NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report 32-1262

Surveyor VI Mission Report

Part I. Mission Description and Performance

Prepared by the Surveyor Project Staff

JET PROPULSION LABORATORY

CALIFORNIA INSTITUTE OF TECHNOLOGY

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Approved by:

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Prepared Under Contract No. NAS 7-100
National Aeronautics & Space Administration

Preface

This three-part document constitutes the Project Mission Report on *Surveyor VI*, the sixth in a series of unmanned missions designed to soft-land on the moon and return data from the lunar surface.

Part I of this report consists of a technical description and a performance evaluation of the systems utilized in the *Surveyor VI* mission. Part I was compiled using contributions of many individuals in the major systems which support the Project, and is based on data evaluation prior to approximately January 1, 1968. Some of the information in this report was obtained from other published documents; a list of these documents is presented in a bibliography.

Part II of this report presents the scientific data derived from the mission and the results of scientific analyses which have been conducted. Part III consists of selected pictures from *Surveyor VI* and appropriate explanatory material.

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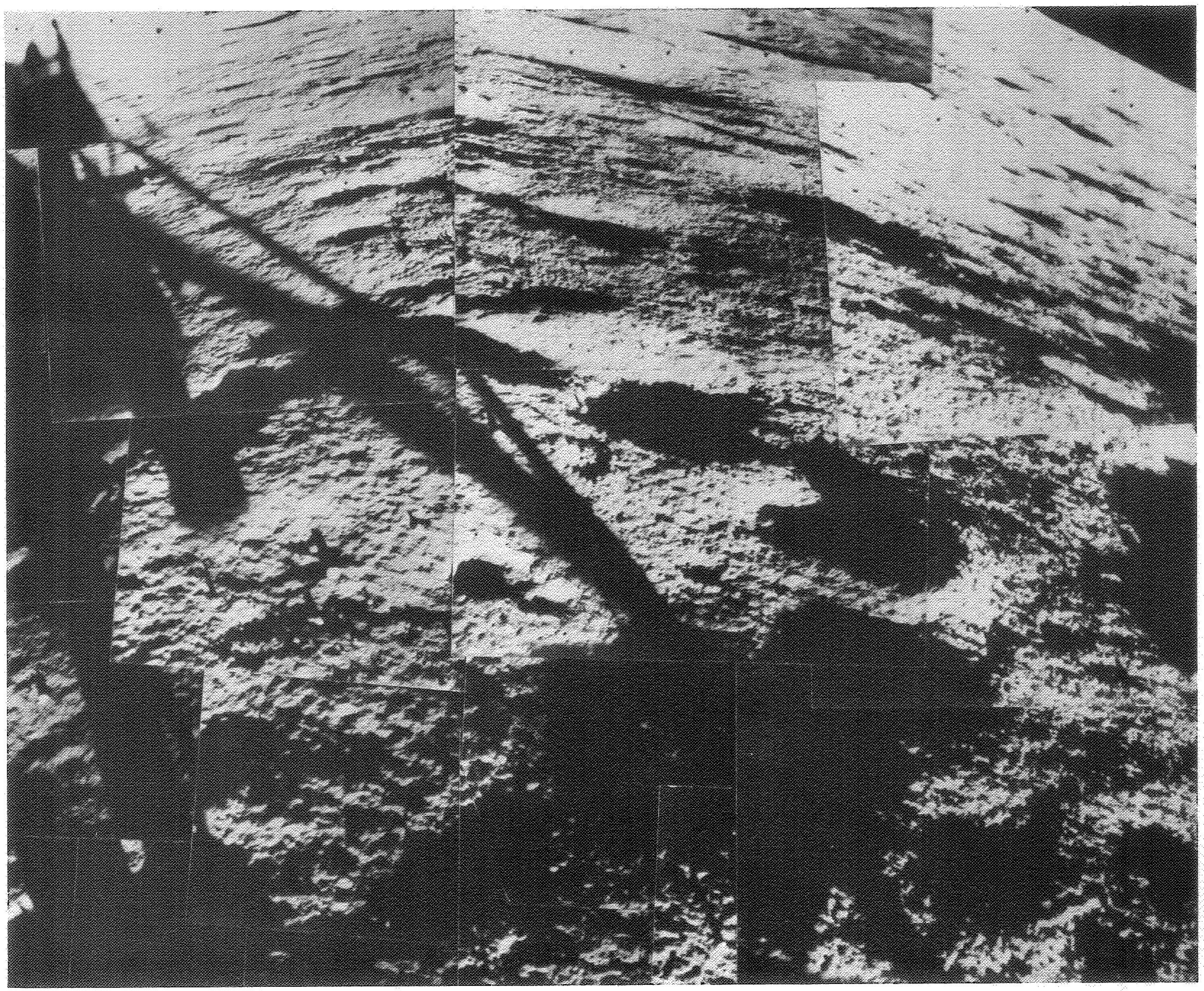
Abstract

Surveyor VI, the sixth in a series of seven unmanned spacecraft designed to soft-land on the moon and return engineering and scientific data, was launched from Cape Kennedy, Florida, on November 7, 1967. All established objectives for the mission were achieved. After a nominal lunar transit phase, a successful soft landing was accomplished on November 10, 1967, in the Central Bay at 0.47 deg north latitude and 1.38 deg west longitude. *Surveyor VI* performed extensive operations before shutdown after sunset of the first lunar day and also was operated briefly the second lunar day. A large quantity of television pictures were provided from the relatively level mare surface on which the spacecraft landed. Other information was also obtained, including data on the chemical composition of the moon provided by an alpha scattering instrument. Unique results were obtained on this mission from a postlanding spacecraft hop experiment, in which the spacecraft moved 8 ft over the lunar surface, and an attitude gas jet firing which caused detectable erosion of the lunar surface.



Wide-angle mosaic of the initial landing taken after *Surveyor VI* hop (Catalog 6-SE-11)

I. Introduction and Summary

Surveyor VI was launched by *Atlas/Centaur* AC-14 from Cape Kennedy, Florida, at 07:39:01.075 GMT on November 7, 1967. After a 12.9-min coast in parking orbit, the spacecraft was accurately injected into a lunar transfer trajectory. The transit phase was very nominal, with the execution of a 10.06-m/sec midcourse correction and the performance of a completely successful soft landing. Touchdown occurred at 01:01:05.467 GMT on November 10, 1967, on a relatively level surface in the Central Bay (Sinus Medii) at 0.47°N latitude and 1.38°W longitude.*

Spacecraft performance throughout the lunar day was excellent, permitting very extensive operations on the lunar surface. Abundant data was obtained from the alpha scattering instrument, television camera, soil magnet, and other spacecraft instrumentation. In addition, a unique experiment was conducted wherein the spacecraft performed a hop to a new location about 8 ft from the original landing site. *Surveyor VI* was shut down on November 26, 1967, about 41 hr after sunset. The spacecraft was turned on again during the second lunar day but could not sustain operations.

*Site based on correlation of *Surveyor* and *Lunar Orbiter* pictures.

The *Surveyor VI* results satisfied all the mission objectives and, with the previous *Surveyor* results, fulfilled the *Surveyor* Project obligation to investigate potential *Apollo* landing sites.

A. *Surveyor* Objectives

The *Surveyor* Project is being conducted to explore the moon with unmanned, automated soft-landing spacecraft which are equipped to respond to earth commands and transmit back scientific and engineering data from the lunar surface.

1. Overall Project Objectives

The overall objectives of the *Surveyor* Project are:

(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 *Apollo*.

(3) To perform operations on the lunar surface which will contribute new scientific knowledge about the moon and provide further information in support of *Apollo*.

2. Results of Previous *Surveyor* Missions

Surveyor I was launched on May 30, 1966, and soft-landed near the western end of the *Apollo* zone of interest at 2.45°S latitude and 43.21°W longitude (based on *Lunar Orbiter III* data). Operations on the lunar surface were highly successful. In addition to a wide variety of other types of lunar surface data, over 11,000 television pictures were received in the course of operations during the first two lunar days. *Surveyor I* exhibited a remarkable capability to survive eight lunar day and night cycles involving temperature extremes of +250 and −250°F.

Surveyor II was launched on September 20, 1966, and achieved a nominal mission until execution of the midcourse velocity correction. One of the three vernier engines did not fire, causing the spacecraft to tumble. Attempts to stabilize the spacecraft by repeatedly firing the verniers were unsuccessful. When nearly all the spacecraft battery energy had been consumed prior to lunar encounter, the mission was terminated shortly after firing of the main retromotor 45 hr after launch. A thorough investigation by a specially appointed Failure Review Board was unable to disclose the exact cause of failure. A number of recommendations were made to assure against a similar failure and to provide better diagnostic data on future missions.

Surveyor III was launched on April 17, 1967, and achieved a successful soft landing, although the spacecraft lifted off twice after initial touchdown before finally coming to rest. Based on correlation with *Lunar Orbiter III* photographs, the *Surveyor III* landing site has been located in the Ocean of Storms at 2.94°S latitude and 23.34°W longitude, about 625 km east of the *Surveyor I* landing site. Important new data was obtained as a result of extensive postlanding operations with a soil mechanics/surface sampler instrument, the television camera, and other spacecraft equipment. The lunar surface characteristics determined by *Surveyor III* confirmed the findings of *Surveyor I* and indicated the suitability of an additional site for *Apollo*, which will utilize final descent and landing system technology similar to that of *Surveyor*.

Surveyor IV was launched on July 14, 1967, and achieved a very nominal mission until the spacecraft radio signal was abruptly lost on July 16, 1967, about 2½ min before expected soft landing. The loss of signal occurred during the main retro phase just 1.4 sec before predicted retromotor burnout, when the spacecraft was about 49,000 ft above the lunar surface and traveling 1070 ft/sec. Exhaustive attempts to reestablish telecommunications with *Surveyor IV* through July 18, 1967, were unsuccessful. A formally appointed Technical Review Board conducted a detailed examination of the *Surveyor IV* mission, but was unable to find evidence of any single or multiple cause for the failure.

Surveyor V was launched on September 8, 1967. A helium leak developed during midcourse correction and persisted although several additional vernier engine firings were made in an attempt to correct the problem. Nevertheless, a completely successful soft landing was achieved as the result of an intensive and resourceful effort by a special HAC/JPL task force team, which assembled during the flight and redesigned the terminal descent sequence by taking full advantage of the inherent flexibility in the basic spacecraft design. In addition to an abundance of television and other data obtained by extensive postlanding operation of the spacecraft, important new data was obtained with an alpha scattering instrument and soil magnet for use in determining the chemical composition and magnetic properties of the lunar surface. Data was also obtained for the first time on the lunar surface erosion effects of firing the vernier rocket engines after landing. The *Surveyor V* results supported and extended the findings of *Surveyors I* and *III* to provide additional assurance of the suitability of the *Surveyor* landing sites for *Apollo*.

3. *Surveyor VI* Mission Objectives

The *Surveyor VI* mission objectives established before launch were as follows:

Primary

(a) Perform a soft landing on the moon.

(b) Obtain postlanding television pictures of the lunar surface.

Secondary

(c) Determine the relative abundance of the chemical elements in the lunar soil by operation of the alpha scattering instrument.

(d) Obtain touchdown dynamics data.

(e) Obtain thermal and radar reflectivity data on the lunar surface.

(f) Conduct a vernier engine erosion experiment.

Surveyor VI satisfied all the mission objectives. Furthermore, the *Surveyor VI* results, combined with the results obtained for the three other mare sites investigated by the previous successful *Surveyor* missions, completely fulfill the *Surveyor* Project obligation to obtain data on potential landing sites for *Apollo*. Thus, the last remaining *Surveyor* spacecraft may be utilized for investigation of a different type of lunar site.

B. Project Description

The *Surveyor* Project is managed by the Jet Propulsion Laboratory for the NASA Office of Space Science and Applications. The Project is supported by four major administrative and functional elements or systems: Launch Vehicle System, Spacecraft System, Tracking and Data System (TDS), and Mission Operations System (MOS). In addition to overall project management, JPL has been assigned the management responsibility for the Spacecraft, Tracking and Data Acquisition, and Mission Operations Systems. NASA/Lewis Research Center (LeRC) has been assigned responsibility for the *Atlas/Centaur* launch vehicle system.

1. Launch Vehicle System

Atlas/Centaur launch vehicle development began as an Advanced Research Projects Agency program for synchronous-orbit missions. In 1958, General Dynamics/Convair was given the contract to modify the *Atlas* first stage and develop the *Centaur* upper stage; Pratt & Whitney was given the contract to develop the high-impulse LH_2/LO_2 engines for the *Centaur* stage.

The Kennedy Space Center, Unmanned Launch Operations branch, working with LeRC, is assigned the *Centaur* launch operations responsibility. The *Centaur* vehicle utilizes Launch Complex 36, which consists of two launch pads (A and B) connected to a common blockhouse. The blockhouse has separate control consoles for each of the pads. Pad 36B was utilized for the *Surveyor VI* mission.

The launch of *Atlas/Centaur* AC-14 on the *Surveyor VI* mission was the sixth operational flight of an *Atlas/Centaur* vehicle, all of which have been very successful. This was the third *Surveyor* mission to utilize the "parking orbit" mode of ascent, wherein the *Centaur* stage burns twice. The first burn injects the vehicle into a temporary parking orbit with a nominal altitude of 90 nm. After a coast period of up to 25 min, the *Centaur* reignites and provides the additional impulse necessary to achieve a lunar intercept trajectory. The parking orbit ascent mode, which will also be used on the last remaining *Surveyor* mission, permits launching for all values of lunar declinations. This allows the design of launch periods which are compatible with favorable postlanding lunar lighting. In contrast, the *Surveyor* direct-ascent missions (*Surveyors I, II,* and *IV*) could utilize only those days (about 8 per month) for which the lunar declination was less than approximately -14 deg.

Surveyor VI was the second mission to utilize the *Atlas* SLV-3C booster, which was uprated from the previous LV-3C configuration by incorporating a 51-in. extension of the tank section, to provide greater propellant capacity, and by increasing the engine thrust levels sufficiently to maintain adequate liftoff acceleration.

2. Spacecraft System

Design, fabrication, and test operations of the *Surveyor* spacecraft are performed by Hughes Aircraft Company under the technical direction of JPL.

Surveyor is a fully attitude-stabilized spacecraft designed to receive and execute a wide variety of earth commands, as well as perform certain automatic functions including the critical terminal descent and soft-landing sequences. Overall spacecraft dimensions and weight of 2200 to 2300 lb were established in accordance with the *Atlas/Centaur* vehicle capabilities. *Surveyor* has made significant new contributions to spacecraft technology through the development of new and advanced subsystems required for successful soft landing on the lunar surface. New features which are employed to execute the complex terminal phase of flight include: a solid-propellant main retromotor with throttlable vernier engines (also used for midcourse velocity correction), extremely sensitive velocity- and altitude-sensing radars, and an automatic closed-loop guidance and control system. The demonstration of these devices on *Surveyor* missions is a direct benefit to the *Apollo* program, which will employ similar techniques.

The first two *Surveyor* spacecraft carried a survey television camera in addition to engineering instrumentation for obtaining in-flight and postlanding data. As an additional instrument, *Surveyors III* and *IV* carried a soil mechanics/surface sampler (SM/SS) device to provide data based on picking, digging, and handling of lunar surface material. On *Surveyors V* and *VI*, an alpha scattering instrument was substituted for the SM/SS device to obtain data from which a chemical analysis of the lunar surface material could be made. A magnet was also attached to one of the spacecraft footpads for missions beginning with *Surveyor IV* to determine magnetic properties of the soil.

3. Tracking and Data System

The TDS provides the tracking and communications link between the spacecraft and the Mission Operations System. For *Surveyor* missions, the TDS uses the facilities of (1) the Air Force Eastern Test Range (AFETR) for tracking and telemetry of the spacecraft and vehicle during the launch phase, (2) the JPL Deep Space Network (DSN) for precision tracking communications, data transmission, processing and computing, and (3) the Manned Space Flight Network (MSFN) and the World-Wide Communications Network (NASCOM), both of which are operated by Goddard Space Flight Center (GSFC).

The critical flight maneuvers and most television operations on *Surveyor* missions are commanded and recorded by the Deep Space Station at Goldstone, California (DSS 11), during its view periods. Other stations which provided prime support for the *Surveyor VI* mission were DSS 42, near Canberra, Australia, and DSS 61, near Madrid, Spain. During postlanding operations on the *Surveyor VI* mission, DSS 42 and DSS 61 also obtained many television pictures, an abundance of alpha scattering data, and other engineering and scientific data. Additional support on a limited basis was provided by DSS 71 (Cape Kennedy) during prelaunch and launch phase, DSS 51 at Johannesburg, South Africa, which provided initial two-way acquisition and coverage during the transit phase, DSS 14 (with a 210-ft antenna at Goldstone) to provide telemetry data during touchdown and to back up DSS 11 during the critical flight phases (midcourse correction and terminal descent) and the postlanding spacecraft hop experiment, and DSS 12 (Goldstone) for additional backup to DSS 11 during terminal descent.

4. Mission Operations System

The Mission Operations System essentially controls the spacecraft from launch through termination of the mission. In carrying out this function, the MOS constantly evaluates the spacecraft performance and prepares and issues appropriate commands. The MOS is supported in its activities by the TDS as well as with special hardware provided exclusively by the *Surveyor* Project and referred to as mission-dependent equipment. Included in this category are the Command and Data Handling Consoles installed at each of the prime DSIF stations and at DSS 71, the Television Ground Data Handling System (TV-GDHS) installed at DSS 11 (TV-11) and the Space Flight Operations Facility (SFOF) (TV-1), and other special display equipment.

C. Mission Summary

Surveyor VI was launched on November 7, 1967, the first day of the selected launch period, from Launch Pad 36B at Cape Kennedy with the *Atlas/Centaur* AC-14 vehicle. Liftoff (L + 00:00) occurred at 07:39:01.075 GMT, only 1 sec after the revised opening of the launch window, which had been delayed during the countdown to achieve improved downrange spacecraft telemetry coverage. Very satisfactory launch phase performance was achieved. Following *Atlas* powered flight on an 83-deg flight azimuth, the *Centaur* first burn injected the spacecraft into a parking orbit with an altitude close to the nominal 90 nm. After a 12.9-min coast period, the *Centaur* was reignited and injected the spacecraft into a lunar transfer trajectory. The injection was very accurate, with the estimated uncorrected impact point only 125 km from the prelaunch target point.

The earth–moon trajectory and major events are depicted in Fig. I-1. Following spacecraft separation, the *Centaur* performed a required retromaneuver sequence to provide increased separation distance from the spacecraft and to miss the moon. After separation, the spacecraft properly executed the automatic antenna/solar panel positioning and sun acquisition sequences. These sequences established the desired attitude of the spacecraft roll axis and insured an adequate supply of solar energy during the coast period.

Telemetry data received in one-way lock by stations of the AFETR and MSFN confirmed a normal mission during the near-earth portion of flight. However, it was not until DSIF acquisition that the required tracking data was obtained to confirm a satisfactory lunar transfer trajectory. As planned, DSS 51 was the first station to establish two-way lock and exercise control of the spacecraft by command. Thereafter, the DSIF stations received and recorded all desired spacecraft data and transmitted necessary earth commands. Nearly continuous two-way coverage of the transit phase was provided, including two-way tracking coverage during the midcourse correction. Extensive use was made of doppler resolver data for the first time on this mission to effectively reduce the standard deviation of the doppler data to about ¼ the value without the resolver.

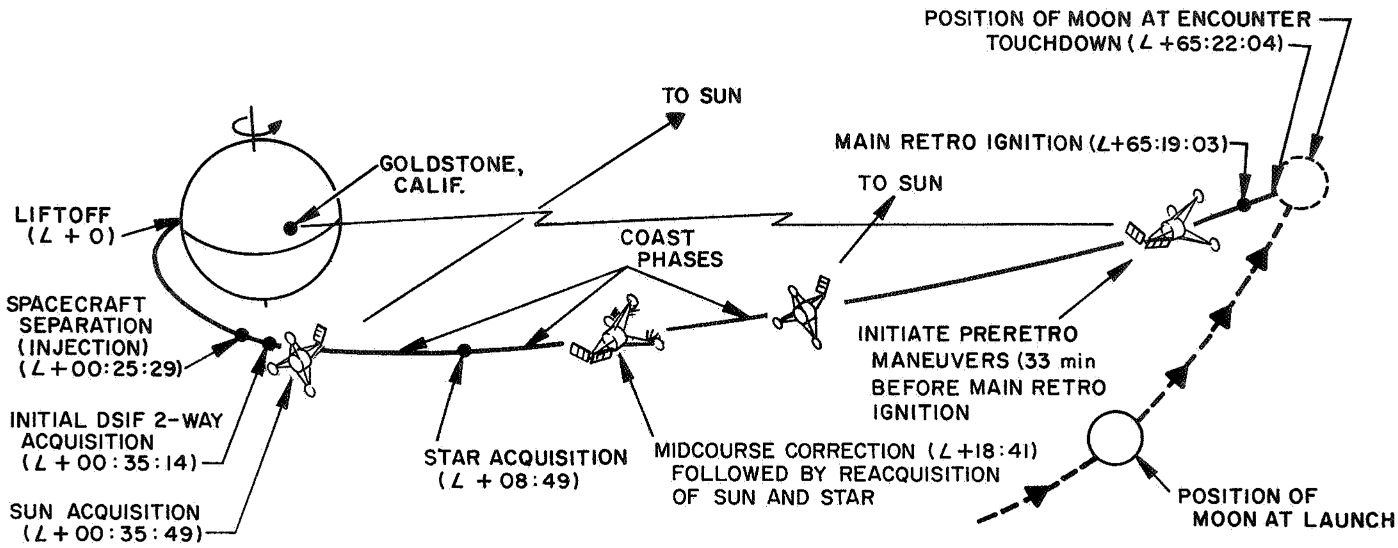


Fig. I-1. Transit phase major events

Spacecraft lock-on with the star Canopus was achieved about 9 hr after liftoff following a spacecraft roll maneuver, during which the Canopus sensor provided data for a star map. The roll maneuver was initiated with the spacecraft transponder on to maintain two-way communications. However, a deeper than anticipated spacecraft omniantenna null caused loss of spacecraft phase lock, which in turn caused a temporary loss of the spacecraft signal by DSS 61. After stopping the roll momentarily to confirm that the spacecraft was performing normally, the roll was resumed in the one-way mode until Canopus lock-on was achieved automatically by the spacecraft. Canopus lock-on provided 3-axes attitude reference, which is required before the midcourse and terminal maneuvers can be executed.

Spacecraft system performance during the coast phases was excellent, with no anomalies occurring. A large number of gyro drift checks were conducted to accurately determine the drift rates, of which the yaw drift rate was above the 1.0-deg/hr specification value. This condition had no adverse effect on the mission because both the midcourse and terminal maneuvers were compensated for the measured drift rates.

During the first "pass" over DSS 11, Goldstone, a roll–yaw maneuver sequence was performed in preparation for a midcourse velocity correction. The final aim point selected in-flight was biased about 0.2 deg east of the prelaunch target point. At 02:20 GMT on November 8, 1967, about 18½ hr after liftoff, a velocity correction of 10.06 m/sec was executed by commanding the vernier engines to burn for a period of 10.25 sec. Only 1.18 m/sec of this velocity change was required to correct to the aim point. The larger maneuver was selected to effect a reduction in the main retro burnout velocity during terminal descent. Shortly after the velocity correction, which corrected the miss distance to within 10.5 km of the aim point, the spacecraft was returned to the coast orientation with sun and Canopus lock by execution of the reverse yaw and roll maneuvers.

In preparation for terminal descent, a roll–yaw–roll maneuver combination was executed, beginning about 33 min before retro ignition, to properly align the retrorocket nozzle in the direction of the velocity vector. Following the attitude orientation maneuver and other preparations commanded from earth, the actual terminal descent sequence was performed completely 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 miles slant range from the lunar surface. After a timed delay of 5.9 sec, which had been preset in the flight control programmer by earth command, the three liquid-propellant vernier engines ignited, followed (after an additional 1.1-sec delay) by ignition of the solid-propellant main retromotor. Performance of the spacecraft through all phases of the descent was very satisfactory, with all events occurring close to the predicted times. The total time of descent from vernier engine ignition until touchdown was 182.5 sec.

The main retromotor provided 8000 to 10,000 lb thrust for almost 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 longitudinal axis along the flight path while maintaining a total thrust level of about 0.9 lunar *g*. When the spacecraft reached 522 ft/sec velocity and 24,730 ft 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 the vernier engines were cut off, causing the spacecraft to free-fall to the lunar surface.

The spacecraft touched down at 01:01:05.467 GMT on November 10, 1967, with a velocity of about 11 ft/sec. A completely successful soft landing was achieved on a relatively smooth and nearly level surface, all three legs making initial contact within a time span of approximately 0.1 sec. The landing dynamics were similar to *Surveyor I*, which also landed on a relatively smooth surface.

The *Surveyor VI* landing site has been identified on *Lunar Orbiter* high-resolution photographs by correlation of features appearing on *Surveyor VI* television frames. This has permitted accurate determination of the site location in the Central Bay (Sinus Medii) at 0.47°N latitude and 1.48°W longitude, approximately 10.5 km from the final aim point and 7.2 km from the final orbit determination using in-flight data. The *Surveyor VI* site is indicated in Fig. I-2 together with landing sites of previous *Surveyor* missions. With the successful soft landing of *Surveyor VI*, four *Surveyor* spacecraft have been successfully landed in the *Apollo* zone of interest on widely spaced mare (with approximately 20 to 25 deg longitude separation).

Communication had been maintained with the *Surveyor VI* spacecraft throughout the terminal descent and landing sequence, making it possible to immediately proceed with postlanding operations. Landing occurred shortly after local sunrise, and the initial assessment of postlanding telemetry data indicated the spacecraft condition to be excellent. Spacecraft performance remained excellent throughout the lunar day, permitting very extensive operations to be conducted.

The first television picture (in 200-line mode) was taken about 50 min after touchdown. A total of 24 high-quality 200-line television pictures were obtained before the spacecraft was reconfigured for transmission of 600-line pictures. The 600-line mode required positioning of the solar panel to receive maximum solar power and precise pointing of the planar array toward the earth to provide maximum signal strength. This reconfiguration was accomplished smoothly, permitting a 360-deg wide-angle survey as well as a special area survey to be accomplished in the 600-line mode before the end of the Goldstone touchdown view period, when it was necessary to transfer to DSS 42 at Canberra, Australia.

Extensive television operations were conducted during each DSIF pass until after sunset. Even during the hot lunar noon period, it was possible to operate the camera at a partial duty cycle by adjusting the antenna/solar panel positioner (A/SPP) to shade the camera and electronics compartments. Although most of the *Surveyor VI* television pictures were obtained during Goldstone view periods, when the special TV Ground Data Handling System could be used to relay the pictures from Goldstone to the SFOF for evaluation in real-time, DSS 42 and DSS 61 (near Madrid, Spain) participated actively in television operations on this mission. *Surveyor VI* obtained a total of about 30,000 pictures, surpassing in quality as well as quantity the pictures received on any previous mission. This performance reflects the significant improvements incorporated in the *Surveyor VI* camera.

The *Surveyor VI* television sequences included (1) a large number of wide- and narrow-angle panoramas of the lunar surface out to the horizon, (2) pictures of stars to aid in spacecraft attitude determination, (3) images obtained with different polarized filters, (4) images obtained with different focus settings to provide a means of determining the distance of objects from the camera, (5) pictures of the visible parts of the spacecraft, (6) pictures showing disturbances of the lunar soil resulting from initial landing, an attitude control gas jet experiment, and a spacecraft hop experiment, (7) pictures showing the amount of material adhering to the bar magnet, and (8) pictures of the solar corona after sunset. Many television pictures were repeated during the lunar day to obtain views under different lighting conditions, and a series of frames was obtained toward the end of the lunar day showing the progression of shadows cast by the spacecraft and lunar surface features.

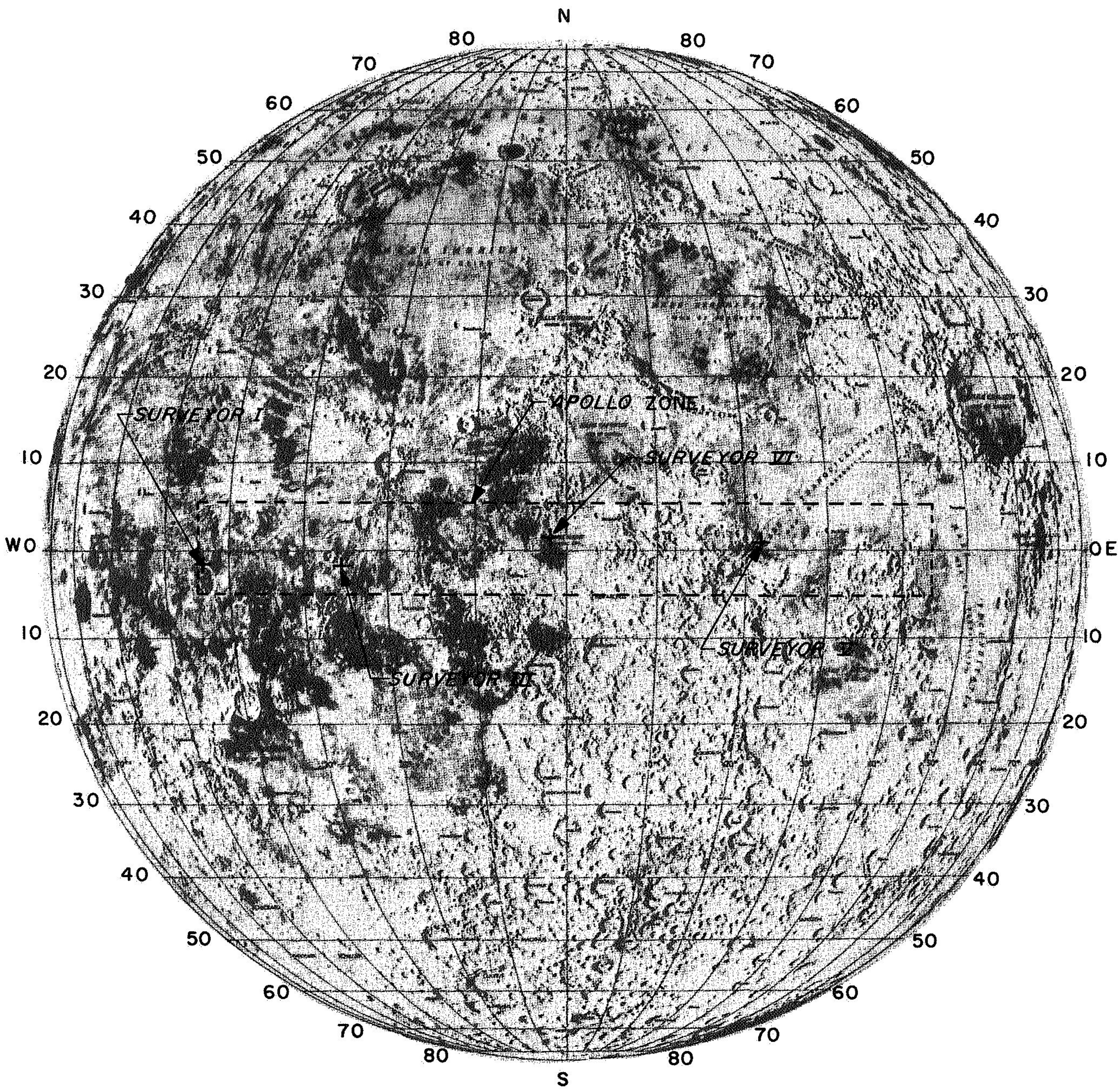


Fig. I-2. Soft-landing sites of *Surveyor VI* and previous *Surveyor* spacecraft

The attitude control gas jets were fired twice about 26 hr after landing, and television pictures indicated that this experiment definitely resulted in disturbance of the lunar surface.

The alpha scattering instrument was also operated extensively to provide sufficient data to obtain a chemical analysis of the lunar surface material, which could be compared with the results obtained on the previous mission for a different lunar mare site. After obtaining calibration data in the stowed position, the sensor head was first deployed to a position about 22 in. above the lunar surface to obtain data on background radiation; then it was lowered to the surface for gathering of lunar back-scattering data. About 30½ hr of alpha and proton back-scattering data was obtained on the surface before the spacecraft hop was performed. After the spacecraft hop, the sensor head was found to be resting on its side. No further data could be obtained for surface analysis. However, an additional 13 hr of data was accumulated for possible use in investigating cosmic proton activity.

Spacecraft engineering data was received frequently throughout the lunar day to enable repeated assessment of the spacecraft condition. In addition, two-way doppler tracking data was obtained whenever possible using the resolver doppler counter.

The vernier propulsion system was monitored very closely, beginning immediately after landing, in order to prepare for the spacecraft hop experiment. A time shortly after lunar noon was selected for performing this experiment, thus permitting sufficient time to obtain substantial results from the other experiments. 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 conducting the experiment.

A very successful spacecraft hop was performed in which the vernier engines operated for 2.5 sec. The planned burn time was 2.0 sec. However, the vernier engines were shut down by the backup earth-command, which was transmitted 0.5 sec after the prime command. During the hop, which lasted approximately 6.1 sec, the spacecraft rose 10 to 15 ft above the lunar surface and landed about 8 ft from its initial location.

Except for the alpha scattering instrument sensor head, which came to rest upside down, the spacecraft survived the hop in excellent condition and it was possible to resume normal operations. From the new location, excellent pictures were obtained of the original landing site, including the erosion of the lunar surface caused by the vernier engines during the hop. The hop experiment caused some lunar surface material to adhere to various spacecraft surfaces, including the magnet. However, there were no detrimental effects. The camera mirror had been commanded to the completely closed position before the hop to maintain clean optical surfaces.

As the first lunar day came to a close, the condition of *Surveyor VI* was excellent. In order to enhance the chances of spacecraft survival through the lunar night, operations near the end of the lunar day were carefully planned to provide nearly a fully charged battery at sunset. This was accomplished while still obtaining desired television pictures and other data.

During the final hours before sunset, narrow-angle surveys and shadow progression sequences were obtained with the camera. Owing to lunar surface unevenness, sunset at the spacecraft occurred about 50 min earlier than predicted at 13:53 GMT, November 24, 1967, during Goldstone view. After sunset, along with solar corona pictures, two star surveys were obtained and two pictures of the spacecraft and lunar surface were taken in earthshine. When the last television picture had been taken, about 6 hr after sunset, the solar panel and planar array were placed in the optimum position for resumption of operations during the second lunar day.

The spacecraft was operated on a low duty cycle as long as possible into the lunar night in order to prevent the battery, which is located in a superinsulated compartment, from cooling to the low temperature of the lunar night environment. Interrogations of the spacecraft were continued periodically during this time for engineering assessment and to obtain thermal data of scientific interest. About 13 hr after sunset, it became evident that some of the thermal switches of the electronic compartments, which are designed to open when the environment cools down, were not opening as expected. This caused the heat losses from the thermal compartments to be greater than planned, requiring the dissipation of extra battery energy in the compartments to maintain temperatures. This forced abandonment of the plan for extended operation into the lunar night and final shutdown of the spacecraft occurred only 41 hr after sunset. By comparison, *Surveyor V* was operated until 115 hr after sunset.

During the second lunar day, *Surveyor VI* responded to a turn-on command at 16:45 GMT on December 14, 1967, but the signal was erratic and provided very little useful engineering data. The signal was finally lost at 19:14 GMT, December 14, 1967, and could not be reacquired during subsequent interrogations.

II. Space Vehicle Preparations and Launch Operations

The *Surveyor VI* spacecraft was assembled and flight-acceptance-tested at the Hughes Aircraft Company (HAC) facility, El Segundo, California. After completion of these tests, the spacecraft was airlifted to the Air Force Eastern Test Range (AFETR), Cape Kennedy, arriving on September 5, 1967.

The *Atlas/Centaur* launch vehicle stages arrived at AFETR on August 26 and 31, 1967, respectively. Prelaunch assembly, checkout, and systems tests were performed successfully at AFETR, and launch was accomplished on November 7, 1967, at 07:39:01.075 GMT, 1.075 sec after the rescheduled opening of the launch window on the first day of the launch period.

A. Spacecraft Assembly and Testing

Tests and operations on the spacecraft were conducted by a test team and data analysis team which worked with the spacecraft throughout the period from the beginning of testing until launch. The test equipment used to control and monitor the spacecraft performance at all test facilities includes (1) a system test equipment assembly (STEA) containing equipment for testing each of the spacecraft subsystems, (2) a command and data handling console (CDC) similar to units located at the DSIF stations (see Section VI-B) for receiving telemetry and TV data and sending commands, and (3) a computer data system (CDS) for automatic monitoring of telemetry during spacecraft testing.

The *Surveyor VI* spacecraft (SC-6) began system testing on March 3, 1967, and passed through the following test phases:

1. Spacecraft Ambient Testing

The ambient test phase consists of initial system checkout (ISCO) and mission sequence tests. During ISCO, each subsystem is first tested for subsystem performance and then is tested for compatibility and calibration with the other spacecraft subsystems. Initial system operational verification is demonstrated by a system readiness test (SRT). The ISCO test phase was completed on May 30, 1967.

During ISCO, several flight hardware failures were encountered. The boost regulator was damaged by a reversal of two connectors, and the RADVS signal data converter (SDC) was damaged by a failure in the ground support equipment. The flight control sensor group (FCSG) was returned to the unit area because of pin retention failures, and the FCSG harness was replaced. A receiver, the alpha scattering instrument electronics, and the RADVS antenna for Beams 1 and 4 were also returned to the unit areas for unit test verification or incorporation of changes resulting from other *Surveyor* spacecraft testing.

The mission sequence (MS) tests are performed to obtain system performance characteristics of the spacecraft under ambient conditions and in the electromagnetic (EM) environment expected on the launch pad and in the flight prior to separation from the *Centaur*. Mission Sequences 1 and 2 are compressed (32-hr) mission runs without the expected EM environment. Mission Sequence 3 is a full (68-hr) mission with plugs out and the expected launch EM environment. This test phase was completed on June 26, 1967.

The only major problem encountered in mission sequence testing was a failure of the FCSG magnitude register to accept commands during MS 2. The FCSG was returned to the unit area and a spare unit was installed on the spacecraft, after which MS 2 was repeated and MS 3 was conducted successfully.

2. Solar–Thermal–Vacuum (STV) Testing

The STV test sequences are conducted to verify proper spacecraft performance in simulated missions at various solar intensities and a vacuum environment. During STV tests, and in the vibration tests which follow, the propellant tanks are loaded with "referee" fluids to simulate flight weight and thermal characteristics.

Preparation for the *Surveyor VI* STV testing began on June 27, 1967. While the spacecraft was being prepared for STV testing, a failed capacitor in the FCSG yaw gyro assembly prevented the gyro from nulling and was replaced. The first test in the solar–thermal–vacuum chamber commenced on July 14 with a Phase A sequence conducted at 112% of nominal sun intensity (high sun). During the Phase A sequence, Vernier Engine 1 ran hotter than expected and the Transmitter B high voltage dropped out in the terminal descent portion of the test. The high-voltage dropout could not be repeated, and the testing proceeded into a Phase B sequence conducted at 87% of nominal sun intensity (low sun). During this sequence, the RF uplink to the spacecraft was lost immediately after the altitude marking radar (AMR) *mark* occurred in the terminal descent portion of the simulated mission. This problem could not be repeated and Phase B was concluded on July 22. The STV chamber was opened (Fig. II-1) to permit replacement of spacecraft Receiver A and Transmitter B. In order to reverify spacecraft performance, a second Phase A sequence was initiated on July 29 but was aborted on July 30 because of recurrence of the uplink problem. This anomaly was repeatable and was traced to "spurs" on the STEA transmitter signal entering the spacecraft receiver. The transmitter was replaced and a third Phase A sequence began on August 2, 1967. This test was completed on August 6 with the only problem being periodic "phase jitter" of spacecraft Transmitter B. The transmitter was replaced and preparations began for the vibration phase of system testing.

3. System Vibration Testing

Vibration tests are conducted in each of the three orthogonal axes of the spacecraft to verify spacecraft integrity and proper functional operation during and after exposure to a simulated launch environment. During these tests, the spacecraft is placed in the launch configuration with the landing legs and omniantennas in the folded position. A vernier engine vibration (VEV) test is also conducted, with the vibration input at the vernier engine mounting points to simulate the vibration environment during the midcourse and terminal descent phases of flight. The spacecraft is checked functionally during each phase, and pre- and postvibration alignment checks are performed to verify that the spacecraft is not degraded by exposure to the vibration environment.

The *Surveyor VI* vibration test phase began on August 8 and ended with completion of VEV on August 30, 1967. During the vibration tests, the solar axis position potentiometer of the antenna/solar panel positioner (A/SPP) registered erratic indications which were caused by the wiper arm lifting off the potentiometer windings. Since the position indication was the same before and after vibration, the potentiometer was not changed. The pin puller used for locking the solar axis during launch also showed some movement during vibration and the pin was modified to provide positive locking. Between the launch vibration and VEV test, an abbreviated mission sequence was performed with "plugs out" under ambient conditions to verify spacecraft performance. After transport preparations, the spacecraft was shipped by air to

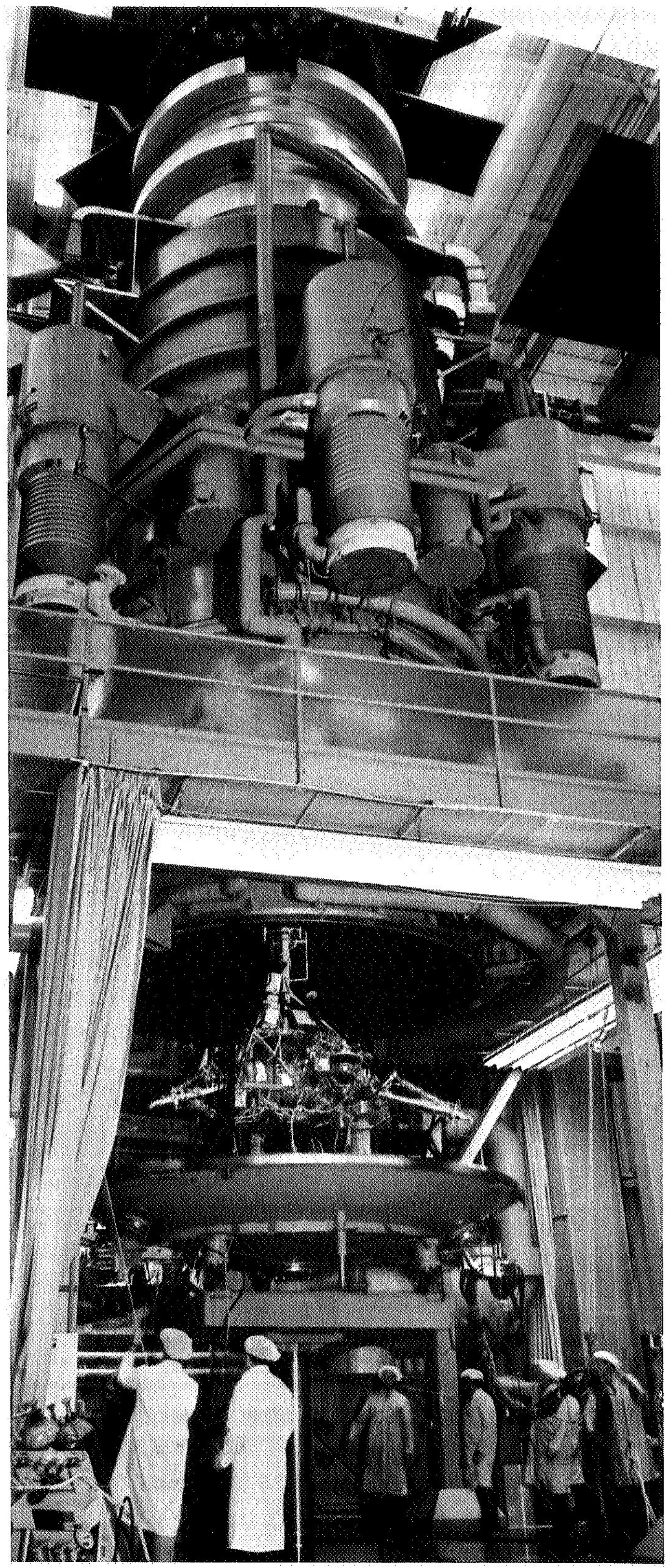


Fig. II-1. Removal of *Surveyor VI* spacecraft from solar-thermal-vacuum chamber after initial tests

AFETR on September 4, 1967. The Combined System Test (CST) phase was eliminated from the *Surveyor VI* spacecraft operations because of schedule limitations.

