVE–SAFE AREA
FA	=	FLIGHT APPROVAL
GFE	=	GOVERNMENT FURNISHED EQUIPMENT
GD/A	=	GENERAL DYNAMICS/ASTRONAUTICS
MPC	=	MANDATORY PRODUCT CONTROL
PSL	=	PAYLOAD SYSTEM LABORATORY
QA	=	QUALITY ASSURANCE
SCF	=	SPACECRAFT CHECKOUT FACILITY
TA	=	TYPE ACCEPTANCE
UCTS	=	UNIT CONSENT–TO–SHIP

Fig. XII-1. Spacecraft flow chart

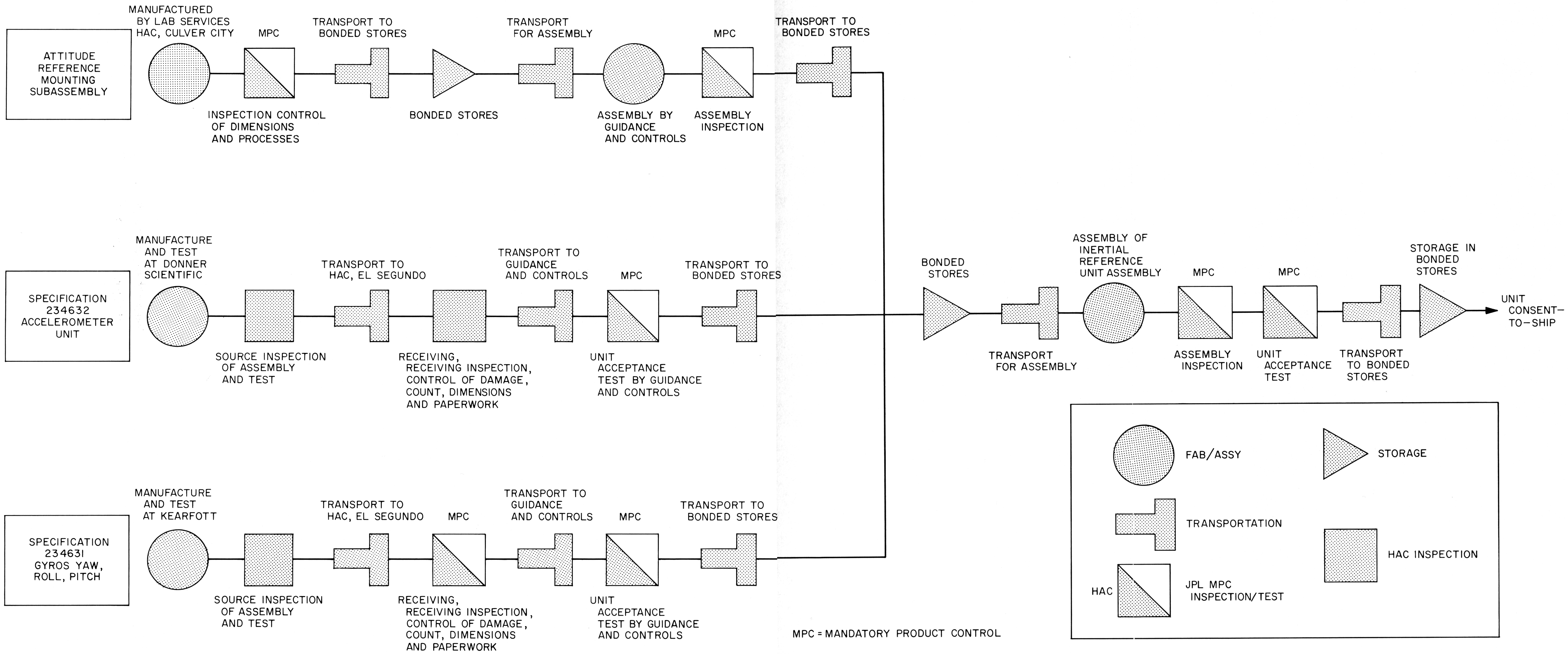

Fig. XII-2. Typical control item flow

Table XII-2. Monthly material review summary — Jet Propulsion Laboratory *Surveyor* quality assurance (representative)

Material review board summary	No. of items MRB	No. of discrepancies dispositioned	Disposition				Discrepancies								Corrective action											
			Use as is	Reworked	Scrap	Suspended action	Improper assembly	Damage	Finish	Dimensions	Functional	Contamination	Other	Total	Reworked	Operator cautioned	Vender notified	Tooling changed	Protection changed	Revised planning	Drawing specification	Procedures changed	Under investigation	Additional documentation	None accomplished	None required
Total this report period Aug '67	89	137	65	24	33	15	4	40	8	9	12	12	52	86		9	1	1			4	3	2	13	2	57
Total reported for— Jul '67	89	127	99	15	0	13	10	16	17	4	29	3	48	64		5	4	1		2	6	3		14	5	36
Total reported for— Jun '67	304	428	291	75	9	53	23	53	59	130	17	22	124	182		26	4	4	2	2	16	11		35	13	81
Total for last 90 days	482	692	455	114	42	81	37	109	84	143	58	37	224	332		40	9	6	2	4	26	17	2	62	20	174
Systems, MRB breakdown																										
Instrument development																										
Total this report period	13	18	13	2	1	2		3	1	1	6		7	13		1						1		2		9
Propulsion																										
Total this report period	24	42	10	1	25	6		16	1	2	3	11	9	21		1								4		17
Mechanical																										
Total this report period	10	11	4	4	2	1	2	2	1	2			4	10		1						2		1		6
Power and guidance																										
Total this report period	8	11	6	1	4			6					5	8										1		8
Telecommunications																										
Total this report period	21	33	15	14		4	1	10	3	2	3		14	21		4		1			4			5		11
MRB, AFETR																										
Total this report period	13	22	17	2	1	2	1	3	2	2		1	13	13		2	1						2		2	6

3. *Control item fabrication.* A 100% inspection by the contractor was required for all control item fabrication, assembly, and test. Monitoring by sampling inspection was performed by AFQA in accordance with their QA plan. To identify and correct poor workmanship practices and to introduce microscopic inspection techniques, JPL-SQA initiated MPC and established mandatory inspection points (Table XII-3) in the fabrication cycles.

All HAC *Surveyor* spacecraft QA plans submitted to JPL for approval prior to June 1963 were rejected for inadequate workmanship standards. These rejections were substantiated by the findings of the JPL QA system and hardware audit (1965) and the NASA-WOO audit (1963). Both of these audits identified workmanship as a major problem. As a result, acceptable workmanship standards were not available for spacecraft and subassembly in the early days of HAC activity on the project. Even after acceptable standards were developed, they were not being utilized in the fabrication of hardware since the majority of the working level personnel were not aware of the approved *Surveyor* workmanship standards.

Based on these findings, which permitted the application of nonuniform standards across the board, JPL QA implemented MPC (early 1965) and established mandatory inspection points in the fabrication cycles to identify and correct poor workmanship practices. This operation further substantiated previous findings regarding workmanship. Equipment coming off the lines was subjected to excessive rework before delivery to inspection because soldering, wiring, and assembly were unsatisfactory. Conditions noted in the MPC operation resulted in a number of operational and hardware changes, which are shown in Table XII-4.

Because of obvious problems with hardware then in production, disassembly inspection of the previously assembled *Surveyor I* spacecraft become necessary to verify level of workmanship and hardware integrity.

Table XII-3. Mandatory product-control inspection points

Type of hardware	Mandatory inspection points
Electronic assembly	Before final closure Note: Additional points are required if installation of subassembly/component masks from view significant areas of the primary assembly, i.e., group of solder joints, critical surfaces
Mechanical assembly subassembly level	Before surface finish process Before any installation obscuring critical assembly of components, i.e., gears, motors, electric wiring, spacing adjustments Any control items that are assembled for functional tests, then disassembled for painting or other thermal finishes and reassembled a final time to produce the control item, will (during each assembly cycle) be inspected before surface-finish process and before any installation obscuring critical assembly of components, i.e., gears, motors, electrical wiring spacing adjustments
Assembly level	Upon completion of final assembly and before start of FA and TA testing
Cables	Before closure and encapsulation of connector backshells (solder joints to be visible)
Command console and spacecraft operational support equipment	Mandatory product control points are required at the finished-drawer or panel level, and on completion at the rack level. However, additional MPC points should be inserted if the construction is such that visibility is obscured at the drawer or rack level
Rework	Mandatory product control requirements for rework are governed by the same conditions as those established for new production. Resubmittal to JPL MPC is required on all rework accomplished after a unit has been previously accepted by JPL. In the case of multiple rework operations, only one submission is necessary, provided all of the rework accomplished can be reinspected at that point. Opening of cover plates and normal disassembly without additional work is not to be considered rework. All rework must be authorized and controlled by proper planning documents with proper MPC points designated
Control item control point	An MPC point at the Culver City facility will be required. At that time JPL QA will review all associated documentation and perform a cursory inspection of the hardware for proper packaging
Spacecraft	An MPC is required to control items prior to installation on the spacecraft. Jet Propulsion Laboratory QA will review documentation, inspect hardware, and monitor assembly testing as necessary to verify conformance to established requirements

Table XII-4. Typical significant action results of mandatory product control

Problem	Action
1. Excessive wear in the spacecraft cabling due to the extended testing. Cables were found to be worn to the point of breakage through work handling of strands	Extensive reinspection and replacement or repair, and ultimate redesign of a number of cables
2. Damaged hardware due to rough handling by personnel brought about immediate pressure applied in all areas and on all levels	Training classes were initiated with attendance required by all personnel involved
3. Assembly handling fixtures were not utilized during critical assembly phases. Due to the physical design of many systems, cable harnesses frequently act as hinges, since they are flexed each time the unit is handled	Various measures, including issuance of stop orders against the hardware, brought about the development of fixtures to maintain hardware integrity
4. Cleanliness problems involving metal fragments and loose bits of solder in flight hardware. Cleaning methods were haphazard and inadequate. Air hoses, used to blow off debris, drove particles into the equipment as often as not	Initiation of vacuum-cleaning methods, preparation and enforcement of housekeeping rules, and movement of certain hardware to properly controlled areas
5. Backside solder problems experienced on the lines indicated that the contractor was unaware of the behavior of molten solder in a confined space, especially of solder extrusion from the backside of a plated-through-hole circuit board with a sealed underside. (When solder is molten, heat expansion causes beads of solder to form off the pads, frequently resulting in short circuits on closely spaced circuit lines)	Resulted in scrapping a number of boards and major rebuilding of many others, as well as devising new assembly techniques to compensate for the problem
6. Investigation revealed that controls on tooling used to position the spaceframe during assembly were not then under calibration or dimensional control. Therefore, effectual resolution of such radical problems in vehicle assembly and test, as weight, balance, and alignment would have been jeopardized	Tooling was provided and spaceframes were subsequently inspected. Dimensional tooling problems were corrected and flight hardware problems were evaluated and subsequently considered in flight configuration analysis
7. Optical alignment capability was identified as a potential problem. Attitude control and midcourse capabilities of the spacecraft are predicted on the precise positioning of the various thrust motors. This positioning is entirely dependent on optical alignment. Investigation revealed that the contractor did not have this capability. No trained personnel were available, procedures were lacking, the optical dock had not been calibrated, optical tooling had not been proofed or calibrated, and no plans existed that would develop capability in this direction	Six months of concentrated effort by JPL QA in conjunction with JPL engineering eliminated the problem. Tooling was modified, procedures were released, and HAC personnel were trained
8. Removal and elimination of in-line splices from cables was another reliability problem. In-line splices, a notoriously weak means of joining two wires, proved to induce hardening breakage when flexed in a cable. Several failures have been induced from the use of in-line splices, a once-recommended method	In-line splices were (in general) eliminated in favor of butt splices

This inspection resulted in the replacement of a number of major control items of substandard workmanship.

The results of mandatory inspections were tabulated and made available to the cognizant HAC QA supervisor for initiation of corrective action and problem resolution. In addition, results of mandatory inspections were reported monthly to show trends by subsystems for hardware and software problems (Table XII-5) and for the overall efficacy (Fig. XII-3) of MPC.

With MPC by JPL, HAC instituted a reassurance activity to reduce the number of defects presented to the customer and to improve workmanship and inspection techniques. As a result of this activity and associated product improvements, the subsystem JPL-MPC points were eliminated as specific hardware came under control, thus permitting JPL-SQA to concentrate the needed attention on spacecraft activities and still provide assurance that subsystem quality was maintained.

g. Environmental and cleanliness controls. Successful operation of the spacecraft during mission performance dictated that design, fabrication, assembly, testing, handling, and launch environments and activities consider and provide for contamination and environmental control.

As a result of these considerations, rigid requirements were imposed to:

(1) Provide certified clean gas, and hardware contamination controls.

(2) Limit area access and govern the use of protective clothing.

(3) Provide approved cleaning sources.

(4) Establish approved work areas by level of cleanliness required for specific items.

Table XII-5. Spacecraft mandatory product control summary

Mandatory product control system	Units Inspected	Units Defective	Hardware: Improper assembly	Bonding encapsulation	Damage	Finish	Identification	Soldering welding	Wiring	Dimensions	Contamination	Other	Software
Instrument development													
Combined[a]	45	2	—	—	1	—	—	—	—	—	—	—	—
Hardware	—	1	—	—	—	—	—	—	—	—	—	—	—
Software	—	1	—	—	—	—	—	—	—	—	—	—	—
Propulsion													
Combined	17	1	—	—	—	—	—	—	—	—	—	—	1
Hardware	—	—	—	—	—	—	—	—	—	—	—	—	—
Software	—	1	—	—	—	—	—	—	—	—	—	—	—
Mechanical													
Combined	206	7	—	2	—	—	—	—	—	—	1	—	5
Hardware	—	3	—	—	—	—	—	—	—	—	—	—	—
Software	—	5	—	—	—	—	—	—	—	—	—	—	—
Power and guidance													
Combined	55	6	—	—	1	—	—	—	—	—	—	—	6
Hardware	—	1	—	—	—	—	—	—	—	—	—	—	—
Software	—	6	—	—	—	—	—	—	—	—	—	—	—
Telecommunications													
Combined	36	4	—	1	1	—	—	—	—	—	1	1	1
Hardware	—	4	—	—	—	—	—	—	—	—	—	—	—
Software	—	1	—	—	—	—	—	—	—	—	—	—	—
Combined units inspected	363	—	—	—	—	—	—	—	—	—	—	—	—
Combined units defective	—	20	—	—	—	—	—	—	—	—	—	—	—
Total this report period	Apr	—	—	3	3	—	—	—	—	—	2	1	13
Total reported for—	Mar	—	4	8	35	2	3	3	—	2	5	15	75
Total reported for—	Feb	—	7	4	22	21	1	5	—	24	2	29	113
Total for last 90 days	—	—	11	15	60	23	4	8	—	26	9	45	201

[a]A combination of defective units can contain more than one defective characteristic.

h. Consent-to-ship/consent-to-launch. A requirement for holding a formal review (CTS/CTL) was established by JPL and NASA (spacecraft only) prior to:

(1) Shipment of each spacecraft to CST/AFETR.

(2) Spacecraft launch.

(3) Shipment of T-2N to Holloman Air Force Base (HAFB).

(4) Shipment of system test equipment assemblies (STEAs) to CST/AFETR.

(5) Shipment of command data consoles (CDCs) to CST/AFETR/tracking stations.

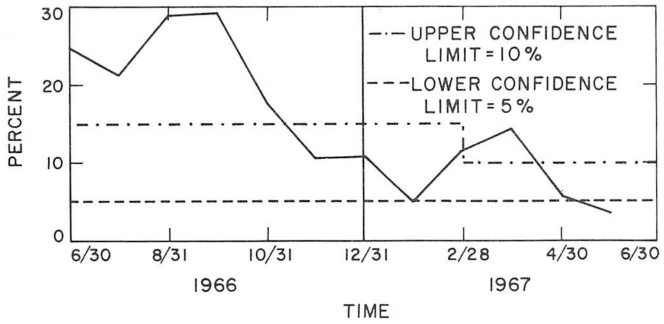

Fig. XII-3. Percent defective trend chart—mandatory product control, flight hardware

Participation in these activities by JPL-SQA included review of CTS data packages with cognizant JPL engineering personnel. These activities were supported by HAC QA and defined by QA documentation.

The T/FR portion of the reliability section of this report describes more completely CTS/CTL activities.

3. Spacecraft Contractor Quality Assurance System

Ultimately, HAC defined the contractual *Surveyor* QA requirements internally by the development of a quality plan and the issuance of SQADs. Implementation of these requirements was a composite, using the regular HAC QA system and developing additional provisions peculiar to the *Surveyor*, including:

(1) Use of *Surveyor* spacecraft program-level quality directives for unified control of all assisting division activities.

(2) Quality assurance chairmanship type approval (TA) test review committee.

(3) Development of a *Surveyor* Program-oriented handbook of workmanship standards.

(4) Quality assurance participation in the engineering change control board.

(5) Establishment of a comprehensive program for tracing parts and materials from end use to origin through QA history records.

(6) Control of nonflight items that interface with flight hardware.

Implementation responsibility by HAC finally evolved into a line (division) quality and project quality structure in which divisional QA was responsible for basic QA activities and program QA was responsible for direction, monitoring, and field activities.

*a. **Organization.*** The *Surveyor* Program QA manager was responsible for all *Surveyor* QA activities, regardless of organizational dividing lines, and maintained coordination with customer and participating quality organizations through scheduled meetings and problem-oriented contacts as cited below.

1. *Quality assurance engineering* was responsible for developing program direction through SQADs, participating in unit and system consent-to-test/ship meetings, and coordinating quality matters with work item technical managers and participating divisions. Other quality engineer responsibilities included membership in the TA test review committee, participation in failure review board (FRB) and engineering change board (ECB) meetings, resolution of program quality problems, and the conduct of product review and quality audits of implementing organizations.

2. *Field operations* was responsible for implementing QA requirements at AFETR, General Dynamics/Convair, and DSIF stations. Field operations participated as follows:

(1) Issued *Surveyor* Quality Assurance Field Operations Procedures (SQAFOPs).

(2) Performed spacecraft inspection.

(3) Monitored spacecraft test and operations.

(4) Inspected support and operations equipment.

(5) Participated in CTL meetings.

(6) Conducted material review.

(7) Performed receiving and shipping inspection, and audited stores and facilities.

3. *Space systems quality assurance section* at El Segundo was the prime implementing quality organization. Responsible for assuring that *Surveyor* requirements were adequately implemented throughout all Culver City, Airport, and El Segundo activities, this section comprised quality engineering, control point and MRB activity, spacecraft and support equipment inspection and test, control item test and inspection, and vehicle buildup inspection groups. These groups provided all necessary QA servcies at HAC, El Segundo, Calif., except for supplier control, receiving, machine shop and special process inspections, which were furnished (under direction of space systems section) by the quality services section.

4. *Culver City quality control department,* under the direction of the space systems section, provided all quality functions for *Surveyor* control item assembly and test at HAC Culver City, Calif., and Airport facilities.

5. *The foregoing structure,* with its division of responsibilities, was evolutionary, with the following key or significant changes occurring:

(1) From mid-1962 to June 1963, organization reporting of program QA organization was done through a QA&R group, then through the reliability and system test department, then to the program manager.

(2) In June 1963 (subsequent to the NASA-WOO audit) QA became a group function under the newly established QA&R department, which reported through system test laboratory to the program manager.

(3) In October 1963, JPL QA requested that HAC establish a program office which would be totally responsible for *Surveyor* quality. In response to this request, the *Surveyor* Program QA department was formed, reporting directly to the *Surveyor* Program manager. This department, in addition to directing and monitoring in-house QA activities, was assigned the in-line field QA and major subcontractor inspection activities.