B. Launch Vehicle Combined Systems Testing at San Diego

Following satisfactory completion of factory acceptance testing of each launch vehicle stage, the *Atlas* and interstage adapter were installed in the Combined Systems Test Stand (CSTS) at San Diego, California, on July 3, followed by the *Centaur* on July 5, 1967. Test sequences in the CSTS culminated in a vehicle Flight Acceptance Composite Test (FACT) on August 16. Several hardware modifications and replacements were completed, and the NASA data review on August 17 and 18 determined the vehicle to be acceptable and ready for shipment to AFETR. The *Atlas* was shipped by air on August 25, followed by the nose fairing and interstage adapter on August 26 and the *Centaur* on August 30, 1967.

C. Launch Operations at AFETR

The major operations performed at AFETR for the launch vehicle and spacecraft are listed in Table II-1.

1. Initial Preparations

Receiving inspection and reassembly of the spacecraft at Building AO began on September 5, 1967. Starting on September 7, a series of Performance Verification Tests (PVT 1 through PVT 4) and television calibrations were performed to verify flight readiness of the spacecraft. Two significant problems were noted during this period. First, the klystron power supply modulator (KPSM) would not pass the ranging test and a spare unit was installed on the spacecraft. The second problem occurred during a functional test of the vernier engines. Vernier Engine 2 failed to shut down at the prescribed time, necessitating replacement of the thrust chamber assembly (TCA). The vernier engine functional test was then repeated successfully.

The *Atlas* and *Centaur* stages of AC-14 were erected on Launch Pad 36B on September 11, 1967, the third day after the launch of *Surveyor V* from the same launch pad. Replacement of pad burnoff items, vehicle checkout, and preparation for the Joint Flight Acceptance Composite Test progressed concurrently, culminating in a successful guidance/autopilot (GAP) test on

Table II-1. Major *Surveyor VI* operations at Cape Kennedy

Operation	Location	Date completed, 1967
SC-6[a] inspection, reassembly and initial checkout	Building AO	Sep 10
AC-14[b] erection (*Atlas* stage)	Launch Complex 36B	Sep 11
AC-14 erection (*Centaur* stage)	Launch Complex 36B	Sep 11
SC-6 vernier propulsion system (VPS) functional test	Building AO	Sep 14
SC-6 Performance Verification Test (PVT) 3	Building AO	Sep 16
SC-6 PVT 4 and TV calibration	Building AO	Sep 20
SC-6 mate to adapter	Explosive Safe Facility (ESF)	Sep 23
SC-6 mate to *Centaur*	Launch Complex 36B	Sep 27
SC-6 spacecraft/DSS 71 compatibility test	Launch Complex 36B	Sep 28
AC-14/SC-6 Joint Flight Acceptance Composite Test (J-FACT)	Launch Complex 36B	Sep 29
SC-6 demate from *Centaur*	Launch Complex 36B	Sep 29
SC-6 decapsulation and demate from adapter	ESF	Sep 30
SC-6 initial alignment	ESF	Oct 4
SC-6 PVT-5 mission sequence	Building AO	Oct 9
SC-6 VPS high-pressure leak test	ESF	Oct 13
SC-6 VPS propellant loading and high-pressure leak test	ESF	Oct 17
AC-14 flight control and propellant tanking integrated test	Launch Complex 36B	Oct 21
SC-6 retromotor installation and final weight, balance, and alignment	ESF	Oct 22
AC-14 FACT (without spacecraft)	Launch Complex 36B	Oct 26
SC-6 PVT-6	ESF	Oct 30
SC-6 encapsulation and System Readiness Test (SRT)	ESF	Oct 31
AC-14 Composite Readiness Test (CRT)	Launch Complex 36B	Oct 31
SC-6 final mate to *Centaur* and SRT	Launch Complex 36B	Nov 1
SC-6 commutator assessment and final SRT	Launch Complex 36B	Nov 7
Surveyor VI launch	Launch Complex 36B	Nov 7

[a]*Surveyor VI* spacecraft designation.
[b]*Atlas/Centaur* vehicle designation.

September 25 and a *Centaur* programmer functional test in the armed configuration on September 26. The short, 18-day time span between the launch of *Surveyor V* and the first mating of *Surveyor VI* to AC-14 on September 27 overcame a severe schedule risk and represents a significant reduction in the planning estimates of launch pad turnaround time.

2. Flight Acceptance Composite Tests and Launch Vehicle Propellant Tanking Test

Following initial testing at the Spacecraft Checkout Facility (SCF) in Building AO, the spacecraft was moved, on September 21, to the Explosive Safe Facility (ESF), where it was prepared for the Joint Flight Acceptance Composite Test (J-FACT). The spacecraft was built up to flight configuration except for a dummy retromotor, nonflight shock absorbers, referee fluids instead of liquid propellants, leg and omni deploy squibs in mufflers, and no flight calibration sources in the sensor head of the alpha scattering instrument. After the spacecraft was mated to the forward adapter, the helium and nitrogen tanks were pressurized, the spacecraft was encapsulated, and an SRT was performed on September 26.

The spacecraft was then transported to Launch Pad 36B and mated to the *Centaur* vehicle on September 27, 1967. After mating, an RF air-link optimization test, an SRT, and a practice countdown were performed independent of the launch vehicle to verify proper operation of the spacecraft and GSE. There were no anomalies. An integrated launch control test to check out the space vehicle countdown sequence was also performed on September 27. On September 28, a compatibility test between the spacecraft and DSS 71, at Cape Kennedy, was performed with no significant problems.

The J-FACT was performed satisfactorily on September 29, with the spacecraft operating in the actual prelaunch environment. This test, involving all systems, covered launch through spacecraft separation and *Centaur* retromaneuver. Spacecraft/*Centaur* separation was simulated by manually disconnecting the adapter field joint electrical connector. The spacecraft temperature was maintained constant during the J-FACT period to permit accurate pressure decay measurements of the vernier propulsion and attitude control systems.

During the J-FACT, between $T - 120$ and $T - 60$ min in the countdown, a launch vehicle electromechanical interference (EMI) test was conducted. The purpose of this test is to verify spacecraft electrical compatibility with the launch complex and vehicle propellant storage and handling systems in lieu of a cryogenic tanking operation. No spacecraft or launch vehicle anomalies were observed during the J-FACT period. However, a minor problem occurred during the countdown when a procedural error in the operation of the Pad Safety Officer's console resulted in a 75-sec hold at $T - 60$ sec. The spacecraft was demated from the *Centaur* and returned to the ESF on September 29.

Following the J-FACT, several *Centaur* flight and GSE equipment components were replaced including: the rate gyro package, the liquid hydrogen (LH_2) and liquid oxygen (LO_2) vent valves, an engine propellant utilization relief valve, an engine yaw actuation rod end assembly, and an engine recirculation pump motor. Several transducers located on both stages and the *Atlas* fuel and oxidizer regulators were also replaced. Satisfactory operation of these components was confirmed by appropriate retest in time to support the *Atlas/Centaur* flight control and propellant tanking integrated test conducted on October 19.

Anomalies occurring during the tanking test included: a loss of one measurement of the *Centaur* insulation panel purge pressure caused by a faulty transducer, an erroneous *Centaur* LO_2 regulator pressure indication caused by miswired instrumentation (corrected during the test), and improper operation of the *Centaur* LH_2 probe which senses the fill level of the liquid in the tank. The LH_2 probe anomaly prevented a test of the LH_2 boiloff rate. Inspection of the launch vehicle following the detanking operation revealed a ruptured *Centaur* forward seal. Posttest analysis of the LH_2 probe problem indicated that the liquid-level sensing elements were miswired at the tank connector. The faulty purge pressure transducer and ruptured sections of the forward seal were replaced, and during a special *Centaur* tanking test conducted on October 21, the miswired condition of the LH_2 probe was confirmed. In addition, a flight level of LH_2 was obtained permitting performance of a LH_2 vent valve lock-up test and the LH_2 boiloff test. The LH_2 control panel and operating procedure was modified to conform to the LH_2 probe wiring, and no vehicle wiring changes were made for the final propellant tanking.

The launch vehicle was then prepared for a Flight Acceptance Composite Test, without the spacecraft, to confirm satisfactory operation of all *Atlas/Centaur* ground and flight electrical systems. Preparations for this test included the installation of a new *Atlas* Range Safety Command (RSC) arming device with diode suppression, replacement of the *Atlas* sustainer engine hydraulic package, and replacement of a *Centaur* nose fairing thruster bottle pressure transducer and a connector pin in the nose fairing disconnect. In addition, the *Atlas* booster staging disconnect lanyard was modified to ensure against failure of the lanyard swivel, and several special tests were conducted to establish acceptability of the *Centaur* rate gyro noise level and to ensure compatibility between the vehicle guidance system and the GSE. The FACT was conducted on October 26, 1967, with no procedure deviations being required or anomalous conditions encountered.

3. Final Flight Preparations

After return to the ESF following the J-FACT operations, the spacecraft was decapsulated, depressurized, and demated from the adapter. The flight helium and nitrogen pressure transducers were calibrated during depressurization. The nonflight items were removed, and initial spacecraft alignment (Fig. II-2) was completed on October 4.

The spacecraft was then transported back to Building AO, where a detailed mechanical inspection was performed, and a spacecraft/TCA functional test was repeated to check out a spare vernier engine. PVT-5, the final spacecraft system verification, was started on October 6 and completed on October 9. During this test, the position telemetry signal of the A/SPP solar axis showed variations during stepping. Special tests were performed which verified a poor wiper arm contact of the potentiometer at the launch position caused by vibration due to the stepping sequence. Since all steady-state indications were normal, the potentiometer was accepted for flight.

The RADVS SDC was removed from the spacecraft for installation of the flight cover and one of the component board assembly bolts was found broken. The bolt was replaced, and the remaining bolts were X-rayed to verify integrity. A fluctuation in the 7-V supply of the alpha scattering instrument electronics was noted during PVT-5. The unit was removed and sent to the University of Chicago, where the problem was corrected by changing two capacitors to provide stable operation. The unit was retested and reinstalled on the spacecraft after propellant loading. The alpha scattering sensor head was removed from the spacecraft during the same period and flight sources were installed. The unit was

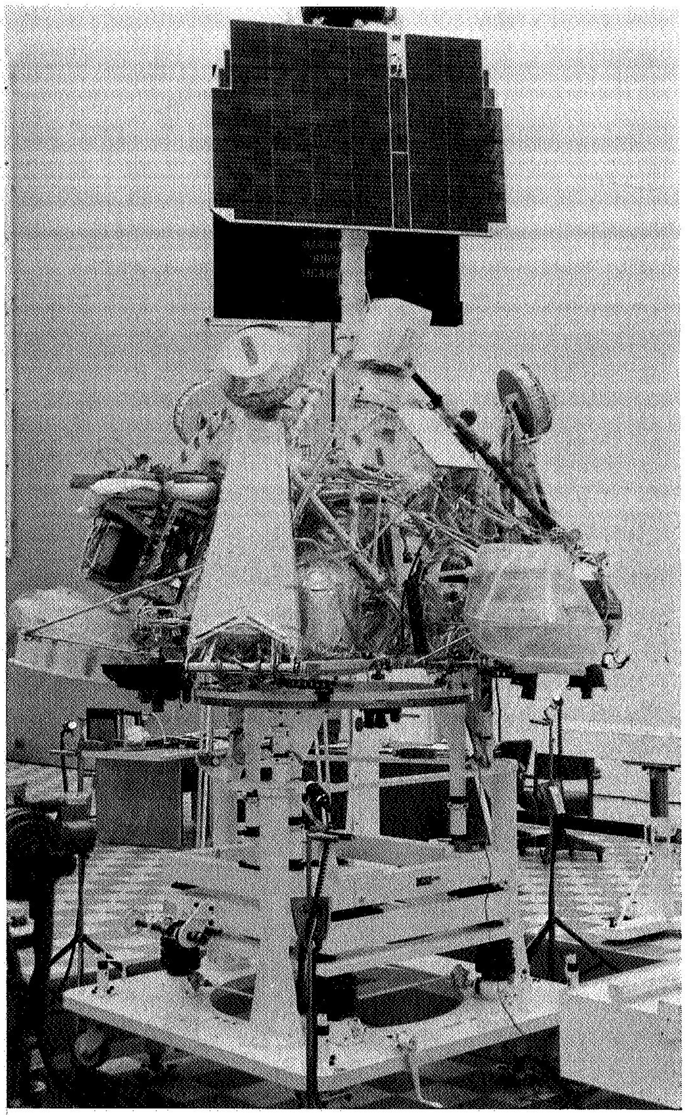


Fig. II-2. *Surveyor VI* spacecraft on alignment stand in Explosive Safe Facility

calibrated in the unit area and installed on the spacecraft just prior to final encapsulation.

On October 10, the spacecraft was moved to the Propellant Loading Building (PLB) of the ESF. The vernier propulsion system (VPS) was loaded with referee fluids, and a high-pressure leak test was performed. The VPS pressure transducers were calibrated during depressurizing. The referee fluids were then down-loaded and the system vacuum-dried. Propellant loading was started on October 16 and completed the next day. After loading, the pad pressure on the tanks was raised to 725 psi to check for leaks. No leaks were detected, and the pressure was reduced to 300 psi.

The spacecraft was then moved to the assembly building in the ESF. Final weight, balance, and alignment were performed, and the spacecraft was mated to the retromotor. PVT-6 was performed in parallel with the alignment operations, and the spacecraft was mated to the forward adapter on October 24, after which the flight shock absorbers and pyrotechnic devices were installed, the helium and nitrogen systems were pressurized for flight, and final inspection and cleaning were performed. The spacecraft was encapsulated within the nose fairing, and an SRT was completed on October 31. On the same day, the launch readiness of all *Atlas/Centaur* electrical and RF systems was verified by the successful performance of a Composite Readiness Test (CRT).

On November 1, the encapsulated spacecraft was transported to Launch Pad 36B, mated to the *Centaur*, and another spacecraft SRT was performed. Also on November 1, a leaking *Centaur* attitude control engine cluster was replaced along with a twisted *Atlas* pneumatic control flex hose. On the following day, with the service tower in the launch position (removed), the spacecraft RF system was calibrated, the retromotor safe-and-arm check was completed, and a simulated countdown was performed.

4. Countdown and Launch

The final spacecraft SRT began at 18:52 GMT on November 6 at a countdown time of $T - 680$ min. At approximately $T - 230$ min, a decay of the upper umbilical boom hydraulic pressure was observed and was traced to internal system leakage. The boom system hydraulic pump was used to maintain operating pressure for the remainder of the countdown.

The spacecraft joined the launch vehicle countdown at 05:22 GMT during the scheduled 60-min hold, which started at $T - 90$ min. The countdown proceeded normally down to the scheduled 10-min hold at $T - 5$ min. The 10-min hold was extended 17 min in order to delay opening of the window so that downrange tracking data coverage would be improved. The countdown was resumed at 07:34 GMT and proceeded smoothly through liftoff (Fig. II-3), which occurred at 07:39:01.075 GMT, November 7, 1967, on a flight azimuth of 82.995 deg.

The countdown included a total of 70 min of planned, built-in holds (the one of 60-min duration at $T - 90$ min and the one of 10-min duration at $T - 5$ min). As a result of the window opening being delayed by the 17-min

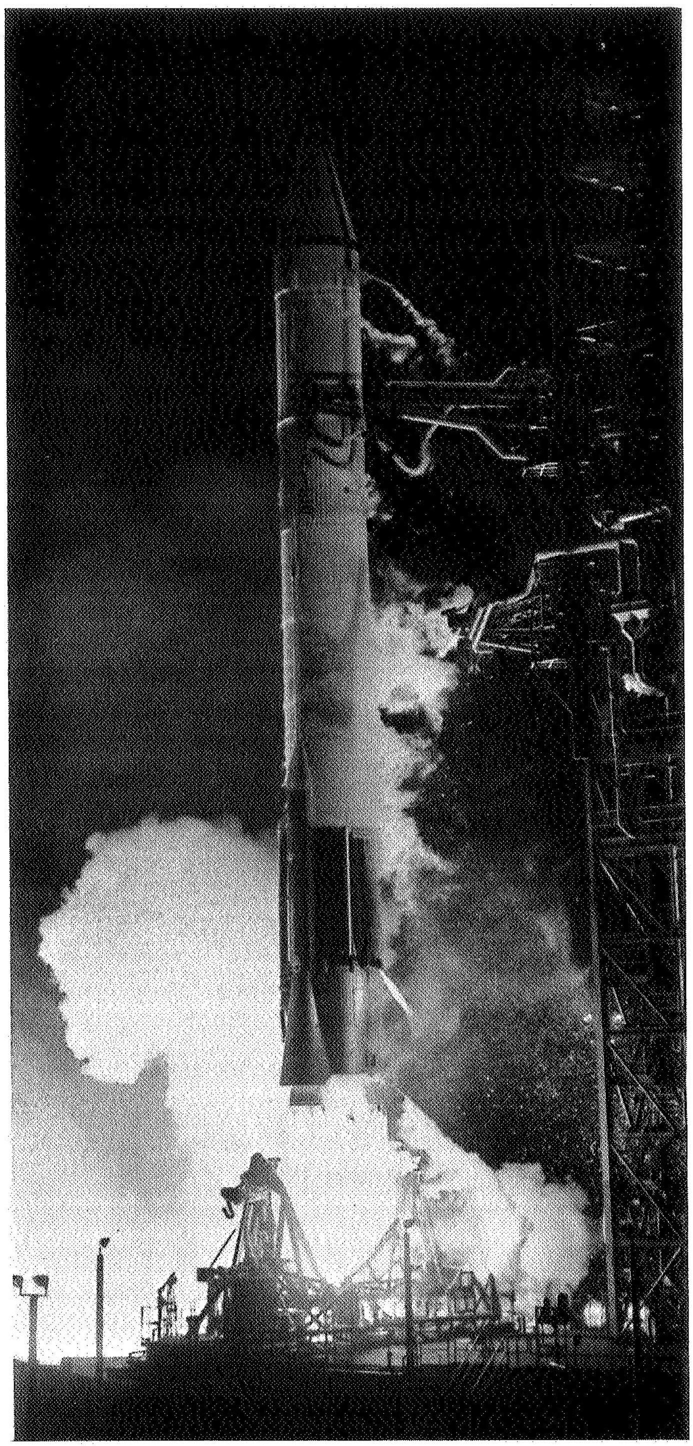


Fig. II-3. ***Atlas/Centaur* AC-14 launching *Surveyor VI***

extension of the $T - 5$ min hold, a 38-min launch window was available on November 7, extending from 07:39 to 08:17 GMT. A countdown time summary is shown in Table II-2.

Table II-2. ***Surveyor VI* countdown time summary**

Event	Countdown time, min	GMT
Started spacecraft SRT	$T - 680$	(Nov 6) 18:52
Started launch vehicle countdown	$T - 335$	(Nov 7) 00:37
Started 60-min built-in hold (BIH)	$T - 90$	04:42
Spacecraft joined launch vehicle countdown	$T - 90$	05:22
End BIH; resumed countdown	$T - 90$	05:42
Started 27-min BIH[a]	$T - 5$	07:07
Resumed countdown	$T - 5$	07:34
Liftoff	$T - 0$	07:39:01.075

[a]An option to extend the BIH an additional 17 min to ensure better downrange spacecraft telemetry data was exercised during the normal 10-min BIH at $T - 5$ min.

All vehicle systems performed satisfactorily throughout the powered phases of flight, and the spacecraft was accurately injected into a lunar transfer trajectory. The *Atlas* sustainer engine cutoff occurred about 3 sec early, while the *Centaur* first burn cutoff occurred approximately 3 sec later than expected. Both of these events were well within tolerance and had no detrimental effects on the mission. Damage to Launch Complex 36B was moderate. The powered flight sequence of events and launch vehicle performance are described in Section III.

The atmospheric conditions during the launch were acceptable with unusually good visibility. Surface winds were 11 knots from 320 deg. Surface temperature was 57°F, with a relative humidity of 69% and a dewpoint of 47°F. Sea level atmospheric pressure was 1019.3 millibars. The cloud cover was 8/10 cirrus above 20,000 ft. The maximum expected wind shear parameter was 10 ft/sec per thousand feet of altitude, occurring between 40,000 and 46,000 ft from 245 deg.

D. Launch Phase Mission Analysis

The *Surveyor VI* launch phase mission analysis activities during the prelaunch planning and countdown stages of the mission were devoted, for the most part, to launch constraint evaluations and launch window definitions. During the near-earth phase of the flight, the activities were centered about obtaining an early and continuing evaluation of the progress of the mission by monitoring information being gathered by each system of the Project.

1. Prelaunch Planning Activities

The initial *Surveyor VI* launch windows for the November 1967 opportunity were derived from the results of the trajectory targeting effort, and were constrained by the maximum allowable launch azimuth corridor as defined by Range Safety and the maximum design parking orbit coast time capability of the launch vehicle. These initial launch windows were considered as maximum windows and published in a launch constraints document for the *Surveyor VI* mission (see Bibliography). Also presented in this document are the preliminary best estimates of the launch window constraints due to Tracking and Data System (TDS) near-earth coverage limitations. When the final near-earth support plans and commitments of the TDS were made available, so that the near-earth coverage capabilities and limitations could be identified, it was possible to establish the final launch windows for the mission.

The final planned launch windows are illustrated in Fig. II-4. Except for closing of the window on the last day, these windows were constrained by TDS coverage limitations. Prior to the opening of the *Surveyor VI* launch period on November 7, the *Saturn 501* launch was rescheduled from that date to November 9, thus relieving but not totally eliminating the impact upon the support for the *Surveyor VI* flight. The AFETR Range Instrumentation Ship *Rose Knot* was committed to support the *Saturn* flight and, therefore, could not be in position to support *Surveyor*. As a result, the closing launch azimuth had to be established at 87.5 deg instead of 90 deg for November 7 in order to ensure coverage of critical launch vehicle events. The planned opening azimuth was 81 deg.

2. Countdown to Liftoff

The countdown on launch day (November 7) proceeded very smoothly except for communication problems with the downrange TDS stations, mostly due to poor RF propagation. However, all stations were in the green by launch time. During the countdown, it was decided to delay the window opening by extending the scheduled hold at $T - 5$ min to launch on a flight azimuth of 83 deg rather than 81 deg. This decision was made because of the expected improved spacecraft telemetry coverage from the AFETR stations for the

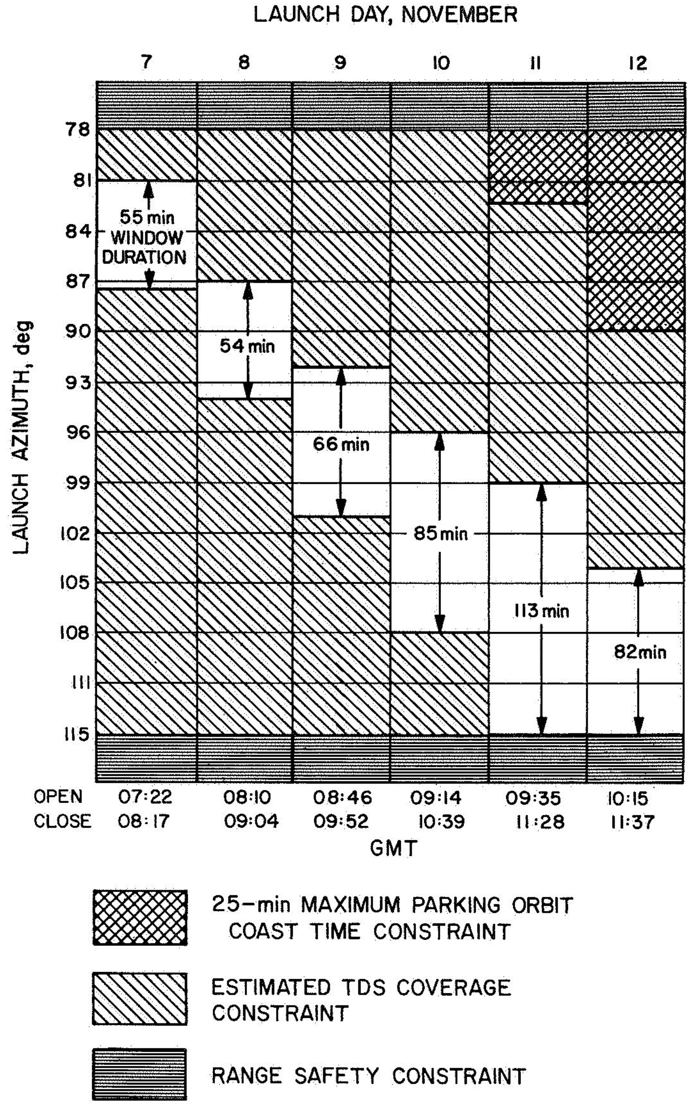


Fig. II-4. Final *Surveyor VI* launch window design for November 1967

83-deg azimuth. There were no unanticipated holds, and *Surveyor VI* liftoff occurred at the revised launch window opening.

3. Near-Earth Phase

The near-earth mission analysis was based on space vehicle Mark Events reported in near-real-time (see Section V, Table V-2), the reported acquisition characteristics of the TDS 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 Cape Kennedy.

The near-earth phase of the flight appeared to be completely nominal with the launch vehicle performing exceedingly well. From information available during the period from liftoff through parking orbit injection, the normalcy of that portion of the flight was readily established.

Although the launch vehicle telemetry data retransmitted in real-time for the *Centaur* second burn period was quite noisy, it did indicate that the second burn prestart and ignition events were nominal. It was not possible to perform an early evaluation of the transfer orbit because of the lack of expected near-real-time *Centaur* C-band tracking data from the *Twin Falls* tracking ship. However, DSS 51 did acquire the spacecraft soon after transfer orbit injection at the nominally expected acquisition time, and the RTCS computed two consistent transfer orbits based on the DSS 51 data. (Refer to Section VII for discussion of transfer orbit determinations.) These orbits indicated a very nominal transfer trajectory. The spacecraft telemetry data received in real-time confirmed that the spacecraft was performing normally.

III. Launch Vehicle System

Surveyor VI was launched very successfully by a General Dynamics/Convair *Atlas/Centaur* launch vehicle (AC-14). Liftoff occurred at 07:39:01.075 GMT on November 7, 1967, from Launch Complex 36B of AFETR at Cape Kennedy, Florida. Launch was via the indirect ascent mode wherein the *Centaur* second stage coasted about 12.9 min in a near-circular parking orbit of about 90 nm altitude before reigniting and thrusting a second time to provide the desired injection conditions. Without a midcourse correction, it is estimated that *Surveyor VI* would have impacted the moon only 126 km (78.3 statute miles) from the prelaunch target point.

This was the sixth operational flight of an *Atlas/Centaur* vehicle and the third to utilize the indirect ascent mode. The other flights utilized the direct-ascent mode, which requires only one "burn" of the second stage to achieve injection. Several design modifications were incorporated in the *Centaur* stage for parking orbit missions in order to achieve propellant control under the low gravity environment during the coast period and to ensure successful second ignition.

The *Surveyor VI* mission was the second *Atlas/Centaur* flight to use the SLV-3C *Atlas*, which has a 51-in.-longer propellant tank section and increased thrust compared with the LV-3C *Atlas* previously used. The SLV-3C *Atlas/Centaur* vehicle with the *Surveyor* spacecraft encapsulated in the nose fairing is 117 ft long and weighs about 325,000 lb at liftoff (2-in. rise). The basic diameter of the vehicle is 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 is illustrated in Fig. III-1. Both the *Atlas* first stage and *Centaur* second stage utilize thin-wall, pressurized, main propellant tank sections of monocoque construction to provide primary structural integrity and support for all vehicle systems. The first and second stages are joined by an interstage adapter section of conventional sheet and stringer design. The clamshell nose fairing is constructed of laminated fiberglass over a fiberglass honeycomb core and attaches to the forward end of the *Centaur* cylindrical tank section.

A. *Atlas* Stage

The first stage of the *Atlas/Centaur* vehicle is a modified version of the *Atlas* used on many previous Air Force and NASA missions such as *Ranger* and *Mariner*. The SLV-3C *Atlas* utilizes an uprated Rocketdyne MA-5 propulsion system, which burns RP-1 kerosene and liquid oxygen in each of its engines to provide a total liftoff thrust of approximately 395,000 lb. The individual

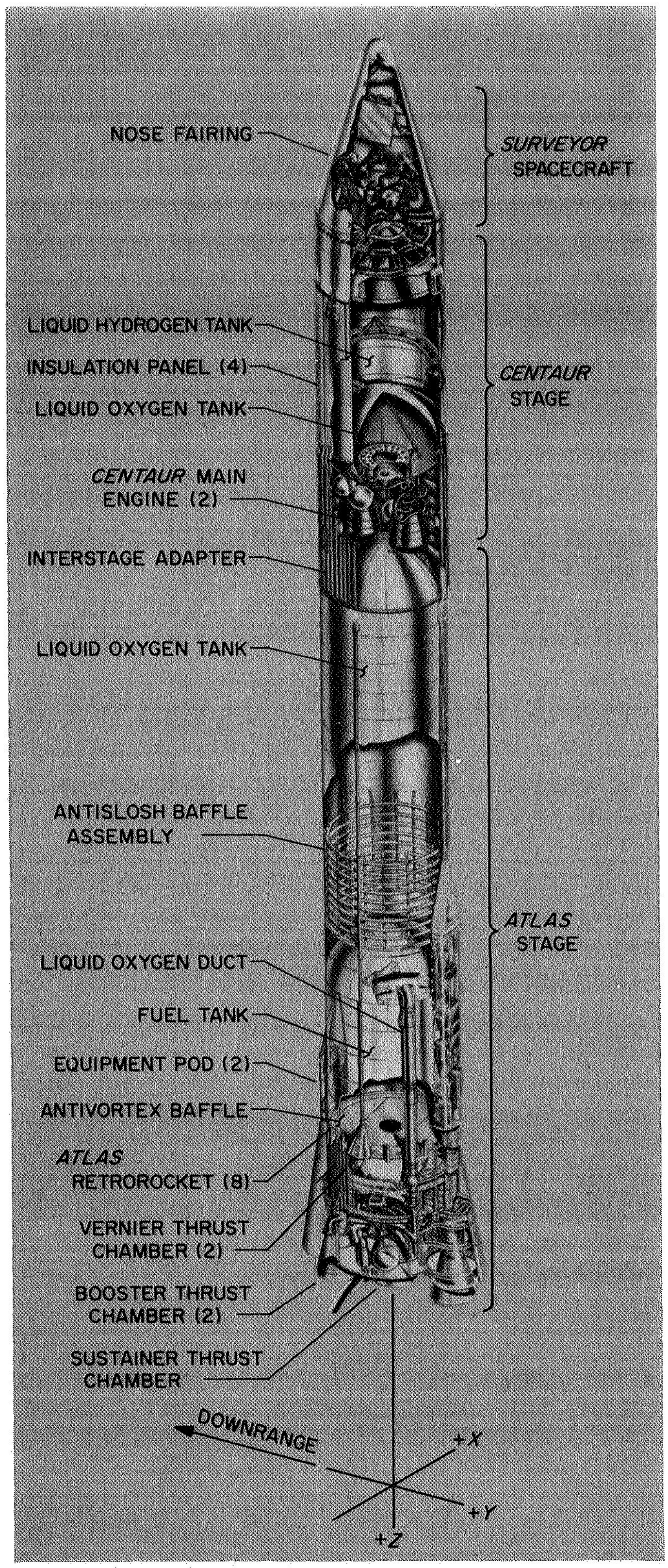


Fig. III-1. *Atlas* (SLV/3C)/*Centaur*/*Surveyor* space vehicle configuration

sea-level thrust ratings of the engines are: two booster thrust chambers at 168,000 lb each, one sustainer engine at 58,000 lb, and two vernier engines at 670 lb each. The 51-in. extension of the SLV-3C *Atlas* tank section permits tanking of approximately 20,000 lb additional propellants.

The *Atlas* can be considered a 1½-stage vehicle because the "booster section," weighing 6000 lb and consisting of the two booster engines together with the booster turbopumps and other equipment located in the aft section, is jettisoned after about 2.6 min of flight. The sustainer and vernier engines continue to burn until propellant depletion. A mercury manometer propellant utilization system is used to control mixture ratio for the purpose of minimizing propellant residuals at *Atlas* sustainer engine cutoff.

Flight control of the first stage is accomplished by the *Atlas* autopilot, which contains 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 also is fed steering commands from the all-inertial guidance set located in the *Centaur* stage. Vehicle attitude and steering control are achieved by the coordinated gimballing of the five thrust chambers in response to autopilot signals.

The AC-14 *Atlas* contained two solid-state telemetry packages for transmission of 201 *Atlas* measurements over two VHF links (229.9 and 232.4 MHz). Both telemetry links utilize each of two antennas 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 are employed on the *Atlas* to provide the Range Safety Officer with means of terminating the flight by initiating engine cutoff, and/or destroying the vehicle.

B. *Centaur* Stage

The *Centaur* second stage is the first vehicle to utilize liquid hydrogen/liquid oxygen, high-specific-impulse propellants. The cryogenic propellants require 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 is 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 are mounted on the forward tank bulkhead in a compartment which is air-conditioned prior to liftoff.

The *Centaur* is powered by two Pratt & Whitney constant-thrust engines rated at 15,000 lb thrust each in vacuum. Each engine can be gimballed to provide control in pitch, yaw, and roll. Propellant is fed from each of the tanks to the engines by boost pumps driven by hydrogen peroxide turbines. In addition, each engine contains integral "boot-strap" turbopumps driven by hydrogen propellant. Hydrogen propellant is also used for regenerative cooling of the thrust chambers. The AC-14 *Centaur* stage utilized RL10A-3-3 main engines, which were improved over those used on direct ascent flights to provide for operation at lower NPSH (net positive suction head) and an increased specific impulse nominal rating of 444 sec.

A propellant utilization system is used on the *Centaur* stage to achieve minimum residual of one propellant at depletion of the other. The system controls 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 nominal oxygen/hydrogen mixture ratio is 5:1 by weight. Special design features were incorporated in the hydrogen tank design for parking orbit missions to insure propellant control during the coast phase. These include (1) an antiswirl/antislosh baffle located at the hydrogen level at the end of the 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 utilizes a Minneapolis-Honeywell all-inertial guidance system containing a navigation computer which provides vehicle steering commands after jettison of the *Atlas* booster section. For wind shear relief during the *Atlas* booster phase, the navigation computer may also be used to generate incremental pitch and yaw corrective signals to supplement the *Atlas* fixed pitch and yaw program. The incremental pitch and yaw programs are selected and fed into the navigational computer prior to launch based on measured wind conditions. The *Centaur* guidance signals are fed to the *Atlas* autopilot until *Atlas* sustainer engine cutoff and to the *Centaur* autopilot after *Centaur* main engine ignition. During flight, platform gyro drifts are compensated for analytically by the guidance system computer rather than by applying corrective gyro torquing signals.

The *Centaur* autopilot system provides 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 employs an electromechanical timer to control the sequence of programmed events during the *Centaur* phase of flight, including a series of commands required to be sent to the spacecraft prior to spacecraft separation. A dual-timer configuration is used to provide for the additional programmed events required on a parking orbit mission.

The *Centaur* reaction control system provides 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 utilizes six individually controlled, fixed-axes, constant-thrust, hydrogen peroxide reaction engines. These engines are 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 contains 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 are installed on the aft bulkhead, with thrust axes parallel to the vehicle axis.* These engines are 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 if necessary.

The *Centaur* stage utilizes a VHF telemetry system with a single antenna transmitting through the nose fairing cylindrical section on a frequency of 225.7 MHz. The telemetry system provides 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. On the AC-14 flight, 161 measurements were transmitted by the *Centaur* telemetry system.

Redundant range safety command receivers are employed on the *Centaur*, together with shaped-charge destruct units for the second stage and spacecraft. This provides the Range Safety Officer with means to terminate the flight by initiating *Centaur* main engine cutoff and destroying the vehicle and spacecraft retrorocket.

*The 3-lb-thrust engines were not installed for the direct-ascent missions.

The system can be safed by ground command, which is normally transmitted by the Range Safety Officer when the vehicle has reached orbital energy.

A C-band tracking system is contained on the *Centaur* which includes a lightweight transponder, circulator, power divider, and two antennas located under the insulation panels. The C-band radar transponder provides 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.

C. Launch Vehicle/Spacecraft Interface

The general arrangement of the *Surveyor/Centaur* interface is illustrated in Fig. III-2. The spacecraft is 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. This encapsulation provides protection for the spacecraft from the environment before launch as well as from aerodynamic loads and heating during ascent. An ablative-type coating (Thermolag) is applied over the nose fairing, *Centaur* insulation panels and interstage adapter to provide added thermal protection.

The spacecraft is 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 pin puller, The following equipment is located on the forward adapter: three separation spring assemblies each containing a linear potentiometer for monitoring separation; a 52-pin

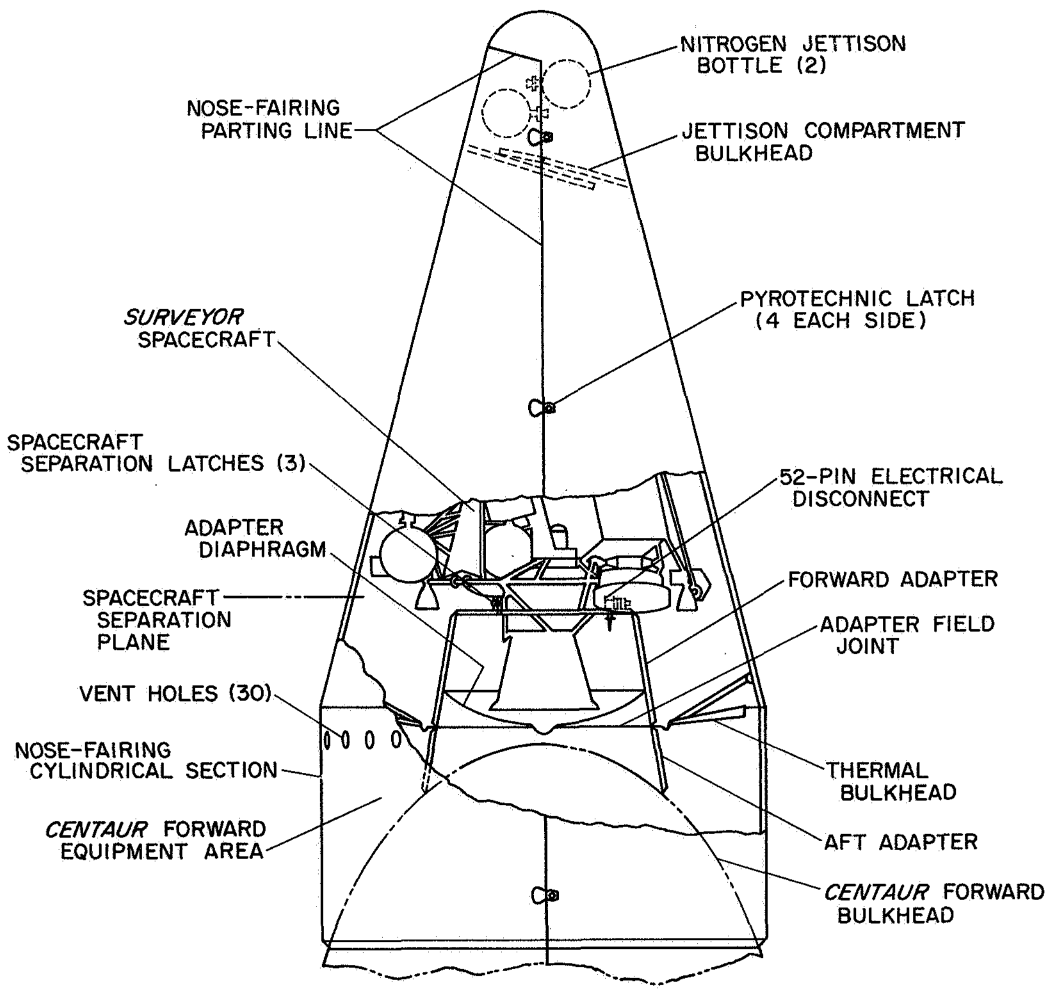


Fig. III-2. *Surveyor/Centaur* interface configuration

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 retromotor; a diaphragm to provide a thermal seal and to prevent contamination from passing to the spacecraft compartment from the *Centaur* forward equipment compartment; and accelerometers for monitoring vibration at the separation plane. As on the previous *Surveyor* mission (AC-13), two high-frequency accelerometers were located on the *Centaur* 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 spacecraft Leg 1 attach point; the other was mounted in the longitudinal direction near Leg 3 attach point. The outputs of both accelerometers were telemetered continuously. On flights prior to AC-13, one accelerometer had been mounted differently in a radial direction on the adapter, and four accelerometers had been installed on the spacecraft side of the separation plane.

The low-drag nose fairing is an RF-transparent, clamshell configuration consisting of four sections fabricated of laminated fiberglass cloth faces and honeycomb fiberglass core material. Two half-cone forward sections are brought together over the spacecraft mounted on the forward adapter. An annular thermal bulkhead between the adapter and base of the conical section completes encapsulation of the spacecraft.

The encapsulated assembly is 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 is bolted to the cylindrical portion of the fairing, the two halves of which are attached to the forward end of the *Centaur* tank around the equipment compartment prior to mating of the spacecraft. Doors in the cylindrical sections provide access to the adapter field joint. The electrical leads from the forward adapter are carried through three field connectors and routed across the aft adapter to the *Centaur* umbilical connectors and to the *Centaur* programmer and telemetry units.

Special distribution ducts are built into the nose fairing and forward adapter to provide air conditioning of the spacecraft cavity after encapsulation and until liftoff. Seals are provided at the joints to prevent shroud leakage except out through vent holes in the cylindrical section. Prior to launch, the shroud cavity is monitored for possible spacecraft propellant leakage by means of a toxic gas detector tube which disconnects at liftoff. Tubes are 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 is rolled away. The spacecraft alpha scattering instrument is also purged by means of a tube which disconnects at liftoff.

The entire nose fairing is designed to be ejected by separation of two clamshell pieces, each consisting of a conical and cylindrical section. Four pyrotechnic pin-puller latches are used on each side of the nose fairing to carry the tension loads between the fairing halves. Nose fairing loads are transmitted to the *Centaur* tank through a bolted joint, which also attaches to the forward end of the *Centaur* insulation panels and contains a flexible linear shaped charge for insulation panel and nose fairing separation. A nitrogen bottle is 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 are located at the base of each fairing half to control ejection, which occurs under vehicle acceleration of approximately 1 g during the *Atlas* sustainer phase of flight.

D. Vehicle Flight Sequence of Events

All vehicle flight events occurred satisfactorily. As has occurred on all previous *Surveyor* missions, the *Centaur* burned longer than expected but well within allowable limits. Predicted and actual times for the vehicle flight sequences of events are included in Table A-1 of Appendix A. Figure III-3 illustrates the major nominal events. The times reported in near-real-time for Mark Events are listed in Section V, Table V-2. Following is a brief description of the vehicle flight sequence of events, with all times referenced to liftoff (2-in. rise) unless otherwise noted. (Refer to Section II-B for a description of the countdown.)

1. *Atlas* Booster Phase of Flight

Hypergolic ignition of all five *Atlas* engines was initiated 2 sec before liftoff. Vehicle liftoff occurred at 07:39:01.075 GMT on November 7, 1967, only 1.075 sec after the revised window opening time on the first day of the launch period. (The window opening had been delayed during the countdown to improve downrange spacecraft telemetry data coverage.) The launcher mechanism is designed to begin a controlled release of the

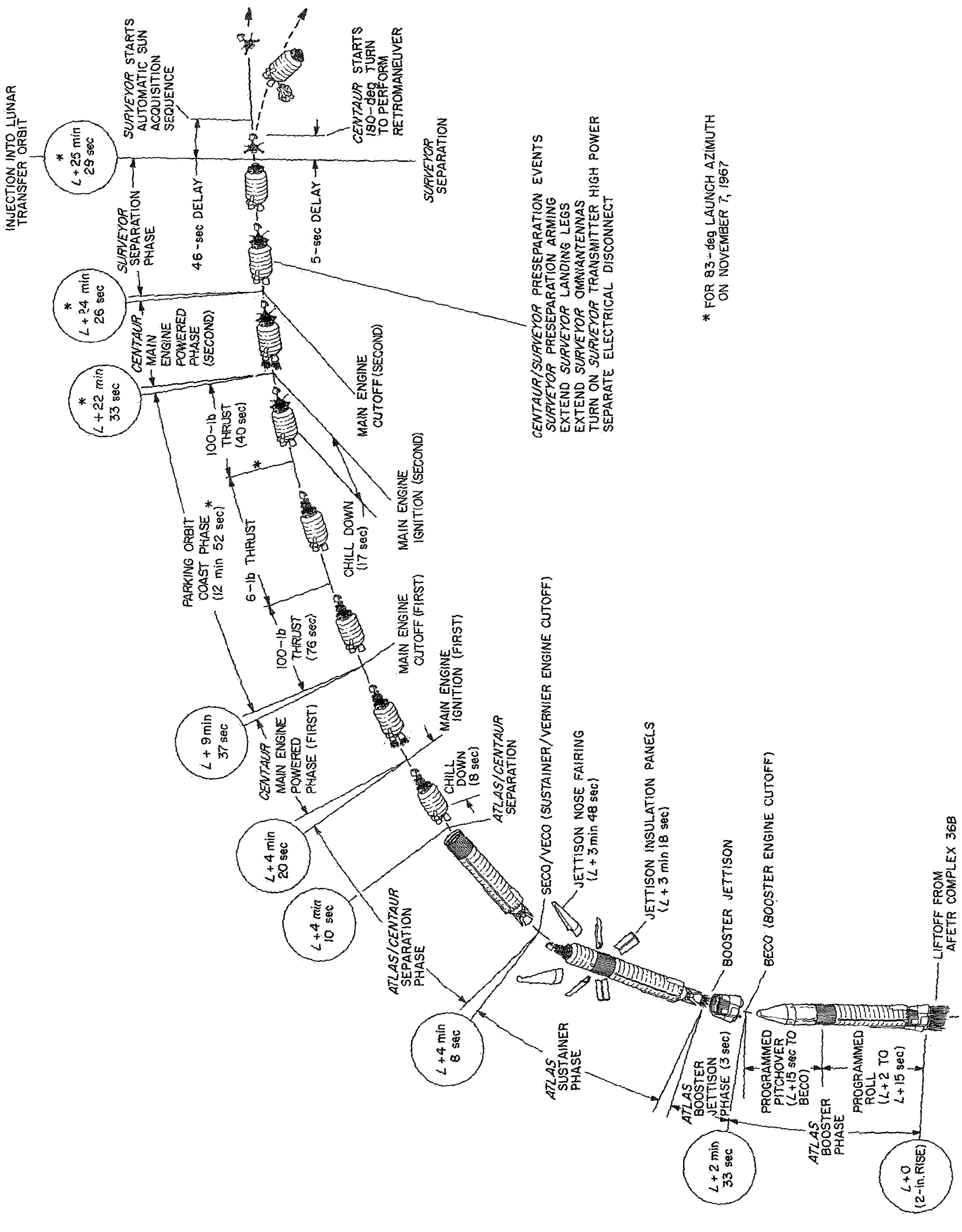


Fig. III-3. *Surveyor VI* launch phase nominal events

vehicle when all engines have reached nearly full thrust. At 2 sec after liftoff, the vehicle began a 13-sec programmed roll from the fixed launcher azimuth setting of 115 deg to a planned launch azimuth of 82.995 deg. The programmed pitchover of the vehicle began 15 sec after liftoff and lasted until booster engine cutoff (BECO), by which time the vehicle had pitched over 71 deg. Incremental yaw program (Code 6) signals from the *Centaur* navigational computer were used. However, no incremental pitch program was required. The yaw program had been selected on the basis of prelaunch wind analysis.

The vehicle reached Mach 1 at 64 sec and maximum aerodynamic loading occurred at 82 sec. During the booster phase of flight, the booster engines were gimballed for pitch, yaw, and roll control, and the vernier engines were active in roll control only, while the sustainer engine was centered.

At 153.3 sec, BECO was initiated by a signal from the *Centaur* guidance system when vehicle acceleration equalled 5.72 *g* (expected value: 5.7±0.08 *g*). At 3 sec after BECO, with the booster and sustainer engines centered, the booster section was jettisoned by release of pneumatically operated latches. Events beginning with BECO occur later on SLV-3C *Atlas* flights because of the increased propellant tank capacity.

2. *Atlas* Sustainer Phase of Flight

At BECO + 8 sec, the *Centaur* guidance system was enabled to provide steering commands for the *Atlas* sustainer phase of flight. During this phase the sustainer engine was gimballed for pitch and yaw control, while the verniers were active in roll. *Centaur* guidance commanded an additional 6 deg of pitchover, resulting in a total vehicle pitchover of about 77 deg by the end of the sustainer phase. The *Centaur* insulation panels were jettisoned by firing shaped charges at 198.0 sec at an altitude of approximately 52 nm, where the aerodynamic heating rate was rapidly decreasing. At 227.3 sec, squibs were fired to unlatch the clamshell nose fairing, which was jettisoned 0.5 sec later by means of nitrogen gas thruster jets activated by pyrotechnic valves.

Other programmed events which occurred during the sustainer phase of flight were (1) the 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 prior to *Centaur* first main engine start (MES 1), and (3) locking of the *Centaur* oxidizer tank vent valve followed by two "burp" pressurizations of the tank.

Sustainer and vernier engine cutoff (SECO and VECO) occurred at 245.9 sec as a result of oxidizer depletion, which was the predicted cutoff mode. Shutdown began with an exponential thrust decay phase of about 1-sec duration due to low oxidizer inlet pressure to the turbopump and resulting loss in turbopump performance. Then, final fast shutdown by propellant valve closure was initiated by actuation of a switch when fuel manifold pressure dropped to 650±50 psi. Also, at the SECO event, the *Centaur* hydrogen tank vent valve was locked and burp pressurization of the tank was begun.

Separation of the *Atlas* from the *Centaur* occurred 1.9 sec after SECO by firing of shaped charges at the forward end of the interstage adapter. This was followed by ignition of eight retrorockets located at the aft end of the *Atlas* tank section to back the *Atlas*, together with the interstage adapter, away from the *Centaur*.

3. *Centaur* First Burn Phase of Flight

The *Centaur* prestart sequence for providing chilldown of the propulsion system was initiated 8 sec before first ignition of the *Centaur* main engines (MES 1). MES 1 was commanded 11.4 sec after SECO, at 257.3 sec. *Centaur* guidance was reenabled 4 sec after MES 1 to provide steering commands during the *Centaur* first burn. The guidance system continued to command vehicle pitchover and, by the end of the first burn, total vehicle pitchover was about 100 deg from the liftoff vertical. Main engine cutoff (MECO 1) was commanded by guidance at 581.2 sec, when sufficient impulse had been delivered for injection into the desired parking orbit. *Centaur* first-burn duration was 323.9 sec, or about 7 sec longer than predicted for the actual launch conditions.

4. *Centaur* Coast Phase of Flight

Coincident with MECO 1, two of the 50-lb-thrust hydrogen peroxide engines were turned on and provided a low level of axial acceleration to overcome transient disturbances to the propellants caused by main engine shutdown. After 76 sec, the 50-lb engines were turned off and two of the 3-lb axial engines were turned on to retain the propellants at the proper location in the tanks. During the parking orbit coast period, the hydrogen peroxide engines also were enabled for attitude control, and pitchover of the vehicle was continued to maintain alignment of the vehicle longitudinal axis approximately with the parking orbit flight path.

The required parking orbit coast time varies with actual liftoff time. For this flight, the coast period (MECO 1 to MES 2) lasted for 771.7 sec (12.9 min). The *Centaur* stage can support coast periods of from 116 sec to 25 min in duration. Initial occurrence of hydrogen tank venting during the coast period was observed 558 sec after MECO 1. Intermittent venting continued until vent valve closure at the start of pressurization for second burn.

At 40 sec before the end of the coast period, the 3-lb engines were turned off and two of the 50-lb engines turned on again until MES 2 to ensure propellant control during the events preceding ignition, which included "burp" pressurization of the propellant tanks, starting of the boost pumps 28 sec before MES 2, and initiation of the prestart (chilldown) sequence 17 sec before MES 2.

5. *Centaur* Second Burn Phase of Flight Through Spacecraft Separation

Second main engine start occurred at 1352.9 sec, followed 4 sec later by *guidance enable* for second burn steering control. After a burn time of 115.7 sec, when sufficient velocity had been attained, the *Centaur* engines were shut down by guidance commanded at 1468.6 sec. *Centaur* second burn duration was about 2.7 sec longer than expected. At main engine cutoff, the hydrogen peroxide engines were enabled again for attitude stabilization.

During the 60.3-sec period between MECO 2 and spacecraft separation, the following signals were transmitted to the spacecraft from the *Centaur* programmer: *extend spacecraft landing gear, unlock spacecraft omniantennas,* and *turn on spacecraft transmitter high power.* An arming signal also was provided by the *Centaur* during this period to enable the spacecraft to act on the preseparation commands.

The *Centaur* commanded separation of the spacecraft electrical disconnect 5.5 sec before spacecraft separation, which was initiated at 1528.9 sec. The *Centaur* attitude-control engines were disabled for 5 sec during spacecraft separation in order to minimize vehicle turning moments.

6. *Centaur* Retromaneuver Phase of Flight

At 5 sec after spacecraft separation, the *Centaur* began a turnaround maneuver 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, which required approximately 110 sec to complete, two of the 50-lb-thrust hydrogen peroxide engines were fired for a period of 20 sec while the *Centaur* continued the turn. This provided initial lateral separation of the *Centaur* from the spacecraft. About 240 sec after spacecraft separation, the propellant blowdown phase of the *Centaur* retromaneuver was initiated by opening the hydrogen and oxygen prestart (chilldown) valves. Oxygen was vented through the engine nozzles while hydrogen discharged directly through the chilldown valves. The oxygen tank pressure remained relatively constant, indicating that liquid oxygen remained in the tank throughout the blowdown. At 117 sec from start of blowdown, the fuel tank pressure decay rate increased, indicating that most of the liquid hydrogen had been expelled at that time.

Coincident with termination of propellant blowdown, a hydrogen peroxide depletion experiment was initiated by firing two of the 50-lb engines. Hydrogen peroxide depletion was indicated about 83 sec after start of the experiment. After 100 sec, the hydrogen peroxide experiment was concluded by energizing the *Centaur* power changeover switch at 2118.9 sec, which turned off all power except telemetry and the C-band beacon.

E. Performance

The *Atlas/Centaur* AC-14 vehicle performance was completely satisfactory, providing injection into the desired parking orbit followed by successful restart and very accurate injection of the *Surveyor VI* spacecraft into the prescribed lunar transfer trajectory.

1. Guidance and Flight Control

Autopilot performance was satisfactory throughout the flight, with proper initiation of programmed events and control of vehicle stability. Vehicle disturbances were at or below expected levels and similar to the previous flight of AC-13. During the *Atlas* phase of flight, the vehicle was easily controlled after *Atlas* autopilot activation at 42-in. motion. Vehicle stability was also satisfactorily maintained during the *Centaur* phase of flight.

The guidance system performed well throughout the flight. A near-circular parking orbit was achieved, having an apogee of 89 nm, a perigee of 85 nm, and a period of about 87.7 min. The spacecraft was injected on a lunar transfer trajectory which would have resulted in an uncorrected impact only 126 km from the prelaunch target point. (Refer to Section VII for a presentation of vehicle guidance accuracy results in terms of equivalent midcourse velocity correction.)

The guidance system provided the programmed incremental yaw signals during booster phase, and all guidance system discrete commands, including BECO, SECO backup, MECO 1, and MECO 2, were generated as planned. Each time guidance was enabled, the initial attitude errors (maximum 5 deg nose right and 1 deg nose up at BECO + 8 sec) were quickly nulled, after which vehicle attitude errors remained small during each of the respective closed-loop steering phases of flight.

The *Centaur* reaction control system performed properly, providing required vehicle attitude control during the coast and retromaneuver phases, the necessary low-level axial thrust for propellant management while in parking orbit, and initial lateral separation from the spacecraft after injection. Disturbing torques occurred as expected owing to boost pump exhaust, main engine chilldown venting, and exhaust impingement from the axially mounted hydrogen peroxide engines. During the *Centaur* lateral thrust period of this flight, valuable data was obtained on hydrogen peroxide engine thrust impingement forces. At approximately 5 sec after the start of *Centaur* retro turnaround, a momentary loss of thrust from one of the 3.5-lb attitude control engines was indicated. This anomaly has been noted on previous missions at about this time, and is attributed to gas bubbles in the hydrogen peroxide feed system.

During the *Centaur* phase of flight, the vehicle is rate-stabilized in roll rather than roll-position-stabilized. Vehicle roll attitude during the *Centaur* powered phase of flight is presented in Fig. III-4.

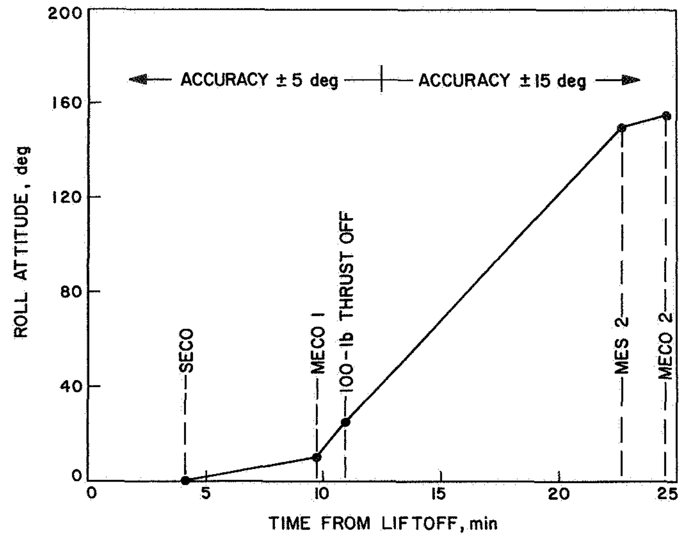


Fig. III-4. *Centaur* clockwise roll attitude relative to local vertical

2. Propulsion and Propellant Utilization

Atlas propulsion system performance was very satisfactory. Normal sustainer cutoff characteristics were exhibited following oxidizer depletion, which had been predicted. A new seal was used for the first time on this *Atlas* at the sustainer engine liquid oxygen elbow-to-dome interface to minimize liquid oxygen leakage. Post-flight studies and Rocketdyne testing had suggested this seal as a source of possible leakage on previous *Atlas* flights and static tests. There were no indications of cryogenic leakage during this flight.

Performance of the *Atlas* propellant utilization (PU) system was also satisfactory. The predicted *Atlas* residuals (propellant above pump) are compared in Table III-1, with preliminary values computed from flight data assuming nominal flow rates from port uncovering times until actual SECO.

The *Centaur* propulsion system performed well during both burn periods. Both burns were longer than predicted (by about 6.9 and 2.7 sec, respectively) but were well within the estimated 3σ tolerances. The cause of the extra burn time is not known at this time. However, the *Centaur* has also burned for periods longer than expected on each of the previous *Surveyor* missions.

The turbine inlet pressure and speed data of the fuel and oxidizer boost pumps exhibited unexpected trends after MES 2 which are indicative of gas flow through the hydrogen peroxide catalyst beds. A similar anomaly was observed during the second burns of AC-12 (*Surveyor III*) and AC-13 (*Surveyor V*), but there has been no apparent effect on main engine performance.

The *Centaur* PU system also performed very well during both burn periods. The predicted and preliminary

Table III-1. *Atlas* propellant residuals, lb

Residuals	Actual	Predicted
Oxidizer	277	256
Fuel	197	199

Table III-2. *Centaur* usable propellant residuals, lb

Usable residuals	Actual	Predicted
MECO 1		
Oxidizer	7382	7250
Fuel	1588	1450
MECO 2		
Oxidizer	651	548
Fuel	170	130

actual *Centaur* usable residuals after MECO 1 and MECO 2 are compared in Table III-2.

Based on a nominal mixture ratio of 5.0:1, the usable residuals would have provided 11.3 sec additional burn time before theoretical oxidizer depletion, with an ultimate fuel residual of approximately 40 lb. Comparing this to the predicted value of 20 lb residual hydrogen indicates a *Centaur* PU system error of 20 lb excess hydrogen.

3. Pneumatic, Hydraulic, and Electrical Power Systems

Operation of the *Atlas* pneumatic system, including the programmed tank pressurization and pneumatic control functions, was properly accomplished throughout the flight. Both *Centaur* propellant tanks were maintained at satisfactory levels during all phases of flight, with normal occurrences of the "burp" pressurization sequence and hydrogen tank venting. The outlet pressure of the control regulator for the *Centaur* engines indicated a gradual increase after MECO 2 followed by a sudden decay to the normal operating level at the start of lateral thrusting. This type of behavior was also observed on previous flights and is attributed to differences in bleed rates between the engines' control and hydrogen peroxide supply regulators.

Special instrumentation added for the SLV-3C flights indicated satisfactory *Atlas* hydraulic system operation similar to the previous *Surveyor* mission. *Centaur* hydraulic system performance was also satisfactory, although the hydraulic pressure decay characteristics for both engines after MECO 1 and MECO 2 were somewhat different from previous flights.

Performance of the vehicle electrical power systems was satisfactory throughout the flight. There were no unexpected voltage demands or transients.

4. Telemetry, Tracking, and Range Safety Command

The *Atlas* and *Centaur* instrumentation and telemetry systems functioned well, with only a few measurement anomalies. Continuous data was obtained over all vehicle flight phases except for a 5-min period during parking orbit coast. Among the measurement anomalies, the oxidizer boost pump inlet pressure transducer responded very slowly to pressure changes at boost pump start before MES 2 and at MECO 2. This anomaly also occurred on the AC-13 flight and may be caused by the freezing of moisture in the transducer line during the parking orbit coast period.

The *Centaur* C-band radar apparently operated normally, although an evaluation of the system can only be made on the basis of received tracking data and station operator logs because the airborne system is not instrumented. Refer to Section V for tracking data coverage.

The *Atlas* and *Centaur* range safety command systems performed satisfactorily. About 16 sec after parking orbit injection (MECO 1), a range safety command to disable the destruct systems was sent and properly executed.

5. Vehicle Loads and Environment

Vehicle loads and thermal environment were within expected ranges throughout the flight. Maximum axial accelerations were 5.72 *g* at BECO during the booster phase and 1.79 *g* just before SECO during the sustainer phase. Longitudinal oscillations during launch were normal, reaching a maximum (0.56 *g* peak-to-peak at 6 Hz) at 0.5 sec and being damped out by 20 sec. Characteristic of the longer SLV-3C *Atlas*, higher-amplitude bending was noted throughout the booster and sustainer phases of flight.

The two high-frequency accelerometers located on the forward spacecraft adapter indicated expected steady-state vibration levels during the flight which agreed well with similar measurements of previous flights. The radially oriented accelerometer indicated a maximum steady-state value during launch of 2.1 *g* (rms) within a 10- to 660-Hz bandwidth. The maximum longitudinal steady-state level indicated during launch was 1.3 *g* (rms) within a 10- to 2100-Hz bandwidth. Low-level, short-duration transients were recorded throughout the launch phase as has been noted on previous flights. The exact cause of these transients is not known, but they are believed to be due to a combination of dynamic and thermal loads in the area of the *Centaur* forward bulkhead and adapter structure. (Also see Section IV-A for a discussion of launch phase vibration environment.)

The *Surveyor* compartment thermal and pressure environments were normal throughout flight. The ambient temperature within the compartment was 81°F at liftoff and gradually decreased to 60°F by $L + 88$ sec as a result of expansion during ascent. The ambient pressure decayed characteristically to essentially zero prior to nose fairing jettison.

6. Separation and Retromaneuver Systems

All vehicle separation systems functioned normally. Booster section jettison occurred as planned, with resulting vehicle rates and high-frequency accelerometer data comparable to previous 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 are 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 from 76 to 84 deg/sec. These values are consistent with rates determined on previous flights.

Normal separation of the nose fairing was verified by indications of 3-deg rotation from disconnect wires which are incorporated in the pullaway electrical connectors of each fairing half. The spacecraft compartment pressure remained at zero throughout nose fairing jettison, with no pressure surge at thruster bottle discharge.

Atlas/Centaur separation occurred as planned. Displacement data obtained with respect to time is in close agreement with expected values and indicates successful *Atlas* retro rocket operation. There was no indication of contact between stages as the *Atlas* cleared the *Centaur.*

At spacecraft separation, data from the extensometers (linear potentiometers) indicates that first motion of all three springs occurred within about 1 msec of each other. Springs extensions to the full 1-in. position were normal and nearly identical, producing a spacecraft separation rate of approximately 1 ft/sec. The spacecraft angular rates resulting from the separation event were small and well within the specified maximum acceptable rate of 3.0 deg/sec. (Refer to Section IV-F for a discussion of angular separation rates as determined from spacecraft gyro data.)

All phases of the *Centaur* retromaneuver were executed as planned. Five hours after spacecraft separation, the *Centaur/Surveyor* separation distance was computed to be about 1650 km, which is well in excess of the required minimum distance of 336 km at that time. The *Centaur* closest approach to the moon was computed to be approximately 28,000 km and occurred at about 15:00 GMT on November 10, 1967.

IV. *Surveyor* Spacecraft

The basic objectives of the *Surveyor VI* spacecraft system were: (1) to accomplish a soft landing near the center of the moon at a new site within the *Apollo* zone of interest; (2) to obtain postlanding television pictures of the lunar surface; (3) to determine the relative abundance of chemical elements at a second lunar mare site using the alpha scattering instrument; (4) to obtain data on radar reflectivity, thermal characteristics, touchdown dynamics, and other measurements of the lunar surface through use of various spacecraft equipment; and (5) to conduct a lunar surface erosion experiment by firing the vernier engines.

Surveyor VI met all of its objectives. Liftoff occurred at 07:39:01.075 GMT on November 7, 1967. After an accurate injection by the *Atlas/Centaur* AC-14 launch vehicle, the spacecraft automatically acquired the sun to begin an almost flawless transit phase. Canopus was acquired about 9 hr after liftoff, and a 10.06-m/sec mircourse correction was performed during the first Goldstone pass. After a close-to-nominal terminal descent, the spacecraft successfully soft-landed, with touchdown occurring at 01:01:05 GMT on November 10, 1967. The landing site is 0.47°N latitude and 1.48°W longitude (based on correlation of *Surveyor* and *Lunar Orbiter* photographs).

After landing, the spacecraft performed very extensive operations, including gathering of an abundance of useful alpha scattering data, taking of a wide variety of high-quality television pictures (about 30,000 received before camera shutdown after sunset), and other scientific and engineering tests and experiments. One of these experiments was the successful performance of a hop in which the spacecraft rose approximately 14 ft above the lunar surface while traveling over the lunar surface to a new position about 8 ft from the original landing site. The spacecraft responded only briefly during the second lunar day on December 14, 1967.

A. Spacecraft System

In the *Surveyor* spacecraft design, the primary objective was to maximize the probability of successful spacecraft operation within the basic limitations imposed by launch vehicle capabilities, the extent of knowledge of transit and lunar environments, and the current technological state of the art. In keeping with this primary objective, design policies were established which (1) minimized spacecraft complexity by placing responsibility for mission control and decision-making on earth-based equipment wherever possible, (2) provided the capability of transmitting a large number of different data channels from the spacecraft, (3) included provisions for accommodating a large number of individual commands from the earth, and (4) made all subsystems as autonomous as practicable.

Figure IV-1 illustrates the *Surveyor* spacecraft in the cruise mode and identifies many of the major components. A simplified functional block diagram of the spacecraft system is shown in Fig. IV-2. The spacecraft design is discussed briefly in this section and in greater detail in the subsystem sections which follow. A detailed

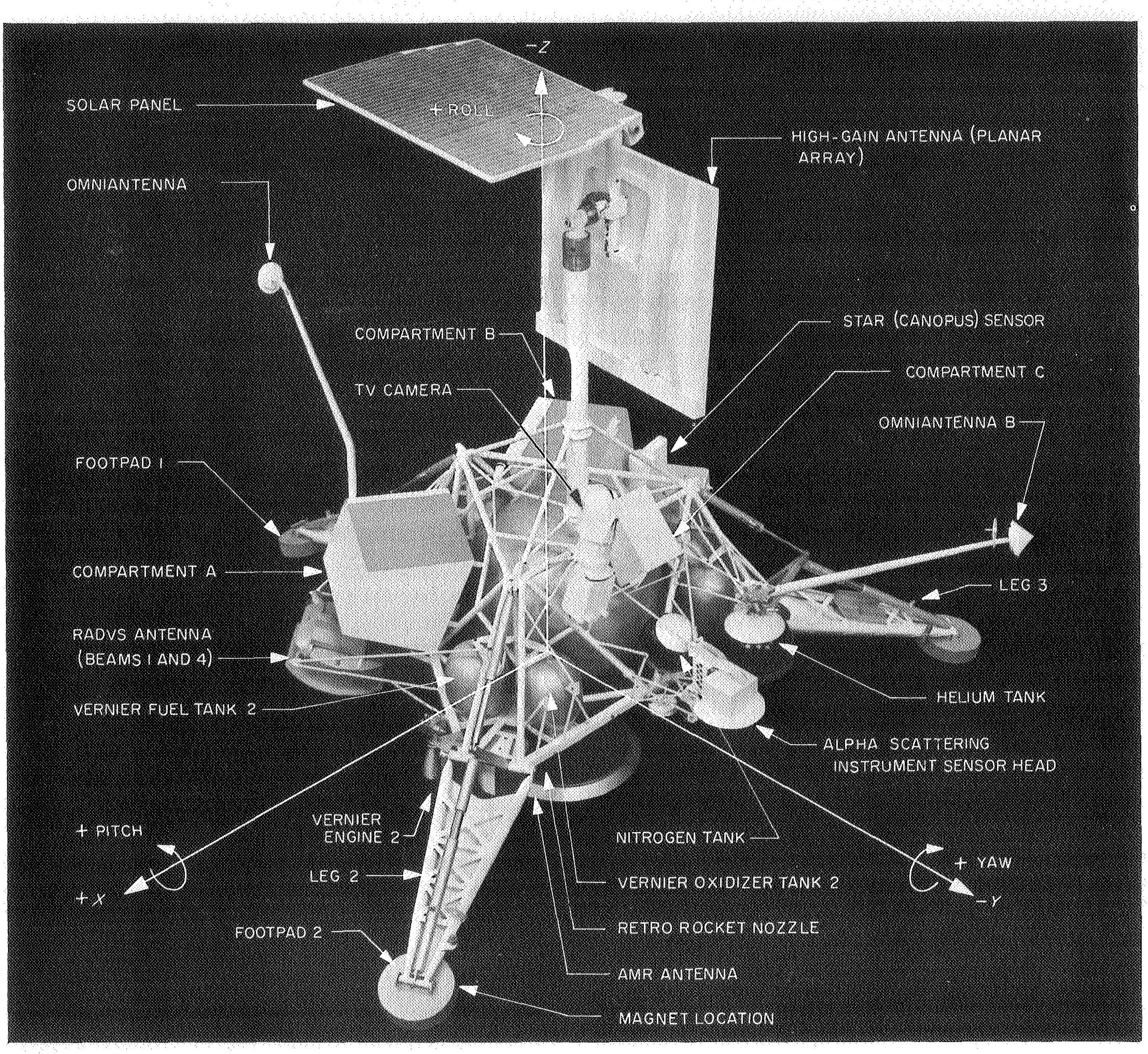


Fig. IV-1. *Surveyor VI* spacecraft in cruise mode

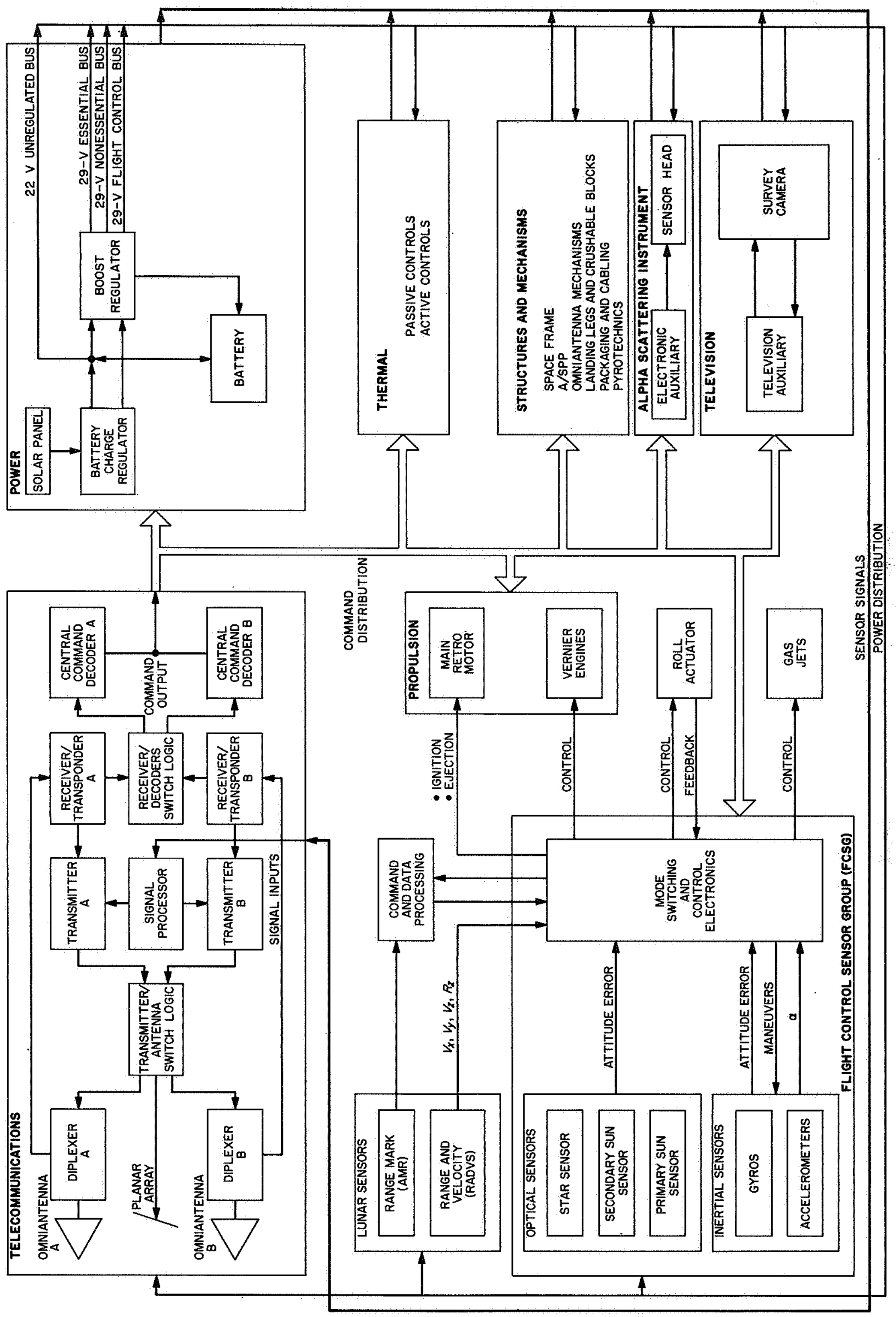


Fig. IV-2. *Surveyor VI* **spacecraft functional block diagram**

configuration drawing of the spacecraft is contained in Appendix B.

1. Spacecraft Coordinate System

The spacecraft coordinate system is an orthogonal, right-hand Cartesian system. Figure IV-3 shows the spacecraft motion about its coordinate axes relative to the celestial references. The cone angle of the earth is the angle between the sun vector and the earth vector as seen from the spacecraft. The clock angle of the earth is measured in a plane perpendicular to the sun vector from the projection of the star Canopus vector to the projection of the earth vector in the plane. The spacecraft coordinate system may be related to the cone and the clock angle coordinate system, provided sun and Canopus lock-on has been achieved. In this case, the spacecraft minus Z-axis is directed toward the sun, and the minus X-axis is coincident with the projection of the Canopus vector in the plane perpendicular to the direction of the sun. The spacecraft +Z-axis is in the direction of the retromotor exhaust, and Leg 1 lies in the Y-Z plane.

2. Spacecraft Mass Properties

The *Surveyor VI* spacecraft weighed 2220.37 lb at separation and 661 lb at initial landing. Center of gravity of

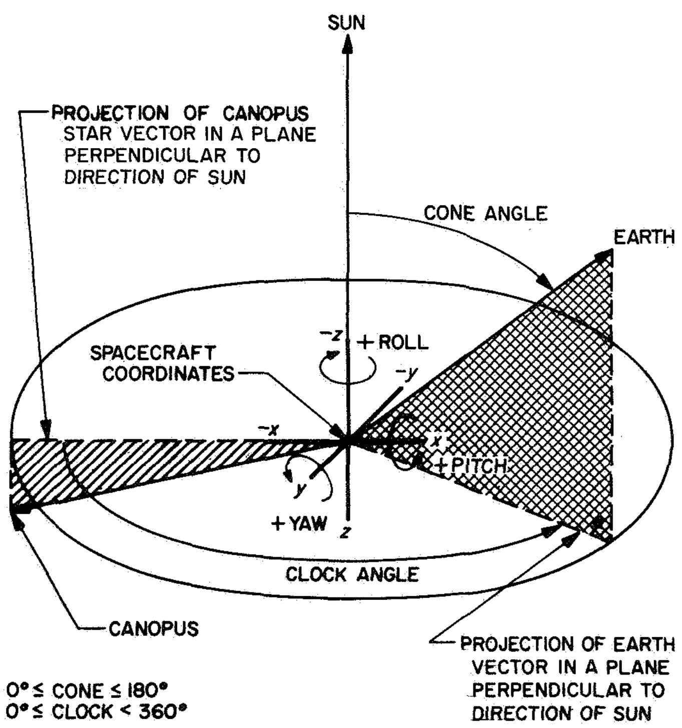


Fig. IV-3. Spacecraft coordinates relative to celestial references

the vehicle is kept low to obtain stability over a wide range of landing conditions. Center-of-gravity limits after *Surveyor/Centaur* separation for midcourse and retro maneuvers are constrained by the attitude correction capabilities of the flight control and vernier engine subsystems during retrorocket burning. Limits of travel of the vertical center of gravity in the touchdown configuration are designed to landing site assumptions and approach angle requirements so that the spacecraft will not topple when landing.

3. Vehicle Structures and Mechanisms

The vehicle structures and mechanisms subsystem provides support, alignment, thermal protection, electrical interconnection, mechanical actuation, and touchdown stabilization for the spacecraft. The subsystem consists of the basic spaceframe, landing gear, crushable blocks, omnidirectional antenna mechanisms, antenna/solar panel positioner, pyrotechnic devices, electrical cabling, thermal compartments, and separation sensing and arming devices.

The spaceframe is the basic structure of the spacecraft and provides a rigid mount for all components of the spacecraft and mounting surfaces and attachments for connecting to *Centaur*. The tripodic landing gear and crushable blocks stabilize the spacecraft and absorb impact energy during touchdown. The omnidirectional antenna mechanisms provide for omniantenna deployment. The antenna/solar panel positioner (A/SPP) supports and positions (around four independent axes) the planar array antenna and solar panel. The pyrotechnic devices mechanically actuate mechanisms, switches, and valves. The electrical cabling interconnects the spacecraft subsystems and components. The thermal compartments provide thermal control for temperature-sensitive components. The separation sensing and arming devices insure that certain critical squib firing circuits will remain disabled until after *Centaur* separation.

4. Thermal Control

Thermal control of equipment over the extreme temperature range of the lunar surface (+260 to −250°F) is accomplished by techniques representing the latest state of the art in the design of lightweight spacecraft. Included are "passive" controls such as superinsulation of electronic compartments and special surface finishes to achieve optimum absorption and emission characteristics, "active" heater systems, and "semiactive" thermal switches.

5. Electrical Power

Surveyor VI used the same simplified, high-efficiency electrical power system that had been redesigned for and successfully utilized on the *Surveyor V* mission. In this system, electrical power for the spacecraft is supplied by a solar panel and a sealed, rechargeable, silver–zinc battery. Power regulation and distribution are provided by a battery charge regulator (BCR) and a boost regulator (BR). The BCR provides solar panel switching functions and battery charge control. The battery is connected to the unregulated bus which supplies power at a voltage between 17.5 and 27.5 V (nominal 22 V) to spacecraft loads not requiring regulation. The unregulated bus voltage is converted by the BR to supply spacecraft loads requiring regulated power. The BR contains (1) a preregulator that provides power at 30 V $\pm 2\%$ for the spacecraft "essential" loads, (2) a nonessential bus that provides power at 29 V $\pm 1\%$ for "nonessential" loads, (3) a flight control regulator to provide power at 29 V for flight control loads, and (4) a shunt regulator which converts excess solar panel power for battery charging. The solar panel can be connected either to the unregulated bus or directly to the preregulator output bus, in which mode the power system operates most efficiently.

6. Propulsion

The propulsion subsystem provides thrust during the midcourse correction and terminal descent phases. The propulsion subsystem, consisting of a bipropellant vernier engine system and a solid-propellant main retrorocket motor, is controlled by the flight control system through preprogrammed maneuvers, commands from earth, and maneuvers initiated by flight control sensor signals.

The three thrust chambers of the vernier engine subsystem provide the thrust for midcourse velocity vector correction, attitude control during main retrorocket burning, and velocity vector and attitude control during terminal descent. Thrust Chamber 1 can be swiveled to provide a thrust vector about the roll axis. Thrust Chambers 2 and 3 are stationary and provide thrust parallel to the roll axis for pitch and yaw control. The thrust of each engine can be differentially throttled over a range of 30 to 104 lb to provide attitude control in pitch and yaw.

The main retromotor is utilized to remove the major portion of the spacecraft approach velocity during terminal descent. It is a spherical solid-propellant motor with partially submerged nozzle to minimize overall length. The motor provides a thrust of 8,000 to 10,000 lb for a duration of about 41 sec.

7. Flight Control

The flight control subsystem provides spacecraft velocity and attitude control during transit and terminal descent and includes the following spacecraft elements: flight control sensor group (FCSG), attitude control jets, attitude control gas supply, and Vernier Engine 1 roll actuator.

The FCSG contains inertial (gyros), optical (Canopus sensor, acquisition sun sensor, primary sun sensor), and acceleration sensors and flight control electronics. The outputs of each of these sensors and radar sensors are utilized by analog electronics to provide commands for operation of attitude gas jets and the spacecraft vernier and main retro propulsion systems. Flight control requires ground commands for initiation of various sequences and performance of "manual" operations. Flight control programming initiates and controls other sequences.

The celestial sensors allow the spacecraft to be locked to a specific orientation defined by the vectors to the sun and the star Canopus and the angle between them. Initial search and acquisition of the sun are accomplished by the secondary sun sensor. The primary sun sensor then maintains the orientation with the sun line.

Of the inertial sensors, integrating gyros are used to maintain spacecraft orientation inertially when the celestial references are not available. Accelerometers measure the thrust levels of the spacecraft propulsion system during midcourse correction and terminal descent phases.

A pair of attitude jets is located on each of the three legs of the spacecraft. The attitude jets provide for angular rate stabilization after spacecraft separation from the *Centaur*, attitude orientation for sun and Canopus acquisition, attitude control during coast phases, and attitude orientation for midcourse correction and terminal descent. The attitude control gas supply provides nitrogen under regulated pressure from a supply tank to the attitude jets.

The three vernier engines are controlled to provide thrust, which can be varied over a wide range, for midcourse correction of the spacecraft velocity vector and controlled descent to the lunar surface. Commands to the vernier roll actuator tilt the thrust axis of Vernier Engine 1 away from the spacecraft roll axis for attitude and roll control during thrust phases of flight when the attitude gas jets are not effective.