(4) In early 1964 contract modification 28 separated quality costs from manufacturing costs and placed them under the control of the *Surveyor* Program QA manager.

(5) In early 1965, the quality engineering staff was developed under contract modification 56, which provided for assumption of unit-level MPC activities from JPL.

(6) In mid-1966, Culver City QC was transferred to the HAC El Segundo division QA department, resulting in a more direct organization structure and a start in providing improved QC direction by use of a more unified QC instruction and procedure system.

(7) In mid-1967, with the phase-out of procured items, supplier QA activities reverted to divisional QA control.

b. System documents. Identification of the project quality requirements to the implementing personnel was of prime importance, especially in this large and diverse company.

Not only did the organizational structure undergo changes, QA requirement (QA plans and SQADs) documents and the implementing QA document systems were in a relatively constant change cycle (e.g., the QA plan had four major revisions; supplier and field operations QA plans had not less than two major revisions). Major reasons for changes were that (1) initially applied systems and concepts were not adequate to assure the necessary quality and performance level; and (2) as hardware problems developed, quality system controls were modified to prevent recurrences.

1. *Quality assurance plan.* Approved by JPL-SQA prior to release, the HAC QA plan described project quality requirements to be met, but was changed several times during the program.

2. *Surveyor quality assurance directives.* Early in 1964, as the project QA office developed, a method of identifying QA requirements and issuing QA direction was implemented by means of *Surveyor* SQADs. These directives carried a sequential identifying number (regardless of subject) until February 1967, when all SQADs were revised and a topical numbering system was established to better correlate with QA plan requirements. All policy and requirements SQADs were approved by JPL, whereas remaining SQADs were subject to JPL review and disapproval.

3. *Project bulletins.* Where possible, the *Surveyor* Project utilized existing engineering bulletins and developed *Surveyor*-oriented bulletins as required. Issued project-wide, these bulletins provided direction on such subjects as drawing control, serialization, bonded stores, materiel review, material and process acceptance requirements (MAPARs), T/FRs, etc.

4. *Divisional implementing documents.* Since each division had its own QA procedures and tier system (e.g., HAC Culver City, El Segundo, and Fullerton, Calif., and Tucson, Ariz.), a profusion of divisional implementing documents accumulated. As previously discussed under audits and inspection, this accumulation created problems induced by the diversity of directions and methods of providing hardware status, documentation, and controls; however, consistency was needed as hardware merged at spacecraft assembly.

The problem of interdivisional policy inconsistency was greatly alleviated: (1) when Culver City control was organizationally transferred to El Segundo, Calif., QA and common divisional QA procedures were issued,

(2) by project review of the implementing documents, and (3) by issuance of an El Segundo, Calif., QA plan.

5. *Supplier quality assurance plan.* The *Surveyor* supplier plan contained the policies and methods that were applied directly by HAC in assuring the conformance of suppliers to their contractual requirements. The QA program provisions for suppliers of *Surveyor* spacecraft equipment and associated ground support equipment (GSE) were imposed by purchase order on suppliers of complex equipment which required a total system of control. These documents were revised as the program developed and in consequence of the audit previously discussed.

6. *Surveyor Quality Assurance Field Operations Procedures.* The SQAOPs provided directions for all HAC QA activities at AFETR, General Dynamics/Convair, and DSIF. This procedure contained plans suitable for direct application by field QA personnel. From experience gained on the TA spacecraft (T-21) critique, and the testing and checkout of the first flight (*Surveyor I*), two major revisions of this document were made.

7. *Workmanship standards and handling specifications.* The HAC *Surveyor Standard Practices Handbook* provided workmanship standards directly to performing manufacturing and inspection personnel.

A material handling, packaging, and storage specification established protective hardware controls that were utilized in the absence of specific engineering criteria.

c. Hardware classifications. At the outset of the program, quality requirements were directed primarily toward flight hardware. However, in order to provide some quality level for other procured hardware, a project bulletin was released to define four hardware classifications to be placed on purchase orders:

Class	Hardware
I	Special test equipment
II	Ground support equipment (STEA, CDC)
III	Spacecraft breadboard hardware
IV	Spacecraft prototype and flight hardware

In 1964, classification by hardware application (above) was extended to include level of quality and documentation requirements. Additionally, Class IV-A (for ambient functional test only) and Class IV-T (for environmental test nonflight usage) were defined to identify hardware built under flight quality requirements but limited to test usage only. Class II quality requirements are defined in SQADs QA requirements, by hardware class, and then formalized in Revision D of the QA plan.

d. Flight hardware controls. The quality requirements for spacecraft and flight hardware were discussed in some detail previously in this report; and since implementation (in general) of these requirements was relatively standard for the aerospace industry, only the more salient features are identified and discussed herein. It should be recognized further that because of system improvement, product improvement, and problem solving, many changes had occurred in detail operations, activities, and requirements. Therefore, some typical examples of critical problems also are considered.

1. *Design reviews.* Quality assurance participation in design reviews, particularly in early program phases, was minimal and workmanship standards were not always adequately considered due to a multiplicity of criteria.

2. *Qualification testing.* Approved parts, materials, and processes were defined for usage. The MAPAR and preferred parts lists were used by inspection and engineering.

Type-approval testing was conducted on control items, and a review was conducted at test conclusion by a TA test review committee (chaired by QA) to determine qualification of design and functional performance. Requirements for TA testing are discussed in the *Environmental Test* portion of this report (Section X-E-1).

3. *Drawing and change control.* The configuration system requirement was clarified in modification 56 to the contract. The function of QA was to inspect to the drawing that manufacturing presented. Two major problem areas were ultimately corrected: use of the advanced release engineering order (EO), which was subsequently changed to emergency EO and approved by the manager of the change control board (CCB); and the undefined required configuration of control items to be installed on the spacecraft, which was corrected by establishing a master index that defined configuration effectivity for each spacecraft.

4. *Receiving inspection.* The most notable task performed by this standard activity was high-reliability receiving inspection for both El Segundo and Culver City, Calif. divisions. Only parts and materials that were

qualified were to be identified as class IV and accepted into bonded stores subsequent to a 100% inspection and/or test.

5. *Control item test.* The primary formal test conducted on a unit was the control item FA test. Depending on the control item, the cognizant QA group either performed or witnessed test operations.

6. *Fabrication and assembly.* Inspections were performed at preplanned points. Problems with inspection criteria and the application of MPC relative to this subject have been discussed in some detail and are discussed further, where applicable, in subsequent sections of this report.

7. *Flight spacecraft assembly and test operations.* A problem of major concern to QA was achieving a sound program of QA participation in system test activities. During the performance of systems tests on T-21 and on the earlier phases of *Surveyor I,* several avoidable defects contributed to spacecraft failures. The problem of defects was emphasized by a miswired cable that caused significant damage to *Surveyor I.*

Since QA, as an independent agency, was responsible for ensuring the delivery of an acceptable product, it became evident (early in 1966) that quality controls in the spacecraft area were inadequate to accomplish this task. To identify the cause and the corrective action required, all operations performed by JPL/HAC QA management were carefully reviewed. The object of this review was to identify those points at which QA operations had to be inserted, modified, or changed in order to fulfill this quality responsibility. From the standpoint of required manpower, the review revealed this to be a large task. The total QA challenge could be broken into the following five major activities, each comprising a significant number of individual operations:

(1) Control of spacecraft handling, support, assembly, and test equipment. Test equipment items numbered in the hundreds, and control of their configuration, quality, storage, and use as related to the various spacecraft operations was vital.

(2) Assurance of spacecraft integrity and configuration through the assembly, test, test preparation, and upgrade phases. Since there were many planned events that modified the assembled configuration, the significance of such assurance was obvious.

(3) Certification of test performance. Most test phases were major in scope and involved several active operators throughout STEA and outlying facilities. Assurance that each step of the procedure was properly executed required QA manloading approaching that of the operators.

(4) Monitoring of data accumulation, reduction, and analysis. This activity was continuous and heightened during performance of major system tests. Assurance that valid data was being obtained and correctly reduced, and that the results satisfied contractual performance obligations, required in-series QA participation.

(5) Review of test procedures. Participation by QA in this area amounted to review of completed procedures against applicable drawings and specifications.

Analysis of the results disclosed by the review revealed the following five key factors:

(1) Manpower deployment of HAC QA&R personnel into most of the previously described areas was so thin that their activities were ineffective.

(2) Some of these areas, such as data analysis, were not being covered at all by HAC QA.

(3) Because of wide deployment of personnel, emphasis could not always be exercised in the right place at the right time.

(4) To have complete QA participation, significant increase in specialized skilled manpower would be required.

(5) A program directive had to be issued to clearly delineate organizational responsibilities, basic operating procedures, and operational restraints.

To reduce the problem recurrence rate, emphasis was required on spacecraft assembly operations as well as test and handling equipment controls.

In early 1966, a program was developed and presented to the HAC program manager and JPL QA management for acceptance. This program included the division of responsibility between the QA department and the systems test department, as well as program direction as to operational restraints. Under this plan, the systems test department assumed the responsibility for assuring spacecraft functional adequacy through test performance, accumulation, reduction, and review of data and procedure reviews. Since the systems test department had no assigned spacecraft or hardware design responsibility (essentially the same as an independent testing agency), that department was not inclined to be biased by performance problems encountered. The QA efforts of HAC were then concentrated in the remaining areas of assuring spacecraft and equipment integrity. All physical operations performed on the spacecraft, including those conducted during test, were subjected to QA review for acceptability and acceptance. Control of operational, handling, support and test equipment was also a QA responsibility.

This separation of departmental responsibilities provided the program manager with two departments responsible for spacecraft quality, each expert in its field. While the program directive established sufficient restraints to ensure that operations were conducted systematically, it also ensured that operations would progress through toll-gate check points for performance assurance. Prior to commencing any operations, approval to proceed was obtained from QA.

Implementation of this program provided an immediate and continuing reduction in the number of defects induced in the systems operations. Detailed implementation procedures were further defined by QA directives.

8. *Environmentally controlled work areas.* A program bulletin governing the types of controls needed to assure vehicle protection was released in January 1963. Simultaneously, indoctrination sessions on cleanliness and protective handling techniques were initiated to which approximately 2000 persons had been exposed by the end of the program.

In December 1964, a revised project bulletin was issued to satisfy the evident need for more definitive and enforceable criteria. In conjunction with this, HAC issued its process 10-22, entitled *Environmentally Controlled Areas General Requirements,* which was completely revised to specify additional areas of control required for *Surveyor* assembly, fabrication, and test; this revision was subsequently updated to keep pace with new requirements for control as they became necessary.

Unlike many HAC programs, *Surveyor* required special environmental controls due to use of thermal control paints and the criticality of particulate matter content of several items (e.g., TV camera, sun and star sensors). Spacecraft assembly and test was accomplished in areas designed to provide absolute control of personnel working on or around the hardware. Requirements for head coverings, smocks, and specially fabricated gloves were enforced in areas where thermal-control finishes were exposed. Limited access to personnel entering or working in the controlled areas and adequate control of aerosol particulate matter were mandatory. Controls were also maintained to ensure that equipment capable of producing oil-type vapors were either fitted with approved filters or vented to the outside. Initiated in the early stages of assembly, these requirements were updated to keep pace with industry advances in measurement techniques and standards.

Quality assurance monitored environmental controls, handling and packaging practices, and personnel disciplines needed to ensure uncontaminated flight hardware.

9. *Control of hazardous equipment.* Several failures were induced in flight electronic control items by inadvertent application of external alternating current to circuitry.

Investigation disclosed that two failures were induced by using faulty soldering irons on spacecraft wiring. The soldering iron tip formed an electrical leakage path to the heater element. This leakage, together with lack of proper grounding of the tip, resulted in the application of an rms voltage to the circuitry.

Another failure was induced when a bench lamp contacted the chassis of an engineering signal processor unit in the unit assembly area. Faulty lamp wiring provided a voltage path to the lamp frame, which had not been properly grounded.

To prevent additional failures, a SQAD was issued requiring daily voltage and resistance ground checks on all electrically operated hand tools, weekly checks on test equipment and appliances, removal of all ungrounded or faulty equipment that could possibly come in physical contact with electronic items, and area logs indicating the items under control and the interval of checks. Several facility changes were made also to provide better grounding capabilities in operating areas, which were audited weekly, for 6 mo, by program office QA engineers to ensure the maintenance of controls.

10. *Control of electrical connectors.* The integrity of contacts in electrical connectors was of serious concern to the program. Extensive mating and unmating of connectors occurring during numerous unit and system tests was the primary reason for this concern. Connector contacts showing excessive contact resistance were detected during control item and system FA testing, and the problem was further emphasized by occasional, unexplained, intermittent spacecraft failures. The need was

apparent for a means of identifying a connector that was in a degraded but functional or intermittently functional state prior to testing.

Efforts were directed toward timely detection of causes that normally contribute to contact failure, i.e., an internally contaminated female socket increasing its contact resistance; a bent male pin which could deform a female socket and result in inadequate retention, etc. Evaluation of a pin-retention tool that had been used by JPL on previous flight projects disclosed two significant problems:

(1) Accuracy of the pin-retention tool was dependent on some relatively uncontrollable variables, such as application technique and coefficient of friction between male and female contacts.

(2) Female sockets with low retention did not necessarily display high contact resistance.

Use of the pin-retention tool was favored by the fact that female contacts with high contact resistance were generally rejected by the tool.

A comprehensive QA program, implemented to control electrical connectors throughout the program, included provisions for procurement activities, receiving inspection, and manufacturing and test activities. Flight hardware and any other equipment that would be connected to flight hardware were placed under controls designed to minimize degradation of connectors. The pin-retention tool and visual acceptance criteria were used to detect defective connectors. Pin-retention testing was conducted at specifically designated points in the manufacture and test phase and upon final mate of connectors prior to launch. Such records as mate/unmate and pin-retention logs were maintained to aid in the detection and disposition of defective or suspect connectors.

A connector program was finalized during the SC-4 test phase. At this juncture, the system provided confidence that all connectors were certified for pin retention just prior to final mate. The program was applied in this manner for all remaining vehicles. Additional information regarding connector problems appears in Section XIII-E-2 of this report.

11. Hardware traceability. Receiving inspection and test records for flight electrical and electronic components, raw stock, and process materials were established at the outset of the program to associate acceptance records and variables data to each part or lot of material. A reference system was utilized whereby the history records of each level of fabrication or assembly would contain the numbers assigned to the records of constituent parts, materials, subassemblies and assemblies of the immediate lower level. The quality history of every phase of fabrication, assembly, and test of a given item could thus be retrieved for problem analysis. Although this system was generally effective, loss of traceability occurred too frequently for toleration.

Investigation revealed several reasons for loss of traceability:

(1) Lack of clear understanding of the traceability requirement by operating personnel.

(2) Carelessness in transferring traceability references to identify tags accompanying hardware or in using assembly history records.

(3) Mixing of parts and lots of parts, or other identifying tags in bonded stores.

(4) Loss of history records due to inadequate control of documentation and files.

To prevent further loss of traceability and to remove those items lacking traceability, the following actions were taken:

(1) A SQAD was implemented to provide a single program definition of the required extent, importance, application, maintenance, and use of traceability (Fig. XII-4).

(2) Items in bonded stores for which traceability had been lost were segregated as discrepant and processed via material review action in accordance with SQAD requirements. Bonded stores methods of packaging and storage were evaluated and changed where necessary to preserve traceability.

(3) Central files of QA history records were established and maintained, and controls were set up to prevent work from progressing beyond points where records were removed and retained.

12. Class IV workmanship controls. The workmanship level of electronic assemblies used for TA testing, T-21, and SC-1 was below specification requirements.

Many electronic assemblies had been subjected to numerous rework and modifications and most had been manufactured in an area where previous experience was primarily with nondeliverable development hardware. Initially, personnel were not assigned on a project basis, a condition which was not conducive to the uniform

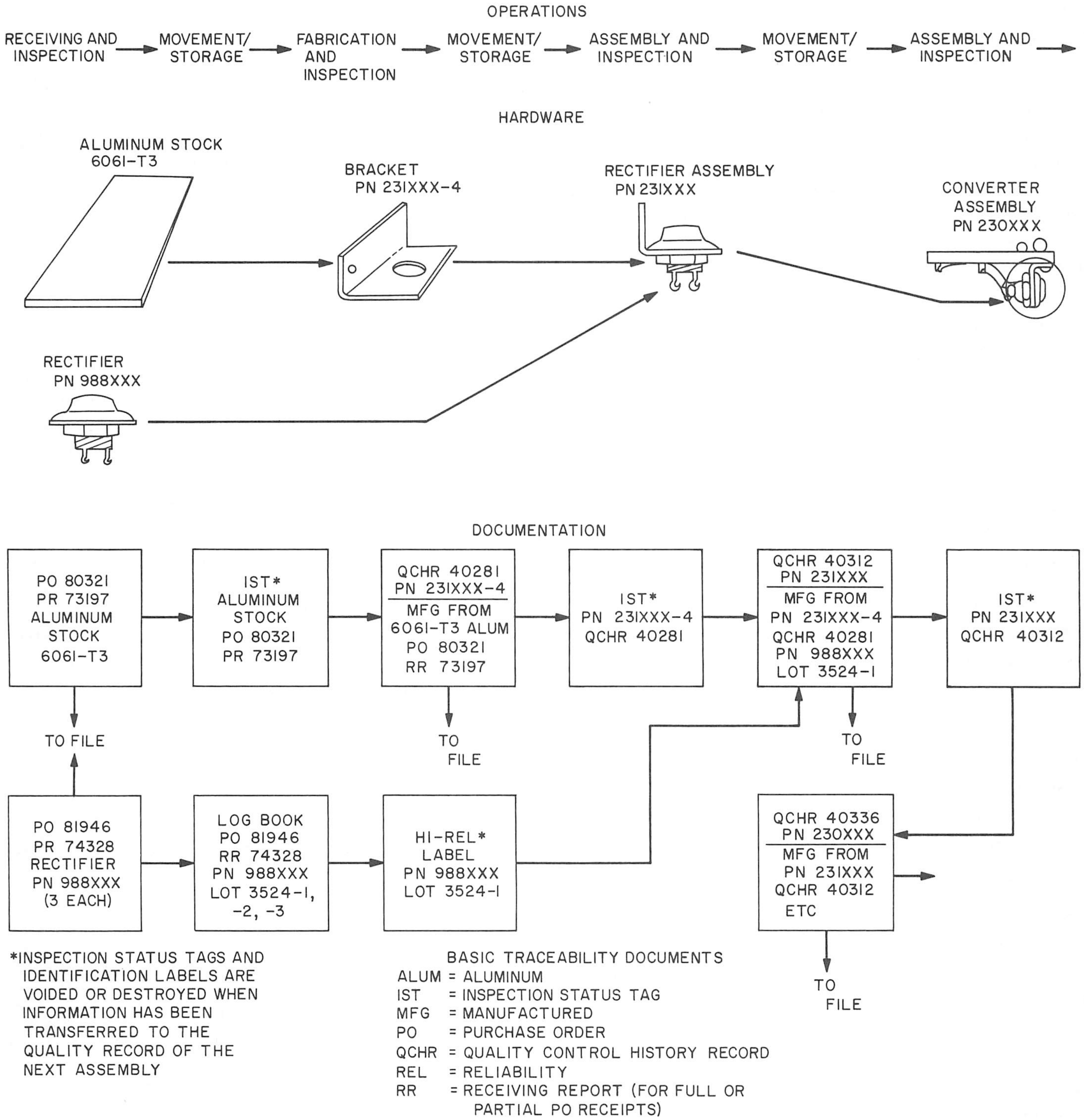

Fig. XII-4. Typical flow—hardware vs traceability

application of *Surveyor* standards. To complicate the situation, *Surveyor* workmanship standards were not always readily available to inspectors and assemblers, nor were inspectors and assemblers adequately trained in the use of those standards.

The detection, isolation of cause, and correction of workmanship problems represented an evolutionary process that periodically required reemphasis because of the number of workmanship defects attributed to the inertial nature of human performance.

Identification of problem areas was facilitated by extensive customer toll-gate inspection and contractor divisional and program office inspections which either supplemented or succeeded customer inspection. Statistical and narrative reports of these inspections were utilized to pinpoint the type and extent of generic problems. In addition, these inspections provided on-the-job training for inspectors and assemblers, which resulted in more critical attention to workmanship as well as the dissemination of valuable information gleaned from experience on other space programs.

Implementation and maintenance of the following positive actions improved workmanship on a continuous basis:

(1) Establishment of a team to review the types of discrepancies and generation of minimum acceptance standards for older work-hardened units.

(2) Solidification of a *Surveyor* assembly and inspection team.

(3) Training of assemblers and inspectors to *Surveyor* standards.

(4) Generation of suitable rework procedures. (Special procedures had to be developed since many of the items were encapsulated or coated.)

(5) Provision of assembly areas with ready access to all applicable standards of workmanship.

(6) Development of visual standards of comparison in the form of photographs and work samples to be utilized by assemblers and inspectors.

(7) Day-to-day emphasis on the importance of workmanship.

(8) Feedback of discrepancy information to assemblers on a personal basis including a description of the acceptance standard as well as the discrepancy.

13. Problems of interest. Problems described in the following paragraphs are representative of QA participation and are not to be construed as a complete listing. Some descriptions are brief since detailed discussions of these problems will be found in other sections of this report.

Conical seals. The consequences of overlooking the critical nature of "ten-cent" items has been well publicized in space programs. Experience on the *Surveyor* Program exemplified this with a conical seal problem. An aluminum seal was required for usage in the propulsion system. When these seals were used, however, a severe gas-leakage problem occurred. Investigation revealed that the supplier of these aluminum seals also made seals similar in appearance but different in material composition, and that these seals were inadvertently received and processed as correct parts.

To prevent continued receipt of improper seals, a positive method of segregation was established at the supplier's facility, including color coding all containers and packaging. All seals which had already been received by HAC bonded stores were reinspected and weighed to isolate those made of wrong materials. A vigorous receiving-inspection program was instituted whereby all incoming seals were weighed, inspected for damage, and individually packaged and identified. To assure maintenance of controls, positive storeroom segregation was established and reuse of seals or use of improperly identified seals was prohibited in assembly areas.

Flux corrosion. Corrosion in the form of green discoloration on solder joints and under hookup-wire insulation was first noted in February 1964 on the *Surveyor* Program. The cause of this corrosion and the corrective action are discussed in detail in Section XIV-E of this report.

Vidicon tubes. Vidicon tubes did not meet procurement specifications in many technical areas; and because of stringent operating requirements and the proprietary nature of some manufacturing processes, satisfactory corrective action was never obtained at the manufacturing source.

To assure the use of the best available flight-worthy tubes, a special receiving test was initiated to select tubes whose usability was based on analysis of detailed performance parameters. All vidicon tubes received were processed through this test program, utilizing both HAC and JPL facilities. Confidence was thus increased and significant time was saved in TV camera assembly and test operations by the early culling of defective or suspect tubes from the system.

Propulsion O-rings. In mid-1965, critical problems emerged involving O-rings used as seals in propellant tank assemblies. Surface defects which adversely affected sealing capability were detected, and mixed O-rings of different material composition were detected in bonded stores.

Surface defect problems were determined to be of manufacturing origin. More than 95% of the procured parts were rejected for pits, cracks, mold contaminants, plastic-fiber impregnation, and excessive material along the mold lines. Satisfactory corrective action was never achieved because approved suppliers refused to establish necessary controls over tooling and manufacturing areas for short production runs. Acceptable O-rings were obtained by observing stringent receiving-inspection procedures.

When the mixture of O-rings was detected in bonded stores, all O-rings were impounded by QA until material composition could be verified and the O-rings separated accordingly. In addition, receiving-inspection procedures were revised to include verification of material composition on all subsequent O-ring procurements.

Printed circuit board—backside solder splash. Short circuits developed in sandwiched printed-circuit-board assemblies. Examination by X-ray revealed metallic bridges between terminals and adjacent terminals and etched circuit lines on the concealed side of the boards.

To isolate the cause and to recommend corrective measures, extensive testing was conducted by engineering and QA. Results of this testing are detailed in Section XIV-E of this report.

Radar altimeter and doppler velocity sensor solder joints. Space limitations of the radar altimeter and doppler velocity sensor (RADVS) system posed problems in manufacturing and process controls. Corrective manufacturing techniques eliminated most major problems; however, defective solder joints continued to plague the RADVS development and manufacturing program. Some joints showed signs of stress fracture whereas other exhibited inadequate wetting action. Investigation by the FRB resulted in several recommendations which were immediately adopted, including improvement in handling practices and inspection techniques to upgrade manufacturing standards. In addition, all units were subjected to an inspection rework and reinspection cycle before delivery to AFETR.

The rework cycle did not eliminate the cause of unsatisfactory wetting of joints, which would have required major redesign of the entire subsystem. Rework did, however, return the solder joints to a known condition wherein the remaining useful life far exceeded test and mission requirements.

e. Aerospace ground equipment and command data console controls. The first two sets of STEAs were fabricated in El Segundo, Calif., well in advance of spacecraft availability. Prereleased engineering drawings supported the manufacture, and inspection was performed at the discretion of the responsible engineer. Inspection criteria were defined as being in accordance with best commercial practices.

The T-21 prototype spacecraft was utilized to verify the capability of the STEAs and their design compatibility with the spacecraft. Miswiring, incorrect parts, and other defects of both engineering and manufacturing origin were corrected at that time. Required design changes were incorporated for future manufacture.

Command data consoles (CDCs) were manufactured by HAC flight test division and transported to El Segundo for integration, test, and shipment to field locations. The CDC destined for use at Goldstone Deep Space Communication Complex (DSCC) was prepared for Air Force inspection and JPL acceptance for shipment. A selloff plan was drafted and coordinated. Since only cursory reinspection was performed by HAC prior to presentation of the CDC to the customer representatives, numerous workmanship discrepancies were documented by the customer. Although the Goldstone CDC was reworked to a point where it could be shipped (April 1964) with open discrepancies considerable action was necessary to improve class II quality. Configuration accountability was also found to be inadequate at this time.

On behalf of the program quality office, a QA engineer was assigned class II responsibility for establishing definitive requirements for adequate class II control. Also established were requirements for the documentation and maintenance of hardware configuration, quality history records, and break and entry controls. Many of the problems encountered during the Goldstone CDC shipment were thus avoided in subsequent class II shipments.

Workmanship standards were established for manufacture of new equipment and for the modification and repair of equipment already in use. To ensure integrity of assemblies undergoing modification, controls were developed for modification kits, i.e., detail instruction and configuration documentation. In addition, class II procurement policies were modified to conform to more rigorous workmanship standards.

Further controls were implemented to assure that class II equipment for flight hardware interface would not incur damage or degradation of the flight hardware. Fit and function checks were required of all class II items prior to use with flight hardware or before shipment to field locations. Electrical connectors were examined periodically and when gases or liquids were used in the spacecraft, they were certified for cleanliness prior to use.

f. Test vehicle controls. The original program concept for test vehicles SD, T-2, S-2, S-6, S-7, S-9, S-10, and S-15 was tailored for engineering control and flexibility. Hardware was to be class III (nonflight breadboard), with specific requirements to be defined by the responsible engineering activity. Since it became necessary to utilize test vehicle data for qualification of class IV (flight) hardware, this concept was not practicable. Lack of class IV controls over test vehicle programs could cast doubt on the validity of data obtained. In most cases, however, it was not necessary that the entire vehicle be a class IV vehicle (i.e., the transmitter's functional ability played no part in demonstrating the propulsion systems capability on the S-6 vehicle).

Each test vehicle program was evaluated to determine pertinent characteristics requiring control. Test vehicles used in determining adequacy of structural stability, thermal, and propulsion characteristics were developed into hybrids. Nonessential areas remained at the class III level, and the significant items were flight quality and were maintained under class IV controls.

There was, however, significant difference in the T-2 program. Developmental (class III) hardware was used to support this program, and many of the problems and gross failures encountered during the T-2 program were of product-quality origin. These problems created concern for the reliability of T-2 test results, which resulted in a second, more extensive program designated T-2N.

The demand that the T-2N culminate in successive soft landings required quality participation on a par with that of flight (class IV) hardware. A QA engineer was assigned full quality responsibility for the T-2N program, including establishing and monitoring the entire T-2N QA program, and applying, whenever possible, the class IV system of controls.

Extensive inspection was imposed on all items of the program. All testing was witnessed and verified by QA, and each completed item underwent an engineering and QA review by the program office before entry into bonded stores. A QA group was established at the HAFB facility to inspect and verify all operations and to assure the display of quality consciousness by all participating organizations.

4. Quality Systems at Other Contractors

Implementation of previously identified QA requirements by other project elements is discussed in this section, with significant aspects highlighted.

a. Launch vehicle. NASA, Lewis Research Center (LeRC) and General Dynamics/Convair (GD/C) were vested with basic quality responsibility for the launch vehicle and associated support equipment. However, spacecraft/launch vehicle compatibility tests and mating operations, detailed in GD/C procedures, were also monitored by both HAC and JPL QA personnel.

Surveyor/Centaur interface hardware, such as the spacecraft electrical disconnect and the *Centaur* adapter, were subjected to stringent QA surveillance. Particular emphasis was given to maintaining cleanliness requirements comparable to those imposed at the spacecraft level. After each significant mating operation and prior to final encapsulation, HAC/JPL QA personnel performed a careful visual inspection of the spacecraft, nose fairing, and adapter, for contamination. Excessive visible particles resulted in immediate inspection by the spacecraft test director to determine the source of contamination and corrective measures required.

b. Television ground data handling system. The QA system implemented at Link (General Precision) in support of fabrication and test of the television ground data handling system (GDHS) procured by JPL was a basic QA plan designed to comply with the concepts of MIL-Q-9858. To ensure contractor compliance with this plan, JPL QA assigned a resident inspector at the manufacturer for the duration of the contract.

Responsibilities of the resident inspector included a comprehensive hardware and software inspection program. Quality assurance instructions generated by JPL QA defined requirements for this program. Flow charts were developed to assist in monitoring system status during all phases of fabrication, assembly, and test. Acceptance checklists were also generated to ensure adequate JPL inspection coverage of all applicable contract provision.

Areas of responsibility, other than at the Link facilities, were given limited JPL QA coverage. Source inspection was provided by JPL QA at various second-tier subcontractors, as required. However, direct JPL QA support of the end product use at Space Flight Operations Facility (SFOF) and at Goldstone Communications Center was provided only upon specific request from JPL engineering.

c. Government-furnished equipment. Several significant spacecraft control items were provided to HAC as GFE. These items, which included solar panels (*Surveyor V, VI,* and *VII*), vernier engines, and the alpha scattering instrument, were subcontracted directly by JPL contractors other than HAC. Quality assurance at JPL played an active role in establishing effective QA systems at each contractor's facilities.

1. Solar panels. The solar panels (*Surveyor V–VII*) were fabricated for JPL at Electro-Optical Systems (EOS), Pasadena, Calif. The QA system implemented by EOS for the *Surveyor* solar panel program was initiated in accordance with applicable portions of JPL and NASA specifications.

Early in the program, an EOS QA engineer was assigned to the project. His responsibilities included MRB activities, interpretation of requirements, customer (JPL) interface, and problem/troubleshooting. Direct inspection functions, however, were performed by EOS line QA organizations.

During the development stage of the solar panel, JPL and EOS QA organizations jointly established an effective QA system. Fabrication and test flow plans were developed that were ultimately formalized on JPL drawings, and formal inspection instructions were prepared to detail the specific inspection operations described in the flow plans.

Problems arose when the first solar panels were delivered to HAC for systems compatibility tests. Numerous discrepancies were recorded by HAC receiving inspection; however, most complaints were directly attributable to inadequate communication and lack of mutual understanding of design and inspection criteria among the EOS/JPL/HAC interfaces. These complaints were resolved through detail briefings of all responsible personnel involved.

Each flight solar panel was accepted by JPL at AFETR after completion of final inspection by EOS personnel assigned to AFETR operations.

2. Vernier engines. Vernier engines were built for JPL by Thiokol Chemical Co., Reaction Motors Division (RMD), Denville, N.J. Originally, RMD was under contract to HAC to design, build, and test the complete *Surveyor* vernier engine propulsion system. This RMD effort was terminated in the spring of 1964, when HAC resorted to in-house fabrication and assembly of the upstream vernier propulsion system and JPL accepted the responsibility of supplying vernier engines to HAC. The RMD effort, under JPL contract, was divided into two distinct categories:

(1) Hardware development and qualifications.

(2) Flight hardware production.

During an initial 60-day effort, RMD demonstrated the capabilities of their engine redesign. This effort, which culminated in the successful development, assembly, and test of a qualifying engine, resulted in a follow-on contract for the production of flight vernier engines.

With the advent of flight hardware production, a QA plan (initiated by RMD in support of the *Surveyor* Program effort) encompassed all levels of responsibility in QA engineering and inspection, and provided for QA participation in product improvement.

Implementation of the QA plan was not totally achieved since QA requirements in such divergent areas as metallurgical samples, environmental controls, and vendor surveillance were not adequately enforced. Moreover, recurring material review discrepancies were not reported promptly and, as a result, appropriate and prompt corrective action was not initiated.

To alleviate the situation imposed by these problems, JPL QA provided full-time source-inspection coverage during the manufacturing cycle. Although the source coverage was unable to totally eliminate the problems, it did provide cognizant JPL personnel with a direct communication link to the RMD facility, which minimized misinterpretation of information and/or requirements.

Other steps taken to increase confidence in the flight engines included modifications to the RMD service contract, which provided for intensified RMD source inspections and for RMD QA engineering support at AFETR.

3. Alpha scattering instrument. The design, development, fabrication, and test of the alpha scattering instrument for *Surveyor* spacecraft was undertaken by the University of Chicago in two distinct phases.

During the original phase, design, fabrication, and assembly of the prototype units were accomplished and assembly of flight hardware was commenced. The QA requirements to be developed during this period were to include the generation and implementation of flow plans and inspection criteria and procedures. This effort was only partially complete when the science instrument development was terminated in August 1965.

Upon resumption of the science instrument development in late 1966, the flight hardware effort was plagued with significant problems resulting from incompleted QA requirements and rigid schedule constraints.

The QA program lacked contamination requirements, calibration and maintenance schedules, inspection, assembly, testing, and handling procedures and flow plans. Problems were further propagated by improperly indoctrinated personnel and poorly enforced inspection criteria.

Recognizing the seriousness of the situation, JPL assigned a QA representative and a technical engineer to the University of Chicago to participate on an as-needed basis.

Due to extensive effort of all personnel involved, both at JPL and at the University of Chicago, resolution of existing problems resulted only in intermediate schedule and cost impacts, without compromising the ultimate performance of flight hardware.

B. Reliability

1. Programs

To help achieve primary mission objectives, reliability responsibilities for the project were organizationally assigned to technical and functional groups such as contractors, JPL design sections, DSN, and the *Surveyor* Project office. To maintain central cognizance over the total project reliability activity, the JPL reliability program office established a reliability program for the JPL *Surveyor* Project office, including audits of all reliability activities and specific tasks that other elements of the project office did not perform.

Reliability tasks were identifiable in all organizational elements of the project (Fig. V-3). Mission-success modeling was performed on the mission level. Failure-reporting and corrective-action programs were established and conducted for each major project element. An extensive and significant reliability program was implemented for the spacecraft. Thus, the reliability programs utilized for the various mission and project elements were different in scope, providing minimal but adequate and effective reliability efforts in response to NASA contractual requirements.

The reliability office interfaced with the respective JPL Spacecraft System sections, major contractors (HAC, spacecraft; RMD, vernier engines; EOS, solar panel; and University of Chicago, ASI), the JPL *Surveyor* MOS and the launch vehicle integration group of JPL *Surveyor* mission analysis and engineering (MA&E).

2. Mission and Major Systems (Excluding Spacecraft)

The major system organizational elements of the *Surveyor* Program are shown in Fig. V-3, which depicts reliability task responsibilities for each element.

a. Mission analysis and engineering. A *Surveyor* mission success evaluation model was developed by the MA&E section which considered three levels of project and flight objectives and allowed for performance measurements, various degrees of mission success, and estimations of success probabilities for each mission based on prior mission experience. Incorporated criteria included critical measures of performance, failure mode analysis, sensitivity analysis, *a priori* estimates, and *a posteriori* evaluations.

b. Launch vehicle system. The launch vehicle system was the responsibility of NASA-LeRC, who through their prime contractor, General Dynamics/Convair, implemented a reliability program. Reliability liaison was conducted primarily through the JPL launch vehicle integration activity, including the *Centaur/Surveyor* interface working group, post-mission critiques, and CTL meetings. In addition, LeRC, JPL, GD/C, HAC meetings were held to resolve specific problems (e.g., LeRC/JPL connector technology interchange meeting, June 2, 1967) and a continuous review of *Atlas/Centaur* documentation was conducted. To ensure an integrated *Atlas/Centaur/Surveyor* space vehicle with a significantly higher level of reliability than is possible through individual checkouts and to reduce the on-stand time for each launch vehicle at AFETR, all space vehicles were originally scheduled for CST at GD/C, San Diego, Calif. *Surveyors I, IV,* and *V,* and T-21 did undergo CST, the results of which were included in the reliability evaluation. In all cases, the integration of the launch vehicle was verified by a CST.

c. Mission operations system. As part of the JPL MOS responsibilities, complete mission profiles were developed to ensure operational readiness and the capability of performing space flight operations from launch through the completion of each individual mission. System reliability criteria were included for the development of data processing, nonstandard operations analysis, human engineering, flight operations, mission-peculiar equipment, and DSN support requirements, i.e., nonstandard flight and lunar operations. A mission operations (MO) problem log was maintained during all operational training tests and flight missions to assure that significant problems associated with MO were considered. Continuous *Surveyor* mission operation surveillance was maintained on those problems under the cognizance of either the HAC trouble and failure system or the DSN discrepancy reporting system. Mission reports were issued that critiqued each mission, including performance and anomalies.

d. Tracking and data system. The JPL tracking and data system utilized the DSN discrepancy reporting system, which provided real-time data stream discrepancy visibility during operations tests and mission "up" periods. Requirements for closure of a discrepancy report included the prior closure of any associated DSIF T/FR and formal review by cognizant technical management. The DSN discrepancy reporting system status reports, issued biweekly, were an input for CTL reviews and project management visibility.