8. Radar

Two radar systems are employed by the *Surveyor* spacecraft. An altitude marking radar (AMR) provides a mark signal to initiate the main retro sequence. In addition, a radar altimeter and doppler velocity sensor (RADVS) functions in the flight control subsystem to provide three-axis velocity, range, and altitude mark signals for flight control during the main retro and vernier phases of terminal descent. The RADVS consists of a doppler velocity sensor, which computes velocity along each of the spacecraft *X*, *Y*, and *Z* axes, and a radar altimeter, which computes slant range from 50,000 ft to 13 ft and generates *1000-ft mark* and *13-ft mark* signals.

9. Telecommunications

The spacecraft telecommunications subsystem provides for (1) receiving and processing commands from earth, (2) providing angle tracking and one- or two-way doppler data for orbit determination, and (3) processing and transmitting spacecraft telemetry data.

Continuous command capability is assured by two identical receivers which remain on throughout the life of the spacecraft and operate in conjunction with two omniantennas and two command decoders through switching logic.

Operation of a receiver in conjunction with a transmitter through a transponder interconnection provides a phase-coherent system for doppler tracking of the spacecraft during transit and after touchdown. Two identical transponder interconnections (Receiver/Transponder A and Receiver/Transponder B) are provided for redundancy. Transmitter B with Receiver/Transponder B is the transponder system normally operated during transit.

Data signals from transducers located throughout the spacecraft are received and prepared for telemetry transmission by signal processing equipment which performs commutation, analog-to-digital conversion, and pulse-code and amplitude-to-frequency modulation functions. Most of the data signals are divided into six groups (commutator modes) for commutation by two commutators located within the telecommunications signal processor. (An additional commutator is located within the television auxiliary for processing television frame identification data.) The content of each commutator mode has been selected to provide essential data during particular phases of the mission (Table IV-1 and Appendix C). Other signals, such as strain gage data which is required continuously over brief intervals, are applied directly to subcarrier oscillators.

Summing amplifiers are used to combine the output of any one commutator mode with continuous data. The composite signal from the signal processor, or television data from the television auxiliary, is sent over one of the two spacecraft transmitters. The commutators can be operated at five different rates (4400, 1100, 550, 137.5, and 17.2 bits/sec) and the transmitters at two different power levels (10 W or 100 mW, minimum). In addition, switching permits each of the transmitters to be operated with any one of the three spacecraft antennas (two omniantennas and a planar array) at either the high or low power level. Selection of data mode(s), data rate, transmitter power, and transmitter-antenna combination is made by ground command. A data rate is selected for each mission phase which will provide sufficient signal strength at the DSIF station to maintain the telemetry error rate within satisfactory limits. The high-gain antenna (planar array) is required for wideband transmission of 600-line television frames.

10. Television

The *Surveyor* television subsystem includes a survey camera and a television auxiliary for final decoding of commands and processing of video and frame identification data for transmission by either of the spacecraft transmitters.

The survey camera is designed for postlanding operation by earth commands to provide photographs of the lunar surface panorama, portions of the spacecraft, and the lunar sky. Photographs may be obtained in either of two modes: a 200-line mode for relatively slow (61.8 sec/frame) transmission over an omniantenna or a 600-line mode for a more efficient and rapid (3.6 sec/frame) transmission over the planar array.

The camera is mounted about 16 deg from vertical and is pointed upward toward a mirror mounted in the camera hood. The mirror together with the hood can be rotated 360 deg and the mirror can be tilted in steps for horizontal and vertical scanning, respectively. Special mirrors are mounted on the spacecraft frame to provide additional camera coverage of areas of interest under the spacecraft.

The camera can be focused over a range from 4 ft to infinity and can be zoomed to obtain photos in narrow

Table IV-1. *Surveyor VI* spacecraft telemetry mode summary

	Data mode	Method of transmission	Data rate	Number of signals		Signals emphasized	Primary use
				Analog	Digital		
Engineering commutator	1	PCM/FM/PM	All	42	40	Flight control, propulsion	Canopus acquisition midcourse maneuver
Engineering commutator	2	PCM/FM/PM	All	80	59	Flight control, propulsion, AMR, RADVS	Transit interrogations, backup for main retro phase
Engineering commutator	3	PCM/FM/PM	All	21	40	Inertial guidance, AMR, RADVS, vernier engines	Backup for vernier descent phase
Engineering commutator	4	PCM/FM/PM	All	75	29	Temperatures, power status, telecommunications	Transit interrogations, lunar operations
Engineering commutator	5	PCM/FM/PM	All	108	50	Flight control, power status, temperature	Midcourse and terminal attitude maneuvers, launch and primary cruise data, lunar interrogations
Engineering commutator	6	PCM/FM/PM	All	47	74	Flight control, power status, AMR, RADVS, vernier engine conditions	Terminal descent thrust phase
Television commutator	7	PCM/FM	Only 4400 bits/sec	13		TV survey camera	TV camera interrogation, TV camera operation
Shock absorber strain gages		FM/PM	Continuous	3		Strain gages	Touchdown force on spacecraft legs
Gyro speed		FM/PM	50 Hz	3		Inertial guidance unit	Verify gyro sync

angle (6.4 deg) or a wide angle (25.4 deg) field of view. A lens iris provides a stop range from $f/4$ to $f/22$. The camera is equipped with a focal plane shutter which normally provides an exposure time of 150 msec but can be commanded to remain open for time exposures. A sensing device, attached to the shutter, will keep it shut if the light level is too strong. The same device automatically controls the iris setting. This device can be overridden by ground command. A filter wheel assembly is located between the mirror and lens. The filter wheel of the *Surveyor VI* camera contained one clear and three polarized filters.

Several design changes were incorporated in the mirror-hood assembly of the *Surveyor VI* camera to provide improved performance and greater reliability.

11. Alpha Scattering Instrument

An alpha scattering instrument was included in the spacecraft system for *Surveyor VI* identical to the one which was successfully employed on *Surveyor V*. This instrument contains a radioactive source of alpha particles, the interaction of which with the lunar surface material produces back-scattering of alpha and proton particles. Detectors are used to determine the characteristics of the back-scattering, from which the relative abundance of chemical elements can be determined. The instrument consists of a sensor head, a deployment mechanism, and associated electronics contained in a thermally insulated compartment. During transit to the moon, the sensor head is held in a stowed position where it is in contact with a standard sample. Following an initial calibration period after landing, the sensor head is deployed—first to a position above the lunar surface where it obtains data on background radiation, and then to a position where it rests on the lunar surface to obtain lunar scattering data. An electronics auxiliary and digital electronics provide command decoding, signal processing, power management, and the electronics interface with the basic spacecraft bus.

12. Instrumentation

Transducers are located throughout the spacecraft system to provide signals that are relayed to the DSIF stations by the telecommunication subsystem. These signals are used primarily to assess the condition and performance of the spacecraft. Some of the measurements also provide data useful in deriving knowledge of certain characteristics of the lunar surface. In most cases, the individual subsystems provide the transducers and basic signal conditioning required for signals provided for data related to their equipment. All the instrumentation signals provided for the *Surveyor VI* spacecraft are summarized by category and responsible subsystem in Table IV-2.

All of the temperature transducers are resistance-type units except for two microdiode bridge amplifier assemblies used in the television subsystem. The voltage (signals) and position (electronic switches) measurements consist largely of signals from the command and control circuits. A strain gage is mounted on each of the vernier engine brackets to measure thrust and on each of the three landing leg shock absorbers to monitor touchdown dynamics.

Table IV-3. Notable differences between *Surveyors V* and *VI*: changes incorporated on *Surveyor VI*

Item	Description
Television	The mirror-hood assembly was redesigned to improve performance and reliability by increasing motor torque output, reducing drive friction and inertial loads, incorporating an improved dust seal, modifying the hood to reduce glare, and making changes to facilitate removal of the assembly without requiring recalibration. In addition, a vidicon gain change was made; the color filters were replaced with polarization filters, and a third auxiliary mirror was added for viewing the lunar surface beneath the spacecraft.
Compartment B	A slit was incorporated in the Compartment B thermal blanket to permit access during STV testing without disturbing the thermal switches.
Signal processor	The manufacturer source was changed for the subcarrier oscillators.
A/SPP	The position potentiometer drives for all axes were improved to eliminate slippage.

Table IV-2. *Surveyor VI* instrumentation

Sensor type	Subsystem location									
	Structures, mechanisms, and thermal control	Electrical power	Propulsion	Flight control	Radar	Telecommunications		Alpha scattering instrument	Television	Total
						Signal processing	Radio and command decoding			
Temperature (thermistors)	33	4	19	9	6		2	2	2	77
Temperature (bridges)									2	2
Pressure		1	3	1						5
Position (potentiometers)	7								6	13
Position (mechanical switches)	10			1						11
Position (electrical switches)	1	2		34	15	4	8		4	68
Current		13						1		14
Voltage (power)		5						2		7
Voltage (signals)				14	7		6	2		29
Strain gages	3		3							6
Accelerometer				1						1
Inertial sensors (gyro speed)				3						3
RF power							2			2
Optical				9						9
Calibration				1		10			1	12
Totals	54	25	25	73	28	14	18	7	15	259

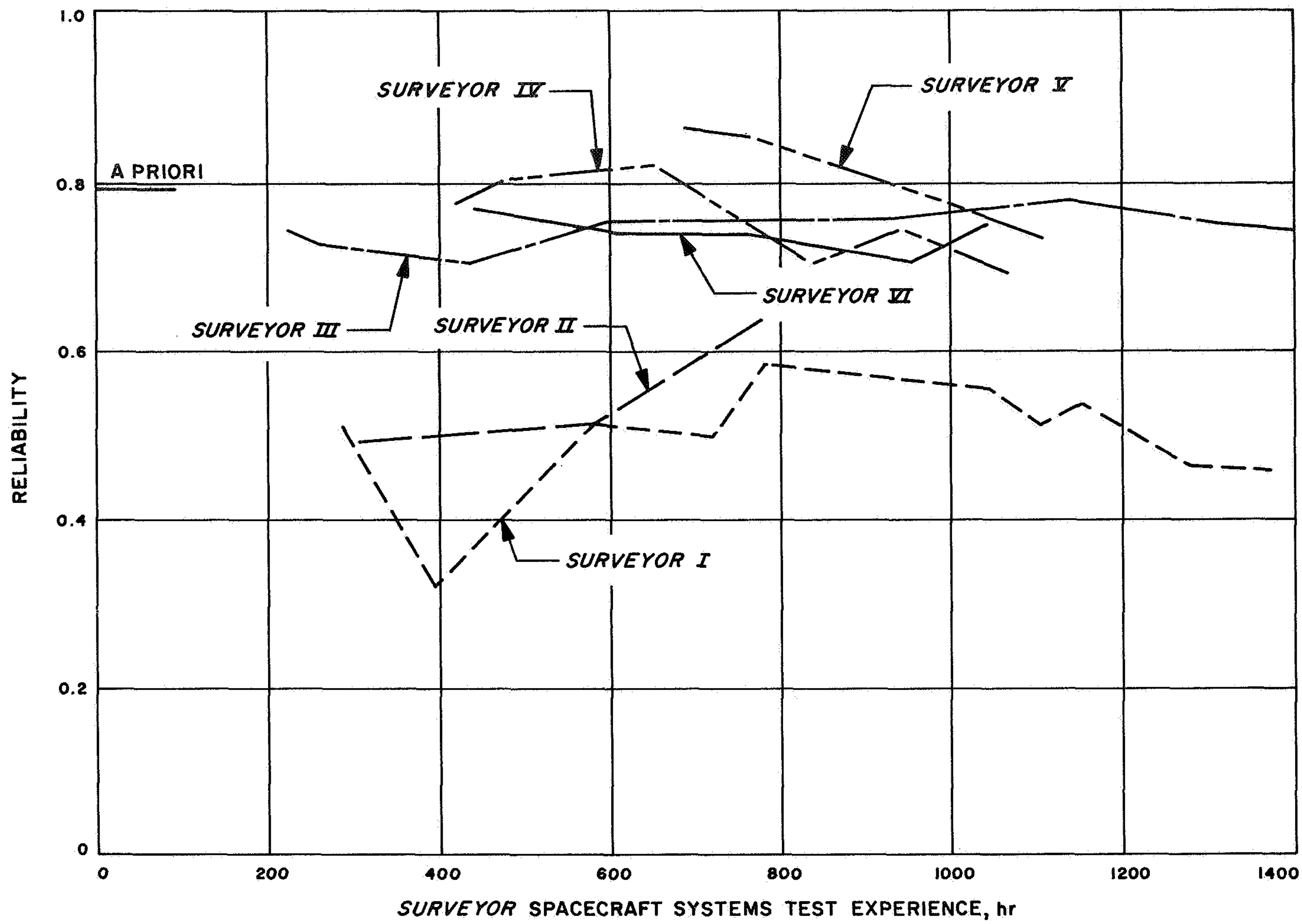


Fig. IV-4. Surveyor spacecraft system reliability estimates

The flight control accelerometer is mounted on the retromotor case to verify motor ignition and provide gross retro performance data. Additional discussion of instrumentation is included with the individual subsystem descriptions.

13. Design Changes

Table IV-3 presents a summary of notable differences in design between the *Surveyor V* and *VI* spacecraft.

14. Spacecraft Reliability

The prelaunch reliability estimate for the *Surveyor VI* spacecraft was 0.70 at 50% confidence level for the flight through landing phases of the mission, assuming successful injection. The reliability estimates for the *Surveyor VI* spacecraft system vs systems test experience are shown in Fig. IV-4. Table IV-4 lists the final reliability estimates for each subsystem. For comparative purposes,

Table IV-4. Surveyor spacecraft subsystem and system reliability estimates

	Surveyor					
	I	*II*	*III*	*IV*	*V*	*VI*
Subsystem						
Telecommunications	0.925	0.944	0.965	0.929	0.987	0.991
Vehicle and mechanisms	0.816	0.868	0.907	0.854	0.853	0.880
Propulsion	0.991	0.991	0.968	0.947	0.934	0.927
Electrical power	0.869	0.958	0.935	0.953	0.985	0.988
Flight control	0.952	0.889	0.971	0.931	0.945	0.940
Systems interaction factor	0.736	0.949	0.967	0.978	0.986	1.0
Overall spacecraft system	0.456	0.658	0.745	0.653	0.722	0.751

estimates for *Surveyor I* through *V* are also shown. The primary source of data for these reliability estimates is the time and cycle information experienced by spacecraft units during systems tests. Test and flight data for the spacecraft used on previous missions was included where there were no significant design differences between the units. In general, a failure is considered relevant if it could occur during a mission.

15. Functional Description of Spacecraft Automatic Flight Sequences

The *Surveyor* spacecraft system has been designed for automatic operation in the following flight sequences:

a. Solar panel deployment and sun acquisition. Immediately upon separation from the *Centaur,* the spacecraft automatically deploys the solar panel to the transit position in a two-step sequence. First, the A/SPP solar panel axis is unlocked and the solar panel is rotated 56 deg from the stowed position to an orientation normal to the spacecraft Z-axis. When this position is reached, the solar panel axis is locked and the A/SPP roll axis is unlocked. The antenna/solar panel combination is then rotated 60 deg about the roll axis and locked in the transit position.

Also upon separation from the *Centaur,* the spacecraft flight control system operates the attitude control jets in a rate-stabilization mode to reduce angular motion imparted to the spacecraft to within deadband limits of ± 0.1 deg/sec. After a 51-sec delay following spacecraft electrical disconnect, which allows time for the spacecraft to reduce the angular motion, a sun acquisition sequence is automatically initiated. First, flight control commands a roll maneuver which permits the sun to be acquired by the acquisition sun sensor, which has a 10-deg-wide by 196-deg-fan-shaped field of view that is centered about the spacecraft minus *X*-axis. Upon initial sun acquisition, the roll is stopped and a positive yaw maneuver is initiated to permit the narrow-view primary sun sensor to acquire and lock-on the sun. A secondary sun sensor is mounted on the solar panel to provide a means of sun acquisition by ground command if the automatic sequence fails.

Upon completion of the solar panel deployment and sun acquisition sequences, the spacecraft coasts with its roll axis (and the active face of the solar panel) held positioned toward the sun by maintaining the sun within the field of view of the center cell of the primary sun sensor. The other axes are held inertially fixed by means of the roll gyro.

b. Canopus acquisition. Some time after sun acquisition, a roll maneuver is initiated by ground command for star verification and Canopus acquisition. As the spacecraft rolls about its Z-axis, the Canopus sensor provides intensity signals of objects which pass through its field of view and have intensities in the sensitivity range of the sensor. Comparison of a map constructed from these signals and a map previously prepared based on predictions permits identification of Canopus from among the signals. After sufficient spacecraft roll to permit Canopus verification, the spacecraft is switched to *star acquisition* mode by ground command to permit the spacecraft to lock-on Canopus automatically the next time Canopus passes through the sensor field of view. Canopus acquisition establishes three-axes attitude reference, which is required before the midcourse and terminal maneuvers can be performed.

c. Midcourse velocity correction. The spacecraft executes each of the desired midcourse attitude maneuvers (roll, pitch, or yaw) upon initiation by ground command. The spacecraft automatically terminates each maneuver when the desired magnitude (previously transmitted by ground command and stored in the flight control programmer) of turn has been accomplished. The maneuvers must be accomplished serially since the flight control programmer can store only one magnitude at a time.

The magnitude of the desired vernier thrust time for midcourse velocity correction is also transmitted to the spacecraft, verified, and stored by the flight control. During the midcourse thrusting phase, the three vernier engines provide a constant spacecraft acceleration of about 0.1 earth g until cutoff is commanded by flight control. During thrusting, the vernier engines are differentially throttled and the roll actuator is controlled to achieve attitude stabilization.

d. Terminal descent sequence. The terminal phase begins with the preretro attitude maneuvers (Fig. IV-5). These maneuvers are commanded from earth in a manner similar to the midcourse maneuvers to reposition the attitude of the spacecraft from the coast phase sun-star reference such that (1) the expected direction of the retro thrust vector will be aligned with respect to the spacecraft velocity vector, and (2) the spacecraft roll attitude will provide the most desirable landed orientation for lunar operations within the RADVS and telecommunications descent constraints. Following completion of the

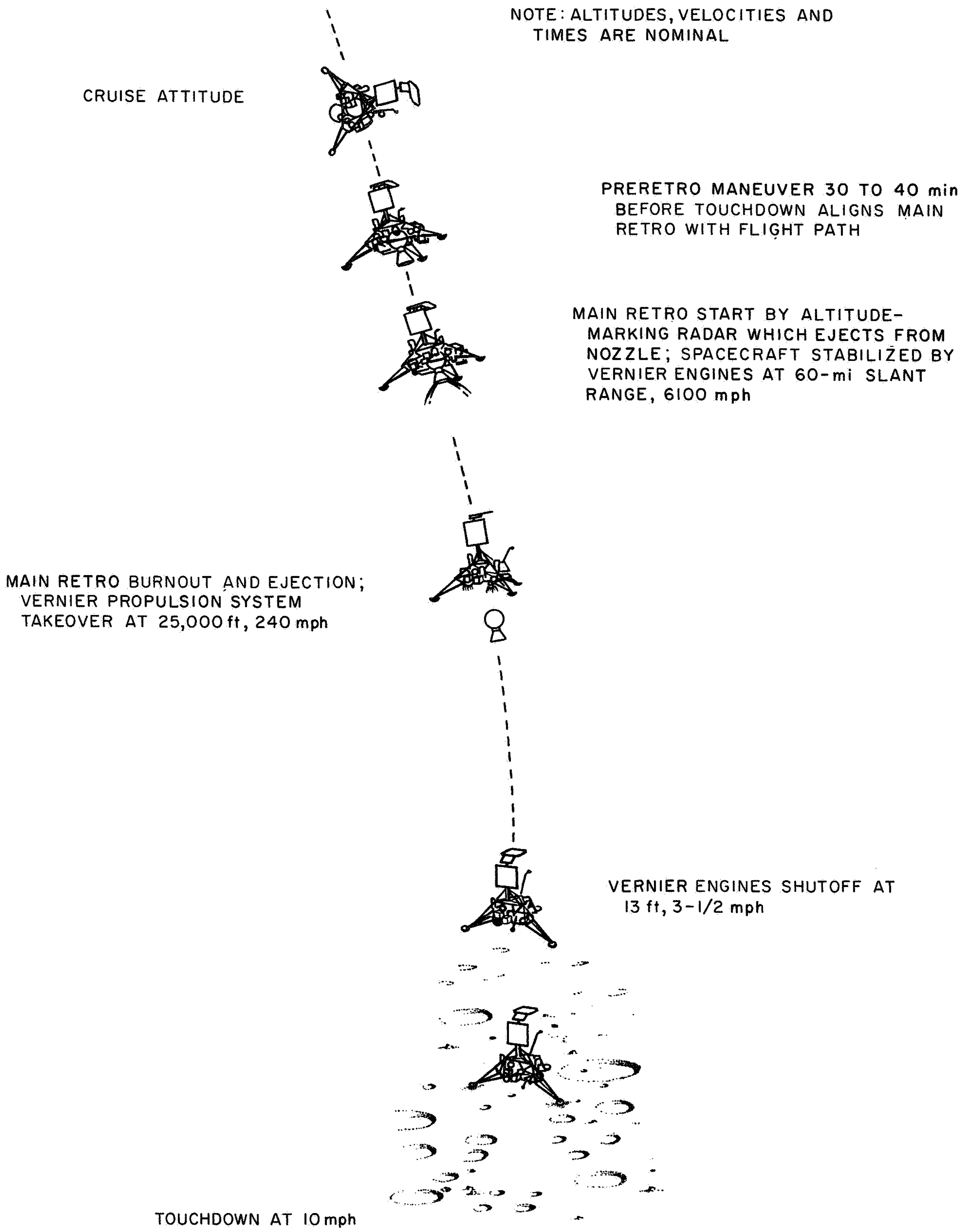


Fig. IV-5. Terminal descent nominal events

attitude maneuvers, the AMR is activated. It has been preset to generate a *mark* signal when the slant range to the lunar surface is 60 miles nominal. A backup *mark* signal, delayed a short interval after the AMR *mark* should occur, is transmitted to the spacecraft to initiate the automatic sequence in the event the AMR *mark* is not generated. A delay between the *altitude mark* and main retromotor ignition has been preset in the flight control programmer by ground command. Vernier engine ignition is automatically initiated 1.1 sec prior to main retro ignition. The main retro phase sequence for a nominal mission is illustrated in Fig. IV-6.

During the main retro phase, spacecraft attitude is maintained in the inertial direction established at the end of the preretro maneuvers by differential throttle control of the vernier engines while maintaining the total vernier thrust at the midthrust level. The main retro burns at essentially constant thrust for about 40 sec, after which the thrust starts to decay. This tailoff is detected by an inertial switch which increases vernier thrust to the high level and initiates a programmed time delay of about 12 sec, after which the main retro motor case is ejected. The main retro phase removes more than 95% of the spacecraft velocity and puts the spacecraft position, velocity, and attitude relative to the lunar surface within the capability of the final, vernier phase.

The vernier phase normally begins at altitudes between 10,000 and 50,000 ft and velocities in the range of 100 to 700 ft/sec. This wide range of vernier-phase initial conditions exists because of statistical variations in parameters which affect main retro burnout. About 2 sec after separation of the main retro case, vernier thrust is reduced and controlled to produce a constant spacecraft acceleration of 0.1 lunar *g*, as sensed by an axially oriented accelerometer. The spacecraft attitude is held in the preretro position until the doppler velocity sensor locks onto the lunar surface. The thrust axis is then aligned and maintained to the spacecraft velocity vector throughout the remainder of the descent until the terminal sequence is initiated (when the attitude is again held inertially fixed). With the thrust axis maintained in alignment with the velocity vector, the spacecraft makes a "gravity turn," wherein gravity tends to force the flight path towards the vertical as the spacecraft decelerates.

The vehicle descends at 0.1 lunar *g* until the radars sense that the "descent contour" has been reached (Fig. IV-7). This contour corresponds, in the vertical case, to descent at a constant deceleration. The vernier thrust is commanded such that the vehicle follows the descent contour until shortly before touchdown, when the terminal sequence is initiated. Nominally, the terminal sequence consists of a constant-velocity descent from 40 to 13 ft at 5 ft/sec, followed by a free fall from 13 ft, resulting in touchdown at approximately 13 ft/sec.

Constraints on the allowable main retromotor burnout conditions are of major importance in *Surveyor* terminal descent design.

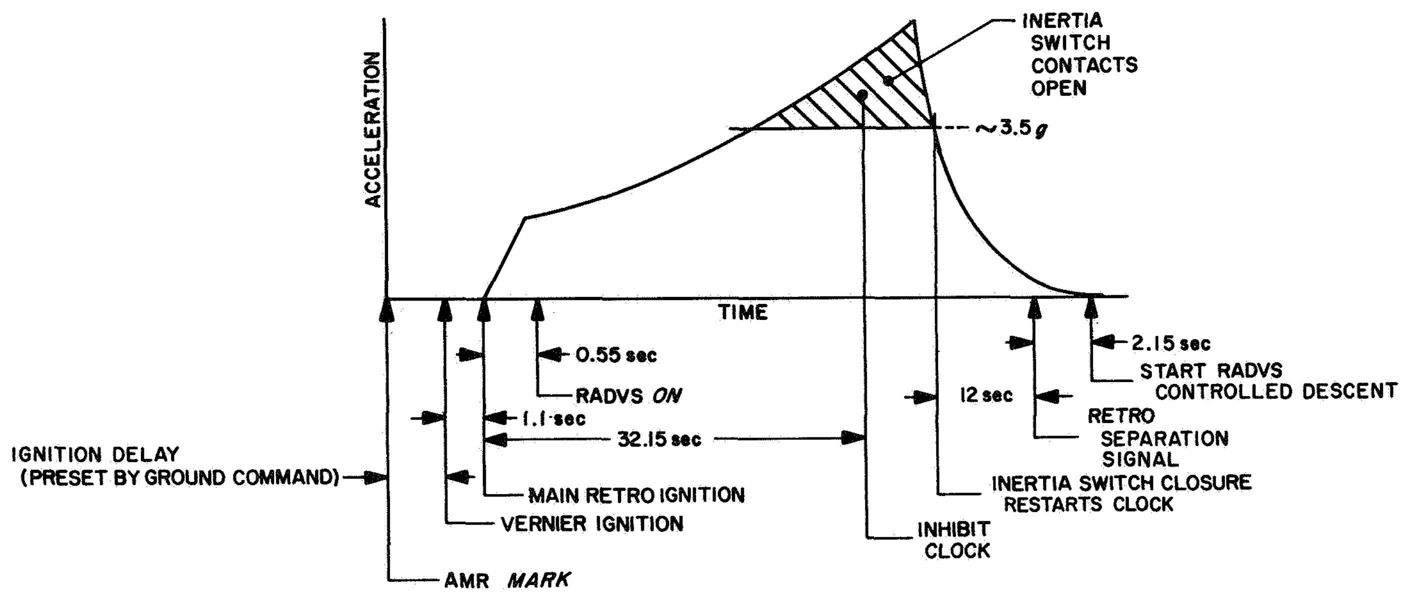


Fig. IV-6. Main retro phase nominal sequence of events

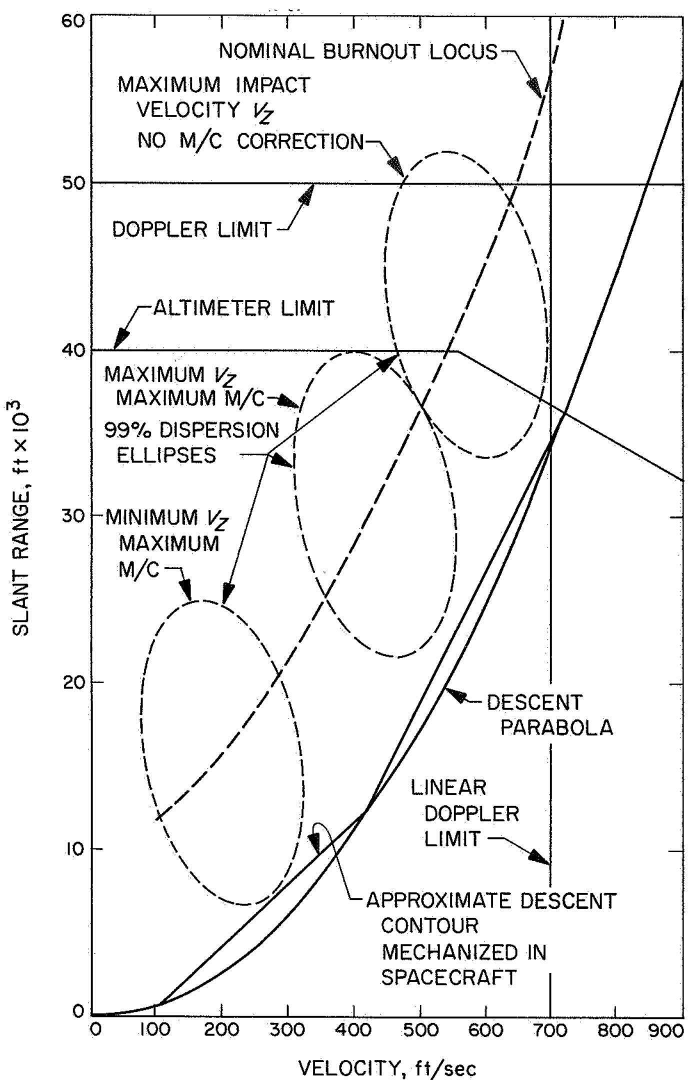


Fig. IV-7. Range-velocity diagram

RADVS operational limitations contribute to constraints on the main retro burnout conditions. Linear operation of the doppler velocity sensor is expected for slant ranges below 50,000 ft and for velocities below 700 ft/sec. The altimeter limit is between 30,000 and 40,000 ft, depending on velocity. These constraints are illustrated in the range-velocity plane of Fig. IV-7.

The allowable main retro burnout region is further restricted by the maximum thrust capability of the vernier engine system. To accurately control the final descent, the minimum thrust must be less than the least possible landed weight (lunar gravity) of the vehicle. The result is a minimum thrust of 90 lb. This in turn constrains the maximum vernier thrust to 312 lb because of the limited range of throttle control which is possible.

Descent at the maximum thrust to touchdown defines a curve in the range-velocity plane below which main retro burnout cannot be allowed to occur. Actually, since the vernier engines are also used for attitude stabilization by differential thrust control, it is necessary to allow some margin from the maximum thrust level. Furthermore, since it is more convenient to sense deceleration than thrust, the vernier phase of terminal descent is performed at nearly constant deceleration rather than at constant thrust. Therefore, maximum thrust will be utilized only at the start of the vernier phase.

The maximum vernier phase deceleration defines a parabola in the altitude-velocity plane. For vertical descents at least, this curve defines the minimum altitude at which main retro burnout is permitted to occur with a resulting soft landing. This parabola is indicated in Fig. IV-7. (For ease of spacecraft mechanization, the parabola is approximated by a descent contour consisting of straight-line segments.)

Main retro burnout must occur sufficiently above the descent contour to allow time to align the thrust axis with the velocity vector before the trajectory intersects the contour. Thus, a "nominal burnout locus" (also shown in Fig. IV-7) is established which allows for altitude dispersions plus an alignment time which depends on the maximum angle between the flight path and roll axis at burnout.

The allowable burnout region having been defined, the size of the main retromotor and ignition altitude are determined such that burnout will occur within that region.

In order to establish the maximum propellant requirements for the vernier system, it is necessary to consider dispersions in main retro burnout conditions as well as midcourse maneuver fuel expenditures. The principal sources of main retro burnout velocity dispersion are the imperfect alignment of the vehicle prior to main retro ignition and the variability of the total impulse. In the case of a vertical descent, these variations cause dispersions of the type shown in Fig. IV-7, where the ellipse defines a region within which burnout will occur with probability 0.99. The design chosen provides enough fuel so that, given a maximum midcourse correction, the probability of not running out is at least 0.99.

The spacecraft landing gear is designed to withstand a horizontal component of the landing velocity. The horizontal component of the landing velocity is nominally zero. However, dispersions arise primarily because of the following two factors:

(1) Measurement error in the doppler system resulting in a velocity error normal to the thrust axis.

(2) Nonvertical attitude due to: (a) termination of the "gravity turn" at a finite velocity, and (b) attitude control system noise sources.

Since the attitude at the beginning of the constant-velocity descent is inertially held until vernier engine cutoff, these errors give rise to a significant lateral velocity at touchdown.

16. Spacecraft System Performance

A summary of *Surveyor VI* spacecraft system performance is presented below by mission phases. Also refer to Sections IV-B through IV-J for spacecraft subsystem performance. Section VI-C presents a chronology of mission operations.

a. Countdown and launch phase performance. The spacecraft performed normally prior to launch. During the countdown, when the command was sent to turn on the AMR heater in accordance with the standard procedure, the command printer did not register the command. Without interrupting the countdown, it was verified that the command had been sent and received by the spacecraft.

Liftoff occurred at 07:39:01.075 GMT on November 7, 1967, with the spacecraft in the standard launch configuration (transmitter low power on, legs and omniantennas folded, solar panel and planar array stowed and locked, etc.). Performance during launch via parking orbit was as expected, and the spacecraft was injected into an accurate transfer trajectory.

During the boost phase of flight, the *Surveyor* was subjected to a variable vibration environment consisting of acoustically induced random vibration and the transient response to discrete flight events. The *Surveyor VI* space vehicle was instrumented with two accelerometers, the output of which was telemetered continuously in order to obtain information on this vibration environment. These accelerometers were located on the spacecraft adapter just below the spacecraft attach points. One accelerometer was oriented in the radial direction at Leg 1 attach point, and the other in the longitudinal direction at Leg 3 attach point. The dynamic environment indicated by these accelerometers may be compared with the dynamic environment of *Surveyor V* as indicated by similar instrumentation, but a comparison cannot be made with the dynamic environment measured on the *Surveyor I, II, III,* and *IV* missions because of differences in accelerometer mounting locations. A comparison between the *Surveyors V* and *VI* dynamic environments is given in Table IV-5.

The spacecraft properly executed all preseparation events initiated by the *Centaur* programmer, which included turning on the transmitter high power, extending the landing legs, and extending the omniantennas. Spacecraft separation from the *Centaur* was also accomplished as desired.

b. Postseparation through star acquisition performance. The automatic solar panel and A/SPP roll-axis stepping sequence was initiated by spacecraft separation and occurred normally. The solar panel was unlocked and required about 336 sec to step through −85 deg and lock in its transit position. Upon locking of the solar panel, the A/SPP roll axis was unlocked, and about 240 sec was required for the A/SPP to roll +60 deg and lock in its transit position.

Spacecraft separation also enabled the cold gas attitude control system, and the small angular rates which had been imparted to the spacecraft were nulled in less than 17 sec. After the 51 sec built-in time delay from electrical disconnect, the sun acquisition sequence occurred properly, beginning with a negative roll through about 264 deg which was terminated upon illumination of the acquisition sun sensor. Then followed a positive yaw turn through about 22 deg, which was terminated upon illumination of the primary sun sensor center cell.

After good two-way telecommunications were initially established by DSS 51 at about L + 35 min and an assessment of spacecraft telemetry indicated all systems to be normal, the spacecraft responded correctly to an initial sequence of earth commands to place the spacecraft in the cruise configuration. In the cruise configuration, the transmitter was operated on low power at 1100 bits/sec with the coast phase commutator (Mode 5) on.

Automatic Canopus acquisition was successfully accomplished about 9 hr after liftoff. However, during the roll maneuver used to generate a star map prior to star acquisition, the DSIF station lost receiver lock owing to a deeper than anticipated omniantenna pattern null. The roll maneuver had been initiated with the spacecraft in

Table IV-5. *Surveyor VI* maximum measured acceleration (*g*) during launch phase events compared with *Surveyor V* data (zero-to-peak unless otherwise noted)

Event	Accelerometer			
	Radial		Longitudinal	
	Surveyor V	*Surveyor VI*	*Surveyor V*	*Surveyor VI*
Liftoff	1.95 rms	2.26 rms	1.26 rms	1.21 rms
Maximum aerodynamic loading	0.61 rms	0.71 rms	0.50 rms	0.55 rms
BECO	2.0	1.0	3.0	2.0
Insulation panel jettison	>10	>10	>10	>10
Nose fairing separation	2.0	2.5	3.0	4.5
SECO	1.0	1.0	1.0	2.0
Atlas/Centaur separation	>10	>10	>10	>10
MES 1	<1.0	<1.0	<1.0	<1.0
MECO 1	1.8	<1.0	3.8	2.8
MES 2	<1.0	<1.0	<1.0	<1.0
MECO 2	3.2	2.7	2.8	4.0
Landing gear extension	4.5	7.0	7.5	11.5
Landing gear lock	4.0	3.5	6.4	5.6
Omniantenna deployment	2.6	2.9	7.2	7.2

high power and transponder (two-way) mode. When the antenna null was reached, the spacecraft receiver lost phase lock and the transmitter automatically switched to narrow-band VCXO, which resulted in a shift in frequency and loss of DSS receiver lock. The roll turn was completed in one-way mode after being temporarily stopped to confirm that the loss of lock was not caused by a spacecraft anomaly.

The total spacecraft roll prior to Canopus acquisition was 652.5 deg. Canopus sensor performance was satisfactory. The ratio of measured-to-predicted Canopus intensity was approximately 0.87.

c. Midcourse correction performance. The attitude maneuvers (+91.9 deg roll followed by +127.3 deg yaw) performed in preparation for the midcourse correction were compensated for gyro drift for the first time on a *Surveyor* mission. Each of the maneuvers was also initiated when the respective attitude control loop limit cycle error was as near to a null as possible to reduce the attitude error. Vernier engine ignition was commanded on about 18½ hr after liftoff for a 10.25-sec period to provide a 10.06-m/sec velocity correction. From orbit determination, the actual magnitude of the velocity change is estimated to have been 10.12 m/sec. Based on prelaunch alignment information and in-flight data, the preignition pointing error was 0.16 deg. The midcourse maneuver corrected the miss distance to within 10.5 km of the final aim point.

The vernier thrust phase was smooth, with small ignition and shutdown transients. Inertial reference was, therefore, retained and reacquisition of the sun and Canopus was accomplished via the standard reverse maneuver sequence.

d. Coast phase performance. Spacecraft performance during the pre- and post-midcourse coast phases was normal. Many engineering interrogations were conducted during the coast phases. Spacecraft data at all times indicated that all subsystems were performing within predicted operational limits.

Solar panel power was slightly higher than expected. As predicted, the solar panel switch tripped six times to prevent excessive battery charging during the early part of the mission when spacecraft loads were small. The battery charge level remained slightly above the programmed management curve, and energy remaining at touchdown was estimated to be 2350 W-hr compared to the nominal value of 2000 W-hr.

Performance of the thermal control system was satisfactory, with no anomalies occurring during transit. Curves of temperature histories of spacecraft components during transit are contained in Appendix D.

Telecommunications system performance was normal throughout transit, although a deeper-than-predicted null was found in the Omniantenna B up-link pattern, causing the above-mentioned loss of two-way lock during Canopus acquisition roll. Following Canopus acquisition, Receiver A/Omniantenna A up-link signal level was higher than predicted, while Receiver B/Omniantenna B signal level was lower than predicted as has occurred on previous missions. Down-link signal strength remained 1 to 2 db higher than predicted throughout the flight.

During the entire transit phase, 14 gyro drift checks were performed. This large number of checks was conducted to accurately determine the gyro drift rates, of which the yaw drift rate was above the maximum specification value of 1.0 deg/hr. This excessive drift rate did not affect the mission because both the midcourse and terminal attitude maneuvers were compensated for measured drift. The final gyro drift rates utilized in terminal maneuver calculations were -0.64 deg/hr roll, -0.00 deg/hr pitch, and $+1.40$ deg/hr yaw.

e. Terminal maneuver and descent performance. The spacecraft properly executed the commanded terminal maneuvers ($+81.62$ deg roll followed by $+111.67$ deg yaw and $+120.55$ deg roll), each of which was initiated when the respective attitude control loop error was near a null position.

The descent sequence was initiated automatically by the *60-mile mark* signal from the AMR, and all the terminal descent events occurred very close to predicted times. Predicted and actual times for the events are listed in Table IV-6. After the desired time delay of 5.9 sec from AMR *mark*, the vernier engines ignited followed 1.1 sec later by main retromotor ignition.

RADVS turn-on occurred at the expected time, and all three velocity beams acquired lock about 30 sec after main retro ignition and remained reliable throughout descent. The RADVS altimeter acquired lock about 1.5 sec after start of the RADVS control phase. Descent during the RADVS control phase until programmed descent contour interception and thence very nearly along the contour is illustrated in Fig. IV-8.

Table IV-6. Predicted and actual times of terminal descent phase events (GMT November 10, 1967 at the spacecraft[a])

Event	Predicted[b]	Actual
AMR *60-mile mark*	00:57:56.06	00:57:55.74
Backup AMR *mark*	00:57:57.3	00:57:57.8
Vernier ignition	00:58:01.96	00:58:01.64
Main retro ignition	00:58:03.06	00:58:02.74
RADVS power on (warmup)		00:58:04.5
RADVS high voltage on		00:58:27
RODVS signal (3 velocity beams in lock)		00:58:32.8
Main retro burnout		
3.5-*g* level (from doppler data)	00:58:42.90	00:58:42.26
3.5-*g* switch closure		00:58:42.35
Vernier high thrust		00:58:52.0
Retro case eject signal		00:58:54.34
Retro case ejected		00:58:54.6
Start RADVS control	00:58:57.05	0058:56.44
RORA signal (altitude beam in lock)		00:58:58.0
Programmed descent segment interception	00:59:26.56	00:59:20.0
1000-ft mark	01:00:42.05	01:00:39.24
10-ft/sec mark	01:00:59.43	01:00:56.34
13-ft mark	01:01:04.98	01:01:02.84
Initial touchdown	01:01:06.76	01:01:04.160

[a]1.297-sec RF delay between spacecraft and DSIF stations.

[b]Based upon the final computer run of the Terminal Guidance Program completed approximately 3 hr before retro ignition.