3. Spacecraft

Hughes Aircraft Co. was the prime contractor for the spacecraft, related operations support equipment, and mission operational support services. An extensive reliability program was conducted by HAC to help achieve basic spacecraft and payload reliabilities of 0.75 and 0.85, respectively, at a minimum confidence level of 80% prior to the fourth flight for the 66-hr normal flight and landing phase only. The HAC reliability program had four identifiable phases; each emphasizing specific reliability functions in direct relation to the stage of the spacecraft development and with increasing technical and/or management rigor.

a. Phase I — Preliminary Design (December 1960–January 1963). The initial reliability program, including development of system block diagram, mathematical models, design review, and iterative studies, was implemented to optimize system redundancy and tradeoffs. (HAC level of effort built up to 25 personnel by January 1962 and averaged 22 personnel over the entire phase.) During phase I, the JPL *Surveyor* Project office was supported by the JPL QA&R office in monitoring and reviewing the HAC reliability program.

b. Phase II — Detailed Design (January 1963–September 1964). First revision of the reliability program impacted subsystem areas primarily, by emphasizing design reviews, conducting design failure mode audits, establishing materials and process acceptance requirements, selecting of subcontractors, training personnel, refining mathematical models, commencing failure reporting effort, and FRB activity. The HAC level of effort was maintained at approximately 20 technical personnel. To provide continuous interface with HAC, a reliability program engineer was assigned to the JPL *Surveyor* Project office in May 1963.

c. Phase III — Fabrication, test/operations and start of system test (September 1964–May 1966). Second revision of the reliability program was consistent with the JPL engineering surveillance of all HAC *Surveyor* activities. There was increased emphasis on mathematical modeling, failure reporting, and ground electronics, as well as continued emphasis on subcontractor liaison, design change, and specification review. A reliability spacecraft test program was included in the revision. There was a significant redirection of the failure reporting effort in early 1965. During this period, when the level of effort in this phase peaked at approximately 40 technical personnel, the *Surveyor* reliability section increased its staff to five engineers: two engineers specialized in electronic parts activities and three engineers were responsible for T/FR, subcontractor, and general audit activities. In addition, a *Surveyor* system reliability engineer was active in system and analysis activity.

d. Phase IV — Systems test, launch, and flight operations (May 1966–February 1968). The third revision of the reliability program emphasized the failure reporting effort, including problem description, failure analysis, and verified corrective action. Reliability audit and mathematical model updating were conducted. Frequent probability of success assessments were made, based upon test data and failure histories. During phase IV, the HAC level of reliability program effort peaked at approximately 35 technical personnel.

To assure adequate implementation of the HAC reliability program, the JPL *Surveyor* Project office exercised technical and management control through the JPL *Surveyor* Spacecraft System engineering reliability group and the *Surveyor* reliability section. The respective task assignments and interfaces with HAC are shown in Fig. XII-5. From May 1966 to February 1968, direct

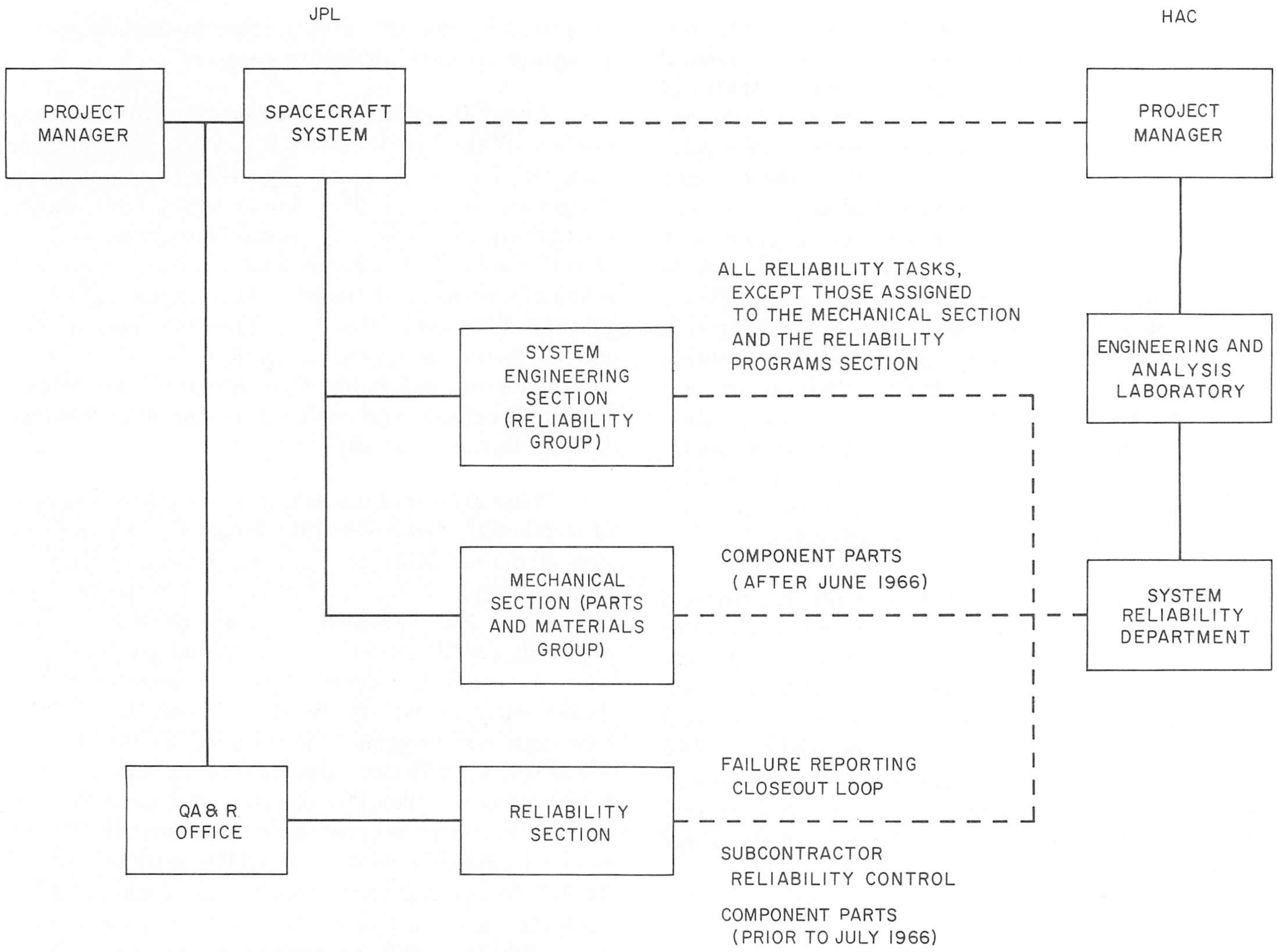

Fig. XII-5. *Surveyor* reliability task assignments and interface

reliability program support was provided by five engineers, including the two parts engineers transferred to the mechanical section during phase IV. In addition, a *Surveyor* system reliability engineer was active in system and analysis activity.

During phases III and IV, 12 cognizant group engineers at JPL, plus their management support, were actively engaged in reviewing and participating in T/FR analyses and closeout actions. This effort was part of their coordination and control-loop responsibility with their respective HAC technical group.

1. Design review. Design review by HAC was an evaluation and appraisal tool utilized to supplement, rather than replace, the primary responsibility of the design group. To assure optimum consideration, implementation, and documentation of all applicable design factors at key points in the design cycle, the depth of the design review varied, generally corresponding to the state of item development and its intended use.

The HAC *Design Review Engineering Procedure* and *Design Review Guide,* both by HAC, were the basis for conducting design reviews, with the major emphasis on design concepts, circuit and mechanical functional adequacy, and packaging techniques. Approximately 235 formal reviews of the Spacecraft System were conducted, with subsequent documentation, including requested and authorized engineering changes. To optimize conceptual and detailed design tradeoffs affecting reliability, reliability personnel worked directly with design personnel to conduct informal design reviews.

From the period 1961 through 1963, a reasonable mixture of HAC design and project personnel participated in the design reviews. Technical representation by

JPL was generally limited to critical subsystems or final reviews. During this period, the effect of attendee recommendations was minimized due to the fluidity of design configuration. The strongest influence exercised by reliability personnel was in the definition of electronic parts and failure mode audits (FMA) stress analysis. All action items noted in design review minutes were not clearly dispositioned.

In direct support of the formal design reviews, FMAs were conducted to evaluate the effect of potential individual equipment failure modes and thereby determine the weakest links in the design. Although some effort was expended on mechanical assemblies, the major effort was directed to electronic systems. The following factors were typically included in a FMA:

(1) All components and/or functional blocks in the circuit.

(2) Component derating factor and generic failure rate.

(3) Component failure mode.

(4) Effect of circuit failure mode on other circuits and the next higher assembly.

(5) Functional block diagram of the circuit.

(6) Length of mission phase. Approximately 32 subsystem FMAs were developed.

Minor design changes were continually reviewed to ascertain their effect on the assessed inherent design reliability. Formal design reviews were reconducted where major design was required, i.e., redesign of the spacecraft power subsystem, transmitter, and the antenna/solar panel positioner (A/SPP), which was implemented on the fifth spacecraft.

Figure XII-6 depicts spacecraft design review activities, illustrates the time sequencing of FMAs into design reviews, and the types of review by program phases.

2. *Component parts and materials control.* A major task of the HAC *Surveyor* reliability department was to provide assurance of parts and materials reliability for all phases of the program. Detailed implementation of the *Surveyor* reliability components program was conducted by the Components and Materials (C&M) Laboratory of the HAC research and development division under the technical and management direction of the HAC *Surveyor* reliability department.

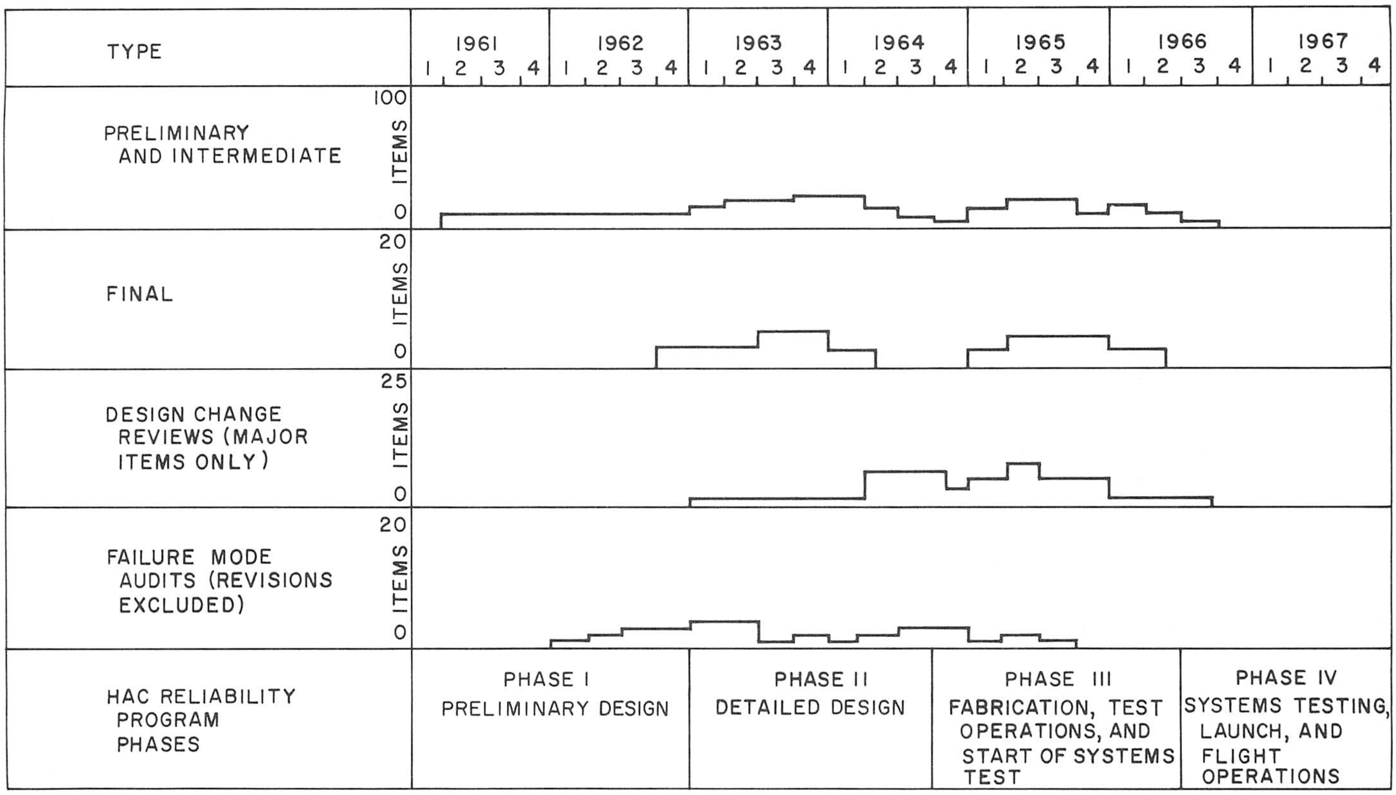

Fig. XII-6. Spacecraft design review activities

The program was directed more heavily toward electronic parts than materials. Furthermore, effort was directed primarily to the use of standard parts rather than nonstandard parts and to specific spacecraft equipment rather than aerospace ground equipment (AGE). The program was oriented toward providing data on:

(1) The capability of parts and materials to survive extended thermal vacuum exposure.

(2) Effects of sterilization techniques on components (requirement later deleted).

(3) Determination of minimum lot and sample sizes to establish confidence in part reliability.

In phase I, the development of preferred parts and materials listing was initiated and qualification of parts was a major effort. Most of the preferred parts choices were based upon military approved parts lists and aerospace experience. Included in the initial *Surveyor* Approved Parts List were approximately 100 preferred parts controlled by the C&M Laboratory through the reliability department and approximately 100 nonstandard parts controlled by project design sections. During this phase, almost two-thirds of the total parts-reliability effort was expended in parts testing (approximately 20% subcontracted to commercial test laboratories). Parts application liaison was performed with HAC design engineers, and C&M engineering participated in design reviews.

The Space Materials Data Book and Approved Materials and Process List issued in phase II included the results of testing connector insert materials, glass melamines, conformal coatings, and magnesium. Improved test chambers and techniques were developed to accurately measure material outgassing. Failed parts diagnosis, as reported through the T/FR system, was initiated (including dissection, X-ray, and special test techniques). In 1964 a rigorous review of 214 components, together with associated data, was conducted.

During earlier phases of the program, the *Surveyor* Project reliability office had accepted part qualification by the C&M Laboratory without performing a technical critique. This was due partly to a HAC internal operating policy wherein the C&M Laboratory had autonomous responsibilities in selecting and approving electronic parts. However, increased emphasis was placed in phase III (approximately a 25% effort) on closer liaison and guidance of the C&M activity.

The JPL *Surveyor* Project office concurred with the need for upgrading the HAC *Surveyor* reliability components program in the latter part of phase III. Electronic parts engineers at JPL initiated an in-depth survey on HAC selection and qualification techniques, part specifications, derating criteria and control procedures. Numerous JPL and HAC electronic part coordination meetings were held to coordinate specific part program requirements and implement the results of the in-depth JPL investigations. The initial JPL survey covered 159 different generic classes of electronic parts. The T/FR reviews disclosed a range of 1–146 recorded problems per part class. In July 1967, a JPL summary review of 79 generic-part types was conducted, utilizing a base of 2185 recorded T/FRs. This review disclosed that only 7.3% of the T/FRs could be identified as primary electronic-parts failures. Of the remaining T/FRs, 27% were identified as misapplication in design, 23% in testing, 16.7% in fabrication, and 26% unassignable.

To minimize primary electronic-parts failures, HAC implemented the following corrective action:

(1) Part screening tests were upgraded.

(2) Close liaison was established with vendors to prevent part manufacturing errors.

(3) All commercial grade parts procured early in the program were purged from stock, and fabricated hardware was reworked.

Phase IV was directed chiefly to the analysis of failed parts and to the associated review and investigation of special part problems. Significant problems investigated were:

(1) Full case capacitor design.

(2) Terminal solder deficiencies.

(3) Transistor (HAC PN 988809) and potentiometer design weaknesses.

(4) Resistor (HAC PN 988644) shorting.

Upgrading of the part control and qualification loops, in addition to problem definition with corrective action, consumed approximately 20% of the total reliability funding.

The effectiveness of the *Surveyor* reliability components program and its associated effort can be measured in part by the single fact that no mission catastrophic

Table XII-6. Total electronic parts per spacecraft by major subcontractor

Part type	HAC[a]	SBRC[b]	RYN[c]	VEC[d]	MRL[e]	EDC[f]	Total
Capacitor	2252	57	1019	204	4	56	3592
Connector	453	2	—	—	—	—	455
Diode	7329	30	986	51	2	32	8430
Resistor	8618	128	2704	340	1	328	12,119
Transformer	168	9	18	—	1	—	196
Coils/inductors	206	2	74	34	4	—	320
Transistor	2761	40	789	68	—	96	3754
Switch	82	—	—	—	—	—	82
Sensistor	—	—	37	34	—	—	71
Miscellaneous	60	3	10	—	—	8	81
Totals	21,929	271	5637	731	12	520	29,100

[a]HAC = Hughes Aircraft Co., El Segundo, Calif.
[b]SBRC = Santa Barbara Research Center, Goleta, Calif.
[c]RYN = Ryan Aeronautical Co., San Diego, Calif.
[d]VEC = Vector Electronic Co., Inc., Glendale, Calif.
[e]MRL = Magnavox Research Laboratories, Torrance, Calif.
[f]EDC = Electro Dynamics Corp., San Diego, Calif.

event could be directly attributable to a primary electronic part failure.*

Developed during the summer of 1966 to assist in configuration control and in understanding the complexity of the Spacecraft System, the electronic parts list shown in Table XII-6 represents the total parts used per spacecraft and is typical of the first four spacecraft.

3. *Trouble/failure reporting system.* The reporting of troubles and failures via T/FRs was initiated on the HAC *Surveyor* Program late in phase I. Evolution of the trouble and failure activity was paced by the amount of effort expended and the stage of hardware development. This segment of the total reliability activity averaged approximately 2% in phase I, 10% in phase II, 25% in phase III, and 63% in phase IV. Figure XII-7 depicts significant actions taken, culminating in a program (Ref. XII-6) which, at the start of phase IV, provided comprehensive visibility of troubles and failures from unit FA testing through MO. Furthermore, a primary control system was provided for categorizing failures as to mission criticality for CTS and CTL reviews.

Progressive modifications to the initial failure reporting system implemented during phase I were:

*The trouble review board (TRB) was unable to identify any single or multiple cause for *Surveyor IV* mission failure.

(1) Establishment of a *Surveyor* Project FRB to review significant T/FRs and plan for automatic processing of T/FRs during phase II.

(2) During phase III, major upgrading of the HAC T/FR system occurred. The primary effort was to verify the adequacy of corrective action prior to T/FR closure and to implement the T/FR system in the areas of special test, AGE, and GSE.

Specific HAC actions taken included:

(a) Formal approval of T/FR closures by reliability, the cognizant senior project engineer, and the program engineering task manager.

(b) Rigorous technical closure with full closed-loop corrective-action implementation.

(c) Extension of the T/FR system to include AGE, GSE, and the T-2 test program.

(d) Definitive management direction in accordance with bulletins and training guides.

(e) Utilization of electronic data processing.

(f) Joint HAC/JPL review of the closure and status of each open T/FR.

(g) Technical effectiveness review of T/FR closures by a high-level technical management team (blue ribbon T/FR review committee) preparatory to CTS and CTL reviews.

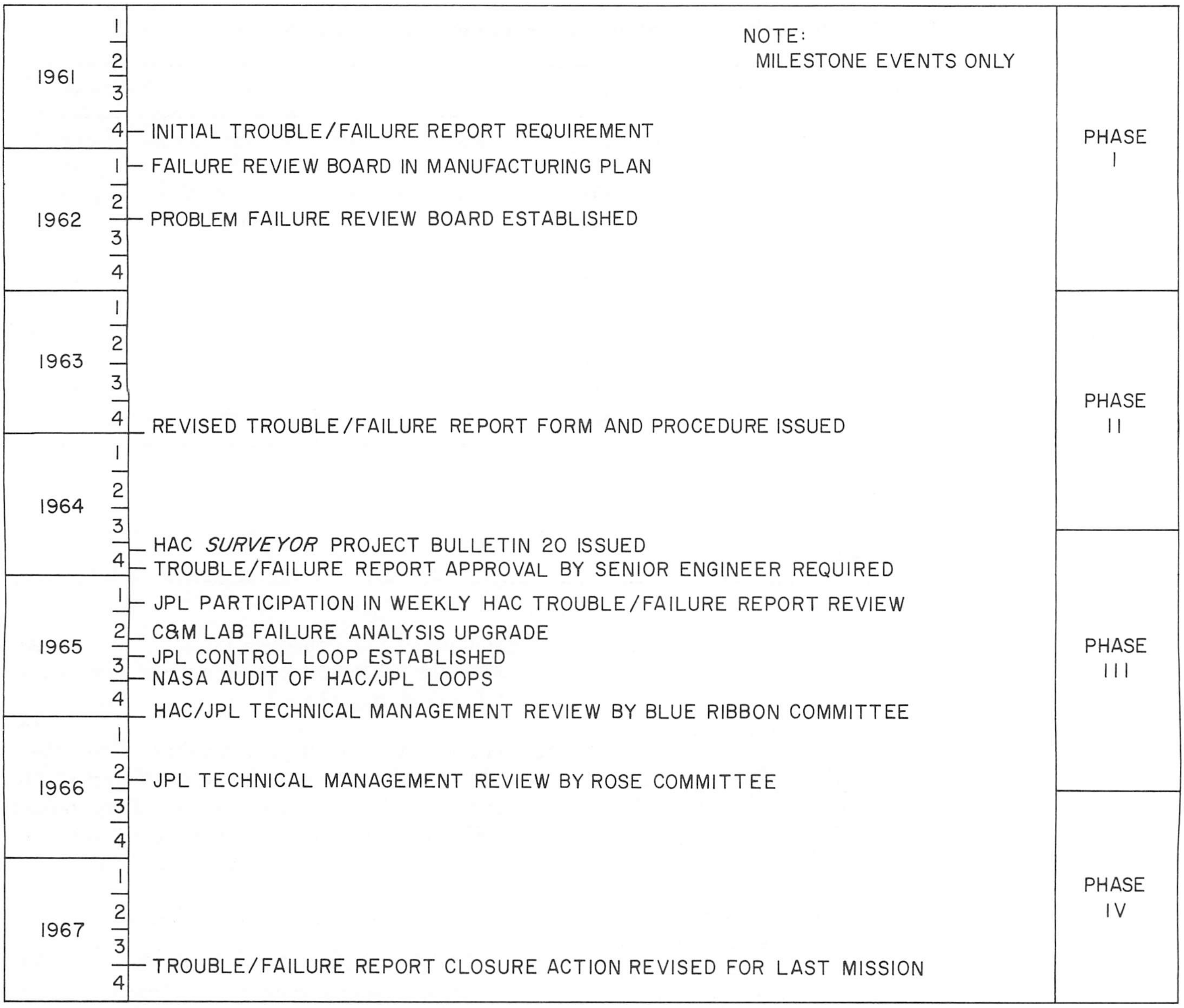

Fig. XII-7. Evolution of spacecraft trouble/failure report program

To supplement the HAC T/FR program, JPL implemented a closed-loop review of HAC T/FR closures requiring formal acceptance of the closure's technical adequacy by the cognizant JPL engineer. Furthermore, a formal review was initiated for HAC acceptance and/or rejection on the basis of technical adequacy of JPL P/FR closures on GFE.

NASA Headquarters *Surveyor* Program office personnel and resident NASA representatives participated in periodic reviews to ascertain adequacy of T/FR discipline in the HAC and JPL activities, and represented NASA in major *ad hoc,* joint T/FR review working groups.

All T/FRs (spacecraft or operational ground equipment) occurring after unit-FA testing were classified as mission-critical (Ref. XII-6). To minimize the probability of mission performance degradation, each T/FR was given individual and immediate attention. Each and every T/FR was considered a potential risk to mission success. The T/FR system built its controls around identification of problems, the real causes of problems, thorough analysis, and positive corrective action. All levels of project management became intimately involved in the adequacy of T/FR closures. Failure and corrective action summaries comprised one of the principal foundations of readiness review board (CTS and CTL) action, where inadequate or incomplete T/FR closures in critical areas became liens against subsequent testing or use of the spacecraft (or test vehicle).

To assist CTS and CTL boards in determining the readiness of a vehicle to be shipped or launched, a grading system was established. This grading system, which applied to each open T/FR as well as to other problems, allowed for development of a simple evaluation scheme that summarized the readiness status for final board decision. Definitions of the grading system are shown in Table XII-1.

Of the 16,834 T/FRs generated on the *Surveyor* Program, 9651 applied to spacecraft, 6205 to operational ground equipment, and 978 to other ground equipment. Further, 16,577 T/FRs were fully closed, 204 were cleared for SC-7 (no impact on SC-7 based on complete technical analysis, but corrective action was not required for hardware not designated as flight hardware), and 53 were open (but dispositioned by the SC-7 CTL board as noncritical).

All problems discussed in CTS and CTL meetings were to be assigned (by the CTS/CTL board) to one of the categories defined in Table XII-7.

The total T/FRs issued on the *Surveyor* Project by HAC are shown in Fig. XII-8. The curves show that manufacturing T/FRs peaked 1 yr prior to spacecraft T/FRs and both decreased steadily after peaking until the end of the program. Until late in 1964, there was not a great deal of management attention to the T/FR control loop and hence pre-1965 T/FRs are regarded as of lesser quality. Post-1965 T/FRs had an improved control loop. Through 1964, only 2650 T/FRs were generated; after January 1965, about 14,200 T/FRs were generated. The first spacecraft was launched in May 1966, when 1½ yr of a controlled loop T/FR system had been in effect.

Many factors influenced the generation of T/FRs during the last three significant years of control. There were at least as many improvements made to increase the quality of T/FR reporting as there were degradations to T/FR reporting.

Increased reporting	Decreased reporting
Real-time system test data printout	An anomaly reporting system came into being as described below
Improved makeup of system test personnel	Repetition of known problems
Sensitivity to suspect failure modes	Specification tolerance widening
Management interest	

Pie charts in Fig. XII-9 compare T/FRs of the seven spacecraft by subsystem. The T/FRs written against each

Table XII-7. Trouble/failure report problem categories

Category	Definition
A	Any unclosed problem which for the following reasons the board considered unrelated to spacecraft performance: (1) Nongeneric problems which occurred on some other spacecraft (2) Problems which are not related to the achievement of such spacecraft performance requirements as follows: (a) Accomplishment of a soft lunar landing (b) Landing accuracy to 50 km with an oblique approach to declared landing sites (c) Interpretable touchdown measurements (d) Television performance as follows: Footpad picture Pan/scan—100 frames Colorimetry—five composite pictures (e) Thermal control—to allow picture taking cited above
A (contd)	(f) Power management—to allow picture taking cited above (g) Ability to receive engineering data on command (h) Ability to perturb the lunar surface on command by means of a surface sampler (i) Ability to acquire on command backward-scattered alpha-particle and proton-particle data from alpha-particle bombardment of lunar surface
B	Problems which require additional work to classify or close out. (The board must assign a closure classification date on each item and no items may remain in this category at CTL)
C	Compilation of all unclosed problems which relate to spacecraft performance requirements as stated in part 2 of category A, above, and cannot be worked off prior to launch

Table XII-8. *Surveyor* spacecraft trouble/failure reports by failure cause (1965 and later)

Failure cause	Number of trouble/failure reports by vehicle								
	T-2N	T-21	SC-1	SC-2	SC-3	SC-4	SC-5	SC-6	SC-7
Design	20	66	121	55	56	38	36	17	10
Manufacturing process	3	17	13	8	9	6	10	2	1
Process engineering	1	9	7	7	5	2	5	0	1
Wear out	4	6	18	12	14	6	6	5	1
Test equipment	0	14	76	25	23	16	17	6	6
Test process	11	28	117	29	32	29	21	5	10
Human error	17	11	48	37	40	27	28	18	19
Workmanship	11	26	49	43	33	14	14	10	11
Wiring error	0	0	5	1	0	1	1	0	2
Rough handling	12	17	17	17	25	13	12	3	7
Environmental effects	10	20	18	15	4	2	7	2	1
Secondary failure	0	12	19	12	8	2	3	4	5
Defective part	5	14	15	11	10	13	13	20	12
Unknown	4	32	41	26	35	19	19	11	21
Not a failure	22	76	127	61	43	11	25	22	17
Pending and unclassified	5	51	26	13	3	6	2	2	2
Totals	125	399	717	372	340	205	219	127	126

subsystem vary considerably by percentage and by absolute value from spacecraft to spacecraft. For example, the mechanisms subsystem (No. 5) has a fairly constant percentage throughout the seven spacecraft. It averages about 7.5 ±2.5% of all T/FRs for each spacecraft. The data processing subsystem (No. 2) varies considerably with each spacecraft (5.5 ±5%).

Table XII-8 presents a variety of comparisons of the seven spacecraft, including T-2N and T-21 tests, and the 15 failure causes. For example, from *Surveyors I–VII*, design cause decreased by a factor of 12, whereas human-error cause decreased by a factor of 2.5. Decrease in design deficiencies is expected as spacecraft testing progresses. Other causes, such as human error, decrease at a lesser rate and become more pronounced at the end of the program. Also of interest, defective part cause is prevalent throughout the seven spacecraft and does not decrease to an absolute zero, even for *Surveyor VII*.

All the spacecraft (including T-21) T/FRs written during ambient, solar-thermal-vacuum (STV) and vibration environments are compared in Fig. XII-10. The trend of decreasing T/FRs in vibration (except for *Surveyor VI*) indicated that vibration, as a part of system FA testing, was questionable as an effective test. As a test tool, however, STV appears to have been necessary for all spacecraft and, by virtue of program success, must have been significant in weeding out defects at the system level.

To provide a review of electronic-parts problems on the spacecraft, Fig. XII-11 shows those parts responsible for the greatest number of T/FRs. Also shown for comparison are the numbers of each generic-part type used in the spacecraft. Of approximately 219 generic-part types on the spacecraft and approximately 50% of total parts on the spacecraft, the 19 shown had 60% of all T/FRs written against them.

Beginning with *Surveyor II* in the summer of 1966, a spacecraft system test anomaly reporting activity sought to develop a quick-reaction method of isolating insignificant problems to minimize the generation of unnecessary T/FRs. Under the anomaly system, the following rules prevailed:

(1) If problems were immediately identifiable as T/FR material, then a T/FR was written; if not, then an anomaly was recorded.

(2) Each problem, malfunction, or discrepancy recorded in the anomaly log was discussed daily at test reviews and resolved as being a true anomaly or a problem/failure.

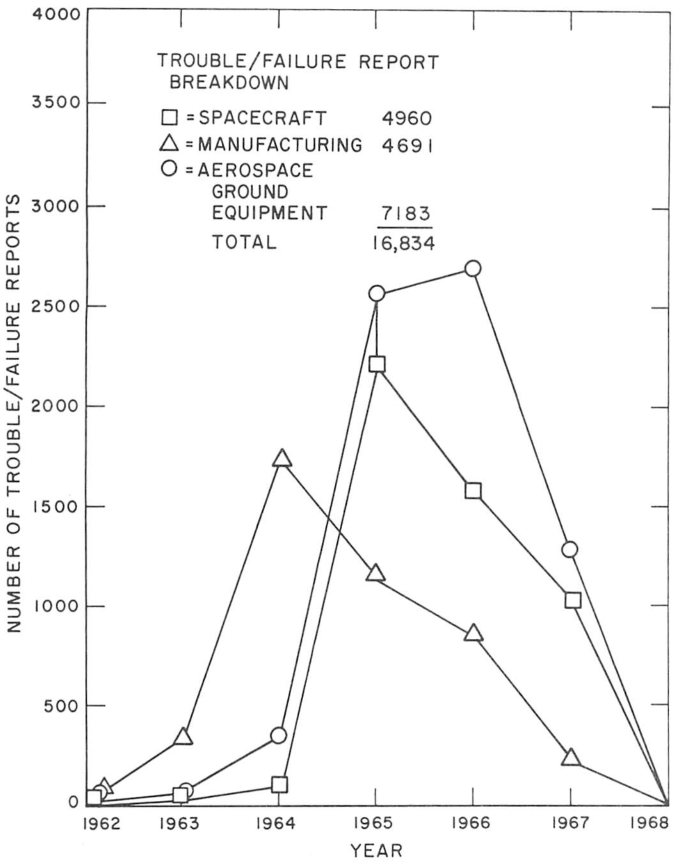

Fig. XII-8. Spacecraft, manufacturing, and aerospace ground equipment trouble/failure reports by year (1962–1968)

(3) If a decision concerning the anomaly's identification and resolution could not be made within 48 hr, then a T/FR was written.

The anomaly system comprised a record (or log) that was modified daily, as needed, and provided a basis for discussions at regular test briefings (these briefings were also called anomaly meetings). Thus, the anomaly system that was developed to report unexplained events or discrepancies, provided one means of generating T/FRs if an analysis of the event showed it to be of T/FR nature. The overall effect of the existence of the system was believed to be negative, however, because it delayed the writing of some T/FRs and caused others not to be written.

A specific example of a delayed T/FR that might never have been written, because it was covered by the anomaly system, was a problem called a long-lines-test problem. On SC-3, during AFETR operations testing, the vernier engines were removed from *Surveyor III* and installed for a test using referee fluids in a hydraulic test laboratory. Long electrical lines connected the spacecraft to the engines. When the spacecraft was exercised by a test procedure to activate the engines, the test was aborted because the electrical resonance of the long lines induced higher actuating voltage than normal to appear at engine solenoids. Initially this problem was called an anomaly and no T/FR was written; later, however, a T/FR was written.

4. *Subcontractor effort.* At the beginning of the program, HAC recognized that basic self-sustaining reliability programs, consistent with HAC *Surveyor* requirements, would be required of subcontractors to provide confidence in procured hardware reliabilities. Early in phase I, reliability assurance program requirements for subcontractors were defined and subsequently included as a subcontractor requirement. During this period, vendor surveys and liaison for reliability were conducted primarily by HAC QA. Early in phase II, evaluation of subcontractor reliability program status disclosed that translation of reliability requirements into action was a major problem area, specifically for the following reasons:

(1) Inadequate inclusion in work statements.

(2) Inadequate subcontractor capabilities.

(3) Inadequate understanding of basic reliability concepts and their practical application.

(4) Inadequate feedback loops from sub-tier vendors to subcontractors.

(5) Inadequate funding.

(6) Inadequate integration in design and engineering activities.

Corrective action was implemented by including technical assessment of product performance and monthly reliability evaluation of subcontractor accomplishments by HAC. Corrective actions initiated in phase II were intensified during phase III by conducting subcontractor reliability training courses that were tailored to correct specific subcontractor deficiencies, by repeated audits, and by reliability approval of subcontractor work statements. During phase IV, the subcontractor effort was reduced to quarterly audits. Major effort was confined to followon subcontracts and hardware corrective-action liaison on mission-critical deficiencies.

5. *Reliability analysis and assessment.* Early in phase I, HAC developed system and subsystem reliability prediction models. The initial intent was to define potential problem areas and provide guidelines for trade-off

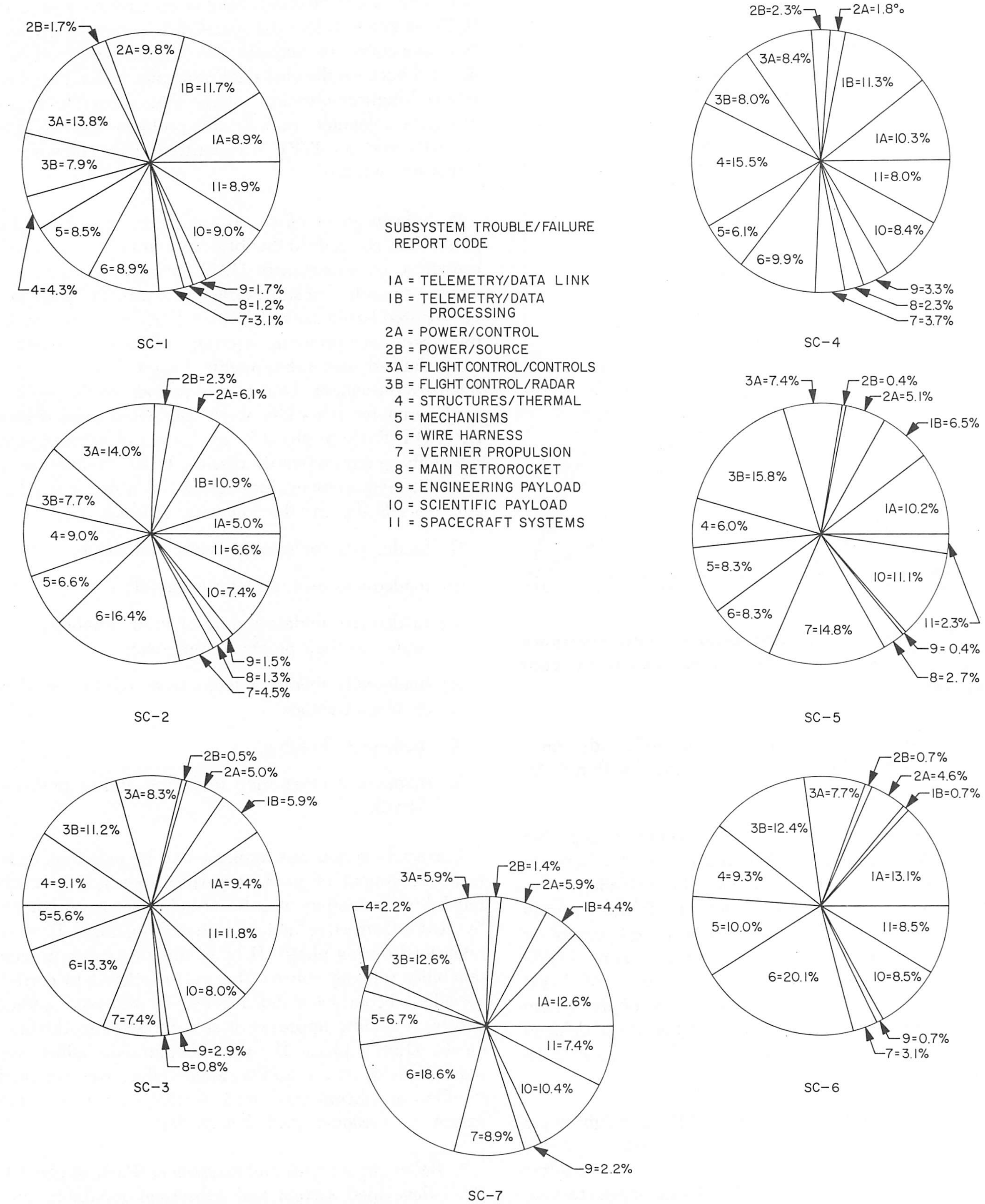

Fig. XII-9. Spacecraft trouble/failure reports by major subsystems (1965 and later)