As indicated by the comparison of predicted and actual terminal descent parameters presented in Table IV-7, spacecraft performance was very nominal except for larger-than-expected main retro thrust misalignment. The thrust misalignment, which was within the maximum allowable limit, caused a larger-than-predicted lateral velocity V_y at start of the RADVS control phase.

f. Landing performance. The spacecraft made initial touchdown at 01:01:05.467 GMT, November 10, 1967, on a relatively smooth and essentially level surface with a velocity of about 11 ft/sec and a final approach angle estimated to be 2.3 deg relative to the landing site. Leg 1 contacted first followed by the nearly simultaneous contacts of Legs 2 and 3 about 0.1 sec later. The shock absorber loads, as indicated by the telemetered strain gage readings (see Section IV-B-2), were very similar to

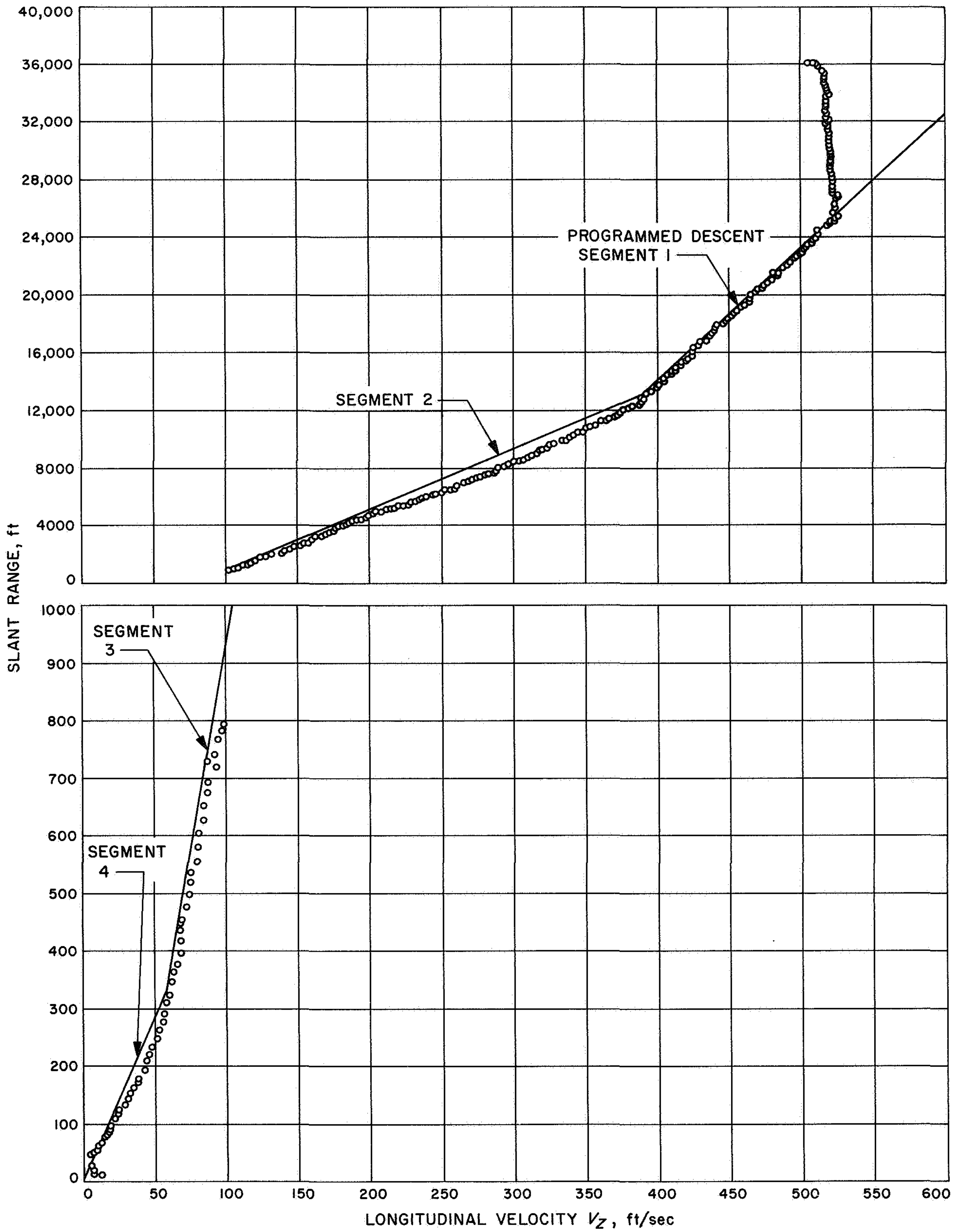


Fig. IV-8. *Surveyor VI* descent profile

Table IV-7. Predicted and actual values of terminal descent parameters

Parameter	Predicted	Actual (best estimate)
Vernier ignition conditions		
Altitude, ft	240,787	239,484
Velocity, ft/sec	8,488.9	8,489.0
Attitude of Z-axis with vertical, deg	24.4	24.5
Main retro phase conditions		
In-plane thrust misalignment, deg	0	0.632
Out-of-plane thrust misalignment, deg	0	0.488
Main retro burn time to 3500-lb-thrust (about 3.8-*g*) level, sec	39.60	39.37
Start RADVS control conditions		
Altitude, ft	37,005	36,625
Longitudinal velocity, ft/sec	468	463
Lateral velocity V_x, ft/sec	36	0
Lateral velocity V_y, ft/sec	105	225
Total velocity, ft/sec	482	515
Flight path angle with vertical, deg	10.8	8.1
Attitude of Z-axis with vertical, deg	25.3	25.4
Programmed descent segment intercept conditions		
Slant range, ft	23,000	24,730
Longitudinal velocity, ft/sec	494	522
1000-ft mark conditions		
Slant range, ft	1000	1000
Longitudinal velocity, ft/sec		106.0
Total velocity, ft/sec	106	106.1
Attitude of Z-axis with vertical, deg	1.3	1.04
10-ft/sec mark conditions		
Slant range, ft	43	50
Longitudinal velocity, ft/sec	8.6	10.0
Attitude of Z-axis with vertical, deg	0	0
Vernier engine cutoff conditions		
Slant range, ft	13	10
Longitudinal velocity, ft/sec	5	5
Attitude of Z-axis with vertical, deg	0	0
Touchdown conditions		
Longitudinal velocity, ft/sec	12.5	11.2
Lateral velocity, ft/sec	0	1.0
Spacecraft weight, lb	663.6	660.9
Flight path angle with vertical, deg	0	2.3
Postlanding conditions		
Roll attitude (spacecraft +X-axis relative to lunar north), deg clockwise		210
Spacecraft tilt, deg		1

those of *Surveyor I*, which also landed on a relatively level surface. The strain gage traces indicate the spacecraft rebounded, with the footpads raising 8 to 11 in. before finally coming to rest. The landed attitude of the spacecraft is shown in Fig. IV-9. A continuous good signal was received during the soft landing. The landing site was determined to be 0.47°N latitude and 1.48°W longitude by correlation of *Surveyor* and *Lunar Orbiter* pictures.

g. Postlanding performance. Engineering assessments revealed the postlanding condition of the spacecraft to be excellent, with no damage having resulted from the landing.

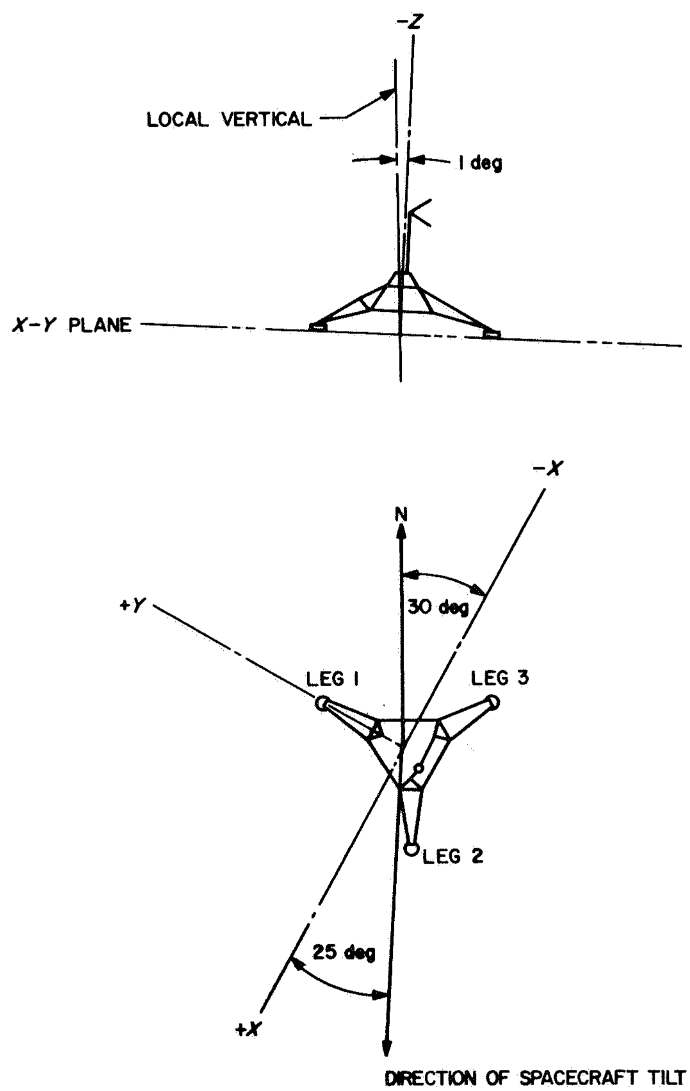


Fig. IV-9. Spacecraft attitude and orientation after initial landing

The spacecraft performed extensive operations very successfully during the lunar day. Only in the period shortly after lunar noon was it necessary to periodically place the spacecraft in a nonoperating mode. This condition is normally anticipated because of high temperatures.

After being unlocked following landing, the A/SPP responded normally to stepping commands. This permitted acquisition of the sun by the solar panel and the earth by the planar array. These are necessary operations to permit transmission of 600-line pictures and to obtain adequate power for continued lunar operation. The A/SPP continued to function properly throughout the lunar day when it was required to (1) reposition the solar panel as sun elevation changed, (2) change the panel shadow patterns for thermal control of critical components such as the camera, alpha scattering sensor head, the compartments, and the vernier propulsion system, and (3) step the planar array for antenna pattern mapping and spacecraft attitude determination. The spacecraft attitude shown in Fig. IV-9 is based upon determinations made from postlanding planar array and solar panel positioning experiments.

The television camera was used extensively and provided excellent-quality pictures. The first television frames returned were 200-line pictures followed, after proper positioning of the A/SPP, by 600-line pictures. The television operations included wide- and narrow-angle panoramas, focus ranging, photometric sequences, star surveys, polarization surveys, shadow progressions, a solar corona sequence, and numerous special area sequences of the lunar surface and parts of the spacecraft. Television pictures also provided data on the amount of lunar soil on the magnet attached to Footpad 2 and the disturbance of the lunar surface material caused by attitude gas jet and vernier engine firings. Before spacecraft shutdown after the first lunar day, about 30,000 pictures had been returned, many of which are of the highest quality yet obtained on a *Surveyor* mission.

An attitude gas jet experiment was successfully performed by firing the gas jets twice about 26 hr after landing. The first of these firings was for a duration of 4 sec. The second firing was for a duration of 60 sec and occurred approximately 24 min after the first. Television pictures indicated there was disturbance of the lunar surface material as a result of this experiment.

About 7½ days after landing, shortly after lunar noon, the spacecraft performed a successful hop. To prepare for this experiment, the A/SPP was used to selectively shade the vernier engines to bring all the temperatures below the 222°F maximum operating limit. The A/SPP was then stowed in a safe position. In order to achieve lateral translation during the hop, in a direction 15 deg to the left of Leg 1 (15 deg from the $+Y$ axis toward the $+X$ axis), the flight control system was preset to achieve a +7-deg pitch attitude (Leg 1 pitched down).

In performing the hop experiment, the vernier engines were fired for 2.5 sec at a total thrust level of about 150 lb. The hop lasted 6.1 sec with the spacecraft rising 10 to 15 ft and translating about 8 ft over the lunar surface. The planned burn time was 2.0 sec. However, the spacecraft rejected the first shutdown command from earth, and the vernier engines were not shut down until the spacecraft received a backup command 0.5 sec later. A small pitch-up moment was caused by slightly delayed shutdown of Engine 1, and this helped to level the spacecraft before landing. Although the hop was greater than expected because of the longer burn time, the spacecraft made a completely successful landing, and no damage resulted from the experiment except for the alpha scattering instrument. Television pictures taken after the hop revealed the alpha scattering instrument sensor head was upside down. This was not unexpected, however, since the instrument design did not permit retraction to a safe position for the hop. Other pictures showed some lunar material was adhering to spacecraft surfaces such as the photometric chart on Omniantenna B boom.

Figure IV-10 illustrates the new spacecraft position resulting from the hop. From the new spacecraft position, excellent pictures, such as the mosaic of Fig. IV-11, were taken of the footpad and crush block imprints made during the initial landing and the surface erosion caused by vernier firing during the hop.

Later on in the lunar day, pressure and temperature data indicated a liquid leak had occurred in Vernier Oxidizer Tank 1. This was to be expected owing to the high temperatures to which the system was exposed over an extended period of time. No leakage of the fuel system was evident during the entire lunar day.

The alpha scattering instrument was also operated extensively to provide an abundance of useful data. About 5 hr of calibration data was obtained in the stowed position before the sensor head was partially deployed to obtain about 6 hr of data on background radiation in the lunar environment. Then the sensor head was lowered to the surface for gathering of lunar back-scattering

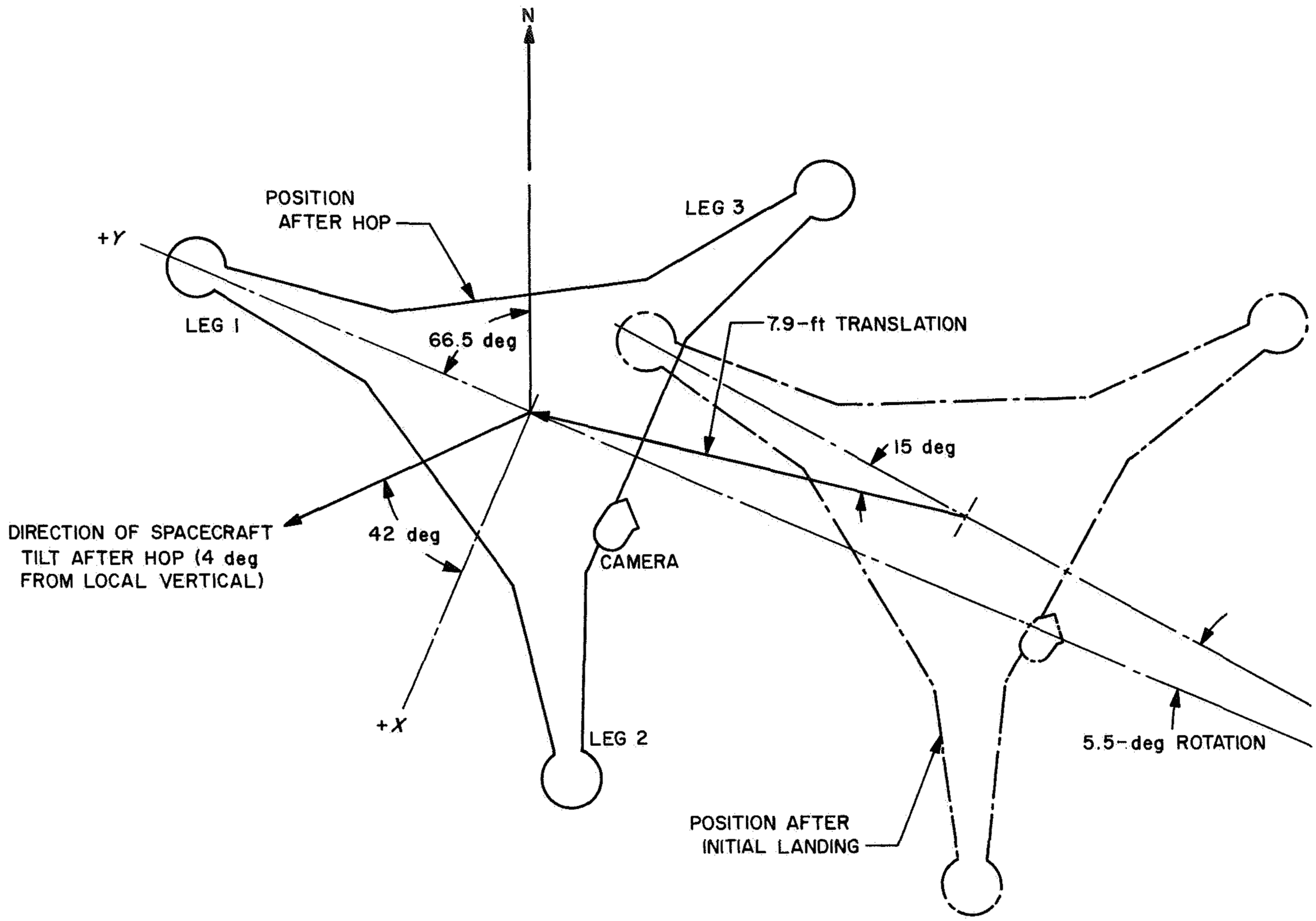


Fig. IV-10. Change in spacecraft position resulting from hop

data. About 30½ hr of alpha scattering data was obtained on the surface prior to the spacecraft hop. After the hop, the alpha scattering instrument was turned on for engineering analysis, but no data useful for scientific analysis could be obtained because the sensor head was on its side.

The spacecraft power subsystem performed perfectly during the entire lunar day and, by careful power management, the battery charge level was brought to a maximum of 165 A-hr before sunset, which is nearly the full-charge condition. For operation during the lunar night, the spacecraft electronics contained in Compartments A and B must be maintained above a lower operating limit (approximately −10°F) to assure reliable operation. Battery capacity is not sufficient to offset heat losses from the compartments to the cold lunar environment and operate the spacecraft through the entire lunar night. However, it was intended to operate the spacecraft as long as possible in order to prevent the battery (located in Compartment A) from cooling to too low a temperature after shutdown.

It had been projected that it might be possible to continue spacecraft operations as long as 200 hr after sunset. However, several Compartment A and B thermal switches did not open. Therefore, the planned battery temperature and power consumption profiles could not be followed, and the spacecraft was turned off only about 41 hr after sunset. When the spacecraft was shut down, the remaining battery charge was estimated to be 102 A-hr.

After sunrise of the second lunar day, *Surveyor VI* responded to a turn-on command at 16:45 GMT on December 14, 1967. The signal obtained was erratic in both signal strength and frequency. Some engineering data was obtained, but signals differed from one mode

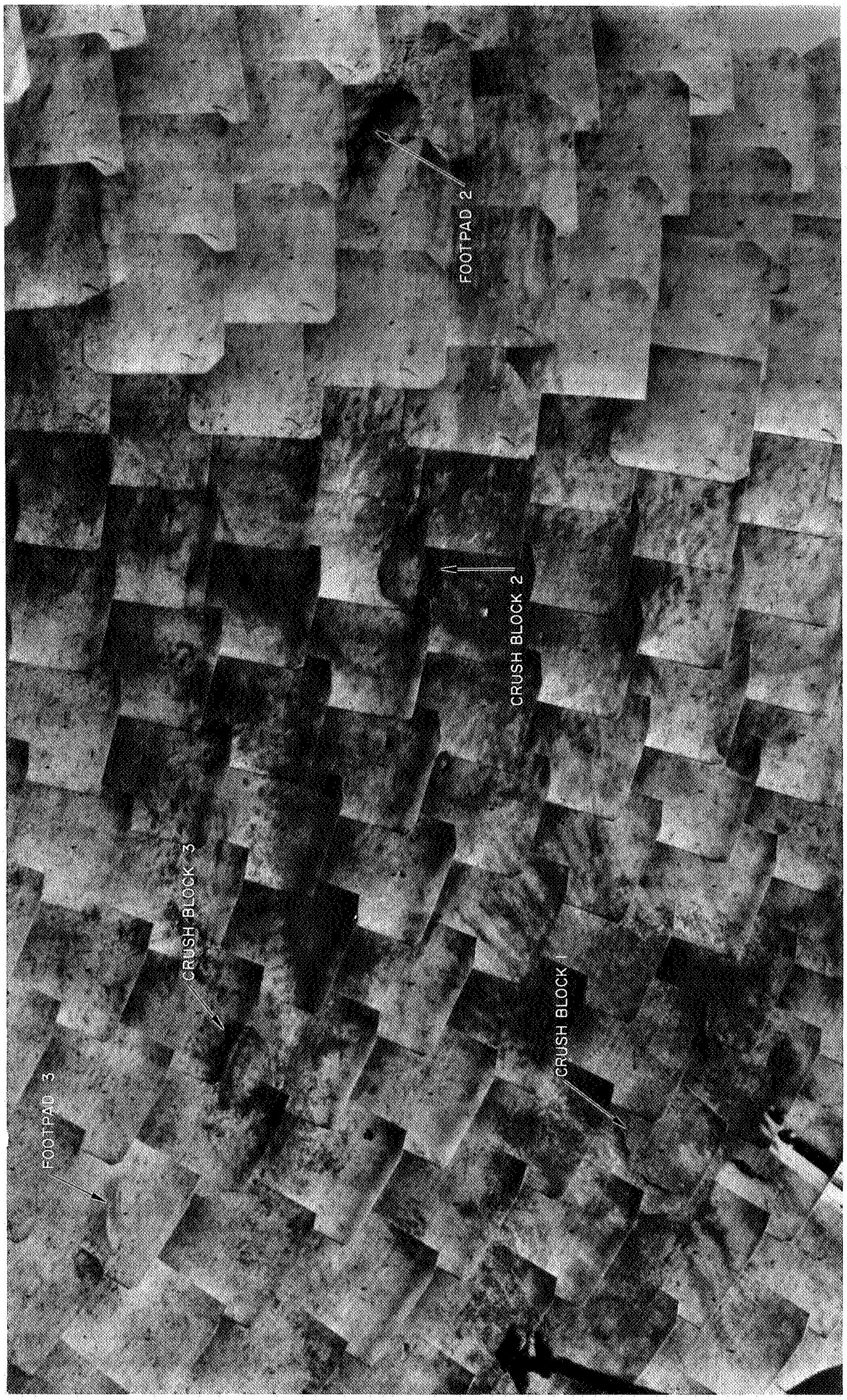


Fig. IV-11. Mosaic of initial landing area taken after spacecraft hop showing imprints of footpads and crush blocks and erosion effects of vernier engine firing (Footpad 1 imprint out of view to lower left) to another. Receiver B was not functioning, and the power status of the battery was not known. The signal was finally lost at 19:14 GMT, December 14, 1967, and subsequent interrogations produced no response.

B. Structures and Mechanisms

The structure and mechanisms subsystem provides support, alignment, thermal protection, electrical interconnection, mechanical actuation, and touchdown shock absorption and vehicle stabilization for the spacecraft and its components. The subsystem includes the basic spaceframe, landing gear mechanism, crushable blocks, omnidirectional antenna mechanisms, antenna/solar panel positioner (A/SPP), pyrotechnic devices, electronic packaging and cabling, thermal compartments, thermal switches, and the separation sensing and arming device.

1. Spaceframe and Substructure

The spaceframe, constructed of thin-wall aluminum tubing, is the basic structure of the spacecraft. The landing legs and crushable blocks, the retrorocket engine, the *Centaur* interconnect structure, the vernier propulsion system, and the A/SPP attach directly to the spaceframe. Substructures are used to provide attachment between the spaceframe and the following subsystems: the thermal compartments, TV, alpha scattering instrument, RADVS antennas, omniantennas, flight control sensor group, attitude control nitrogen tank, and the vernier system helium tank.

During the *Surveyor VI* mission, there was no indication of any failures or anomalies attributable to structural malfunctions.

2. Landing Gear Subsystem

a. Configuration and functional description. The landing gear subsystem consists of three landing leg assemblies and three crushable honeycomb blocks attached to the spaceframe (Fig. IV-12). Each leg assembly is made up of a tubular inverted tripod structure with a shock absorber, an A-frame, a lock strut, and a honeycomb footpad.

During launch, the legs are folded for stowage under the shroud. Shortly before spacecraft separation, upon command from the *Centaur* programmer, the legs are released by squib-actuated pin pullers and deployment is initiated by leaf-type kickout springs located near the release pin pullers. Torsion springs at the hinge axes force the legs to the fully deployed positions, where they are automatically latched.

Upon landing, each leg pivots about its hinge axis while the landing loads are absorbed by compression of the upper member of the tripod, which is a combined shock absorber and spring assembly. These shock absorber columns are instrumented with strain gages. Axial loads measured by the strain gages are telemetered to provide continuous analog traces during the terminal descent and touchdown phase.

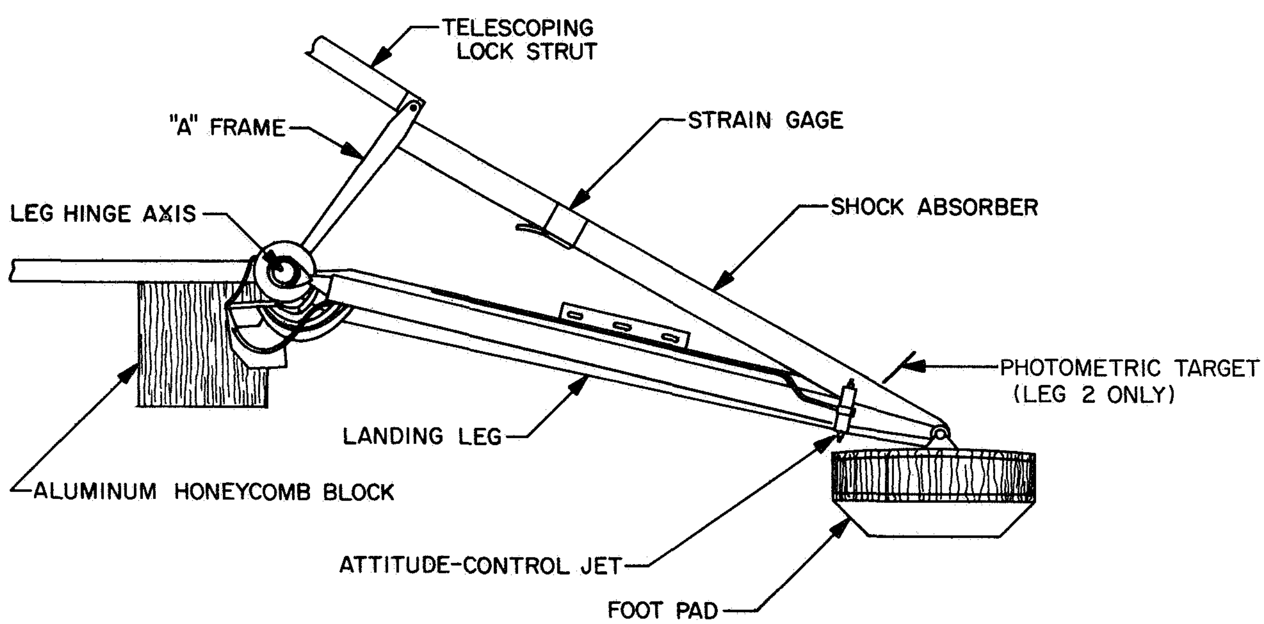


Fig. IV-12. Landing leg assembly

For a vertical landing on a hard, level surface at a velocity in excess of about 8.5 ft/sec, the legs will deflect sufficiently to cause the crushable blocks to contact the surface. Energy will then be further dissipated through penetration of the surface by the blocks or by crushing of the blocks if the surface bearing strength exceeds 40 psi. Crushing of the honeycomb footpads is not expected for landing velocities below about 11.5 ft/sec or for surface bearing strength less than about 10 psi.

b. Transit and lunar landing performance. The performance of the landing gear subsystem on the *Surveyor VI* mission was satisfactory in every respect. The legs were deployed in response to a command from the *Centaur* and latched into the landing positions.

Figure IV-13 shows the strain gage force histories of the three shock absorbers throughout the landing phase. The peak axial forces in the shock absorbers and times of initial impact of the footpads are given in Table IV-8. The forces are considered accurate within ±80 lb, the times within ±2 msec.

An evaluation of the data in Table IV-8, together with other engineering telemetry and postlanding video data, has resulted in the following reconstruction of events during the landing sequence: The 10-ft/sec descent velocity *mark* signal was generated 7.85 sec prior to initial ground contact when the spacecraft was at an altitude of 45±4 ft. Immediately following this *mark*, the spacecraft was slowed to a constant descent velocity of 4.6±1.5 ft/sec (nominal: 5.0±1.5), which was maintained until the 13-ft altitude *mark* was generated 1.24 sec before initial touchdown, at an altitude of 9.8±1.0 ft (nominal: 13±4.5). At this time, all three vernier engines were cut off, resulting in a free-fall period, during which time the spacecraft vertical velocity component increased to 11.2±1.1 ft/sec (nominal: 12.6±2.4) when Leg 1 first contacted the lunar surface. Angular spacecraft motions during the constant-velocity descent were very small. However, the following angular deflections were indicated between the reference attitude at the 10-ft/sec *mark* and the final resting position: pitch, −2.2±0.1 deg; yaw, −0.2±0.1 deg; roll,

Table IV-8. Shock absorber forces and times of footpad impacts for initial lunar landing

	Leg 1	Leg 2	Leg 3
Maximum shock absorber axial force, lb	1590	1810	1590
Time of initial footpad impact, GMT (November 10, 1967)	01:01:05.467	01:01:05.490	01:01:05.560

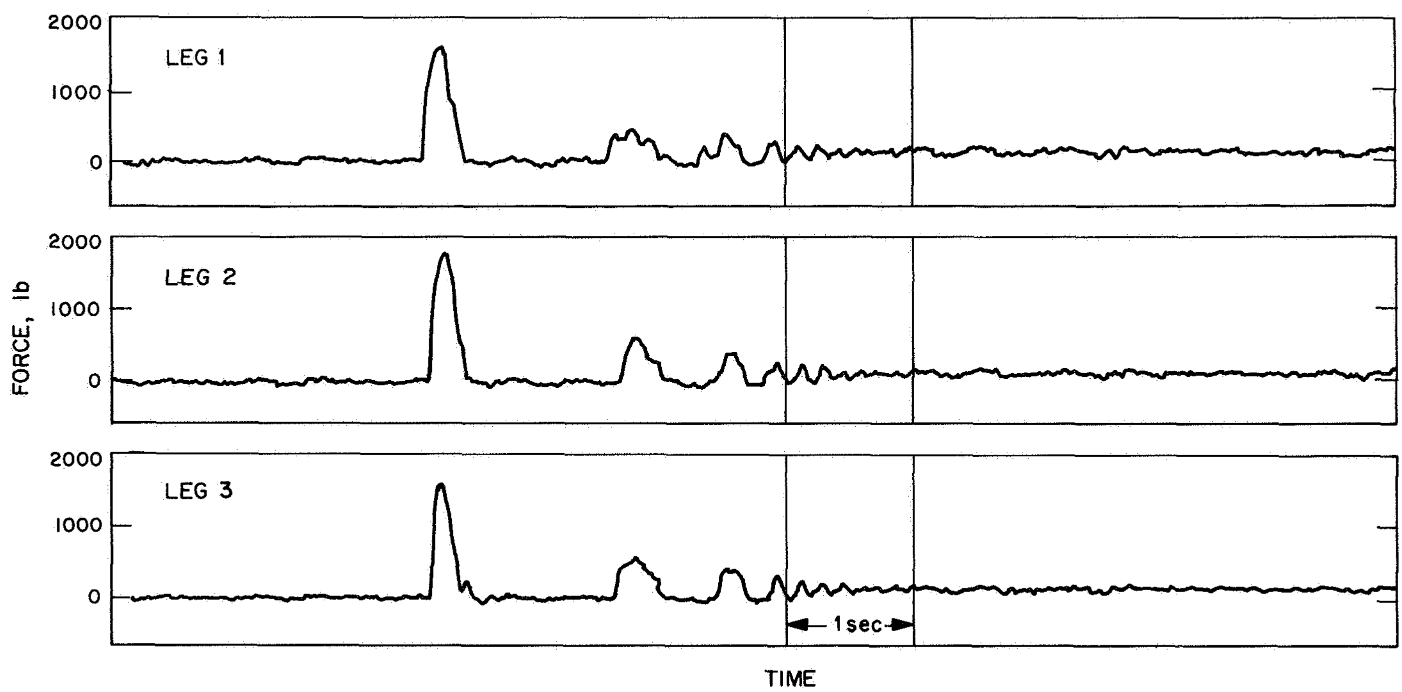


Fig. IV-13. Loads on spacecraft shock absorbers during initial landing

+0.7±0.1 deg. The landed weight of *Surveyor VI* at the time of initial lunar landing is estimated to have been 660.9 lb (earth-weight).

The shock absorber force histories (Fig. IV-13) exhibit an initial high-force period of approximately 0.25-sec duration (very similar to those of *Surveyor I*), followed by a 1.0- to 1.2-sec-long zero-force reading. This reading indicates that the spacecraft rebounded, raising the footpads by 8 to 11 in. from their initial impact positions.

The relative timing of the leg impacts indicates an angle of approximately 2.3 deg between the plane containing the three footpad pivot points at the time of Leg 1 contact and the plane determined by the three footpad imprints. The spacecraft position after landing has been determined to have a pitch angle of 0.0 deg and a yaw angle of 0.8 deg with respect to the lunar vertical. Hence, the spacecraft flight path angle at first touchdown is estimated to have been the indicated 2.3 deg in a direction approximately 25 deg to the left from the pitch direction (looking down), with Leg 1 tilted downward. Accordingly, the horizontal spacecraft velocity is estimated to have been approximately 1.0±0.2 ft/sec in the above given direction (25 deg to the left of Leg 1).

The shock absorbers were not locked after landing, in order to permit shock absorption capability for the post-landing spacecraft hop experiment. The legs were locked prior to sunset and no anomalous leg deflections were observed upon entering the lunar night.

c. Postlanding spacecraft hop experiment. The spacecraft hop experiment was performed approximately 177.5 hr after initial landing. A translation toward the Leg 1 direction was desired; hence, the flight control system was preset such that the closed-loop control system would acquire a +7-deg pitch attitude (i.e., Leg 1 tilted downward by 7 deg) as soon as possible after liftoff. The vernier engines were ignited and commanded to sustain nominal midthrust (i.e., a combined thrust of 150 lb ±20%) until shutdown after 2.5 sec. The horizontal component of the tilted thrust vector provided the desired motion toward Leg 1.

The three strain gages, indicating the axial shock absorber loads, were monitored in addition to commutated gyro and other telemetry signals. The strain gage force histories throughout the hop experiment are shown in Fig. IV-14. Liftoff and landing are clearly evident; engine ignition and shutoff were established from the commutated telemetry.

Using all available telemetry data, an estimated trajectory of the spacecraft hop was obtained analytically. Since the thrust indications are not very accurate, event timing was used as a basis for the trajectory calculation. Three constant thrust phases were assumed: first, 0.5 sec with a constant positive pitch thrust moment; second, 0.5 sec with the reversed thrust moment; and finally, 1.5 sec of moment-free thrust. The combined thrust was assumed constant throughout the burning. The thrust level and differential thrusts were then established such that the indicated timing and pitch angle were satisfied. The pitch angle indicated by the pitch gyro telemetry signal was 6.4±0.15 deg. Based on this value and the timing indicated in Fig. IV-14, the following numerical values describing the translation trajectory were obtained:

Combined thrust, lb	174±3.5
Individual thrust during positive pitch moment phase, lb	
Engine 1	42.9±1.5
Engine 2	61.0±1.5
Engine 3	70.1±1.5
Individual thrust during negative pitch moment phase, lb	
Engine 1	73.1±1.5
Engine 2	55.0±1.5
Engine 3	45.9±1.5
Acceleration at liftoff, (ft/sec)^2	3.17±0.17
Maximum elevation, ft	14.3±1.8
Vertical landing velocity, ft/sec	12.3±0.8
Horizontal landing velocity, ft/sec	1.8±0.2
Total horizontal translation, ft	8.4±0.3

After engine cutoff, the pitch motion was reversed and a negative yaw motion introduced due, probably, to uneven engine cutoff, resulting in a +2-deg pitch and −3.5-deg yaw position shortly before touchdown, relative to the attitude of the spacecraft prior to the hop. Also, a negative roll motion was introduced at engine cutoff, resulting in a net relative roll angle of −4.3 deg after completion of the hop.

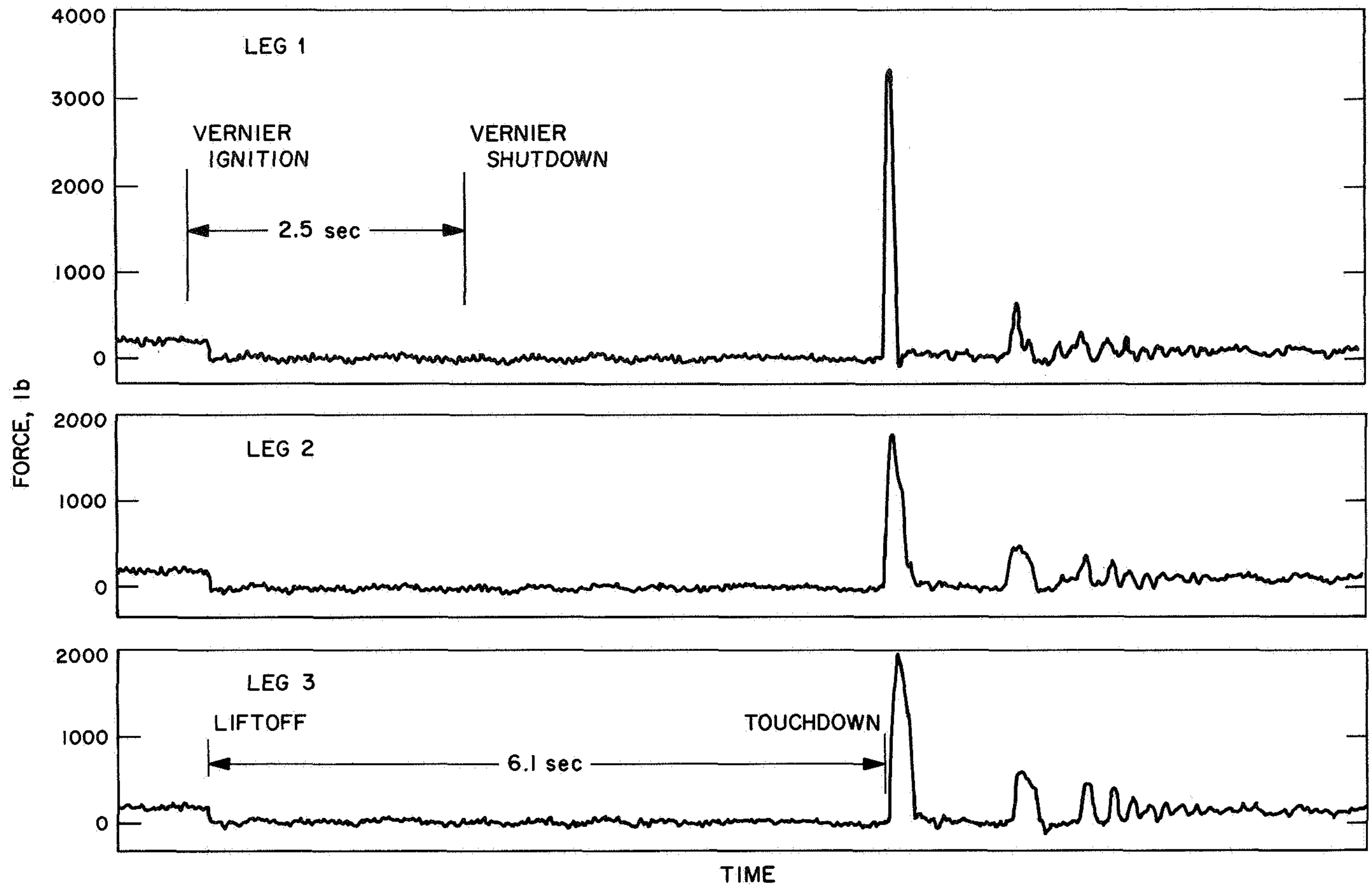


Fig. IV-14. Loads on shock absorbers during spacecraft hop experiment landing

Leg impact timing and maximum shock absorber axial loads for the hop are given in Table IV-9.

The high force encountered by Leg 1, as compared to Legs 2 and 3, can be explained by the horizontal velocity in this direction. However, a high friction force is required to obtain the indicated level in a landing simulation. Therefore, it is assumed that Leg 1 encountered some obstruction constraining its lateral motion during the landing, probably in the form of a local surface unevenness. Approximately 1.5 lb (earth-weight) of vernier propellants were consumed during the translation firing, resulting in a spacecraft weight of 659.4 lb at the end of the hop.

Table IV-9. Shock absorber loads and times of footpad impacts for postlanding hop experiment

	Leg 1	Leg 2	Leg 3
Maximum shock absorber axial force, lb	3350 ± 180	1760 ± 90	1940 ± 90
Time of initial footpad impact, GMT (November 17, 1967)	10:32:11.205	10:32:11.215	10:32:11.235

3. Omnidirectional Antennas

The omnidirectional antennas are mounted on the ends of folding booms hinged to the spaceframe. Pins retain the booms in the stowed position during launch. Squib-actuated pin pullers release the booms upon command from the *Centaur* shortly before spacecraft separation. A leaf-type kickout spring on each omniantenna boom initiates movement. Torsion springs continue the movement to the fully deployed positions, where the booms are automatically locked. On *Surveyor VI,* as on *Surveyor V,* Omniantenna B was painted black to reduce television picture glare. The paint had little or no effect on the thermal control of the spacecraft.

The omniantenna booms were extended by *Centaur* command, and both antennas were locked in the landing or transit position as indicated by telemetry.

4. Antenna and Solar Panel Positioner (A/SPP)

The A/SPP supports and positions the high-gain planar array antenna and solar panel. The A/SPP has four axes of rotation: roll, polar, solar, and elevation (Fig. IV-15). Stepping motors rotate the axes in either direction in response to ground commands or during automatic deployment following *Centaur*/spacecraft separation. This permits orienting the planar array antenna toward the earth and the solar panel toward the sun. Each ground command gives approximately 1/8 deg of rotation about the roll, solar, and elevation axes and approximately 1/15 deg about the polar axis.

The solar axis is locked with the solar panel in a vertical position for stowage in the nose fairing during launch. After spacecraft separation from the *Centaur*, the roll and solar axes are repositioned and locked such

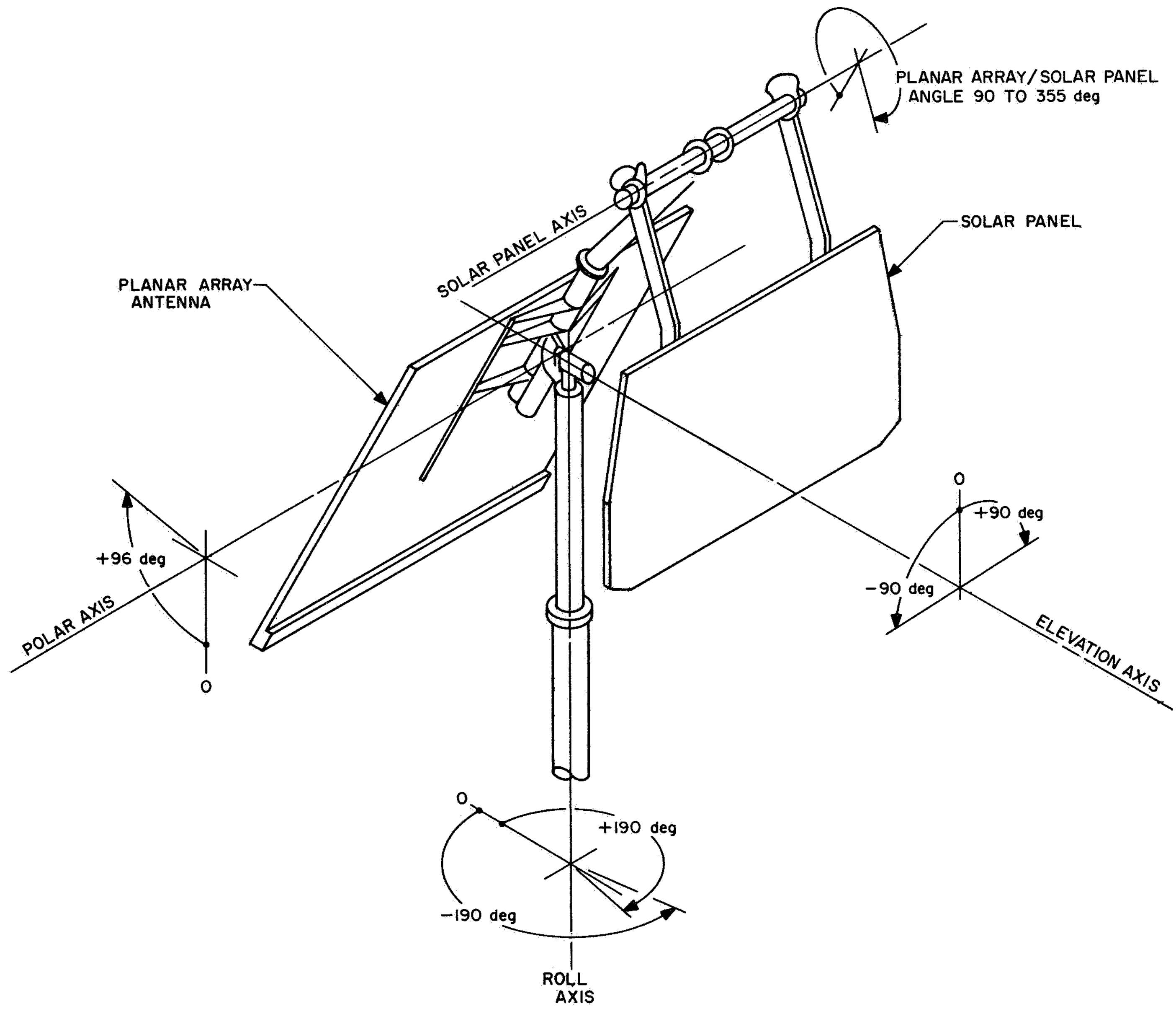


Fig. IV-15. Antenna/solar panel configuration

that the planar array is parallel to Leg 1, and the solar panel is perpendicular to the A/SPP mast. The A/SPP remains locked in this position until after touchdown, at which time the roll, solar, and elevation axes can be released in any order desired to reposition the A/SPP for solar panel acquisition of the sun and planar array acquisition of the earth. Following initial acquisition, the A/SPP is stepped periodically during the lunar day to maintain desired solar panel and planar array positioning relative to the sun and earth. Data from potentiometers on each axis can be used to determine spacecraft lunar orientation.

Surveyor VI used the same A/SPP configuration as *Surveyor V*. The A/SPP was redesigned for *Surveyor V* to increase the structural and operational margins and to incorporate new high-torque stepping motors.

After the *Surveyor VI* spacecraft separated from the *Centaur*, the A/SPP operated as expected during the auto-deploy sequence, and the roll and solar axes relocked.

After touchdown, the following earth command sequence was used to acquire the sun and earth and to determine the spacecraft attitude. The roll- and solar-axes pin pullers were fired (the elevation axis was left locked for later accurate attitude determination and spacecraft hop capability). A first attempt was made to acquire the earth by moving the roll and polar axes and comparing the observed planar array antenna gain with the known antenna pattern. This initial attempt to acquire the earth and sun was successful and was accomplished in 50 min. The solar panel power and sun sensor signals were used as a means of lock-on indication. The sun was acquired very close to its predicted position, indicating that the spacecraft was on a fairly level surface.

Later, after the initial sequence of 600-line TV pictures, the A/SPP was also used to make more accurate attitude determinations in the following manner. The A/SPP was stepped to obtain a precise sun sighting with the secondary sun sensor. A precise earth sighting was obtained by stepping the planar array slowly through the earth position in two orthogonal A/SPP axes and observing potentiometer positions corresponding to maximum DSIF received signal strength. The potentiometer readings were then used in a computer program to obtain an accurate attitude determination.

The A/SPP was repositioned many times during the mission to cool the propulsion system prior to the spacecraft hop experiment. Just prior to the hop, it was moved to a safe position with the polar and solar panel axes in direction of translation, the planar array tilted down 20 deg from horizontal, and the solar panel in approximately the launch position. This was to reduce the possibility of structural damage to the system. The A/SPP apparently survived the hop with no damage. After the hop, the elevation axis was unlocked so that planar array alignment with the earth could be optimized, while allowing the solar panel to be moved so that desired battery charge current could be maintained.

Throughout the lunar day, the A/SPP stepping efficiency appeared to be 100%. During lunar noon, the temperature of the planar array reached a maximum of 296°F, which is 16° above the predicted maximum. The A/SPP was not commanded at this time. The number of degrees of movement of each axis at the end of the first lunar day (November 24) were: solar, 1246; polar, 190; elevation, 27; and roll, 2022.

5. Thermal Compartments

Three thermal compartments (A, B, and C) house the thermally sensitive spacecraft electronic equipment. Compartment C, which was added for *Surveyor V* and also used on *Surveyor VI*, is an insulated and heated box that houses the additional electronics needed to operate the data system associated with the alpha scattering instrument. The main compartments, A and B, contain the components identified in Table IV-10. These components are arranged on thermal trays which distribute heat from the electronic components throughout the compartments.

Table IV-10. Thermal compartment component installation

Compartment A	Compartment B
Receivers (2)	Central command decoder
Transmitters (2)	Boost regulator
Battery	Central signal processor
Battery charge regulator	Signal processing auxiliary
Engineering mechanisms auxiliary	Engineering signal processor
Television auxiliary	Low data rate auxiliary
Thermal control and heater assembly	Thermal control and heater assembly
Auxiliary battery control	Auxiliary engineering signal processor

Each main compartment contains a thermal control and heater assembly to maintain the temperature of the thermal tray above a specified temperature (above 40°F for Compartment A and above 0°F for Compartment B). The thermal control and heater assembly is capable of automatic operation, or may be turned on or off by earth command.

Each main compartment employs thermal switches (nine in Compartment A and six in Compartment B) which are capable of varying the thermal conductance between the inner compartment and the external radiating surface. The thermal switches maintain thermal tray temperature below +125°F. Like the switches used on *Surveyors IV* and *V*, the switches used on *Surveyor VI* were set to open at 35±10°F.

An insulating blanket consisting of 75 sheets of 0.25-mil-thick aluminized mylar is installed between the inner structure and the outer protective cover of each of the main compartments. The thermal blankets and protective covers for Compartments A and B, which were redesigned for *Surveyor V* to be made in two pieces, were also used on *Surveyor VI*. One piece covers the back and bottom of the compartments; the other piece covers the top, ends, and front. The redesign provides easier access to the electronic components during all phases of systems checkout and much simpler final compartment installation. The blanket for Compartment A, as redesigned, allows easier cable access into the electronics during STV tests.

6. Thermal Switch

The thermal switch is a thermal-mechanical device which varies the conductive path between an external radiation surface and the top of the equipment tray of each compartment (Fig. IV-16). The switch has two contact surfaces which are ground to within one wavelength of being optically flat. One surface is coated with a conforming substance to form an intimate contact with the mating surface. To reduce the contact sticking condition experienced on the *Surveyor I* and *III* missions, the coating surface, which consists of a room-temperature-curing (RTV) silicone compound, is vacuum-baked at 300°F to drive off the volatiles. The coating surface is also charged with molybdenum disulfide, which serves as a parting agent. These processes were employed in the hope that they would increase the opening reliability of the thermal switches significantly above that of the switches used on *Surveyors I* through *III*. Actuation of the thermal switch is accomplished by four bimetallic elements located at the base of the switch. These elements are connected mechanically to the top of the compartment so that switch actuation is controlled by compartment temperature.

The external radiator surface is such that it absorbs only 12% of the solar energy incident on it and radiates 74% of the heat energy conducted to its surface. When the switch is closed and the compartment is hot, the switch loses its heat energy to space. When the compartment gets cold, the switch contacts are designed to open about 0.020 in., thereby opening the heat-conductive path to the radiator and thus reducing the heat loss through the switch to almost zero. Four Compartment A and three Compartment B switches are instrumented with temperature sensors attached to the radiators. The temperature sensors provide an indication of switch actuation for those switches which are monitored.

On the *Surveyor VI* mission, the thermal switches were closed during flight and kept the electronics at or below the maximum allowable temperature at all times. All the thermal switches also remained closed during the lunar day after landing.

Based on a heat balance analysis made on each compartment, it was determined that at least eight of the nine thermal switches on Compartment A and two of the six switches on Compartment B were stuck as the spacecraft entered the lunar night. At spacecraft shutdown 41 hr after sunset, six switches on Compartment A and one on Compartment B remained closed. At least six of the switches (three on each compartment) which were open at shutdown had opened out of the specified temperature range.

7. Pyrotechnic Devices

The pyrotechnic devices installed on *Surveyor VI* are listed in Table IV-11. All the squibs used in these devices are electrically initiated, hot-bridgewire, gas-generating devices. Qualification tests for flight squibs included demonstration of reliability at a firing current level of 4 or 4.5 A. "No fire" tests were conducted at a 1-A or 1-W level for 5 min. Electrical power required to initiate pyrotechnic devices is furnished by the spacecraft main battery. Power distribution is through 19.0- and 9.5-A constant-current generators in the engineering mechanism auxiliary (EMA).

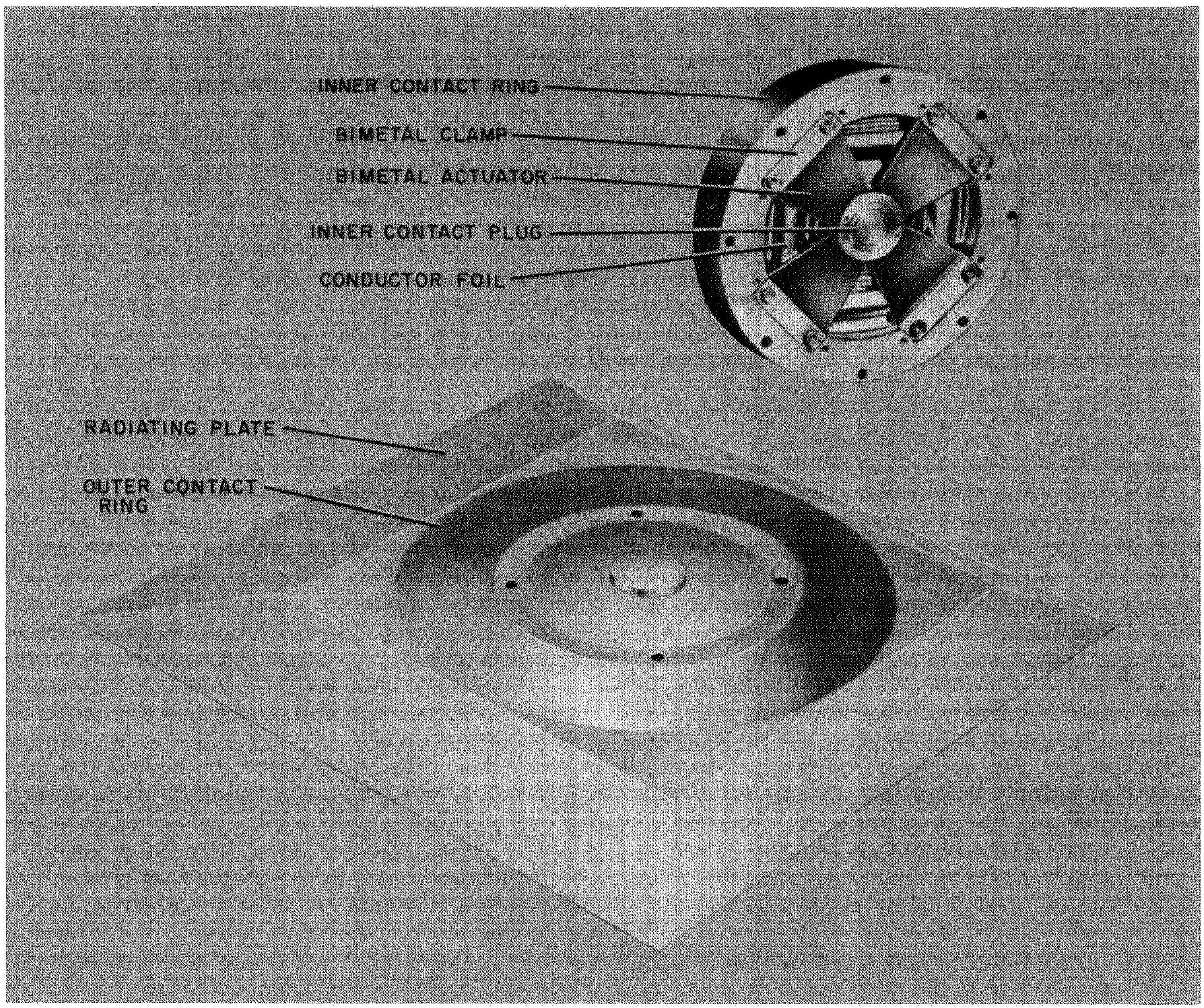


Fig. IV-16. Thermal switch

Table IV-11. Pyrotechnic devices

Type	Location and use	Quantity of devices	Quantity of squibs	Command source
Pin pullers	Lock and release Omniantennas A and B	2	2	*Centaur* programmer
Pin pullers	Lock and release landing legs	3	3	*Centaur* programmer
Pin pullers	Lock and release planar antenna and solar panel	7	7	Separation sensing and arming device and ground station
Pin pullers	Lock and release Vernier Engine 1	1	1	DSIF station
Pin pullers	Alpha scatter unlatch and lower mechanism	2	2	DSIF station
Separation nuts	Retro rocket attach and release	3	6	Flight control subsystem
Valve	Helium gas release and dump	1	2	DSIF station
Pyrotechnic switches	EMA Board 4, RADVS power on and off	4	4	DSIF station and flight control subsystem
Initiator squibs	Safe-and-arm assembly retromotor initiators	1	2	Flight control subsystem
Locking plungers	Landing leg, shock absorber locks	3	3	DSIF station
		27	32	

All the taper pin assemblies associated with the A/SPP pin pullers were of the new design first incorporated on *Surveyor V*. They were all made heavier in conjunction with the A/SPP redesign, which was required in order to obtain a positive structural margin. The basic operating characteristics of the pin pullers were not changed, although the axial degree of freedom of the pins used to lock the solar axis during launch was limited to keep them from backing out due to vibration loads encountered during the launch phase of the flight.

All pyrotechnic devices functioned normally upon command. Mechanical operation of most of the squib-actuated locks, valves, switches, and pins are indicated on telemetry signals as part of the spacecraft engineering measurement data.

The shock absorber lock pins were not fired at the normal time in order to maintain the postlanding capability to fire the vernier engines. The command to fire the locks was transmitted soon after the spacecraft hop was performed, and the devices functioned normally as indicated by the lack of leg sag as the lunar night approached.

8. Electronic Packaging and Cabling

The electronic assemblies provide mechanical support for electronic components in order to ensure proper operation throughout the various environmental conditions to which they are exposed during a mission. The assemblies (or control items) are constructed utilizing sheet metal structure, sandwich-type etched circuit board chassis with two-sided circuitry, plated-through holes, and/or bifurcated terminals. In general, each control item consists of only a single functional subsystem and is located either in or out of the three thermally controlled compartments, depending on the temperature sensitivity of the particular subsystem. Electrical interconnection is accomplished primarily through the main spacecraft harness. The cabling system utilizes lightweight, minimum-bulk, and abrasion-resistant wire which is constructed of extruded teflon with a dip coating of modified polyamide.

C. Thermal Control

The thermal control subsystem is designed to provide acceptable thermal environments for all components during all phases of spacecraft operation.

1. Configuration and Functional Description

Spacecraft items with close temperature tolerances were grouped together in thermally controlled compartments. Those items with wide temperature tolerances were thermally decoupled from the compartments. The thermal design fits the "basic bus" concept in that the design was conceived to require minimum thermal design changes between missions. Monitoring of the performance of the spacecraft thermal design is provided by engineering temperature sensors which are distributed throughout the spacecraft. On the *Surveyor VI* spacecraft, 76 temperature sensors were located within the subsystems as follows:

Flight control	9
Mechanisms	3
Radar	6
Electrical power	4
Transmitters	2
Television	4
Vehicle structure	27
Propulsion	19
Alpha scattering instrument	2

The spacecraft thermal control subsystem is designed to function in the space environment, both in transit and on the lunar surface. Extremes in the environment as well as mission requirements on various components of the spacecraft have led to a variety of methods of thermal control. The spacecraft thermal control design is based upon the absorption, generation, conduction, and radiation of heat.

The radiative properties of the external surfaces of major items are controlled by using paints, by polishing, and by using various other surface treatments. Reflecting mirrors are used to direct sunlight to certain components. In cases where the required radiative isolation cannot be achieved by surface finishes or treatments, the major item is covered with an insulating blanket composed of multiple-sheet aluminized mylar. This type of thermal control is called "passive" control.

The major items whose survival or operating temperature requirements cannot be achieved by surface finishing or insulation alone use heaters that are located within the unit. These heaters can be operated by external command, thermostatic actuation, or both. The thermal control design of those units using auxiliary heaters also includes the use of surface finishing and insulating blankets to optimize heater effectiveness and to minimize the electrical energy required. Heaters are considered "active control."

Items of electronic equipment whose temperature requirements cannot be met by the above techniques are located in thermally controlled compartments. Each of the main compartments (A and B) is enclosed by a shell covering the bottom and four sides and contains a structural tray on which the electronic equipment is mounted. The top of each compartment is equipped with a number of temperature-actuated switches (nine in Compartment A and six in Compartment B). These switches, which are attached to the top of the tray, vary the thermal conductance between the tray and the outer radiator surfaces, thereby varying the heat-dissipation capability of the compartments. When the tray temperature increases, heat transfer across the switch increases. During the lunar night, the switch opens, decreasing the conductance between the tray and the radiators to a very low value in order to conserve heat. When dissipation of heat from the electronic equipment is not sufficient to maintain the required minimum tray temperature, a heater on the tray supplies the necessary heat. The switches are considered "semiactive."

Examples of units which are controlled by active, semiactive, or passive means are shown in Fig. IV-17.

2. *Surveyor VI* Performance

Surveyor VI did not experience any thermal anomalies during the transit phase of flight. Of the 74 temperature sensors monitored during transit, 69 held within 10°F of their predicted nominal steady-state values. Vernier Oxidizer Line 1 ran hotter than on all previous spacecraft but remained within specified limits. Plots of selected transit temperatures are contained in Figs. D-1 through D-8 of Appendix D.

After *Surveyor VI* landed, the temperatures of most components were as expected except for the alpha scattering instrument, which was warmer than anticipated. From the time of landing until the spacecraft hop was performed, the helium check and relief valves and certain propellant tanks were shaded as far as possible by A/SPP positioning. Prior to the hop experiment, other critical components were also shaded by suitable positioning of the A/SPP to reduce the temperatures below their upper operating limits. These additional units included the vernier engines, flight control electronics, and gyros. The shading of critical vernier propulsion units was continued after the hop until a helium leak made any further vernier firings impossible. Selected postlanding temperatures for the first lunar day are plotted in Figs. D-9 through D-11 of Appendix D.

The battery, located in Compartment A, was maintained below 75°F during the entire lunar morning.

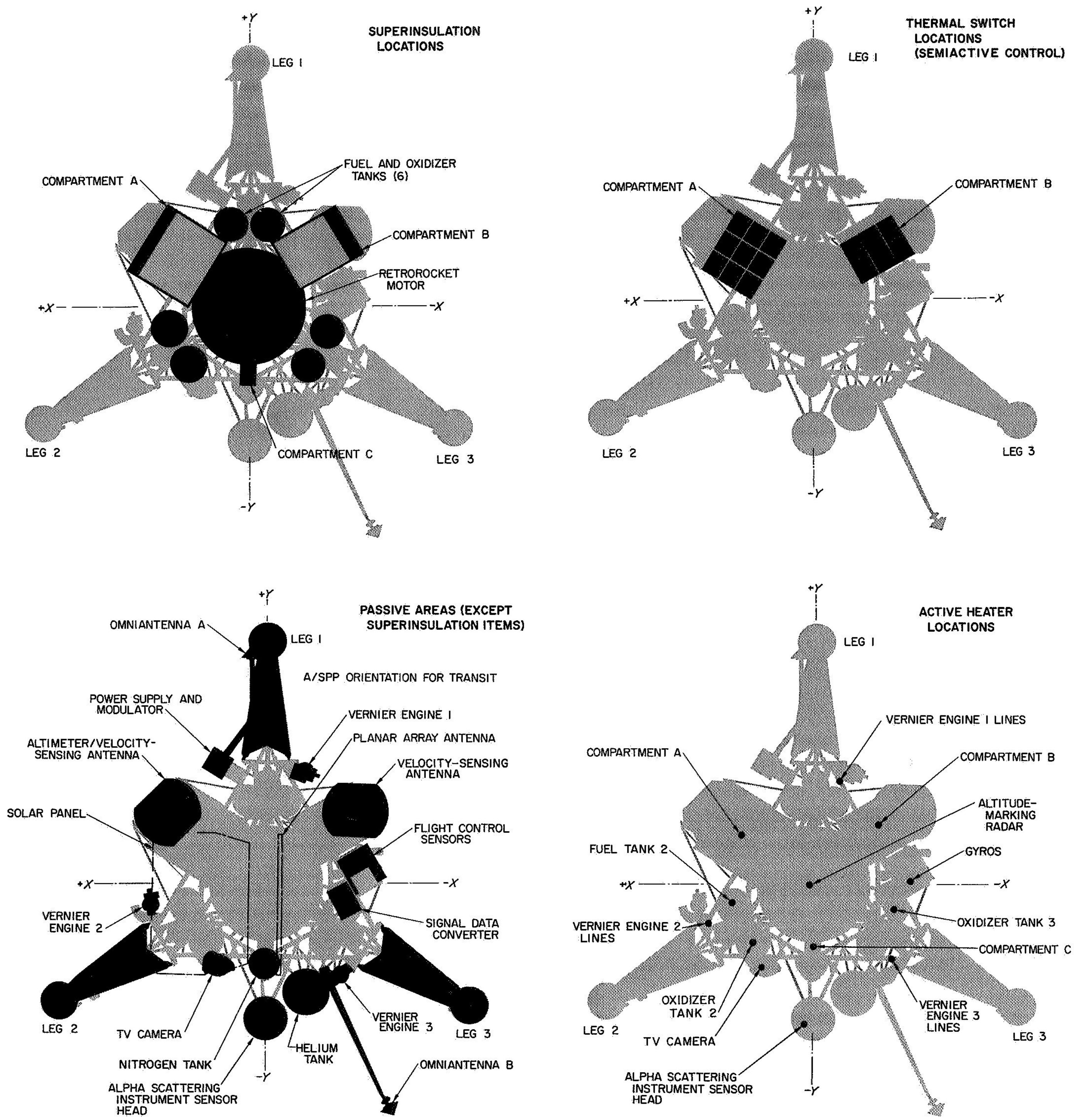


Fig. IV-17. Thermal design

During the afternoon, when Compartment A could not be shadowed without loss of earth-lock, the main battery reached 115°F twice. Compartment B never exceeded its operational limits, the maximum temperatures observed being 115°F for the lower thermal tray and 111°F for the upper thermal tray.

The television system was maintained within operational temperature limits throughout the entire lunar day, although it was necessary to go to the standby mode during the lunar noon period. Television camera thermal performance was the best yet experienced, the camera demonstrating the capability to operate almost continuously with interruptions for engineering interrogations on approximately 2-hr intervals.

After sunset, in accordance with an established operating plan designed to maximize the chance for spacecraft survival of the lunar night, Compartments A and B were allowed to cool to 20 and −15°F, respectively. Many of the thermal switches on both compartments were stuck closed, and the battery power required to maintain these temperatures was too great to allow 200 hr of night operation as had been projected. Under a modified plan, the battery temperature was allowed to fall toward zero, with the intention of turning off the spacecraft temporarily when the battery reached 0°F. The last interrogation was about 40 hr into the night when the battery read +1°F; at the next attempted interrogation, the spacecraft did not respond.

Since so many thermal switches were stuck closed when the spacecraft was last interrogated, no accurate estimate was possible of the warmup profile for the second lunar morning. Presumably, the thermal switches opened during the night, allowing a partial recovery of the spacecraft on December 14, 1967, during the second lunar day (see Section VI-C, Mission Operations Chronology).

D. Electrical Power

The electrical power subsystem is designed to store, convert, and distribute electrical energy. Effective first on the *Surveyor V* mission, the power subsystem was redesigned to provide better matching characteristics between power sources and power conditioning units. By this means, a number of improvements were introduced into the subsystem which increased the capability, reliability, and overall power performance.

1. Power Subsystem Description

A block diagram of the subsystem used beginning with the *Surveyor V* mission is shown in Fig. IV-18.

The spacecraft system derives its energy from two sources, a solar panel which converts solar radiation energy to electrical energy and a rechargeable battery, of which only one is employed in the redesigned subsystem.

Several improvements made in the solar panel design for *Surveyor V* and later missions provide increased efficiency and improved performance of the subsystem. A series-parallel cell arrangement was incorporated to obtain maximum power output at 30.5±2 V during transit. In this arrangement, open-circuit voltage of the solar panel ranges from 65 V after sunrise on the lunar surface to 32 V at lunar noon. Flat-cell rather than shingle-cell mounting is used, thus providing more uniform bonding of the cells to the substrate for better performance over a wider temperature range.

The lower output voltage of the redesigned solar panel permits use of a simplified battery charge regulator (BCR) and switching of the solar panel directly to: (1) the 30-V preregulated bus in the boost regulator (BR) during transit, or (2) the 22-V unregulated bus during lunar operations, when the temperature is higher and the solar panel voltage is lower. The power subsystem is in its most efficient mode of operation when the solar panel is connected to the 30-V preregulated bus. (This mode of operation was not possible in the previous design.) The unregulated bus is used for charging of the battery and for distribution of current from the BCR and battery to the unregulated spacecraft loads and the BR. The voltage on the unregulated bus can vary between 17.5 and 27.5 V, with a nominal value of 22 V. The BCR also has an *off* mode in which the solar panel is left floating. The *off* mode can result from earth command or automatically if the battery voltage exceeds 27.3 V or battery pressure exceeds 65 psia. The automatic turnoff feature of the BCR can be enabled or disabled by earth command.

The BR converter boosts the unregulated bus voltage, the output of which is used by the BR preregulator to supply 30 V to the preregulator bus. The essential loads are fed by the preregulated bus through a series diode which drops the preregulated bus voltage to the essential bus voltage of 29 V. The preregulator bus also feeds

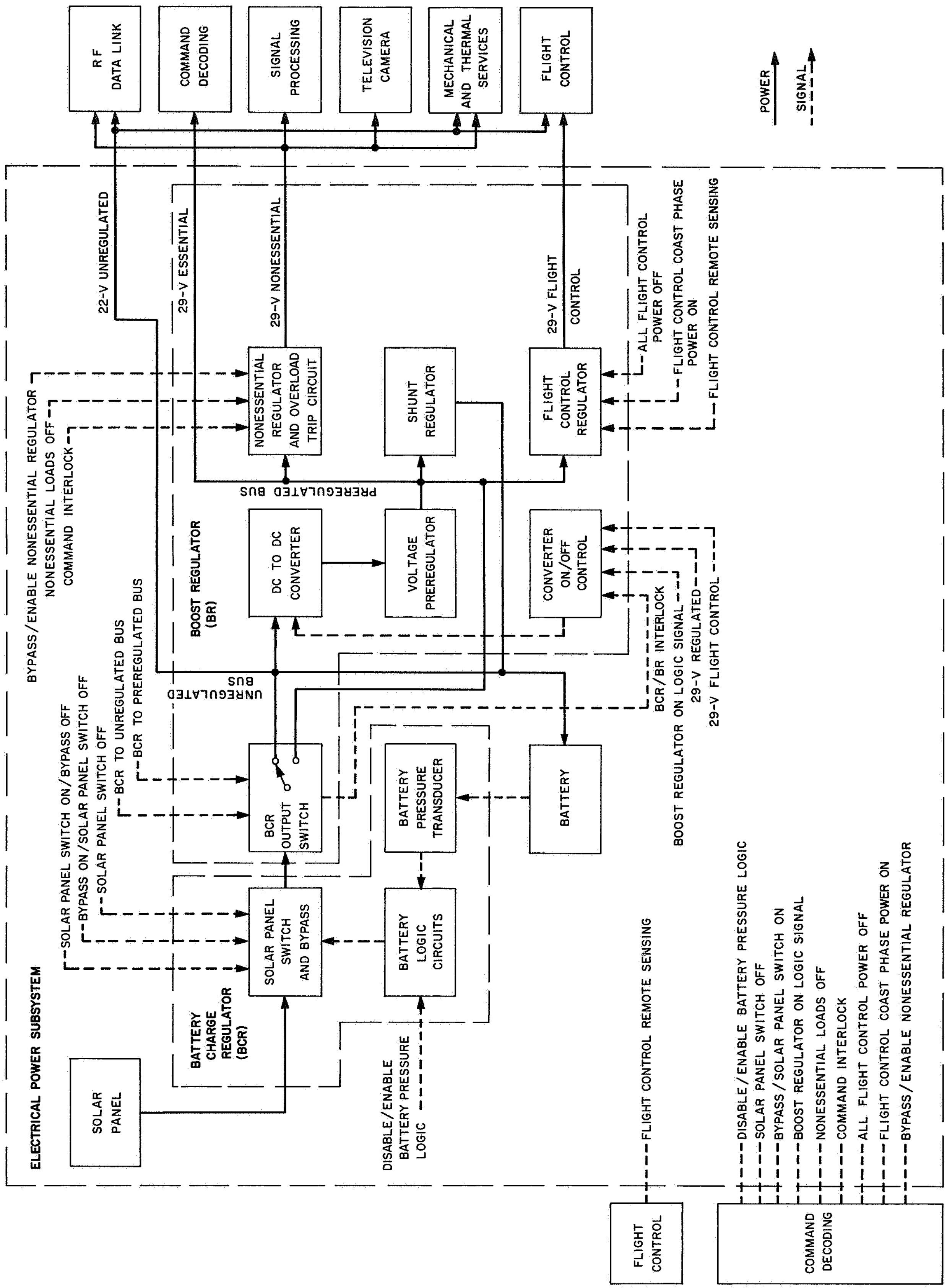


Fig. IV-18. Functional diagram of *Surveyor VI* electrical power subsystem

the flight control regulator and the nonessential regulator, which in turn feed the flight control and nonessential buses, respectively. These regulators can be turned on and off by earth commands. The nonessential regulator has a *bypass* mode which is used to connect the preregulated bus directly to the nonessential bus if the nonessential regulator fails.

The BR operates at higher efficiency as a result of the simplified BCR. The higher efficiency is especially pronounced during transit because all the regulator current does not have to pass through the preregulator. In addition, a shunt regulator is connected between the preregulated bus and the battery to supply excess solar panel current to the battery and prevent excess solar panel voltage when the spacecraft current demands are low during transit. In the previous design, the shunt output was connected to battery ground so that no battery charging was possible.

In the redesigned power subsystem, the maximum battery storage capacity is 180 A-hr. The battery provides about 2000 W-hr during transit, the balance of the energy being supplied by the solar panel. Nominal expected battery energy remaining at touchdown is approximately 2000 W-hr.

2. Power Subsystem Performance

Performance of the electrical power subsystem was very satisfactory during the transit and postlanding phases of the *Surveyor VI* mission.

Expected and actual values of several power system parameters during the transit phase are presented in Table IV-12. The battery energy at liftoff was 3608 W-hr (165 A-hr), which was the predicted nominal value. Following solar panel deployment after spacecraft separation, solar panel voltage was 30.5 V with a current of 2.51 A. The BCR was connected to the preregulated bus and remained in that position until after touchdown. Solar panel output power was about 4% above nominal, which can be attributed in part to greater-than-nominal solar intensity for the September 1967 launch period. The solar panel switch tripped out automatically owing to high bus voltage about 9 hr after liftoff. About 10 min later, the *solar panel on* command was transmitted and the switch stayed on. During the next 6 hr, the switch tripped out 5 more times. This was normal operation of the BCR overvoltage-protection circuit resulting from battery charging during periods of minimum heater loads, and had been predicted from results of spacecraft tests conducted in a solar–thermal–vacuum chamber. Average solar panel current was 2.88 A at an average of 30.2 V during transit. Average BCR output current during the coast phase was 2.88 A compared to 2.56 A predicted. Although the BCR efficiency is not measured directly in flight, it was estimated to be close to the expected value of 94.6%. During high power operation, the regulated loads were 3.9 A vs the predicted value of 3.3 A. The BR efficiency of low- and high-power modes was 81% and 86%, respectively.

Table IV-12. Electrical power performance during transit

Parameter	Time period	Predicted	Actual
Battery energy, A-hr	Liftoff	165	165
	Touchdown	100	107
Average solar panel power, W	Coast phases	84	87
Average BCR output current, A	Coast phases	2.56	2.88
BCR efficiency, %	Coast phases	94.6	94.6
Regulated loads, A	High power	3.3	3.9
BR efficiency, %	Low power	80.5	81
	High power		86

The average battery discharge current was 0.21 A during the first coast period; this was approximately 0.2 A lower than predicted owing to higher-than-anticipated solar panel output. Battery temperature averaged 76°F prior to midcourse, and battery pressure remained low, never exceeding 14.8 psia during the entire transit phase. The battery charge level followed the predicted profile closely, although the charge level remained slightly above predicted as illustrated in Fig. IV-19. Battery energy remaining at touchdown was estimated to be 2350 W-hr (107 A-hr).

After touchdown the power subsystem continued to perform satisfactorily. The two requirements imposed on the power system during lunar day operations are: (1) to supply the power demands to perform the engineering and scientific operations, and (2) to recharge the battery to assure a fully charged battery prior to lunar sunset.

Typical spacecraft regulated current requirements to support postlanding operations, including alpha scattering instrument and television camera operation, are shown in Fig. IV-20.

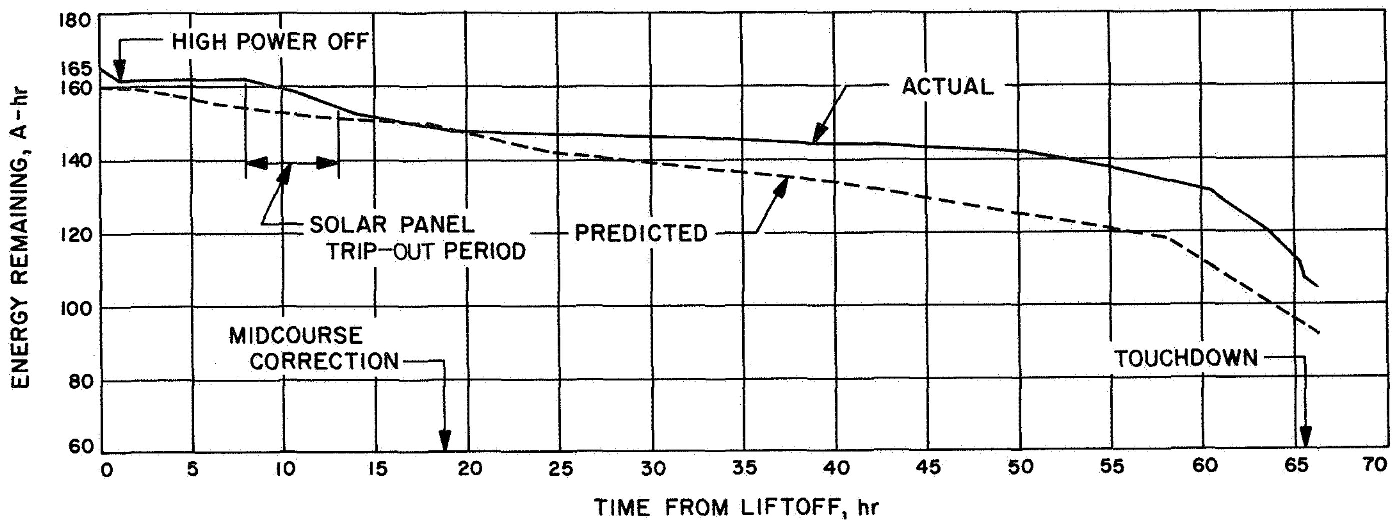


Fig. IV-19. Calculated battery energy profile during transit

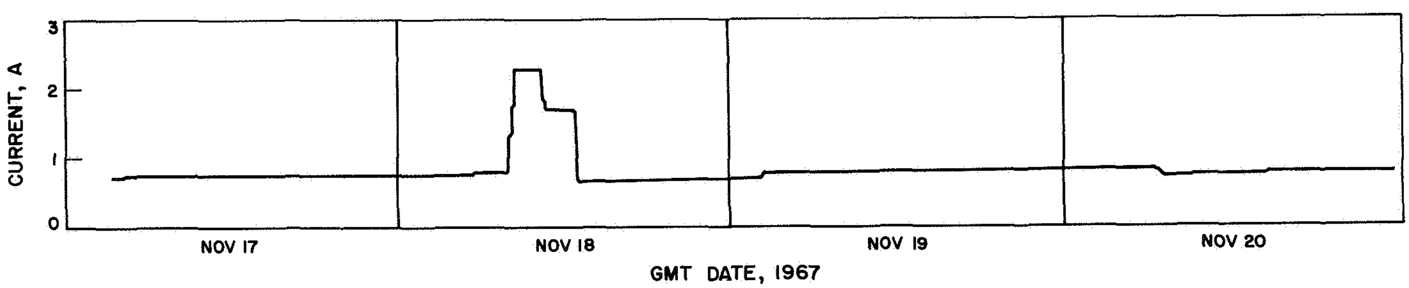


Fig. IV-20. Typical postlanding regulated current requirements

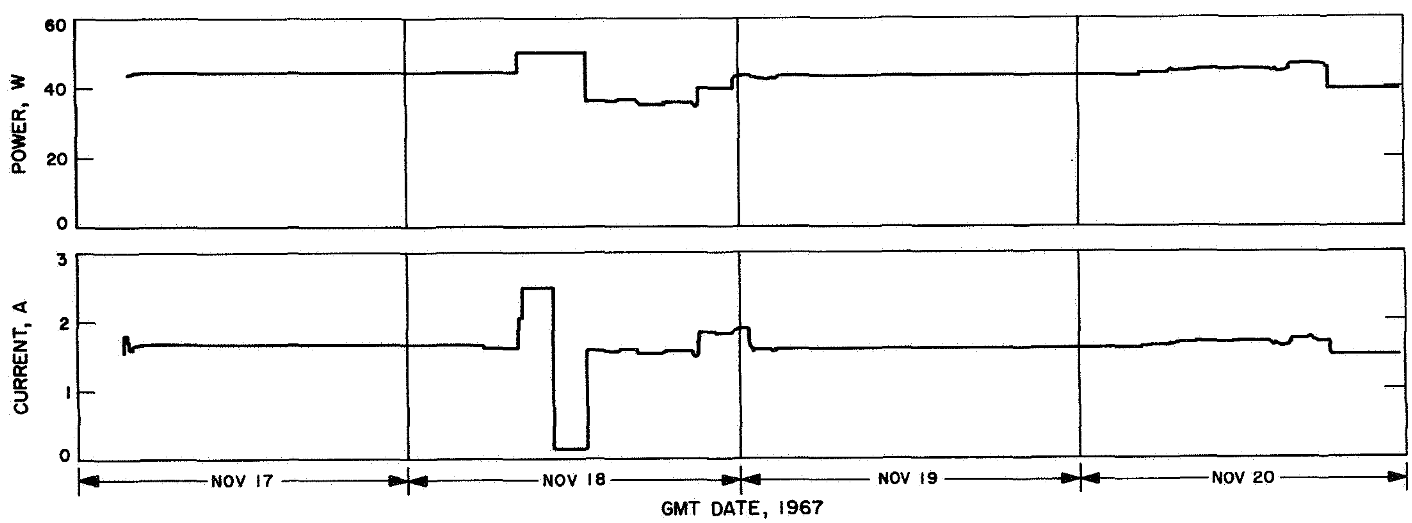


Fig. IV-21. Typical postlanding solar panel current and power output

Solar panel performance continued to be satisfactory throughout the lunar day. In addition to supplying power to the spacecraft for lunar operations, the solar panel was utilized to shade the compartments, the alpha scattering electronics and sensor head, and TV camera in order to provide a more comfortable environment. Figure IV-21 shows the solar panel current and the calculated power output of the panel during a typical period of the lunar day. With about 67% charge remaining in the battery at touchdown, the solar panel was not positioned directly on the sun until the last 24 hr of the lunar day. The solar panel was positioned at all times from approximately 15 to 45 deg ahead of the sun to prevent solar panel current from exceeding the desired battery charge rate.

No problems were encountered during performance of the spacecraft hop experiment. The battery was the sole source of power during this period and performed as anticipated. Battery performance during the first lunar day is shown in Fig. IV-22. The battery reached a maximum charge of 165 A-hr before sundown.

The prime consideration during lunar night operation was to spend the battery energy wisely in order to accomplish the following:

(1) Maintain the battery temperature at +20°F as long as possible by dissipating power in Compartment A until the battery was depleted to 45 A-hr. The 45 A-hr limit was determined from the results of lunar night survival tests. It represented (1) the energy storage level necessary to supply the spacecraft constant load requirements during cooldown and warmup of the battery to and from the temperature at which it would cease to provide power (zero-volt level), and (2) the minimum energy level necessary to prevent cell reversal.

(2) Prevent the battery temperature from dropping below −175°F. In tests performed at JPL, three batteries survived after exposure to a simulated lunar night environment at a temperature of −175°F.

Several Compartment A thermal switches were inoperative. Therefore the battery temperature and power consumption plan was abandoned and spacecraft shutdown occurred only about 40 hr after sunset.

During the brief operation of *Surveyor VI* the second lunar day, sufficient usable data was not obtained to assess the condition of the power system.