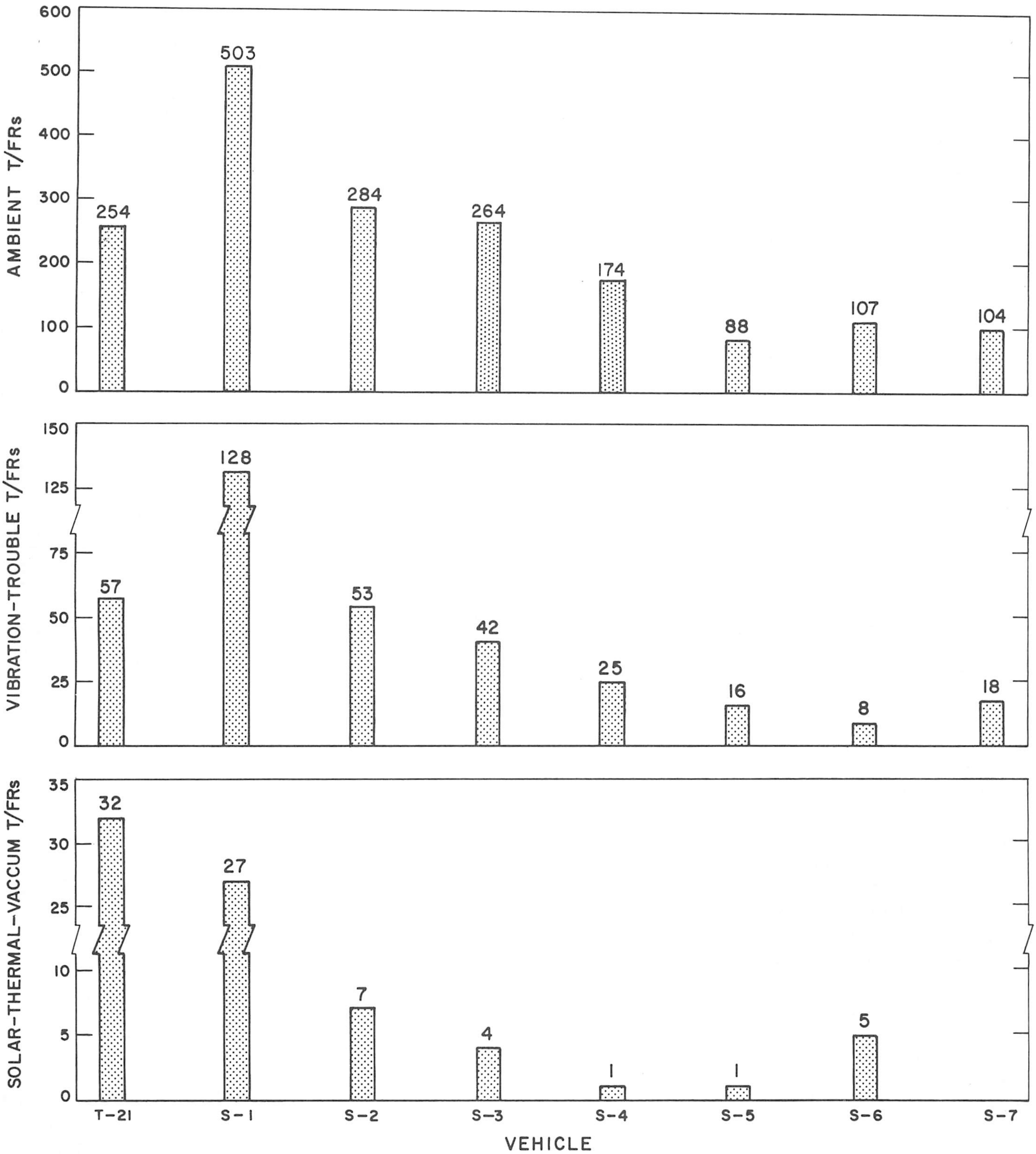

Fig. XII-10. Spacecraft trouble/failure reports by ambient, solar-thermal-vacuum, and vibration environments

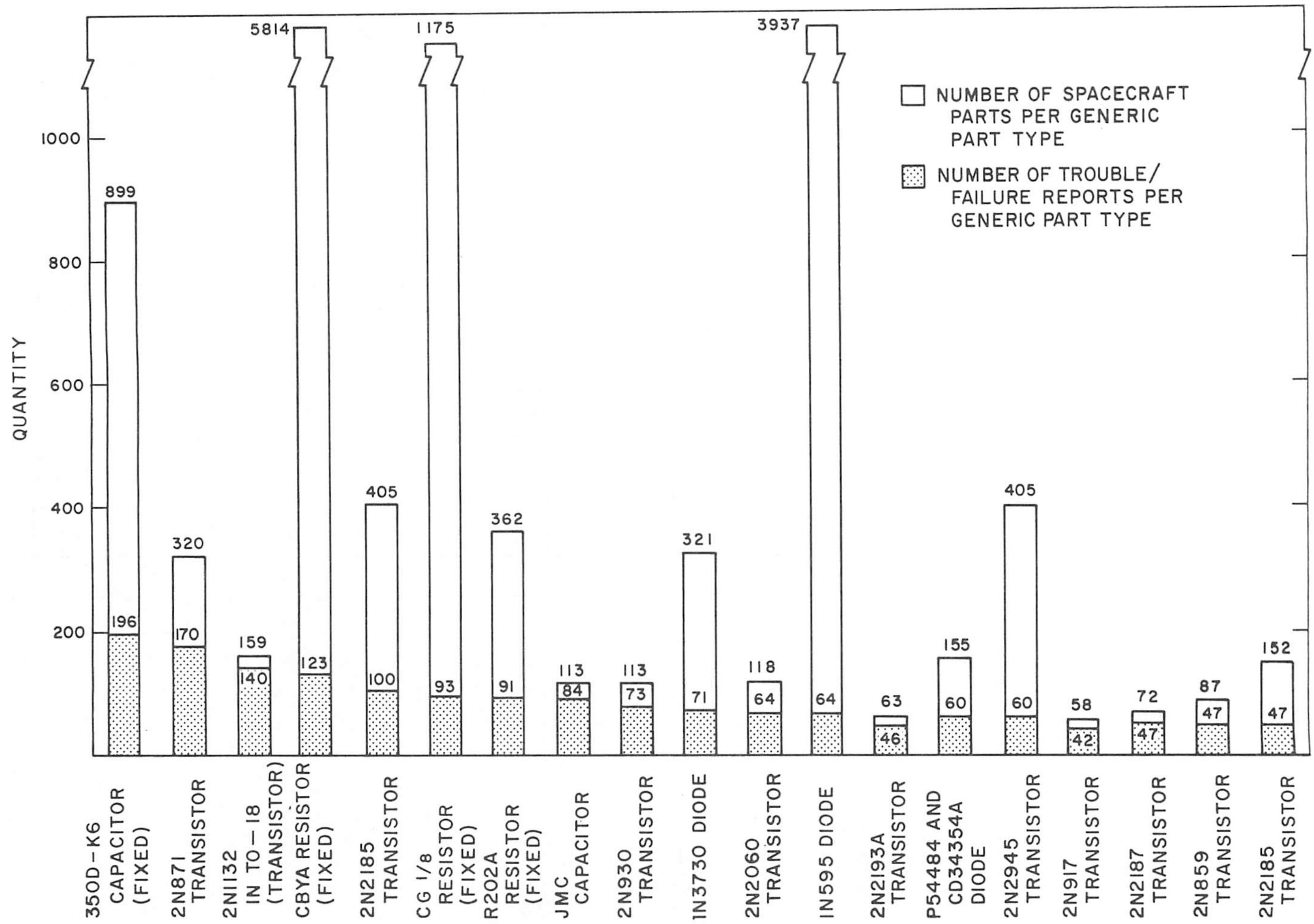

Fig. XII-11. Electronic parts (generic-type) trouble/failure reports

studies at the subsystem level. In addition, these models provided the basis for subsequent test-program planning, reliability assessment, redundancy studies, and a guide for failure-mode analysis. Approximately 10% of the reliability effort was directed toward the application of mathematical and statistical concepts during the preliminary design phase. During phase II, major effort was to provide confidence that major assemblies and subsystems would perform their required functions. A variety of data was analyzed from such sources as HAC, subcontractors, JPL, and other governmental agencies.

To determine the effectuality of design changes and/or corrective actions, the following types of analysis were included:

(1) Degree of variability, considering mean and standard deviation limits of critical performance parameters.

(2) Interface tolerance capability.

(3) Determination of optimum test sample size.

(4) Confidence limits on test results.

(5) Correlation studies.

(6) Goodness of fit tests (statistical) to verify basic failure distribution.

(7) Growth curves.

(8) Significant difference tests (student t, F, etc.)

Except for significant problem areas requiring critical analysis, personnel from cognizant hardware areas conducted these analyses under the direction of *Surveyor* reliability. During this detailed design phase, the analyses comprised approximately 5% of the total reliability effort.

During phase III, HAC reapportioned the spacecraft reliability requirement to the various spacecraft systems.

Test plans were reviewed to assure optimization of testing to meet the 80% confidence requirement. Computer utilization was expanded to provide a near real-time data processing capability. Data reduction and analysis was expedited while covering a large integrated test program. Special studies were conducted:

(1) Sustained vernier system thrust imbalance.

(2) Probabilities of success for all essential spacecraft mission operations that were subsequently utilized in the development of nonstandard mission operation procedures. Unit failure rate histories for each subsystem were initiated by serial number to assist in making CTL decisions. During this phase, approximately 2.5% of the total reliability effort was directed toward analysis.

During phase IV, HAC provided limited direct analytical support for systems test, launch and mission operations. Support included:

(1) Analysis of inertial reference gyroscope problems.

(2) Analysis of g error curve changes in inertial reference accelerometers, which subsequently allowed utilization of possibly degraded mission data.

(3) Investigation of the effect of digital subsystem transient voltage pulses on HAC PNs 988809-1 and 988809-2 PND chopper transistor life characteristics.

(4) The study of each *Surveyor* approved part to determine whether any spacecraft application exceeded its specific capabilities.

The effort expended on *a priori* reliability predictions range from 2% during phase I to 20% in phase IV. Predictions were based on:

(1) System models and piece part counts by generic types in phase I.

(2) Detailed designs, FMA, and limited test data analysis in phase II.

(3) Spacecraft assessments, subsystem service life inventories and growth curves in phases III and IV.

Prior to the fourth engineering test flight, the reliability of the basic spacecraft and the A-21 payload, with a minimum achieved confidence level of 80%, was shown to meet the specified reliability requirements. The spacecraft reliability assessment was based primarily on spacecraft mission simulation test data, plus three actual mission flights. After completion of each mission, a postflight reliability analysis of spacecraft performance (throughout the test and mission phases) was prepared and included in HAC Flight Performance Flight Reports.

Spacecraft reliability prediction and assessment, as used for mission commitment, was based on specific flight hardware histories. These data included time and cycle experience, weighted failures, all relevant T/FRs, reliability growth curves, and a service-life inventory depicting remaining useful life for each spacecraft subsystem designated for flight use.

6. *Reliability testing.* During phase I, a reliability assurance testing program was initiated to provide valid measurements on achievable reliability. This program was intended to supplement the scheduled developmental, type-approval, flight-acceptance, T-21 system and T-2 system tests, thereby providing sufficient durations of test time and/or cycles to expose latent design deficiencies and establish a base for spacecraft evaluation prior to launch, utilizing the equivalent mission concept (EMC). To maximize the early feedback of design and reliability data and to minimize proof-test costs, 20 selected critical items of spacecraft equipment were scheduled for testing as early as possible.

After peaking at about 5% of the total reliability effort in phase II, the effort was cancelled in phase III. Cancellation was due primarily to:

(1) The unavailability of test hardware.

(2) Schedule slippage that prevented obtaining timely data.

(3) Continual redesign of basic spacecraft equipment.

(4) Economy measures.

Approximately one-third of the testing was completed on the following subsystems:

(1) Separation sensing and arming device.

(2) Solar panel.

(3) Attitude jets.

(4) Canopus sensor.

(5) Auxiliary battery.

(6) Main battery.

The program detected several latent design deficiencies, however, including:

(1) Canopus sensor: gas leaks due to inadequate hermetic sealing and the shearing off of four studs holding the electronic package during vibration.

(2) Auxiliary battery: procedural error in electrolyte filling that could cause battery destruction.

(3) Main battery: failure of five batteries in vibration, requiring structural improvement of the silver screen plate.

To demonstrate the reliability of one-shot devices, the concept of margin testing was utilized to reduce prohibitively large sample sizes. Variables of critical functional output parameters were measured to determine the margin of safety; for example, squib testing included ball travel and brisance tests on detonators, as well as peak output pressures.

7. *Reliability training and indoctrination.* Early in phase I, it was recognized that HAC personnel would require reliability orientation to assure the achievement of program objectives. A series of lectures on reliability techniques was conducted, and pertinent space applications data were disseminated. By mid-phase II, over 2300 *Surveyor* personnel (1600 HAC and 700 subcontractor) had attended formal training courses. Training was oriented primarily toward the project goals and objectives, the need for high quality and reliability, and the impact of only one significant failure on a flight mission. Although both formal and informal training was planned for the duration of the program, subsequent activities were confined primarily to continual working contact with design groups and subcontractors; issuance of project bulletins, directives and guides; and encouragement of personnel to participate in professional societies, formal university courses, and corporate-sponsored reliability courses. Some special *ad hoc* classes were conducted to provide advanced training for key personnel.

Coordination was maintained with QA for certification of personnel, conducting NASA-sponsored soldering and workmanship courses, and conducting special courses when required (example, connector training course in early phase IV to reduce mission-critical failures due to human error).

4. Government Furnished Equipment

As part of the total *Surveyor* Program, JPL was responsible for furnishing the Spacecraft System contractor (HAC) with three major flight subsystems, i.e., solar panel, alpha scattering instrument, and the thrust chamber assembly. The service life and reliability of JPL-furnished equipment were the responsibility of JPL; however, HAC was responsible for establishing the reliability goals for such GFE as was consistent with the Spacecraft System requirements. In addition, prior to actual inclusion of any GFE as designated flight hardware for a mission, HAC reviewed the documentation and inspected the hardware. As the Spacecraft System contractor, HAC included GFE as part of the spacecraft for CTL review.

a. Solar panel. The solar panels for *Surveyors V–VII* were redesigned by JPL and subcontracted to EOS for manufacture and qualification testing under JPL technical direction. The flight panels were delivered to HAC, the *Surveyor* Systems contractor, as GFE. During the manufacturing cycle, a reliability program was negotiated with EOS in response to HAC reliability system requirements. The failure reporting activity was in effect during all program testing phases, and EOS completed all elements of the reliability program plan in a satisfactory manner and met HAC system requirements.

Technical direction of EOS by JPL involved a full-time resident engineer and QA inspector, plus consultant specialists who attended weekly status meetings to review the program status and resolve problems that needed management direction. For further details on the solar panel, see Section XVI-C-1.

Solar panel reliability was predicted using a JPL-defined mathematical model. Comparison of the *a priori* estimate with spacecraft requirement is as follows:

	Mission phase	Reliability Required	Reliability Predicted
I	Sixty-six-hour nominal flight and landing	0.989	0.999
II	Touchdown to touchdown + 80 hr (20 hr of predawn in operating mode)	0.999	0.999
III	Lunar daylight (15 days, sunrise to sunset)	0.996	0.999

The solar panel reliability program comprised five major elements:

(1) Performance prediction during mission.

(2) Failure mode analysis.

(3) Design review.

(4) A failure reporting system.

(5) Control of parts, materials, processes, and subcontractors.

An FMA of the *Surveyor* solar panel design for *Surveyors V–VII* was conducted as part of the subcontracted reliability program. Solar panel failure modes were analyzed to determine their effects on panel performance.

A number of failure modes on the electrical solar panel design were examined, such as open and shorted cells, isolation diode failure, cabling failure, and kovar resistance changes. The analysis determined that, in general, two or more failures or combinations of failures would have to occur before the solar-panel power output would be seriously affected. For redesigned solar panel discussion, see Section XVI-B-1.

Two design reviews were held to review the JPL *Surveyor* solar panel design for *Surveyors V–VII,* with JPL personnel, the systems contractor, and the subcontractor in attendance. The first design review meeting was held June 24, 1966, and covered the interface requirements, description of mechanical and electrical designs, justification for the design, and qualification requirements. The second design review was held January 31, 1967, and presented the results and status of the qualification program for the substrate and solar panel, the status of the hardware, and a review of the documentation.

Upon implementation of the JPL P/FR system, in conjunction with the EOS P/FR system, 28 solar panel predelivery problem/failures were reported. Problem/failure exposure allowed for management control, along with the necessary assurance of corrective-action adequacy.

Of the problems reported in Fig. XII-12, 21% were attributed directly to human errors associated with difficult fabrication techniques. Typical discrepancies were delaminations, loose sensors, and crazed paint. Design problems, which constituted 29% of all problems, were

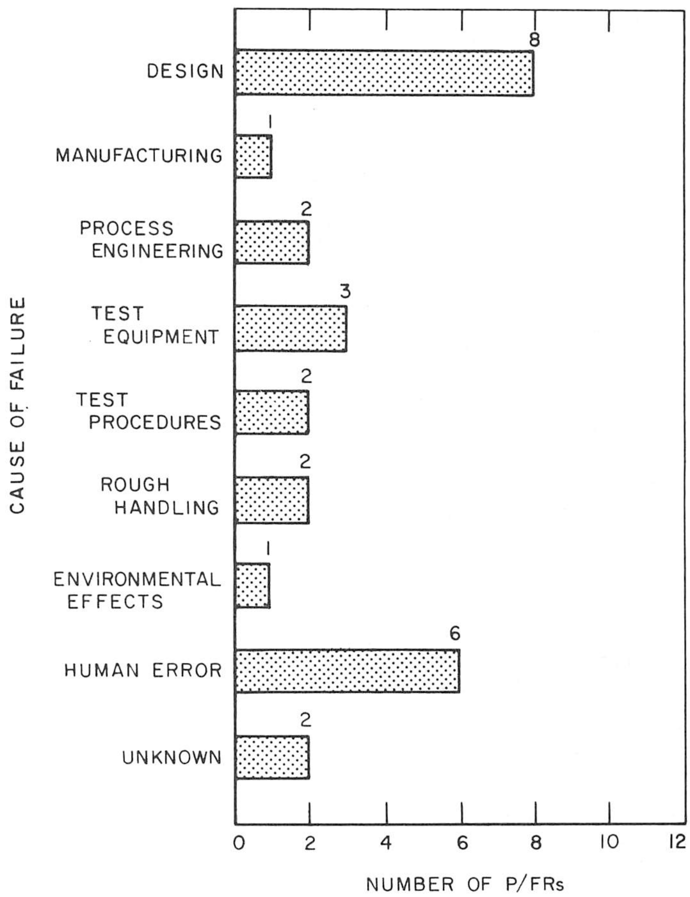

Fig. XII-12. Surveyor GFE solar-panel P/FRs listed by failure cause

primarily test failures on the first manufacturing prototype.

Figure XII-13 shows inspection responsible for initiating 29% of the P/FRs. Most of these inspection problems were latent defects brought to light by FA test environmental stresses.