E. Propulsion

The propulsion subsystem supplies thrust force during the midcourse correction and terminal descent phases

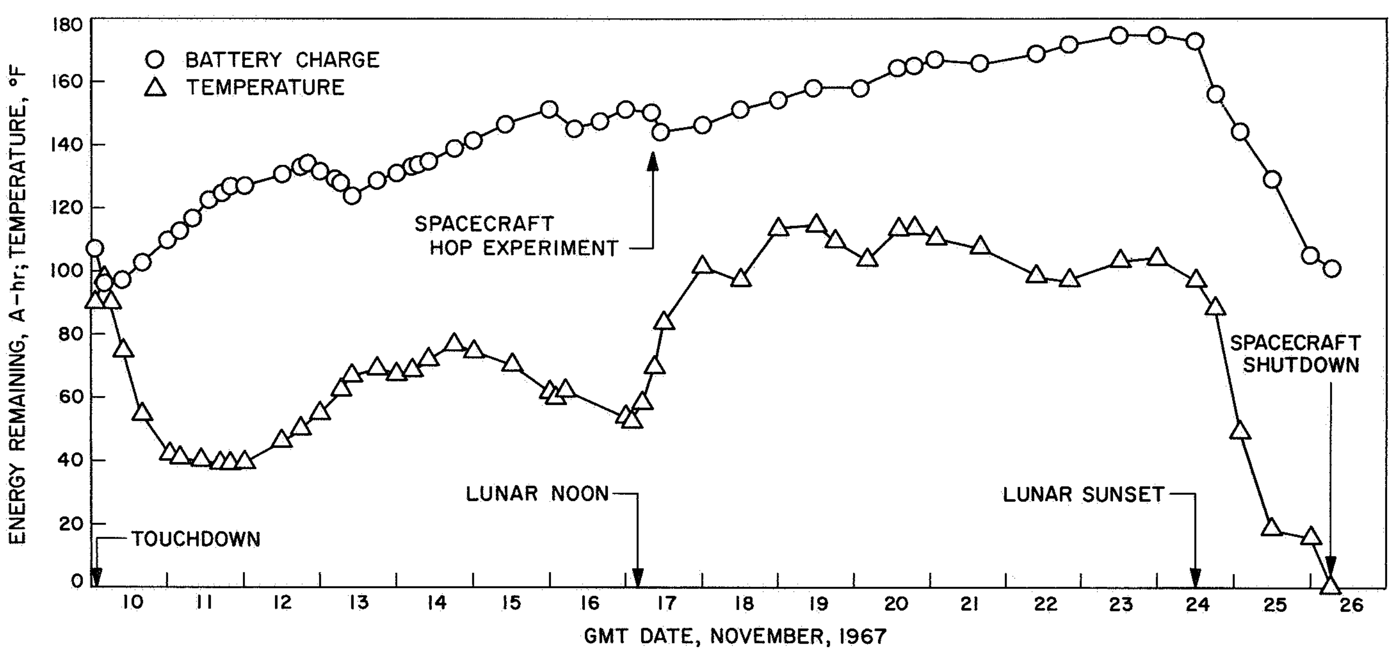


Fig. IV-22. Postlanding battery temperature and energy level

of the mission. The propulsion subsystem consists of a vernier engine system and a solid-propellant main retrorocket motor. The propulsion subsystem is controlled by the flight control system through preprogrammed maneuvers, commands from earth, and maneuvers initiated by flight control sensor signals.

1. Vernier Propulsion

The vernier propulsion subsystem supplies the thrust forces for midcourse maneuver velocity vector correction, attitude control during main retrorocket motor burning, and velocity vector and attitude control during terminal descent.

a. Vernier propulsion description. The vernier engine system consists of three thrust chamber assemblies and a propellant feed system, as shown in the schematic of Fig. IV-23. The feed system is composed of three fuel tanks, three oxidizer tanks, a high-pressure helium tank, propellant lines, and valves for system arming, operation, and deactivation.

Fuel and oxidizer are contained in six tanks of equal volume, with one pair of tanks for each engine. Each tank contains a teflon expulsion bladder to permit complete and positive expulsion and to assure positive control under zero-g conditions. The oxidizer is nitrogen tetroxide (N_2O_4) with 10% by weight nitric oxide (NO) to depress the freezing point. The fuel is monomethyl hydrazine monohydrate ($MMH \cdot H_2O$). Fuel and oxidizer ignite hypergolically when mixed in the thrust chamber. The total minimum usable propellant load is 178.3 lb. The arrangement of the tanks on the spaceframe is illustrated in Fig. IV-24. Propellant freezing or overheating is prevented by a combination of active and passive thermal controls, utilizing surface coatings, multilayered blankets, and electrical and solar heating. The propellant tanks are thermally isolated to insure that the spacecraft structure will not function as a heat source or heat sink.

Propellant tank pressurization is provided by the helium tank and high-pressure valves assembly (Fig. IV-25). The high-pressure helium is released to the propellant tanks by activating a squib-actuated helium release valve. A single-stage regulator maintains the propellant tanks at a nominal working pressure of about 730 psi. Helium check and relief valves are located in a separate package on the spaceframe and are connected by a single line to the helium tank and high-pressure valves assembly. This represents a slight modification of the system used on early missions (*Surveyor I* through *IV*), in which the low-pressure valves were mounted directly on the high-pressure valves assembly. On the *Surveyor V* and *VI* system, a filter was also added to the inlet part of each relief valve to protect against contamination. The check valves allow the flow of pressurizing helium to the propellant tanks but prevent the back flow of helium and propellant vapors from the propellant tanks to the pressure regulator or between fuel and oxidizer tanks. Helium relief valves relieve excess pressure from the propellant tanks in the event of a helium pressure regulator malfunction.

The thrust chambers (Fig. IV-26) are located near the hinge points of the three landing legs on the bottom of the main spaceframe. The moment arm of each engine is about 38 in. Engine 1 can be rotated ± 6 deg about an axis in the spacecraft X–Y plane for spacecraft roll control. The Engine 1 roll actuator is unlocked by the same command that actuates the helium release valve. Engines 2 and 3 are not movable. The thrust of each engine (which is monitored by strain gages installed on each engine mounting bracket) can be throttled over a range of 30 to 104 lb. The specific impulse varies slightly with engine thrust.

Changes were made to obtain additional vernier propulsion system data on *Surveyor V* and *VI* by adding a pressure transducer to Fuel Line 2 and temperature sensors to each of the fuel lines.

b. Vernier propulsion performance. The *Surveyor VI* vernier propulsion system performance was completely nominal during midcourse correction and terminal descent. The vernier propulsion system also performed successfully in the spacecraft hop experiment shortly after lunar noon.

Vernier propulsion system parameters during launch and the premidcourse coast period were all normal and well within their allowable limits. After completion of the pre-midcourse attitude maneuvers, vernier pressurization was commanded in accordance with the normal mission sequence. Vernier propellant tank pressurization is accomplished by firing the helium release squib and allowing high-pressure helium gas into the inlet port of the helium pressure regulator. Pressurization occurred normally; the regulator locked-up at 764 psia, its normal lock-up pressure, which is well below the 825 psia relief valve cracking pressure. The propellant tank and helium tank pressures were stable and showed no signs of leakage.

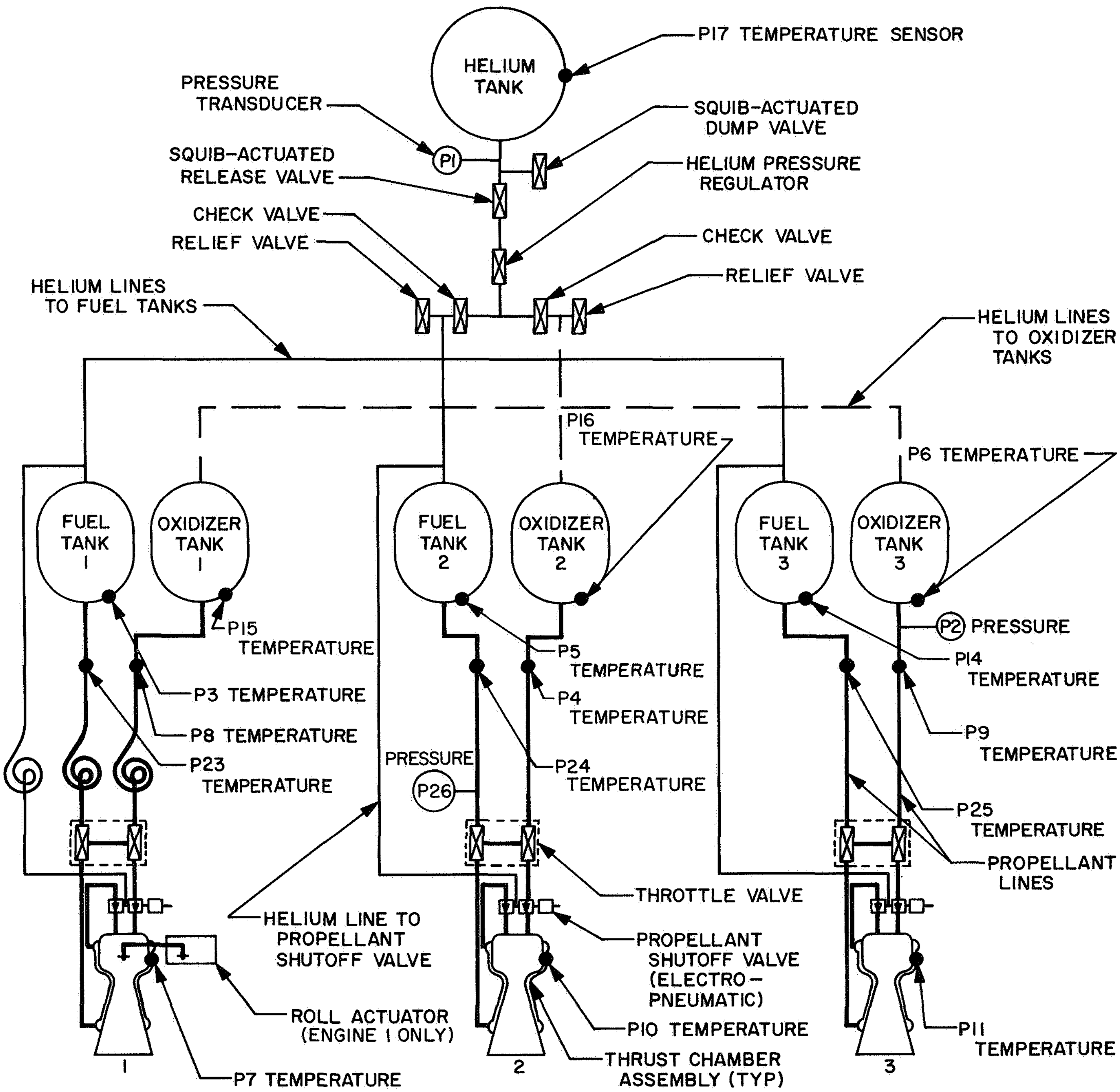


Fig. IV-23. Vernier propulsion system schematic

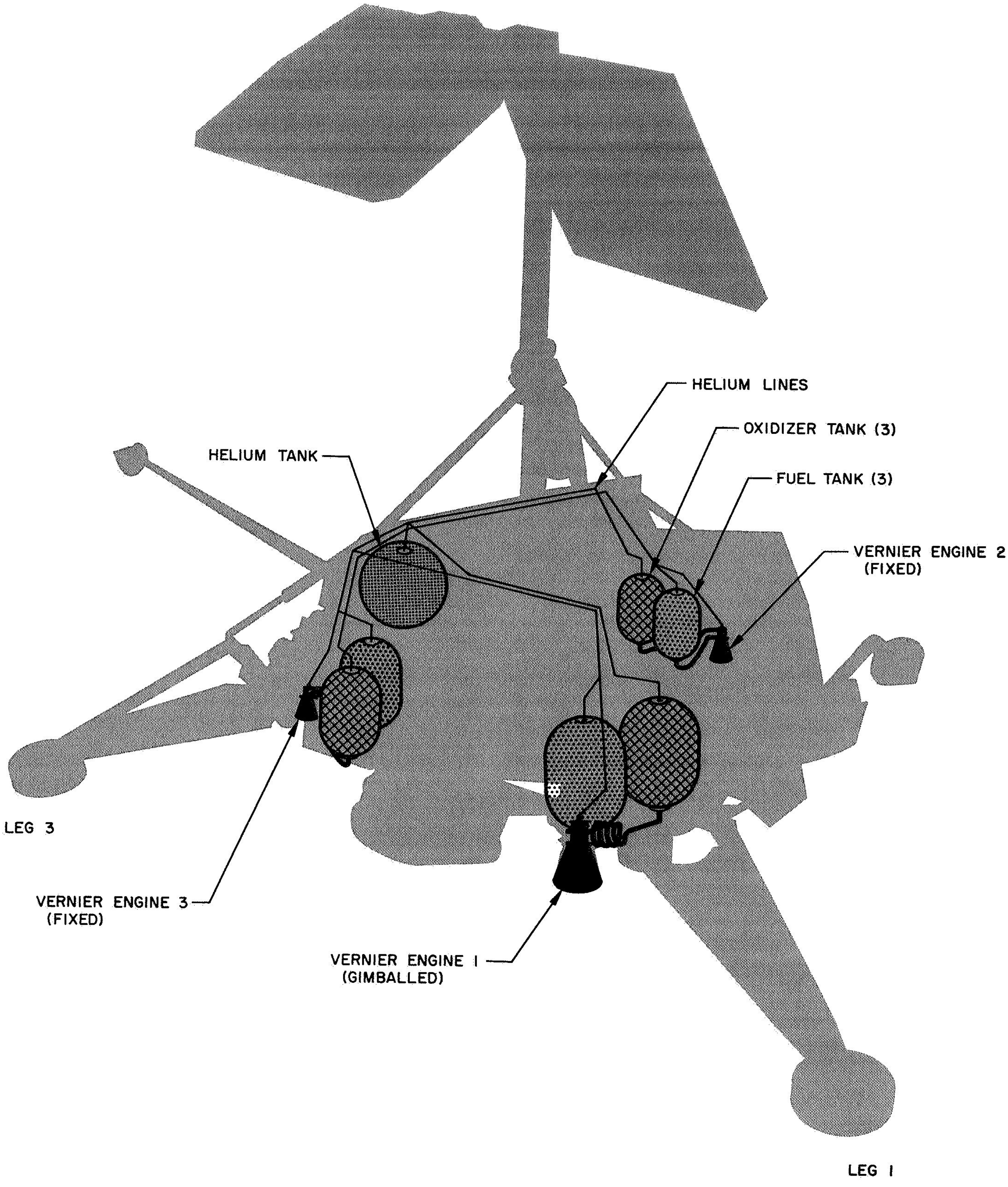


Fig. IV-24. Vernier propulsion system installation

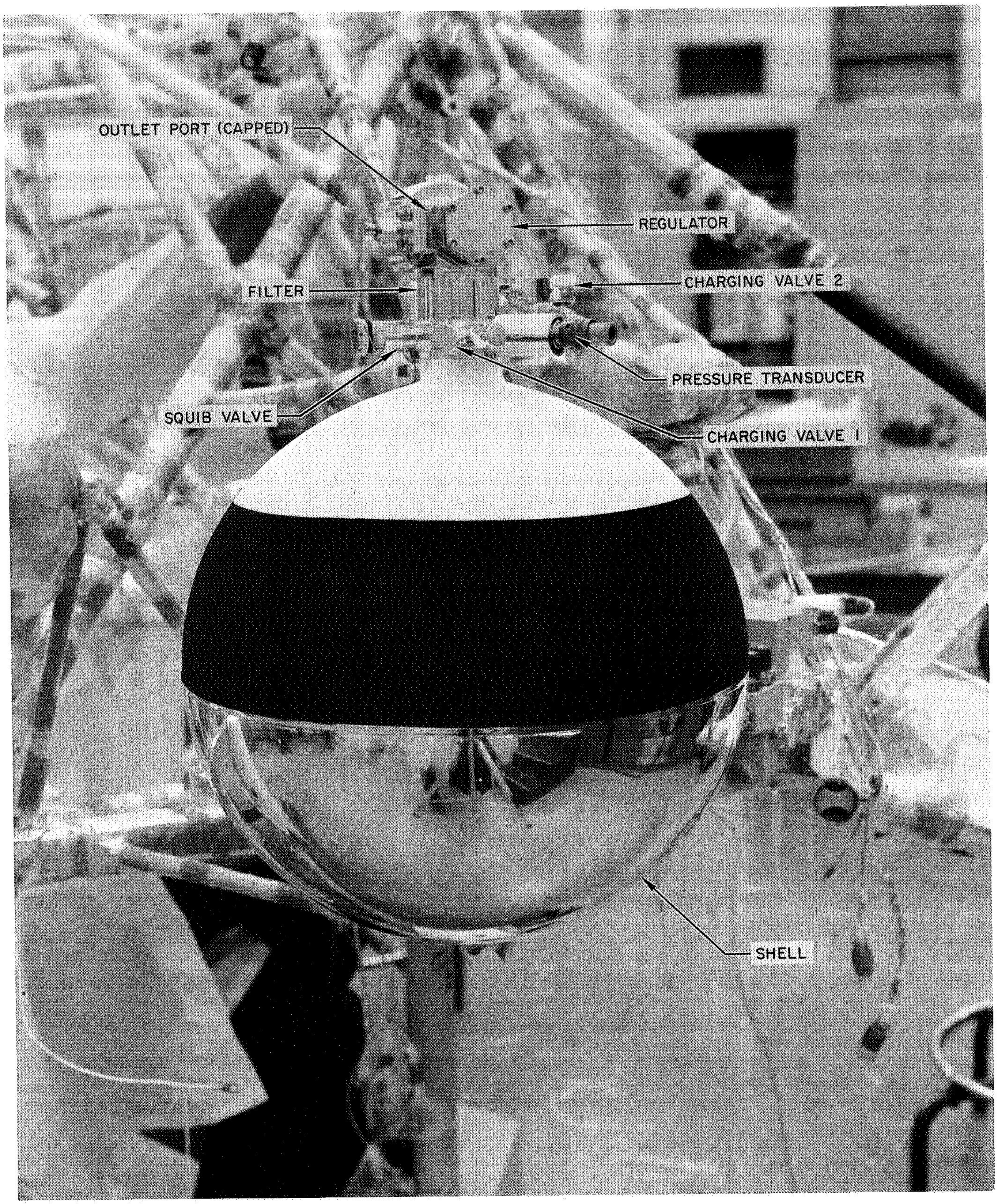


Fig. IV-25. Helium tank assembly

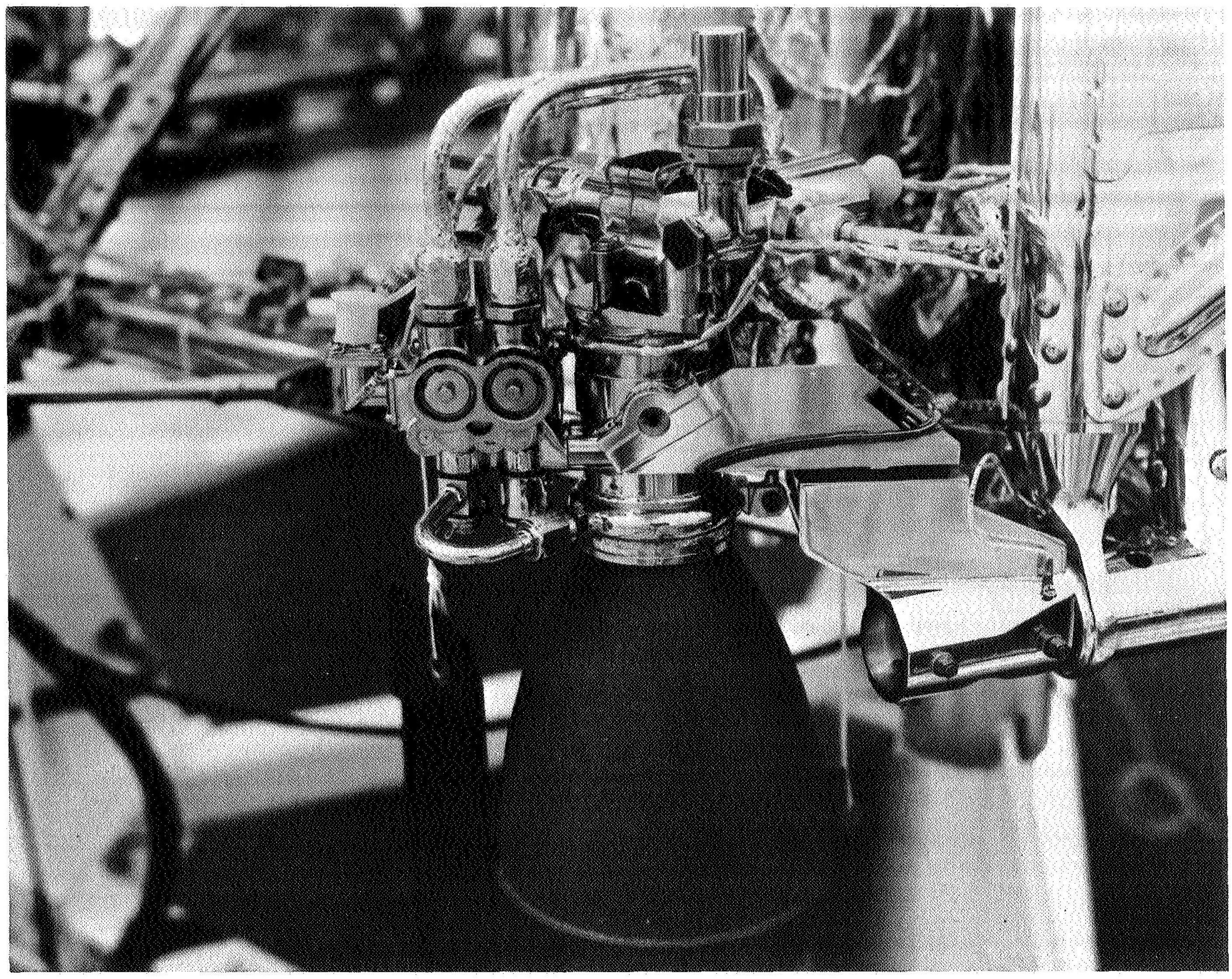


Fig. IV-26. Vernier engine thrust chamber assembly

A 10.3-sec midcourse correction was conducted on November 8 at 02:20:03 GMT. Propulsion system performance was completely nominal. In Table IV-13 engine thrust levels are compared with thrust command and strain gage data for the midcourse and terminal phases of the mission. Note that the predicted and actual thrust commands agree within 4 lb. The thrust data indicates that the vernier propulsion system and the flight control system were functioning nominally. Engine shutoff at completion of the midcourse correction was very smooth as shown by the maximum spacecraft attitude excursion of considerably less than 1 deg. This is well below the 10 deg allowable.

During the post-midcourse phase, vernier propulsion system parameters continued to be normal and well within allowable limits. Figure IV-27 shows temperature extremes during the transit and lunar phases of the *Surveyor VI* mission.

Vernier ignition for terminal descent occurred at 00:58:03 GMT on November 10. Ignition of the vernier and main retromotor was very smooth as indicated by the gyro data. Vernier engine thrust commands and gyro data were also very smooth during the main retro phase and indicated stable spacecraft performance. Differential throttling commands during the main retro phase

Table IV-13. Vernier engine thrust levels, lb

Event	Engine 1	Engine 2	Engine 3
Midcourse correction			
Actual	77.8	71.6	71.4
Predicted	76.8	73.2	72.0
Terminal descent phase			
Vernier ignition (before main retro ignition)			
Actual	68.0	64.5	65.0
Predicted	68.5	65.3	64.2
Retromotor case separation			
Actual	93.5	90.5	90.5
Predicted	93.4	88.9	87.1

indicated an average steady-state main retro disturbance moment of approximately 230 in.-lb shortly after ignition. This steady-state moment decayed to zero by the end of the main retro phase due to asymmetric vernier propellant usage. The disturbing moments were well below the 1800 in.-lb allowable moment about the pitch or yaw axis. Main retromotor burnout and case separation occurred smoothly. Radar controlled descent was nominal, and vernier engine cutoff occurred just above the surface.

Detailed examination of the entire transit phase indicates vernier propulsion system performance was completely nominal and well within allowable limits. As shown in Table IV-14, adequate amounts of propellant remained at landing to perform a unique postlanding spacecraft hop experiment using the vernier propulsion system.

Shortly after landing, special precautions were taken to maximize vernier propulsion system life. On the *Surveyor V* mission, Oxidizer Tank 1 developed a liquid leak about 7 days after landing. On *Surveyors I* and *III*,

Table IV-14. Vernier propellant usage, lb

Total propellant loaded		184.7
Unusable propellant	2.1	
Midcourse correction usage	8.5	
Terminal descent usage	139.1	
Usable propellant remaining at landing		35.0
Spacecraft hop experiment usage	1.4	
Usable propellant remaining after spacecraft hop		33.6

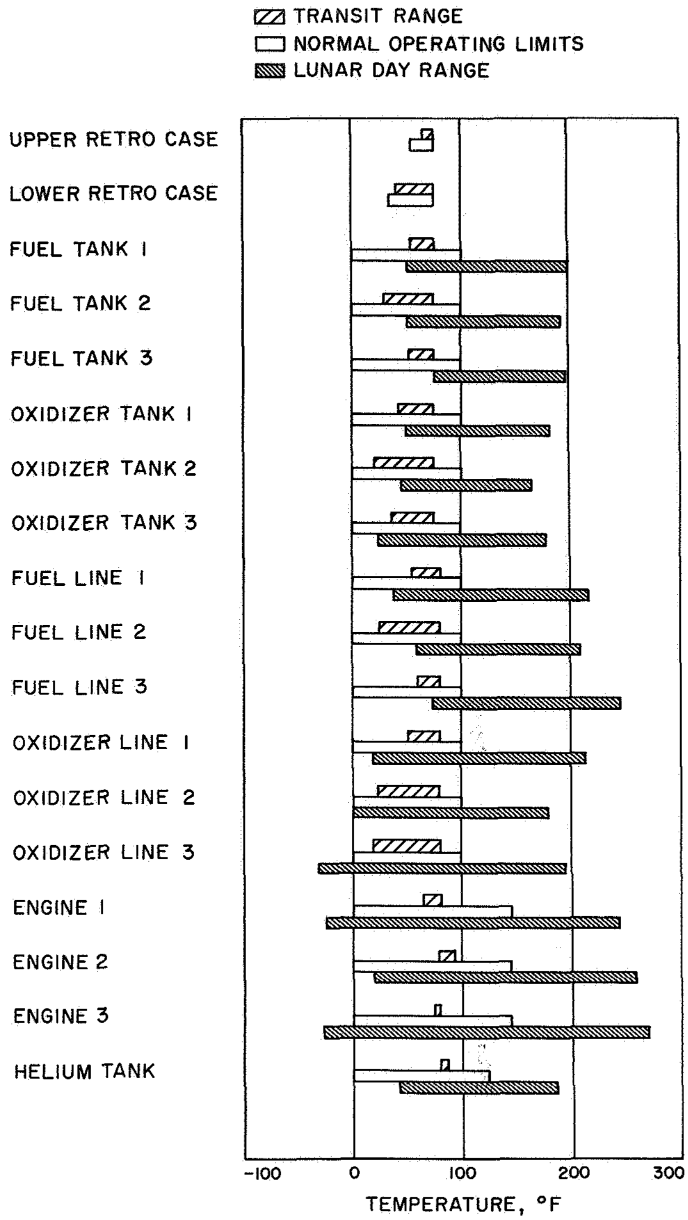


Fig. IV-27. Propulsion system temperature ranges during transit and lunar day

oxidizer system leakage occurred before lunar noon. On the *Surveyor VI* mission, the high-gain antenna and solar panel were used to shade the relief valves and hot oxidizer tanks in an attempt to extend the life of the vernier propulsion system. The optimum positioning was determined using the 1/5-scale model shown in Fig. IV-28 and a collimated light source. By simulating the actual landed attitude and sun elevation, the spacecraft shading possibilities were carefully evaluated. In most instances a compromise between alpha scattering instrument, television and propulsion system shading had to be made.

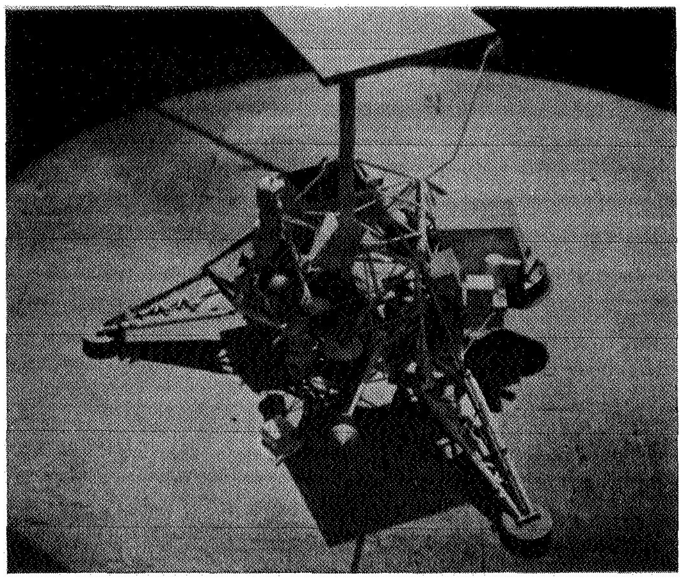


Fig. IV-28. One-fifth-scale model used to optimize antenna and solar panel positions for spacecraft shading

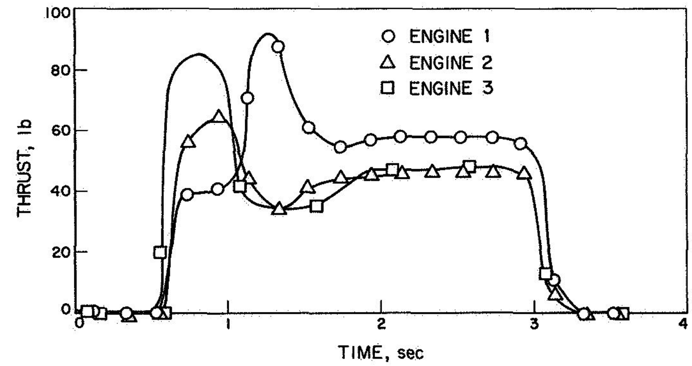


Fig. IV-29. Vernier engine thrust values during spacecraft hop experiment

The relief valves were kept continuously shaded using this method. This reduced their temperature by an estimated 100°F and prevented their failure. Oxidizer Tanks 2 and 3 were on the sunlit side before lunar noon. By shading these tanks whenever possible, Oxidizer Tank 2 was prevented from exceeding 160°F, and Oxidizer Tank 3 was prevented from exceeding 180°F. On *Surveyor V,* Oxidizer Tank 1 leaked when it reached 193°F.

Near lunar noon, when the sun was almost directly overhead, all three engines could be shaded with the high-gain antenna and solar panel, although not all at the same time. After lunar noon, it was impossible to shade Engine 1 because of the sun elevation. For this reason, the lunar noon period was the last opportunity to perform the spacecraft hop experiment until late in the lunar day. Since it was considered unlikely that the vernier propulsion system would remain leak-tight until late in the lunar day and since the major science experiment requirements had already been satisfied, it was decided to perform the spacecraft hop experiment.

The vernier engines were brought below the 220°F temperature limit by alternately shading each of the engines. At the time of ignition, the engine temperatures were all below 210°F. Figure IV-29 shows the thrust-vs-time curve for each of the vernier engines. The planned profile utilized a 2.0-sec firing at 150 lb total thrust with flight control set for an initial pitch offset of +7 deg. The vernier firing was normal except that the firing duration was 2.5 sec rather than 2.0 sec due to spacecraft rejection of the first shutdown command from earth and acceptance of the second, backup, shutdown command. All engines shut down normally. Engine 1 appears to have shut down slightly later than Engines 2 and 3. This caused the spacecraft to pitch up about 3 deg before landing. Since the gyro null position during flight was with Leg 1 pitched down 7 deg, this small cutoff dispersion actually helped level the spacecraft before landing.

The possibility of performing a longer (100 to 300 ft) hop late in the lunar day was also investigated. This would have required a firing of 30- to 60-sec duration. Because of the predicted high initial engine temperatures and engine heating due to combustion during firing, it appeared doubtful that the engines would shut off on command. Failure to shut off is possible at high engine temperatures owing to differential thermal expansion in the engine solenoid valves, which could prevent valve closure as required for engine shutdown. Based on this concern, hopping trajectories which involved several bounces were investigated to ensure that the spacecraft would be close to the surface at engine shutoff. This would maximize the probability of success for a long translation.

Approximately 9 days after landing, the oxidizer tank pressure began dropping, and further consideration for a second lunar hop experiment was abandoned. Figure IV-30 shows the Oxidizer Tank 1 temperature at the beginning of the pressure drop. These characteristics are very much different from those observed on *Surveyor V.* The *Surveyor V* liquid leakage caused temperature drops

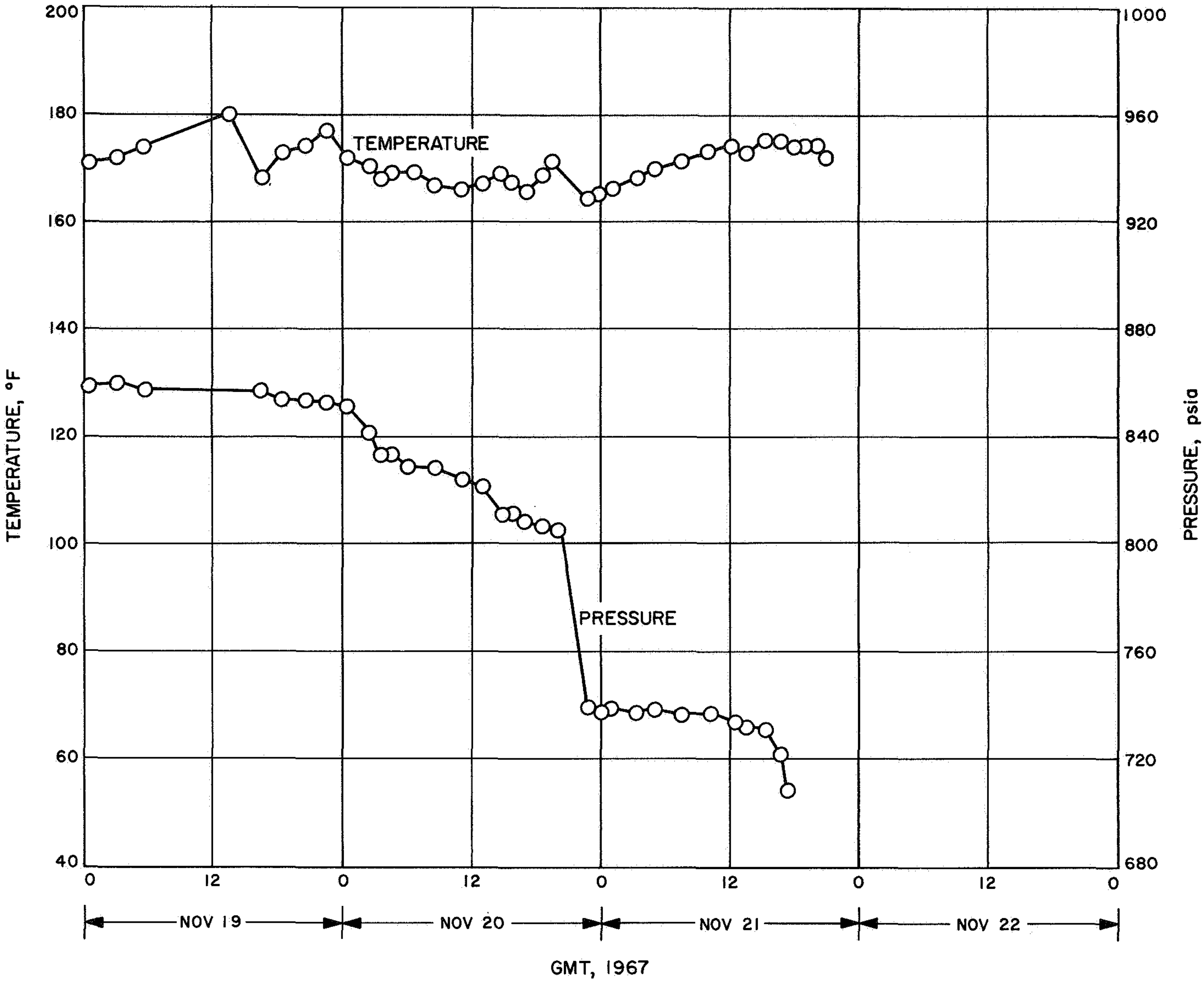


Fig. IV-30. Oxidizer Tank 1 temperature and pressure during postlanding liquid leak

as high as 100°F of the tank and surrounding hardware. As noted in Fig. IV-30, the *Surveyor VI* temperature drops were only 10 to 15°F. A detailed investigation indicated that Tank 1 could not have been partially shaded during this period and, therefore, the temperature changes must have been associated with leakage. After a detailed analysis, it was determined that a liquid leak of Oxidizer Tank 1 had occurred, but that it did not result in large, rapid temperature changes because it was much slower than the *Surveyor V* leak. Tank cooldown curves at lunar sunset, shown in Fig. IV-31, also confirmed the absence of most of the liquid oxidizer from Tank 1. The fuel system held pressure during the entire lunar day and no leakage was evident.

2. Main Retrorocket Motor

The main retrorocket performs the major portion of the deceleration of the spacecraft during terminal descent.

a. Retromotor description. The main retromotor is a spherical solid-propellant unit with a partially submerged nozzle to minimize overall length (Fig. IV-32). The motor utilizes a carboxyl-terminated polybutadiene composite-type propellant and conventional grain geometry.

The motor case is attached at three points on the main spaceframe near the landing leg hinges, with explosive nut disconnects for postburnout ejection. Friction clips

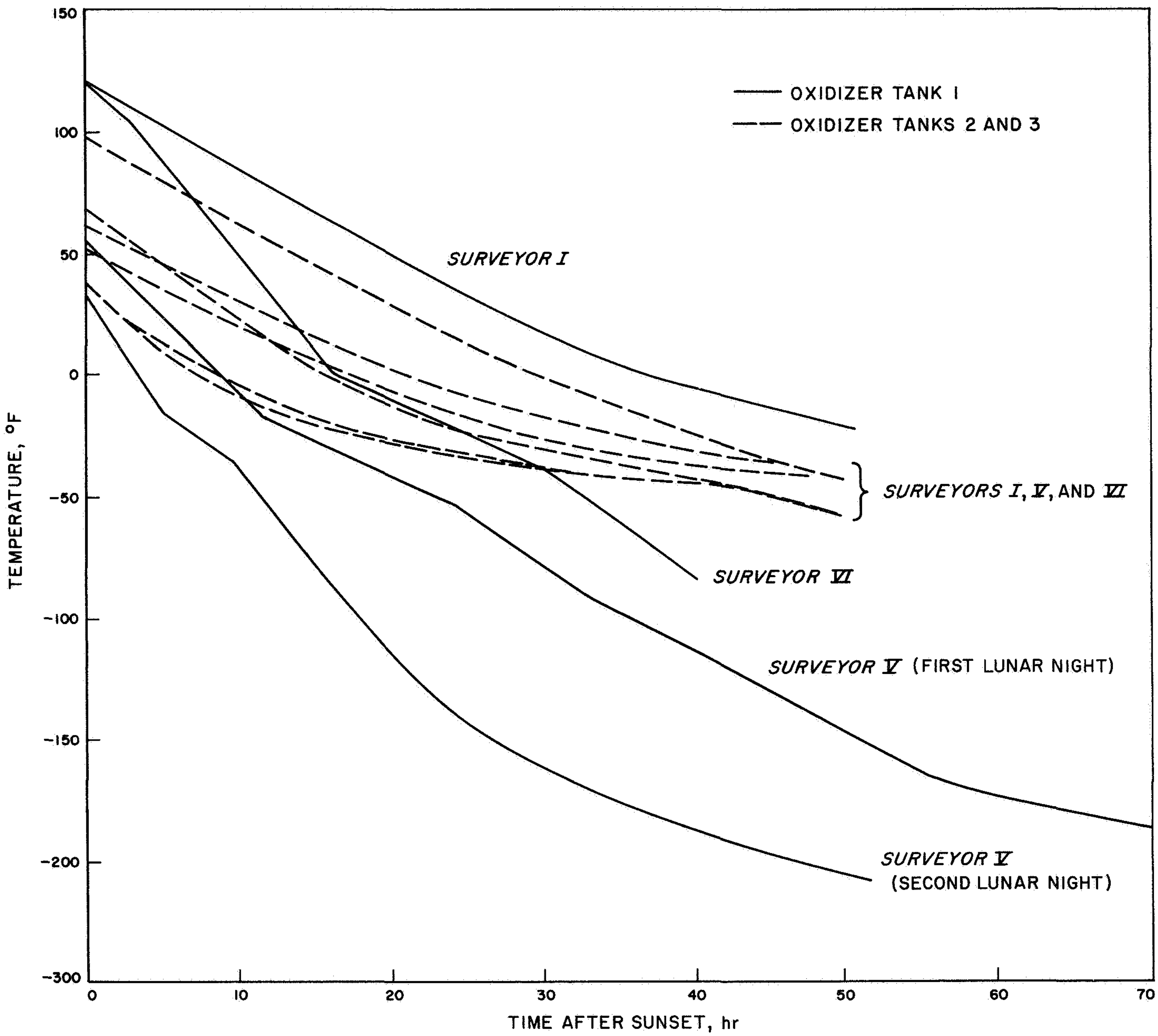


Fig. IV-31. Oxidizer tank temperatures of *Surveyors I, V, and VI* after lunar sunset

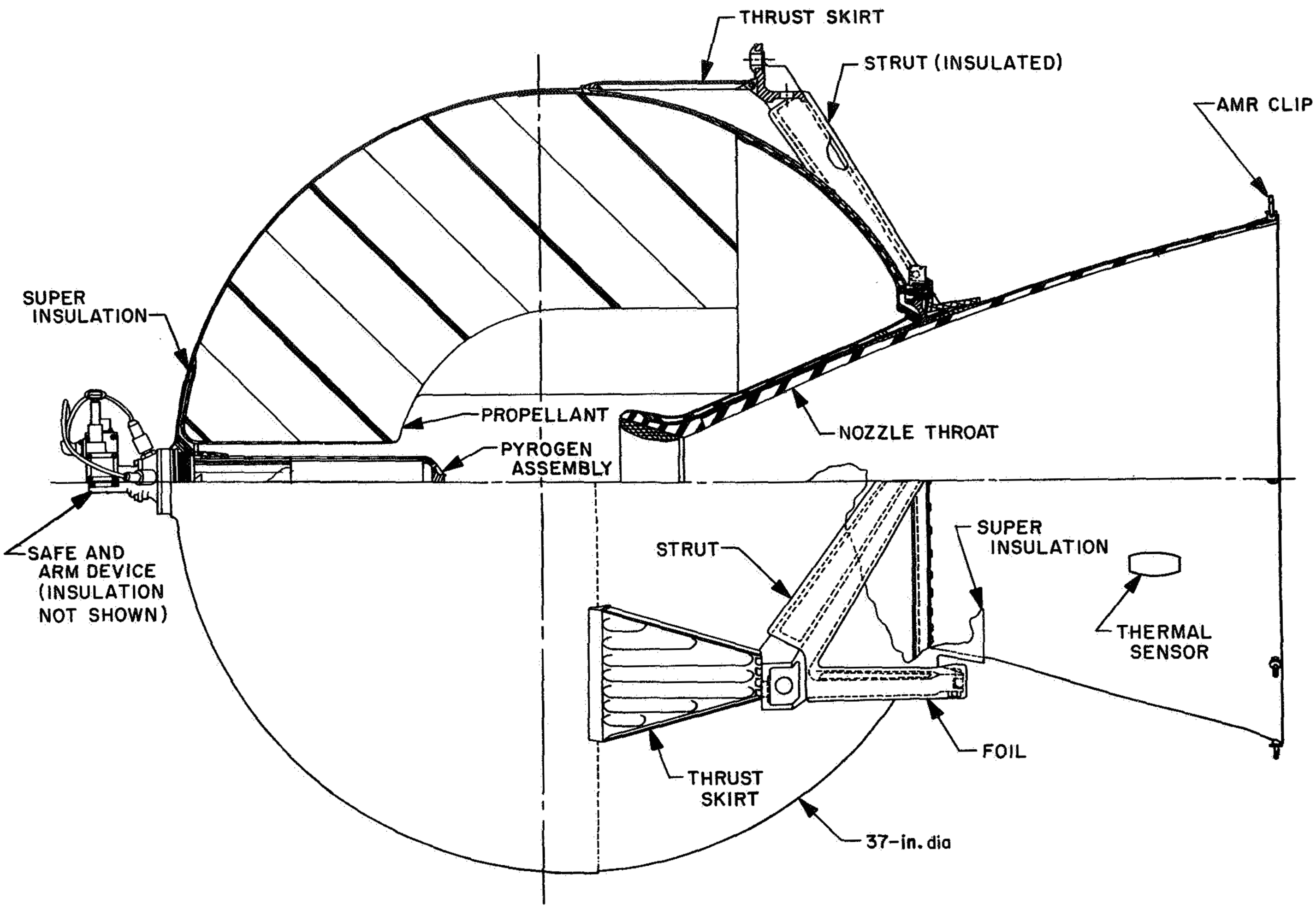


Fig. IV-32. Main retrorocket motor

around the nozzle flange provide attachment points for the altitude marking radar (AMR). The *Surveyor VI* retrorocket, including the thermal insulating blankets, weighed approximately 1400 lb. This total included 1251 lb of propellant. The thermal control design of the retrorocket motor is completely passive, depending on its own thermal capacity and insulating blanket (21 layers of aluminized mylar plus a cover of aluminized teflon). The prelaunch temperature of the unit is 70±5°F. At terminal maneuver, when the motor is ignited, the propellant will have cooled to a thermal gradient with a bulk average temperature of about 50 to 55°F.