The effectiveness of testing under thermal-vacuum conditions is illustrated in Fig. XII-14. Approximately 35% of the problems were detected during thermal vacuum. Three of the five vibration P/FRs were caused by human error in running the test equipment.

b. Alpha scattering instrument. The alpha scattering instrument was built by the University of Chicago for JPL and supplied to HAC as GFE. The contract with the University of Chicago did not call for a formal reliability program but did require a reporting of flight equipment failures on JPL P/FR forms. All P/FRs gen-

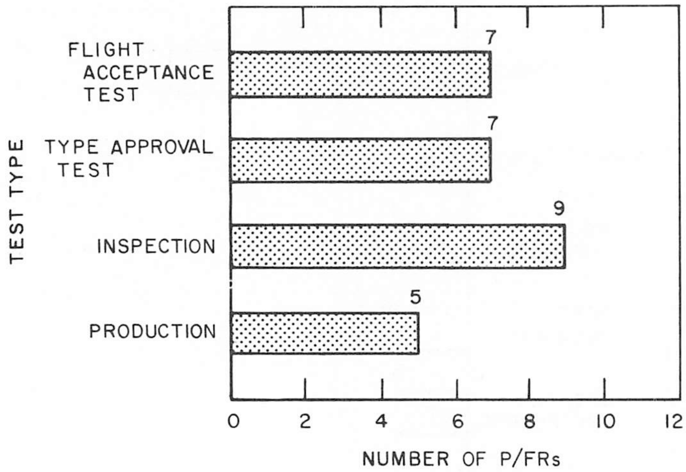

Fig. XII-13. *Surveyor* GFE test-type solar panel P/FRs

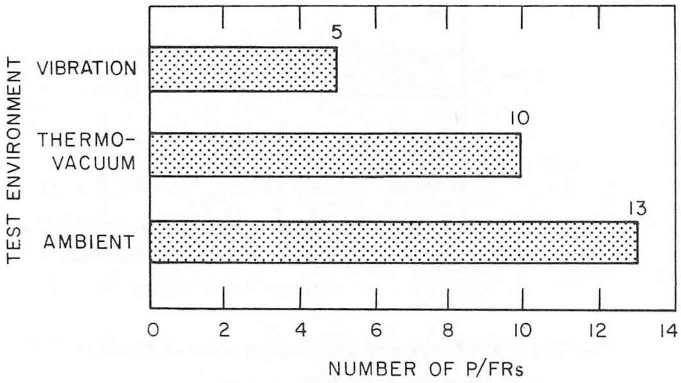

Fig. XII-14. Environmental-test P/FRs of *Surveyor* GFE

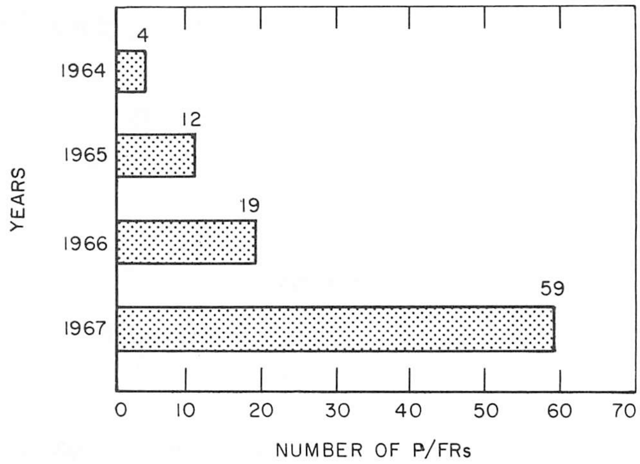

Fig. XII-15. *Surveyor* alpha scattering instrument problem/failure reports by year

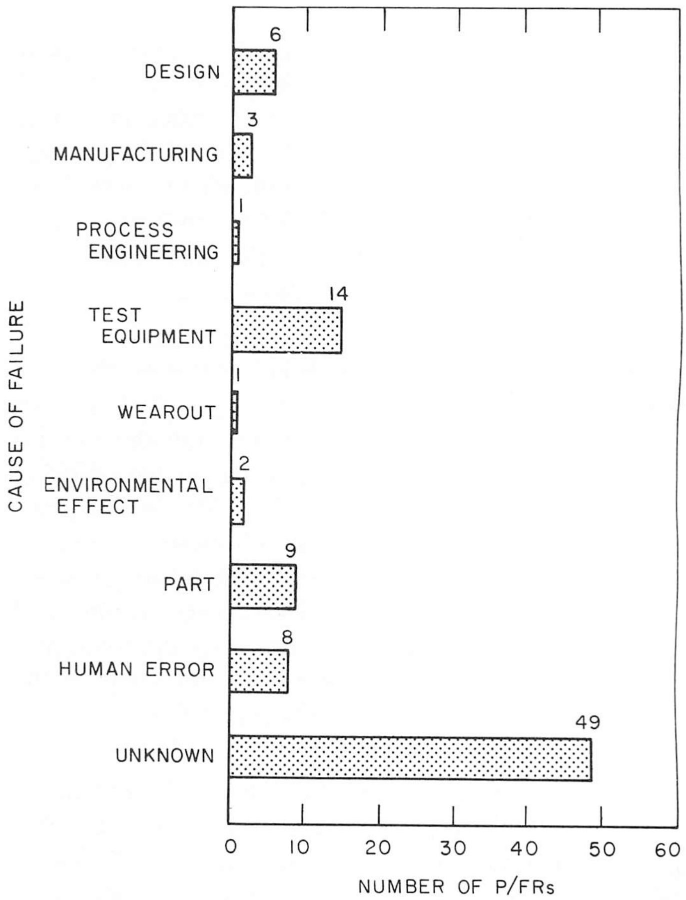

Fig. XII-16. *Surveyor* alpha scattering instrument problem/failure reports by failure cause

erated were reviewed for closeout by JPL and all FA test P/FRs were given to HAC for review and concurrence.

Significant aspects reflected in Figs. XII-15 through XII-18 are the continuing failure of the alpha scattering semi-conducting detectors during:

(1) Production and FA testing.

(2) Thermal-vacuum environment.

(3) The complete test phase of the program.

The basic failure cause was unknown. When some of the early P/FRs reflected that the alpha scattering instrument detectors were in trouble, a new design was instituted; however, the new detectors also were subject to a heavy failure rate. Many approaches were tried to reduce this failure rate, but this state-of-the-art problem was not amenable to solution. It is of interest to note that in spite of the high rate of detector failure during the test program, only one problem was observed during lunar operation: apparently the space vacuum conditions

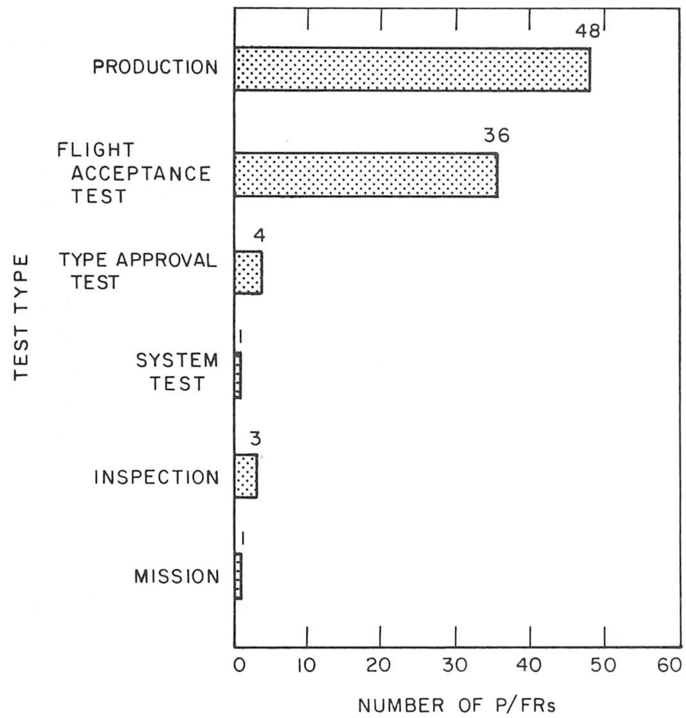

Fig. XII-17. *Surveyor alpha scattering instrument problem/failure reports by test type*

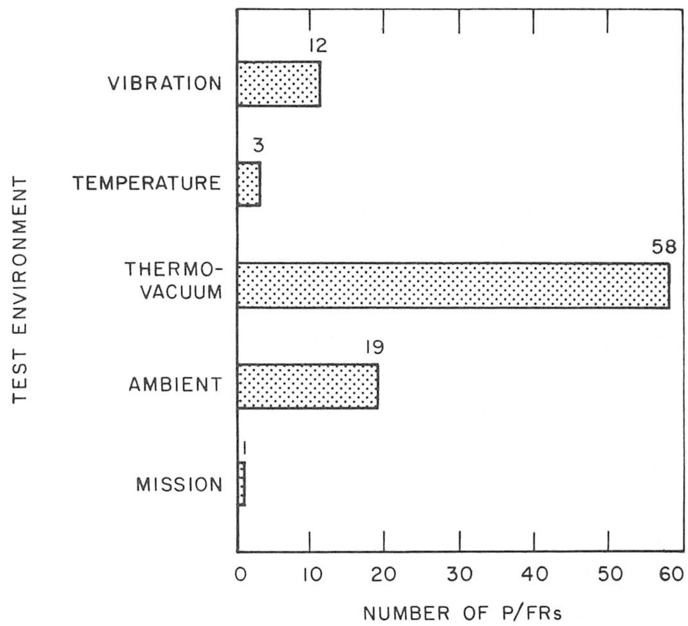

Fig. XII-18. *Surveyor alpha scattering instrument problem/failure reports by test environment*

exhibit some cleaning effect on the detectors, but this phenomenon is not understood. For further details on the alpha scattering instrument, see Section XXII-D.

As shown in Fig. XII-15, the number of P/FRs increased considerably during 1967. This was due to extensive production and FA testing conducted in 1967. A less rigorous failure reporting system was enforced at the University of Chicago in the early developmental phases of the program.

The 49 cause-unknown P/FRs shown in Fig. XII-16 reflect the large number of alpha scattering detector failures. Extensive investigations were conducted, but the exact cause was not determined.

Production and FA testing, the two high points on the Fig. XII-17 chart, were generally attributable to the alpha scattering detector failures. Not shown in Fig. XII-17 are 13 P/FRs written in systems testing and systems readiness testing at HAC, bringing total system test problems to 14.

As shown in Fig. XII-18, most failures occurred during the thermal-vacuum environment testing, due primarily to failure of the alpha scattering detectors. The most common failure mode was evident when the alpha scattering detector became noisy during temperature changes. Thermal vacuum served as effective screening, however, in isolating those detectors with latent defects.

c. Vernier engines. Initial reliability requirements for the vernier engines were imposed by HAC early in phase I (June 1961), by including a detail specification in the HAC subcontract with the Thiokol Chemical Corp., Reaction Motors Division (RMD). Development of the model TD-280 engine by RMD was terminated by HAC in the latter part of phase II (March 1964), due to serious thermal problems.

Concurrent with RMD's development of the TD-280 engine for HAC, JPL initiated a parallel development engine program, which was conducted by TRW, Inc., (Space Technology Laboratories, Inc.). An integrated reliability program was in effect throughout the development program for the cavitating venturi engine.

Upon termination of the HAC/RMD TD-280 engine program, JPL initiated a temporary reinstatement (60-day development program) of RMD, as the engine supplier, to provide an opportunity to assess RMD's progress toward expeditious resolution of all remaining engine problems. Based on engine failure histories and an in-depth design review, several basic design changes were proposed and evaluated by test. See Section XVII-A-4 for detailed discussion.

In July 1964, RMD was reactivated as the engine supplier under direct JPL contract to develop and produce the RMD model TD-339 engine, and RMD implemented a reliability program that was an amplification of prior activities. Since the TD-339 engine was an evolvement of the prior TD-280 engine, all prior test and failure data for unmodified TD-280 hardware elements were utilized to maximize the sample size for reliability achievement estimates. Also included by RMD were post-initial acceptance data from flight engines, as obtained from JPL, HAC, and subsequent RMD rework operations. The reliability estimate for one engine, prior to the *Surveyor I* mission, was 98.9% at the 80% confidence level; the final estimate was 99.0% at the 80% confidence level (Ref. XII-7).

To maintain full visibility of all engine problems and adequacy of corrective action after initial delivery of flight hardware, JPL coordinated and was responsible for an interorganizational failure reporting system (Fig. XII-19), which included the respective RMD, HAC, and JPL systems. As discussed in Subsection B-3, HAC and RMD T/FR closures were formally reviewed by JPL, and JPL P/FR closures were formally reviewed by HAC.

Figures XII-20 through XII-23 present graphically the 65 problems (P/FRs) that occurred after initial delivery by RMD of 28 engines designated for flight use. When an engine failed after delivery to HAC and was returned to RMD for rework and corrective action, a P/FR was written. Production block A, consisting of 11 engines, delivered May 1965–August 1965, had a total of 16 discrepancies; whereas production block B consisting of 16 engines, delivered February 1966–May 1966, had a total of 39 discrepancies. Spare component hardware, not assigned to flight engines, had 10 discrepancies.

Because of the probability that *Surveyor II* failure was caused by one of the vernier engines, additional testing and tighter operational constraints were imposed on that hardware. These requirements resulted in the initiation of additional P/FRs. As shown in Fig. XII-20, the majority of failures were of minor mission effect. Figure XII-21 illustrates the distribution of P/FRs by

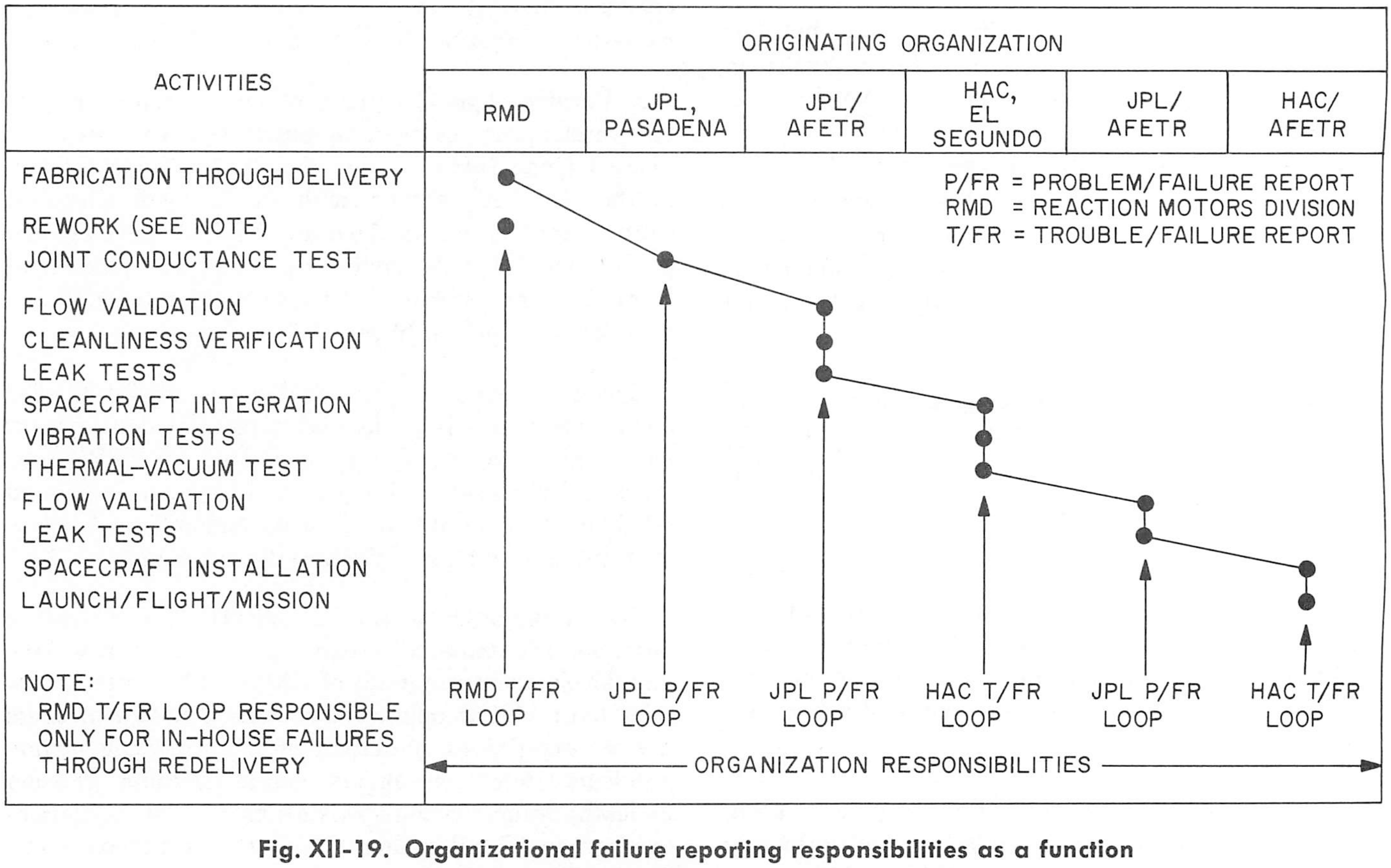

Fig. XII-19. Organizational failure reporting responsibilities as a function of post-acceptance vernier engine hardware flow

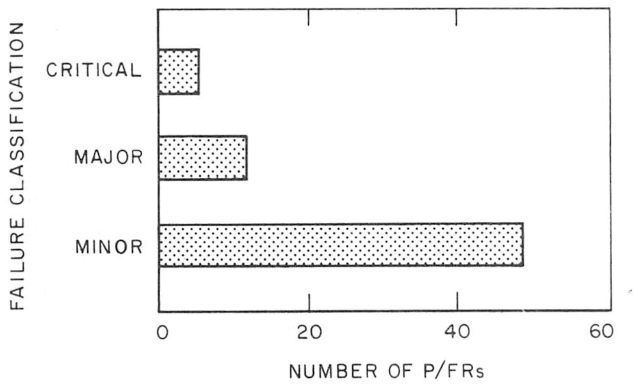

Fig. XII-20. ***Surveyor*** **vernier engine problem/failure reports by failure classification**

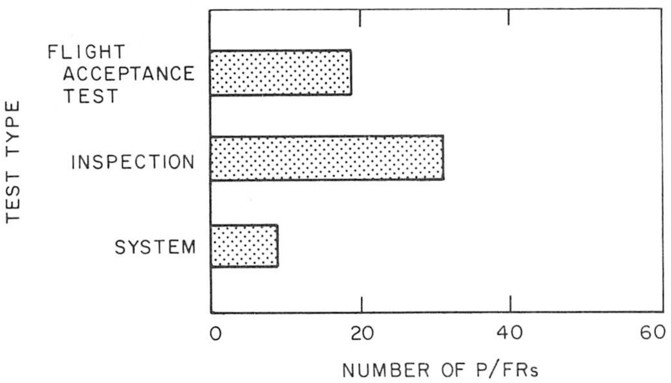

Fig. XII-21. ***Surveyor*** **vernier engine problem/failure reports by test type**

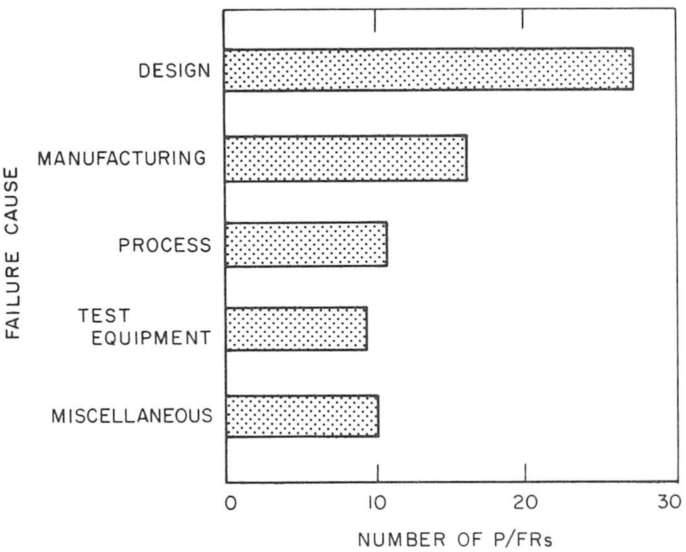

Fig. XII-22. ***Surveyor*** **vernier engine problem/failure reports by failure cause**

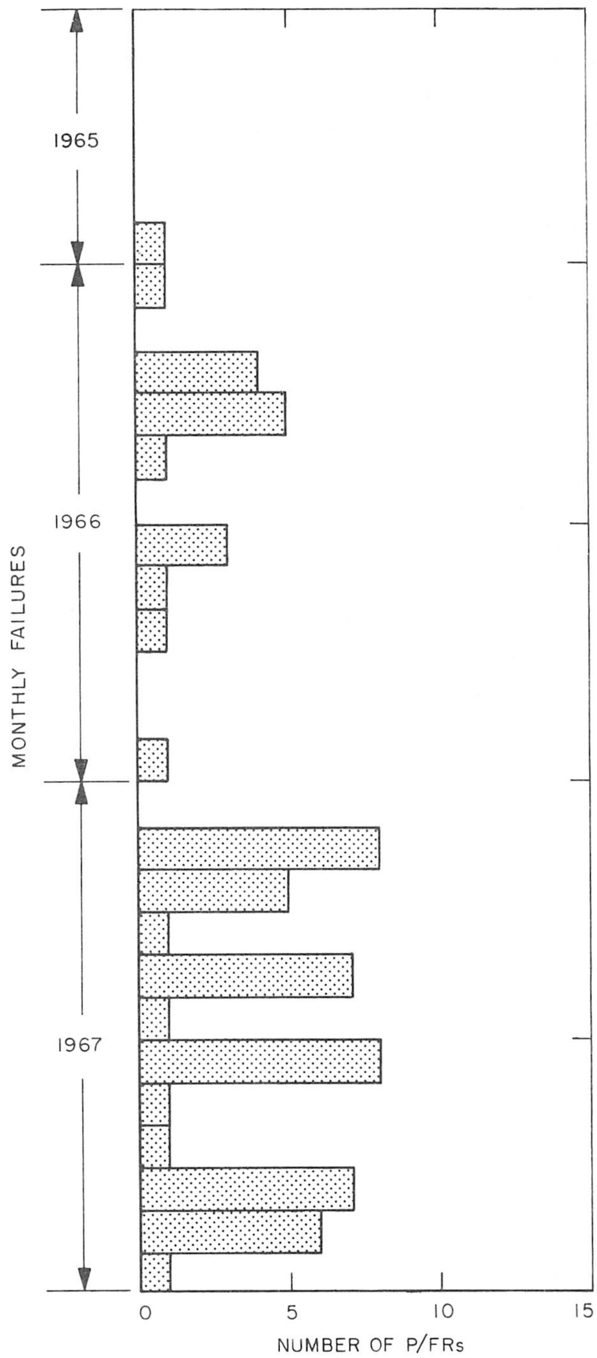

Fig. XII-23. ***Surveyor*** **vernier engine problem/failure reports by month**

test type. The rework category comprises nearly 50% of all P/FRs. Only 18 of 65 P/FRs, or 28% of the problems, were noted during FA testing. Almost 50% were noted during rework, where design and manufacturing defects were still being detected. The most frequent type of failure was design-induced (Fig. XII-22); probably inherent in the prior TD-280 vernier engine design, the design-induced failure was not recognized until more stringent operational requirements were imposed as a result of the *Surveyor II* failure. Most failures attributed to manufacturing and process causes were associated primarily with the residue of malfunctions (nickel salts due to inadequate postfiring flush and other contamination). Figure XII-23 depicts a calendar correlation of PFR generation with preparatory launch activities and shows the effect of more stringent testing during calendar year 1967.

5. Aerospace Ground and Operational Support Equipment

Upon inclusion of definitive reliability program requirements for AGE and OSE in midphase III, the HAC spacecraft reliability program was extended to include ground equipment. Before this, the reliability program had given little attention to ground equipment since all criteria were oriented to design objectives only, including: (1) 5-yr service life, (2) 1-wk mean time between random failures, and (3) the assurance that no mission was to fail as a result of ground system malfunction.

Of major impact was the application of the formal failure reporting system with its rigorous technical closure requirement. Mathematical models for ground command equipment were developed, supplementing the spacecraft mathematical models to provide better visibility of total system probability of success and to provide data for spares provisioning.

To provide better assessment of total system probability of success and provide data for spares provisioning, the necessary confidence in the ground equipment capabilities, a reliability upgrade program was conducted, with effective implementation, prior to the fifth mission; in addition, a maintenance program was defined and implemented. Equipment upgrade was required due to nonspacecraft design and fabrication standards, including the use of nonqualified parts and commercial subsystems.

Appendix

Surveyor Quality Requirement Summary

A. Contractor Quality Plan

Contractors shall be required to prepare a quality plan delineating the method of fulfilling the contractual quality requirements. Quality plans, directives, procedures, etc., shall be submitted to the JPL *Surveyor* Project Quality Assurance Office for review and/or approval. The plan shall contain as a minimum:

(1) A flow chart showing the quality program in terms of what, where, and how, including the various inspection and test points, related quality operations, and applicable documents which establish controls and acceptance criteria.

(2) Delineation of quality requirements and referencing of applicable implementing documents.

(3) Organizational structure, showing the various elements and areas of responsibility necessary to ensure compliance to the quality requirements.

(4) Proposed additions or revisions to current quality system policies, procedures, and organization needed to meet the requirements.

B. Documentation Control

A system shall be established to control all documents affecting the *Surveyor* Project. It will provide for review, release, approval, issuance, change control, and removal of obsolete documentation. Operational and project level documents affecting product quality shall be submitted for *Surveyor* Project QA review prior to their final approval and release.

C. Design and Development

Quality shall be a major consideration in design and development. A system shall be established for reviews of design, drawings, processes, specifications, test procedures and plans, etc., and shall require QA participation.

A system shall be established to perform qualification and requalification of parts, materials, and assemblies. Tests shall be to approved procedures with results documented. A qualification status list of the various articles shall be maintained.

The component part program shall be controlled to ensure compliance to reliability requirements for component parts selection, qualification and acceptance.

D. Configuration Control

Drawing and change control system shall be established for control (release, approval, and change) of drawings and specifications, and provide a valid hardware configuration requirement and status.

Articles shall be identified by a unique part number and change status. Serialization of all subassemblies and assemblies, qualified articles, Hi-Rel, preferred parts, etc., shall be accomplished as a minimum.

E. Subcontractor Controls

A system shall be established for assuring delivery of products that conform to established quality requirements. The system shall provide for selection of qualified suppliers, review of procurement documents and verification of supplier's performance and documented evidence of compliance, as follows:

(1) Supplier shall be selected from approved supplier source list substantiated by quality records and/or quality survey reports. A supplier rating system shall be instigated to provide a continuous and current status of supplier product quality.

(2) Procurement documents shall be reviewed by quality to ensure inclusion of appropriate and compatible and technical requirements, i.e., applicable drawings, specifications, receiving inspection instructions, certifications, special packaging, etc.

(3) Source inspection will be performed on items of a critical or complex nature for which a valid inspection and/or test cannot be performed upon receipt at the contractor's facility.

(4) Receiving inspection shall be used to ensure compliance of delivered products to procurement requirements. Definitive acceptance criteria will be established with provisions for periodically verifying chemical and physical composition of materials.

(5) All items and materials shall be segregated as to whether acceptance or rejection indicates inspection status and provide for a means of traceability to procured lot or part. These controls shall be maintained through all phases of fabrication, assembly, inspection and test.

F. Quality Program Audits

Audits of the quality system will be conducted by contractor quality assurance to ascertain compliance and adequacy of quality requirements, effectiveness of the system, and quality of the product. Corrective action will be initiated to more effectively enhance product quality and to improve quality system implementation.

G. Control of Government Property

A method shall be established to account for, inspect, control, store, and return government property.

H. Control of Fabricated Articles

A system shall be established to ensure that quality and supporting documentation are maintained during the fabrication, assembly, and test of the spacecraft system and its supporting equipment, and shall contain as the minimum the quality considerations outlined below.

1. Inspection and Test

Inspection and test shall meet the following requirements:

(1) Formal planning will be used to define sequence of operations/processes, with appropriate inspection points and acceptance criteria denoted.

(2) Workmanship criteria and test procedures will be developed to clearly define particular methods and acceptance criteria.

(3) Hardware inspection and test verification shall be performed at designated points to ensure compliance with QA requirements.

(4) Appropriate records will be maintained by quality, verifying status and acceptability of products.

2. Manufacturing Control

This aspect of QA shall have the following requirements:

(1) Cleanliness and contamination controls will be established in accordance with the nature of the work.

(2) Processes shall be controlled and procedures established to delineate techniques, equipment, material, and inspection requirements.

(3) A system shall be established for certifying machines, equipment, and personnel performing such special processes as welding, X-ray, magnetic particle inspection, soldering, etc.

3. Acceptance

The system shall provide for incremental acceptance by contractor QA and customer. This should include mandatory inspections (MPC), UCTS, CTT at system test phases, spacecraft CTS, CTL, and final customer acceptance.

I. Nonconforming Material

A system shall be established to identify, document, segregate, disposition, and prevent recurrences of discrepant items by identifying negative trends, and by initiating corrective action.

As a minimum requirement, the MRB consists of the following personnel: the responsible design engineer, quality assurance engineer and customer representative. Concurrence of disposition by the Board must be unanimous.

J. Inspection, Measuring and Test Equipment

A system shall be established to perform and control the calibration and periodic maintenance of all inspection, measuring and test equipment used to determine article conformance and acceptability and production and test tools, jigs, fixtures, etc., and shall provide for procedures, identification, traceability to National Bureau of Standards, records, recall, and calibration integrity.

K. Inspection Stamps

A system shall be established to issue, control, and define usage of inspection stamps. Inspection stamps will be used to designate accomplishment of inspection and document validation.

L. Preservation, Packaging, Storage, and Shipping

Procedures for preservation, packaging, storage, and shipping of parts, assemblies and systems will be developed to minimize degradation of the various items.

M. Statistical Sampling

Statistical techniques used for inspection and acceptance shall conform to MIL-STD-105, or MIL-STD-414 requirements. Approval of specific sampling plans derived from these or the use of alternate plans is required from JPL *Surveyor* Project QA.

N. Training and Certification of Personnel

A program shall be established to train and certify personnel engaged in special processing, and to promote and instill quality mindedness, and upgrade technical skills of personnel responsible for quality.

O. Data Reporting and Corrective Action

A corrective action system shall be implemented to provide for positive and prompt action to prevent recurring discrepancies in reporting, monitoring, and analyzing nonconformances and test failures.

A system shall also be established to file all information relating to quality history. Articles submitted for acceptance will be accompanied by a data package denoting its history, status, and configuration.

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Glossary

AAL	automatic acceleration limiter
ABC	auxiliary battery control
A/D	analog-to-digital
A/DC	analog/digital converter
AESP	auxiliary engineering signal processor
AFC	automatic frequency control
AFETR	Air Force Eastern Test Range
AGC	automatic gain control
AGE	aerospace ground equipment
AMR	altitude marking radar
APC	automatic phase control
ARPA	Advanced Research Projects Agency
AS	alpha scattering
ASACG	Alpha Scattering Analysis and Command Group
ASFOD	assistant spaceflight operations director
ASI	alpha scattering instrument
A/SPP	antenna/solar panel positioner
AVS	altitude velocity sensor
BCD	binary-coded decimal
BCR	battery charge regulator
BDA	Bermuda station
BECO	booster engine cutoff
B/R	booster-regulator
CCD	central command decoder
CC&S	central computer and sequencer
CCSLL	cross-coupled side-lobe logic
CDC	command data console
CDS	computer data system
CPC	command preparation and control
CPCG	command preparation and control group
CRO	Carnarvon (Australia) radar designation; cathode-ray oscilloscope
CRT	composite readiness test
CSP	central signal processor
CST	combined systems test
CSTS	combined systems test stand
CTL	consent-to-launch
CTS	consent-to-ship
CW	continuous wave
DC	direct command
DET	design evaluation test
DFR	dual frequency receiver
DIS	data input subsystem
DPS	data processing system
DSIF	Deep Space Instrumentation Facility
DSN	Deep Space Network
DSS	Deep Space Station
DVS	doppler velocity sensor
ECA	engineering change authorization
ECO	engineering change order
ECP	engineering change proposal (plan)
ECR	engineering change request
ECU	engineering conversion unit
EDR	engineering data reduction
EMA	engineering mechanism auxiliary
ESF	explosive-safe facility
ESP	engineering signal processor
FA	flight acceptance (test, testing)
FACT	flight acceptance composite test
FAR	flight altitude radar
FCSG	flight control sensor group
FPA	flight path analysis
FPAA	flight path analysis area
FPAC	flight path analysis and command
FSG	Flight Support Group
G&A	general and administrative
GCF	ground communications facility
GD/A	General Dynamics/Astronautics
GD/C	General Dynamics/Convair
GDHS	ground data handling system
GHE	ground handling equipment
GSE	ground support equipment
HAC	Hughes Aircraft Company
HSD	high-speed data
HSDL	high-speed data line
IF	intermediate frequency
I/O	input/output
IRIG	interrange instrument group
ISCO	integrated systems checkout
J-FACT	joint flight acceptance composite test
KPSM	klystron power supply modulator
KSC	Kennedy Space Center
LASR	Laboratory for Astrophysics and Space Research
LeRC	Lewis Research Center
LF	low frequency

Glossary (contd)

LOC	launch operations console
LOS	loss of signal
LP	launch platform, launch pad
LV	launch vehicle
LVPS	low-voltage power supply
MAC	maneuver analysis and command
MACG	maneuver analysis and command group
MAPAR	material and process acceptance requirements
MC	midcourse
MCDR	media conversion data recovery
MCDRS	media conversion data recovery subsystem
MCFR	media conversion film recording
MCFRS	media conversion film-recording subsystems
MDL	master data library
MECO	main engine cutoff
MES	main engine start
MO	mission operations
MOPS	mission operations paging system
MOS	Mission Operations System
MPC	mandatory product control
MPS	main power switch
MRB	Materiel Review Board
MS	mission sequence
MSA	mission support area
MSFC	Marshall Space Flight Center (Huntsville, Ala.)
MSFN	Manned Space Flight Network
MTGOP	midcourse and terminal guidance operation program
MTGS	midcourse and terminal guidance system
MVA	mechanism vehicle assembly
NAA	narrow-angle acquisition
NACA	National Advisory Committee for Aeronautics
NAMG	narrow-angle Mars gate
NASC	National Aeronautics and Space Council
NASCOM	NASA Communications Network (worldwide)
NAS—NRC	National Academy of Sciences—National Research Council
NBVCXO	narrow-band voltage-controlled crystal oscillator
NCAR	National Center for Atmospheric Research (Boulder, Colo.)
NSF	National Science Foundation
NSSCC	National Space Surveillance Control Center (Cambridge Research Center)
OAMS	orbital attitude maneuvering system
OAO	Orbiting Astronomical Observatory
OART	Office of Advanced Research and Technology (NASA)
OAT	operating ambient temperature
OCC	operations control center
OCR	optimum-charge regulator
OGE	operational ground equipment
OLF	orbital launch facility
OLO	orbital launch operations
OLV	orbital launch vehicle
OMR	operations management report
OMSF	Office of Manned Space Flight (NASA)
ORT	operational readiness test
OSDP	on-site data processing
OSDR	on-site data recovery
OSE	operational support equipment
OSFD	Office of Space Flight Development (NASA)
OSFR	on-site film recorder
OTA	optical tracking aid
OTC	overload trip circuit
PA	performance analysis (group)
PCC	pneumatic control console
PCE	power conditioning equipment
PCM	pulse-code modulation
PCS	pressure control system
PCU	power control unit
PD	project document
PDP	project development plan
PE	project engineer
PERT	program evaluation and review technique
PFM	pulse-frequency modulation
P/FR	problem/failure report
PIPS	post-injection propulsion system
PLAS	postlanding attitude determination
PLM	pulse-length modulation
PM	phase modulation
PMR	project management report

Glossary (contd)

PPL	preferred parts list
PPM	pulse-position modulation
PPS	primary propulsion system
PRB	problem review board
PRD	program requirements document
PRF	pulse-repetition frequency
PSD	power spectral density
PSNR	power signal-to-noise ratio
PSP	program support plan
PTM	proof test model
PU	propellant utilization
PVT	performance verification test
PWM	pulse-width modulation
RA	radar altimeter
RADVS	radar altimeter and doppler velocity sensor
RCS	reentry control system
RDS	receiver decoder selector
RIG	radio inertial guidance
RIS	Range Instrumentation Ship
RMD	Reaction Motors Division, Thiokol Chemical Corp.
RODVS	reliable-operate doppler velocity sensor
RORA	reliable-operate radar antenna
ROTI	recording optical tracking instrument
RPI	relay position indicator
RSC	range safety command test
RSPWG	range safety and planning work group
RSS	radio subsystem
R-T	receive-transmit
RTC	real-time command
RTCF	real-time computer facility
RTCS	Real-Time Computer System, Kennedy Space Center
RTG	radioisotope thermoelectric generator
RTV	reentry test vehicle
RV	reentry vehicle
SAA	S-band acquisition aid
SADL	Sterilization Assembly Development Laboratory
SAF	Spacecraft Assembly Facility (JPL or AFETR as defined)
SBR	Santa Barbara Research (Santa Barbara, Calif.)
SBRC	Santa Barbara Research Center (Goleta, Calif.)
SCCF	Spacecraft Checkout Computer Facility
SCF	Spacecraft Checkout Facility
SCO	subcarrier oscillator
SCR	silicon-controlled rectifier
SDA	systems data analysis
SDAT	systems data analysis team
SDC	signal data converter
SDCC	simulated data conversion center
SECO	sustainer engine cutoff
SET	solar energy thermionic (conversion system)
SFO	Space Flight Operations
SFOC	Space Flight Operations Center; Space Flight Operations Complex
SFOD	Space Flight Operations Director
SFOF	Space Flight Operations Facility
SFOG	Space Flight Operations Group
SFOP	Space Flight Operations Plan
SFOS	Space Flight Operations System
SFT	system functional test
SIRD	support instrumentation requirements document
SM/SS	soil mechanics/surface sampler
SMAC	soil mechanics analysis and command (group)
SOC	*Surveyor* Operations Center
SOCP	*Surveyor* on-site computer program
SP	signal processor
SPA	signal processing auxiliary
SPAA	spacecraft performance analysis area
SPAC	spacecraft performance and analysis command
SPODP	single-precision orbit-determination program
SPP	solar panel positioner
SRT	systems readiness test
SRV	system readiness verification
SS&A	separation, sensing, and arming
SSAC	space science analysis and control
SSACA	space science analysis and control area
SSB	single sideband
SSD	subsystem decoder
SSE	standard sequence of events
ST	systems test
STCDS	systems test complex data system
STEA	system test equipment assembly
STL	Space Technology Laboratory
STP	standard temperature and pressure
STS	space telecommunications system
STV	solar-thermal-vacuum

Glossary (contd)

TA	type approval (test, testing)
TCA	thrust chamber assembly
TCP	telemetry and command processor
TCR	temperature control reference
TCVA	thrust control vector assembly
TDA	tracking data analysis
T&DA	Tracking and Data Acquisition (office)
TDH	tracking data handling
TDP	tracking data processor
T&DS	Tracking and Data System
T&FA	trend and failure analysis
T/FR	trouble/failure report
TOP	test and operations plan
TPAC	television performance analysis and command
TSAC	television science analysis and command
TSP	telemetry simulation program
TWT	traveling-wave tube
VCO	voltage-controlled oscillator
VECO	vernier engine cutoff
VEV	vernier engine vibration
VFSW	variable-frequency sine wave
VPS	vernier propulsion system
VSWR	voltage standing-wave ratio
VTVM	vacuum tube voltmeter
WBVCXO	wide-band voltage-controlled crystal oscillator