The AMR normally triggers the terminal maneuver sequence. When the retro firing sequence is initiated, the retrorocket gas pressure ejects the AMR. The motor operates at a thrust level of 8000 to 10,000 lb for approximately 39 sec at an average propellant temperature of 55°F.

b. Retromotor performance. Beginning at a uniform launch temperature of approximately 70°F, the retromotor cooled at the predicted rate to a thermal gradient condition, with a bulk average temperature of 52.5°F at the time of ignition. At this temperature, the burn time to the 3500-lb thrust level (approximately 3.8-g level) at tailoff was predicted to be 39.60 sec. The postflight doppler data analysis indicates a burn time of 39.37 sec to the 3500-lb thrust level, or a difference of 0.6%. Main retro thrust-vs-time traces, reconstructed from both accelerometer and doppler data, are shown in Fig. IV-33. These compare very well with the predicted trace which is shown. The maximum thrust developed was 9700 lb compared to a prediction of 9650 lb. The total impulse delivered, based on analysis of velocity increment data, was approximately 0.1% higher than predicted.

Modulation of vernier engine thrust commands and the output of the gyros was small, which indicated a

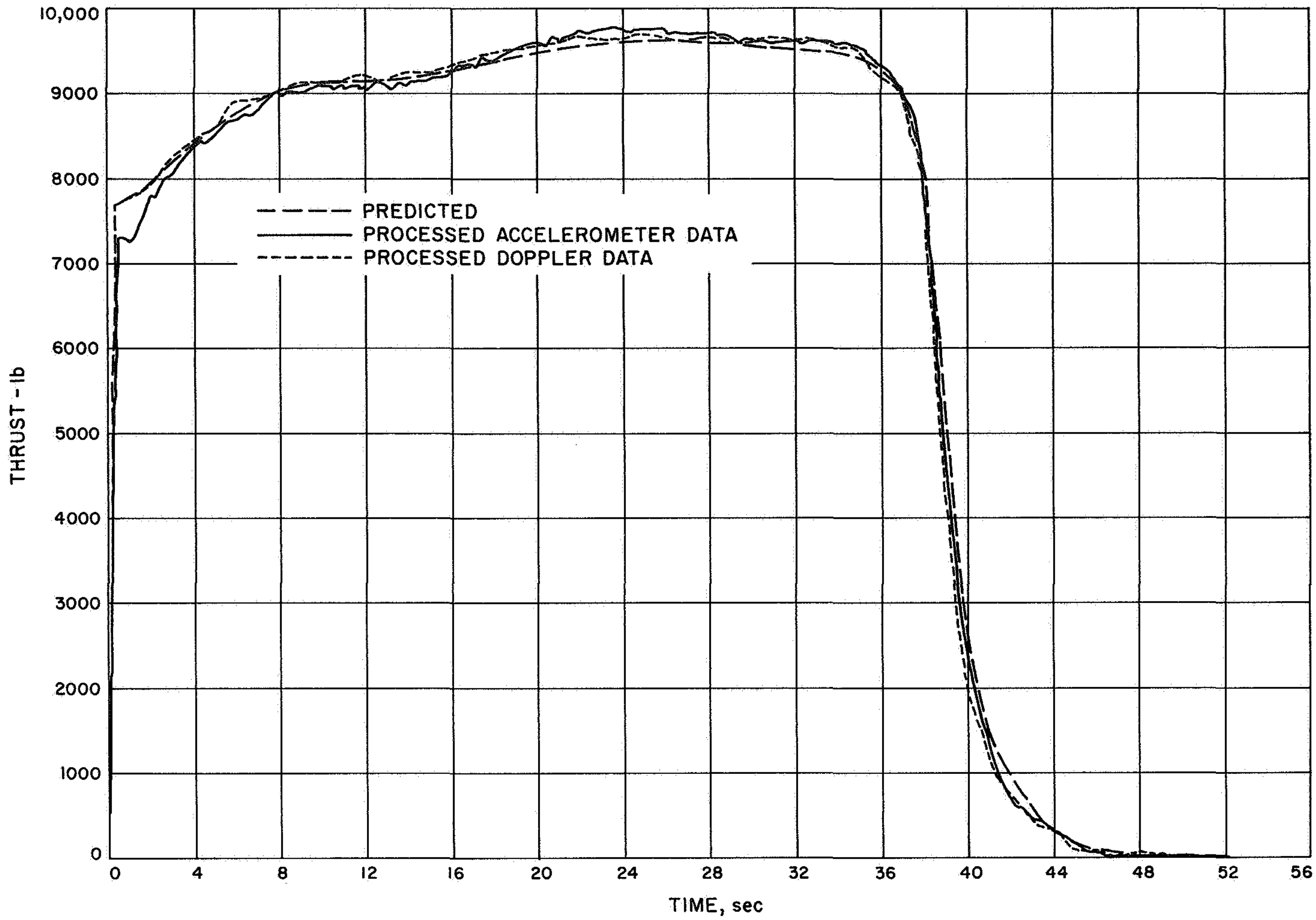


Fig. IV-33. ***Surveyor VI*** **main retromotor thrust vs time**

stable spacecraft during the main retro phase. Main retromotor ignition produced a short-duration attitude torque of approximately 720 in.-lb and a roll torque of approximately 36 in.-lb. The vernier engines operated nearly uniformly, with steady-state disturbing torque gradually decaying from an average of about 230 in.-lb shortly after main retro ignition to essentially zero by main retro burnout. A 180-in.-lb torque occurred briefly at about 30 sec after ignition. These torques were all well below the control capability of approximately 1800 in.-lbs. The maximum roll torque correction required after retro ignition was approximately 12 in.-lb. Assuming all this torque was produced by the main retromotor, the roll torque was well below the 90 in.-lb capability of the spacecraft attitude control system.

Ejection of the retromotor occurred without difficulty. A small spacecraft acceleration disturbance (0.6 *g*) and small shifts in the vernier engine thrust commands suggest that the retromotor may have bumped the spacecraft during ejection.

F. Flight Control

The flight control subsystem provides spacecraft velocity and attitude control during transit from the time of spacecraft separation from the *Centaur* vehicle to spacecraft touchdown on the lunar surface. The basic flight control functions include:

(1) Attitude stabilization and orientation during transit.

(2) Midcourse velocity correction based on ground commands.

(3) Retro ignition and ejection and vernier descent control for soft-landing of the spacecraft.

1. Flight Control Description

The flight control subsystem consists of reference sensing elements, flight control and mode control electronics, and vehicle control elements functionally arranged as shown in Fig. IV-34.

The principal references used by the spacecraft are inertial, celestial, and lunar; each is sensed respectively by inertial, optical, and radar sensors. The control electronics process the reference sensor outputs, earth-based commands, and the flight control programmer and decoder outputs to generate the necessary control signals for use by the vehicle control elements. The vehicle control elements consist of the attitude-control cold-gas-jet activation valves and gas supply system, the vernier engine throttlable thrust valves and controllable gimbal actuator, and the main retro igniter and retro case separation pyrotechnics.

The gas-jet attitude-control system is a cold gas system using nitrogen as a propellant. This system consists of a gas supply system and three pairs of solenoid-valve-operated gas jets interconnected with tubing (Fig. IV-35). The nitrogen supply tank is initially charged to a nominal pressure of 4600 psia. Pressure to the gas jets is controlled to 40±2 psia by a regulator.

Vehicle response in attitude, acceleration, and velocity is controlled as needed by various "control loops" throughout the coast and thrust phases of flight, as shown in Table IV-15. Upon separation of the spacecraft from the *Centaur*, stabilization of the spacecraft tipoff rates is achieved through activation of the gas jet system and use of rate feedback gyro control (*rate* mode). After rate capture, *inertial* mode is achieved by switching to position feedback gyro control.

Because of the long duration of the transit phase and the small unavoidable drift error of the gyros, celestial references are used to provide the desired attitude of the spacecraft. Following a 51-sec delay after spacecraft electrical disconnect from the *Centaur*, a flight control timer automatically initiates the sun acquisition sequence by commanding a negative roll maneuver. The sun is

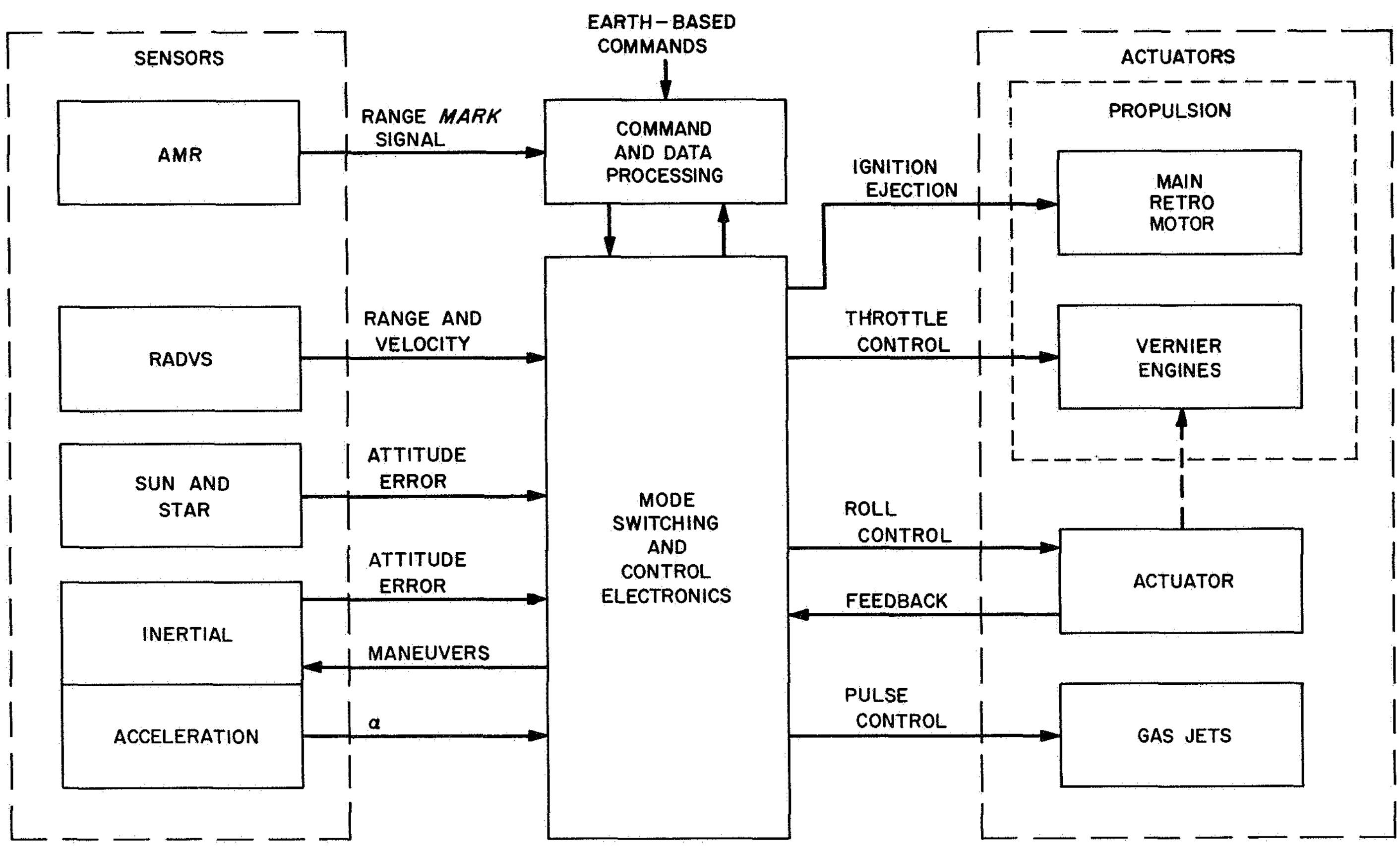


Fig. IV-34. Simplified flight control functional diagram

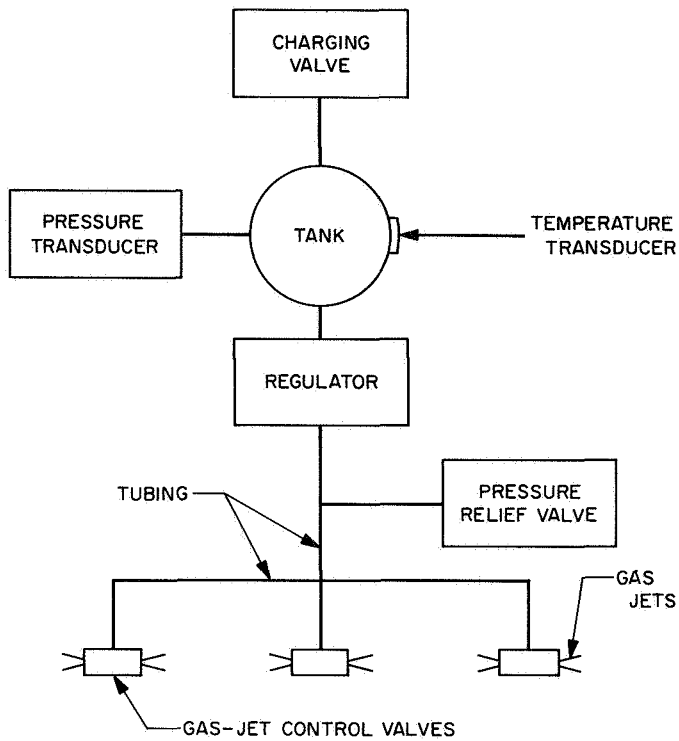


Fig. IV-35. Gas-jet attitude control system

first acquired by the acquisition sun sensor, which has a 10-deg-wide by 196-deg-fan-shaped field of view that includes the spacecraft Z-axis and is centered about the minus X-axis. The roll command is terminated after initial sun acquisition, and a positive yaw command is automatically initiated which allows the narrow-view primary sun sensor to acquire and lock-on the sun. A secondary sun sensor, mounted on the solar panel, provides a backup for manual acquisition of the sun if the automatic sequence fails. Beginning with the *Surveyor V* spacecraft, a separation latch was incorporated in the flight control programmer which, upon initiation of the automatic sun acquisition sequence, inhibits subsequent false triggering of the automatic sun acquisition sequence.

Automatic Canopus acquisition and lock-on are normally achieved after initiation of a roll by command from earth. This occurs because the Canopus sensor angle is preset with respect to the primary sun sensor prior to launch for each mission. Star mapping for Canopus verification is achieved by commanding the spacecraft to roll while the spacecraft maintains sun lock.

The transit phase is performed with the spacecraft in the celestial-referenced mode except during initial rate-stabilization, midcourse and terminal descent maneuvers, and gyro drift checks, when the inertial mode is used.

Table IV-15. Flight control modes

Control loop	Flight phase	Modes	Remarks
Attitude control loop			
Pitch and yaw	Coast	Rate Inertial Celestial	Gas jet matrix signals
	Thrust	Inertial Lunar radar	Vernier engine matrix signals
Roll	Coast	Rate Inertial Celestial	Leg 1 gas jet signals
	Thrust	Inertial	Vernier Engine 1 gimbal command
Acceleration control loop			
Thrust axis	Thrust (mid-course)	Inertial (with accelerometer)	Nominal 3.22 ft/sec^2 Minimum 4.77 ft/sec^2 Maximum 12.56 ft/sec^2
	Thrust (terminal descent)	Inertial (with accelerometer)	
Velocity control loop			
Thrust axis	Thrust	Lunar radar	Command segment signals to 43-ft altitude Constant 5-ft/sec velocity signals to 13-ft altitude
Lateral axis	Thrust	Lunar radar	Lateral/angular conversion signals

Midcourse velocity correction capability is provided by means of the vernier engine throttle valves, a precision timer, and an accurate acceleration sensing device. The difference between the command acceleration level and the output from the accelerometer provides an error signal which is used to command the throttle valves for the required thrust level. The timer controls the duration of vernier thrusting in accordance with a time interval preset by earth command.

The terminal maneuver descent sequence has been described in detail in Section IV-A-15. The flight control subsystem provides initial orientation of the main retromotor thrust axis and automatic sequencing after the lunar reference is first established by a signal from the AMR. Most of the approach velocity is removed by the solid-propellant retromotor during the initial phase of terminal descent. Spacecraft attitude during this phase is inertially stabilized using the gyros and differential throttling of the vernier engines.

During the vernier phase of descent, which follows main retro burnout and ejection, a sophisticated flight control technique is utilized that includes the use of radars to obtain range and velocities. The range information is used by the flight control subsystem during the vernier phase of descent to compute longitudinal velocity commands for controlling the total vernier engine thrust to achieve the desired descent profile (approximately constant acceleration). The velocity data is applied to the attitude control loop to produce a near-gravity turn during descent by aligning the spacecraft thrust axis with the velocity vector. The velocity data is also used to generate error signals for the velocity control loop.

2. Flight Control Performance

The flight control subsystem performed in a very nominal manner for all phases of the *Surveyor VI* mission. Included were unique operations such as a hop on the lunar surface and pre-midcourse attitude maneuvers that were compensated for gyro drifts.

a. Launch phase. Prior to launch, some question was raised about the performance of the Canopus tracker, which had a star intensity signal considerably lower than observed on previous spacecraft. However, since the secondary sun sensors were also reading lower, it was concluded that the *Surveyor VI* nose fairing was simply more lightproof than previous fairings.

Spacecraft separation from the *Centaur* was normal. At the time of separation, spacecraft telemetry data was marginal and contained multiple parity errors. However, the following approximate separation rates were extracted from the data:

Roll	+0.1 deg/sec
Pitch	+0.1
Yaw	+0.1

The tipoff rates were well within the capabilities of the attitude control loop, and rate stabilization was easily achieved.

b. Sun acquisition. The sun acquisition sequence was automatically executed, resulting in primary sun sensor center cell lock-on. The acquisition sun sensor was illuminated after a negative roll of 264 deg. This was followed by a positive yaw of 22 deg, which established sun-lock. The sun acquisition sequence required about 9 min and 32 sec.

c. Canopus acquisition. A *sun-and-roll* command was sent to the spacecraft about 6 hr after liftoff to initiate the star mapping and Canopus acquisition sequence. Analysis of star intensity signals resulted in positive identification of the earth, Deneb, and Canopus. Table IV-16 is a summary of the objects that appeared in the field of view of the Canopus sensor.

As noted in Table IV-16, a loss of data occurred at a roll angle of 143 deg. This is attributed to an antenna pattern null which caused loss of spacecraft receiver phase lock in the transponder mode and shift of the transmitter to narrow-band VCXO frequency, resulting in loss of signal at DSS 61. *Sun mode on* was commanded shortly after loss of signal to stop the roll and permit an assessment of telecommunications. After determining that the telecommunications system had responded in a normal manner to the antenna null, the roll was restarted with the transponders off. Another loss of data occurred for 8.5 deg, beginning at 488 deg of roll as had been predicted because of an antenna null.

A Canopus lock-on signal occurred when the spacecraft was rolling past the earth and Canopus the first time. It was therefore decided to send the *sun and star* command after rolling past the earth for the second time and before the second appearance of Canopus. The spacecraft was switched to the *sun and roll* mode 70 deg before receipt of the Canopus automatic *lock-on* signal, which occurred at 652.5 deg of roll. Spacecraft roll motion was damped out to the start of optical deadband limit cycling in approximately 45 sec. The spacecraft roll rate based on time of Canopus lock-on was 0.5014 deg/sec.

The Canopus sensor functioned in a nominal manner during the entire mission. As for the previous three spacecraft, the *Surveyor VI* sun filter was calibrated at the level of 0.8 × Canopus. The ratio of measured Canopus intensity to predicted intensity was about 0.87.

d. Midcourse maneuvers and velocity correction. The maneuver combination selected to orient the spacecraft in the desired direction for the midcourse velocity correction consisted of a +91.9-deg roll maneuver followed by a +127.3-deg yaw maneuver. The actual maneuver magnitudes were verified as +91.8 deg of roll and +127.3 deg of yaw.

Table IV-16. *Surveyor VI* star map results

GMT[a] (November 7, 1967)	Roll angle, deg	Object in field of view	Angle from Canopus, deg		Peak telemetered intensity, V		Comment
			Indicated	Predicted	Telemetered	Predicted	
15:50:23	0		−298				Start roll
15:51:06	21.5	Earth	−276.5	−275	3.7		
15:54:52	134.5	Deneb	−163.5	−163.9	0.72	0.55	
15:55:09	143		−155				Loss of data
15:58:45	251		− 47				Stop roll
16:14:26	251		− 47				Restart roll
16:15:51	293.5		− 4.5				Canopus *lock-on* signal
16:16:00	298	Canopus	0	0	3.49	4.01	
16:18:44	380	Earth	82	85	3.3		
16:21:31	463.5	Object 1	165.5		0.65		
16:21:37	466.5	Object 2	168.5		0.60		
16:22:20	488		190				Loss of data
16:22:37	496.5		198.5				Return of data
16:25:29	582.5		284.5				*Sun and star* mode on
16:27:49	652.5	Canopus	354.5				Canopus *lock-on* signal
		Canopus			3.79	4.33	Canopus signal after acquisition
		None			0.42	0.46	

[a]Times are based on bulk printer in SFOF and are ±1 sec.

As on previous spacecraft, an attempt was made to reduce the optical sensor control loop contribution to the pointing error by initiating the pre-midcourse attitude maneuvers at the limit cycle null points. This was successfully accomplished, the roll maneuver being initiated with only −0.048 deg of roll gyro error. The yaw maneuver was initiated with −0.03 and 0.0 deg of pitch and yaw gyro error, respectively.

This was the first time that the pre-midcourse attitude maneuvers were corrected for gyro drifts. In calculating the amount of correction, it was estimated that the roll and yaw attitude control loops would be in the inertial mode for approximately 0.17 hr and 0.14 hr, respectively.

For the midcourse velocity correction, a quantity of 103 bits (equivalent to 10.25 sec of burn time) was entered into the magnitude register. Vernier engine ignition occurred at approximately 02:20:03 GMT on November 8. Ignition was smooth, with maximum pitch and yaw attitude changes of −0.08 deg and −0.17 deg, respectively. The actual burn time was 10.242 sec.

The total indicated thrust based on the vernier engine thrust commands was approximately 225.0 lb. The expected level was 222 lb based on a separated weight of 2220 lb.

Following the midcourse velocity correction, the reverse attitude maneuvers were commanded, resulting in reacquisition of the sun and Canopus.

e. Gyro drift measurements. Fourteen gyro drift checks were made during the mission. Three were roll-axis-only checks and eleven were checks of drift about all three axes. Four of the gyro drift checks were made prior to the midcourse correction and provided the following drift rates, which were used to compensate the premidcourse maneuvers for drift.

Roll	−0.55 deg/hr
Pitch	0.0
Yaw	+1.2

The preterminal attitude maneuvers were compensated for the following gyro drift rates:

Roll	−0.64 deg/hr
Pitch	0.0
Yaw	+1.4

f. Nitrogen gas consumption. The calculated weight of the nitrogen contained in the attitude control system at launch was 4.56 lb, based on a telemetered tank pressure of 4686 psi and a tank temperature of 80.4°F.

Prior to the terminal maneuvers, 4.12 lb of nitrogen remained in the nitrogen tank, indicating a somewhat economical nitrogen consumption of 0.44 lb.

g. Terminal maneuvers and descent. As in the case of the pre-midcourse attitude maneuvers, an attempt was made to initiate the terminal maneuvers at the limit cycle null points. The roll maneuver was initiated within approximately −0.06 deg of null, while the pitch and yaw optical errors at the start of yaw were 0.0 deg and +0.2 deg, respectively. The magnitudes of the maneuvers were verified to be +81.82 deg of roll, +111.71 deg of yaw and +120.55 deg of roll.

A total vernier thrust level of 201±5 lb was selected for the main retro burn period. A vernier engine ignition delay of 5.875 sec (following AMR *mark*) was entered into the magnitude register about 16 min before AMR *mark*. After *thrust phase power on* was commanded from earth and confirmed by the spacecraft, all terminal descent events beginning with generation of the AMR *mark* signal occurred automatically, as planned, at close to predicted times (refer to Table IV-6). A plot of slant range vs vertical velocity during terminal descent is presented in Fig. IV-8.

Peak attitude transients at vernier engine ignition were −0.77 deg roll, −0.17 deg pitch, and −0.08 deg yaw. During main retromotor burn, the mean attitude errors were −0.15 deg roll, +0.08 deg pitch, and −0.03 deg yaw. Peak roll actuator position at main retromotor ignition was +1.08 deg, while the mean value during main retro burn was +0.125 deg.

The changes in spacecraft attitude at touchdown were approximately −2.3 deg in pitch and −0.21 deg in yaw. *Thrust phase power off* was commanded and verified 36 sec after touchdown, and *flight control power off* was commanded and verified 60 sec after touchdown.

h. Postlanding spacecraft hop experiment. The planned spacecraft hop experiment called for an initial pitch gyro precession of −7.0 deg, to impart a +7.0 deg pitchover of the spacecraft after liftoff so that lateral motion would occur during the hop in the direction of Leg 1. Logic in the flight control programmer was arranged to throttle the vernier engines to produce a total of 150 lb of thrust. Duration of the engine firing was to be controlled by taped radio commands from earth to provide a firing time of 2 sec.

Vernier engine ignition was observed at 10:32:05.0 GMT, November 17, 1967, and engine shutdown occurred at 10:32:07.5. The thrusting period lasted about 2.5 sec instead of the planned 2 sec, because the first command to terminate firing was apparently not accepted by the spacecraft. However, the second command, transmitted 0.5 sec later, did stop the thrusting.

Approximately 1 sec after ignition, the spacecraft completed its +7-deg pitchover maneuver and maintained a stable attitude for the remainder of the thrusting period. At vernier engine shutdown, small residual angular rates were integrated by the spacecraft until at touchdown the following gyro errors existed:

Pitch	−4.3 deg
Yaw	−3.4
Roll	−2.5

The spacecraft completed the hop at 10:32:11.2 GMT, after a total flight time of 6.1 sec. The only anomaly occurring was the 0.5-sec extra burn time. The additional thrusting time produced a higher and longer trajectory than originally planned. However, the larger touchdown impact had no apparent ill effects on the spacecraft.

The hop experiment demonstrated for the first time flight control capability to control the spacecraft in stable flight over the lunar surface. Some immediate benefits derived from the hop include: stereo photographs of the lunar surface, more complete examination of the footpad imprints formed on the lunar surface during initial landing, and study of vernier engine blast effects on the lunar surface. This pioneering effort could also pave the way towards more sophisticated translations to be used on future spacecraft.

G. Radar

Two radar devices, the altitude marking radar (AMR) and the radar altimeter and doppler velocity sensor (RADVS), are employed on the *Surveyor* spacecraft for use during the terminal descent phase.

1. Altitude Marking Radar

The AMR provides an altitude *mark* signal for automatic initiation of the spacecraft terminal descent sequence.

a. AMR description. The AMR (Fig. IV-36) is a conventional noncoherent radar employing a pulsed magnetron, single antenna, duplexed mixer, crystal-controlled solid-state local oscillator, wideband intermediate frequency (IF) amplifier, noncoherent detector, and video processing circuitry. Dynamic range is extended by automatic gain control (AGC) of the IF amplifier; AGC voltage is telemetered and provides an indication of received signal power. The video circuitry is of special design to provide the *mark* signal with high accuracy and reliability at a slant range from the lunar surface that can be preset between 52 and 70 miles (60 miles for *Surveyor VI*). Two fixed, adjacent range gates continuously examine the video signal. Their outputs are continuously summed and differenced. When the sum exceeds a fixed threshold and the difference simultaneously crosses zero with positive slope, the *mark* signal is generated. The sum threshold is set for an extremely low probability of marking on noise (false *mark*) throughout the operating time, while video integration plus a very substantial radar gain margin insure a high probability of marking successfully.

Two separate ground commands, whose timing is controlled, are required to fully activate the AMR. The first signal, called simply *AMR on,* commands on the primary power to the AMR, which includes all internal power except high voltage to the transmitter. The video signal is inhibited from reaching the marking circuits until the second command, thus eliminating any residual probability of false marking on noise during this warmup interval. The second signal, called *AMR enable,* commands

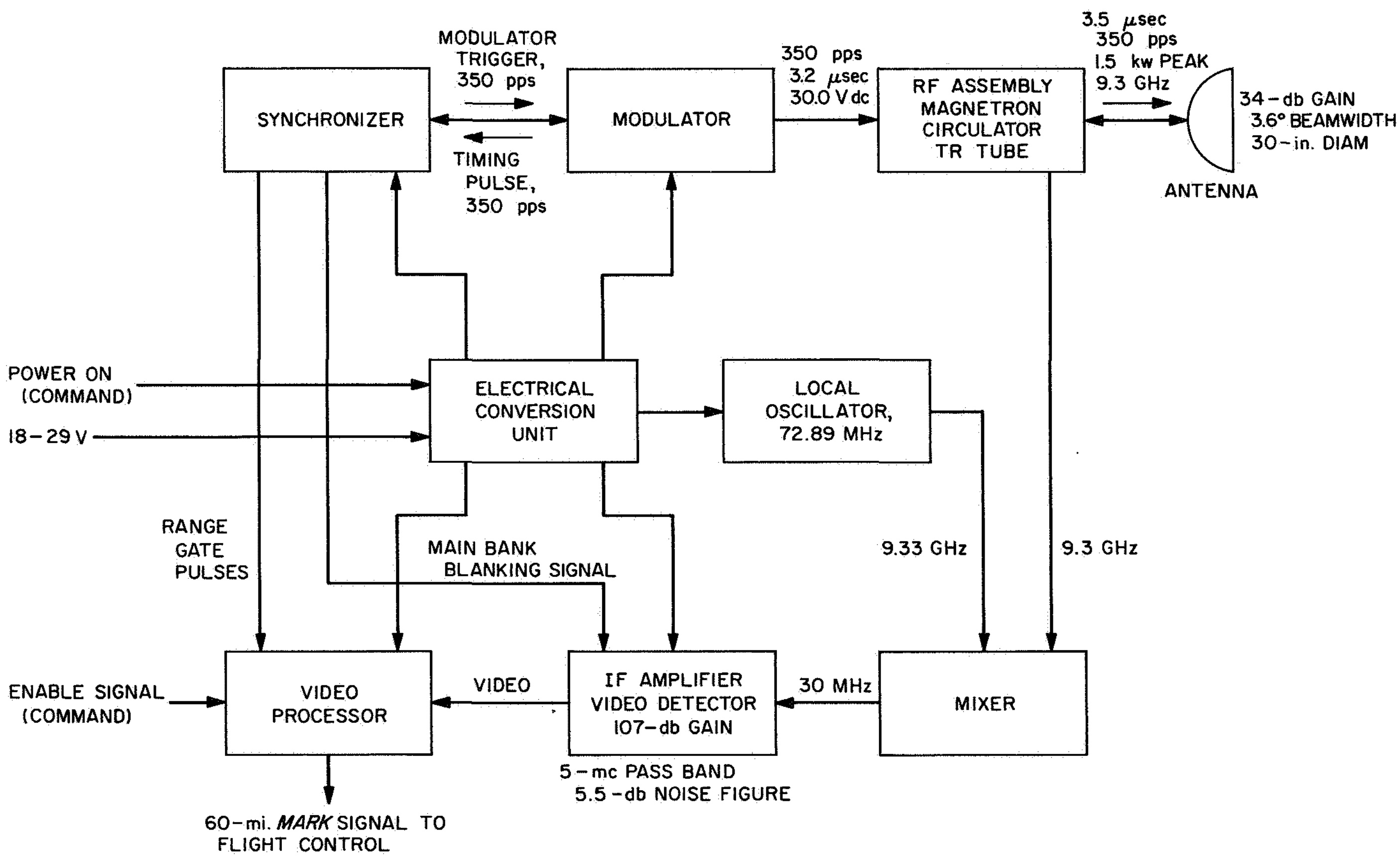


Fig. IV-36. Altitude making radar functional diagram

on the transmitter high voltage and also removes the video inhibit. This enabling function is timed not only for favorable thermal conditions at the expected marking time but also to preclude premature marking on second-round echoes at much longer ranges.

AMR signals which are telemetered include three digital signals and three analog signals plus analog temperature data. The three digital signals (R-1, R-11, FC-64) are confirmations of on-board discrete events; they confirm, respectively, that prime power has been applied (*AMR on*), that high voltage and video enabling have been applied (*AMR enable*), and that the self-generated slant range trigger (*AMR mark*) has occurred. The three analog signals (besides temperature) are magnetron current (R-12), AGC voltage level (R-14), and late gate detected video voltage level (R-29). The AGC not only confirms receiver response to RF return, but is also useful in evaluating terrain reflectivity. The magnetron current confirms pulsing of the magnetron after *enable,* and is useful primarily as a transmitter failure mode indication. The late gate signal, primarily a receiver failure mode indication, normally confirms the presence of gated video signal rising quickly to a peak at the time of *mark* and decaying quickly thereafter.

The AMR mounts in the retro rocket nozzle and is retained by friction clasps around the nozzle flange, with spring washers between the AMR and the flange. When the retro rocket is ignited, the gas generated by the ignitor develops sufficient pressure to eject the AMR from the nozzle. The AMR draws 22-V dc power through a breakaway plug that also carries input commands, the output *mark* signal, and telemetry information.

b. AMR performance. On the *Surveyor VI* mission, the AMR functioned normally in all respects. The 60-mile *mark* was generated at the proper altitude and started the automatic terminal sequence.

The AMR AGC curve shown in Fig. IV-37 represents the signal power being received at the AMR feedhorn vs time to main retro ignition. It can be seen that the received power increased smoothly and evenly as the spacecraft approached a slant range of 60 miles. At a return power level of -75.3 db, the 60-mile *mark* was initiated. The return power at *mark* compares to within 1 db of the calculated signal return for proper *mark.* The predicted return power for *mark* is -74.5 db, based upon a pulse width of 16.2 μsec and an approach angle of 25 deg.

AMR temperatures were as expected and compared favorably to those of previous missions. AMR temperatures are not telemetered during terminal descent, so the last temperatures are those just prior to AMR *enable* and are given below:

Antenna, edge	+2°F
AMR electronics	+19°F

2. Radar Altimeter and Doppler Velocity Sensor

The RADVS (Fig. IV-38) functions in the flight control subsystem to provide three-axis velocity, range, and altitude *mark* signals for flight control during the main retro and vernier phases of terminal descent. The RADVS consists of a doppler velocity sensor (DVS), which computes velocity along the spacecraft X, Y, and Z axes, and a radar altimeter (RA), which computes slant range from 40,000 to 13 ft and generates 1000-ft and 13-ft *mark* signals. The RADVS comprises five assemblies: (1) klystron power supply/modulator (KPSM), which contains the RA and DVS klystrons, klystron power supplies, and altimeter modulator, (2) altimeter/velocity sensor antenna, which contains Beams 1 and 4 transmitting and receiving antennas and preamplifiers, (3) velocity sensing antenna, which contains Beams 2 and 3 transmitting antennas and preamplifiers, (4) RADVS signal data converter (SDC), which consists of the electronics to convert doppler shift signals into dc analog signals, and (5) interconnecting waveguide. The RADVS is turned on at about 50 miles above the lunar surface and is turned off at about 13 ft.

a. Doppler velocity sensor description. The doppler velocity sensor (DVS) operates on the principle that a reflected signal has a doppler frequency shift proportional to the approaching velocity. The reflected signal frequency is higher than the transmitted frequency for the closing condition. Three beams directed toward the lunar surface enable velocities in an orthogonal coordinate system to be determined. The RADVS beam orientation is shown in Fig. IV-39.

The KPSM provides an unmodulated DVS klystron output at a frequency of 13.3 GHz. This output is fed equally to the DVS 1, DVS 2, and DVS 3 antennas. The RADVS velocity sensor antenna unit and the altimeter velocity sensor antenna unit provide both transmitting and receiving antennas for all three beams. The reflected signals are mixed with a small portion of the transmitted frequency at two points ¾ wavelength apart for phase determination, detected, and amplified by variable-gain

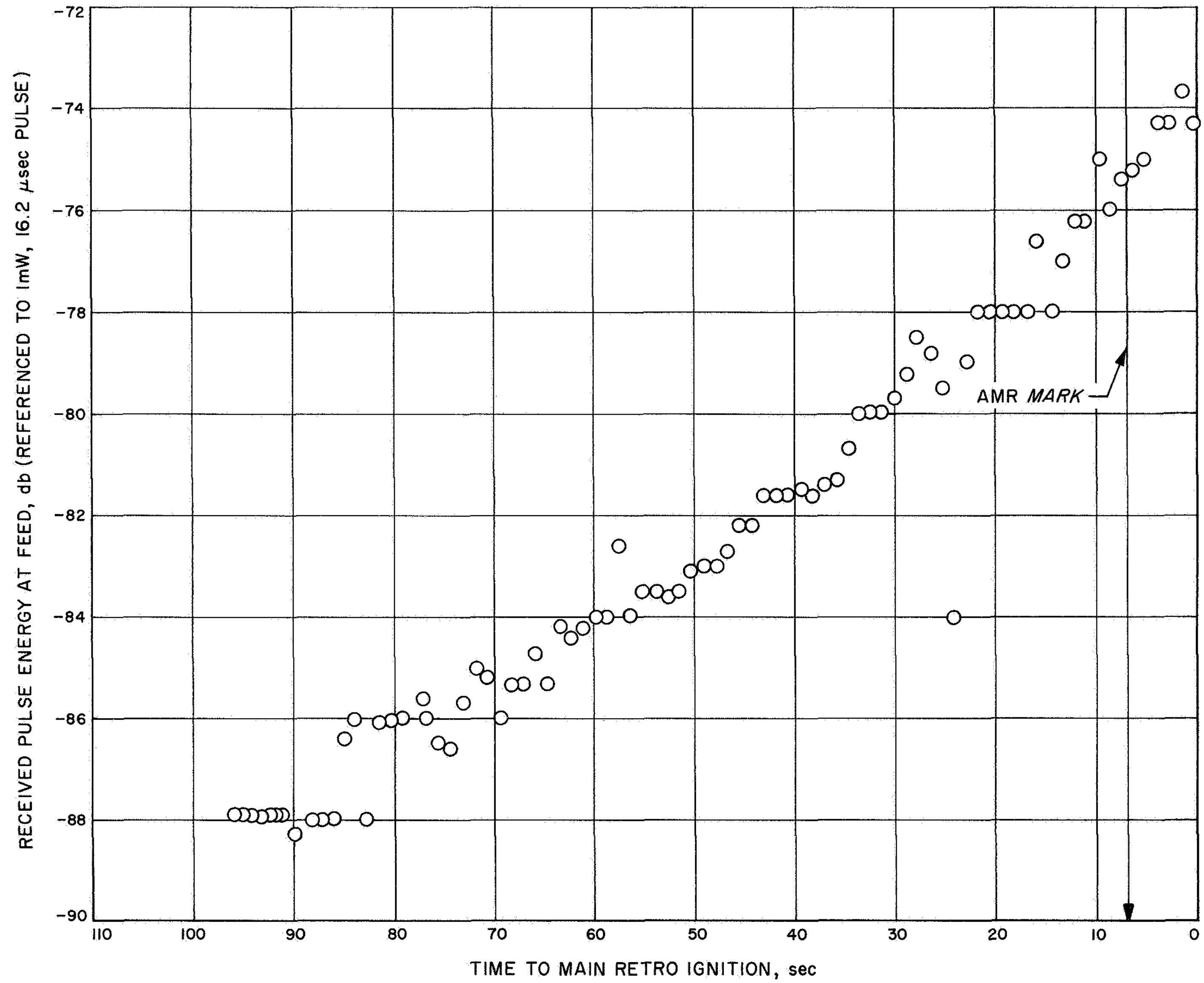


Fig. IV-37. Altitude marking radar AGC

amplifiers providing 40, 65, or 90 db of amplification, depending on received signal strength. The preamp output signals consist of two doppler frequencies, shifted by ¾ transmitted wavelength, and preamp gain-state signals for each beam. The signals are routed to the trackers in the RADVS signal data converter.

The D1 through D3 trackers in the signal data converter are similar in their operation. Each provides an output which is 600 kHz plus the doppler frequency for approaching doppler shifts. If no doppler signal is present, the tracker will operate in search mode, scanning frequencies between 82 kHz and 800 Hz before retro burnout, or between 22 kHz and 800 Hz after retro burnout. When a doppler shift is obtained, the tracker will operate as described above and initiate a lock-on signal. The tracker also determines amplitude of the reflected signal and routes this information to the signal processing electronics for telemetry.

The velocity converter combines tracker output signals D_1 through D_3 to obtain dc analog signals corresponding to the spacecraft X, Y, and Z velocities; $D_1 + D_3$ is also sent to the altimeter converter to compute range.

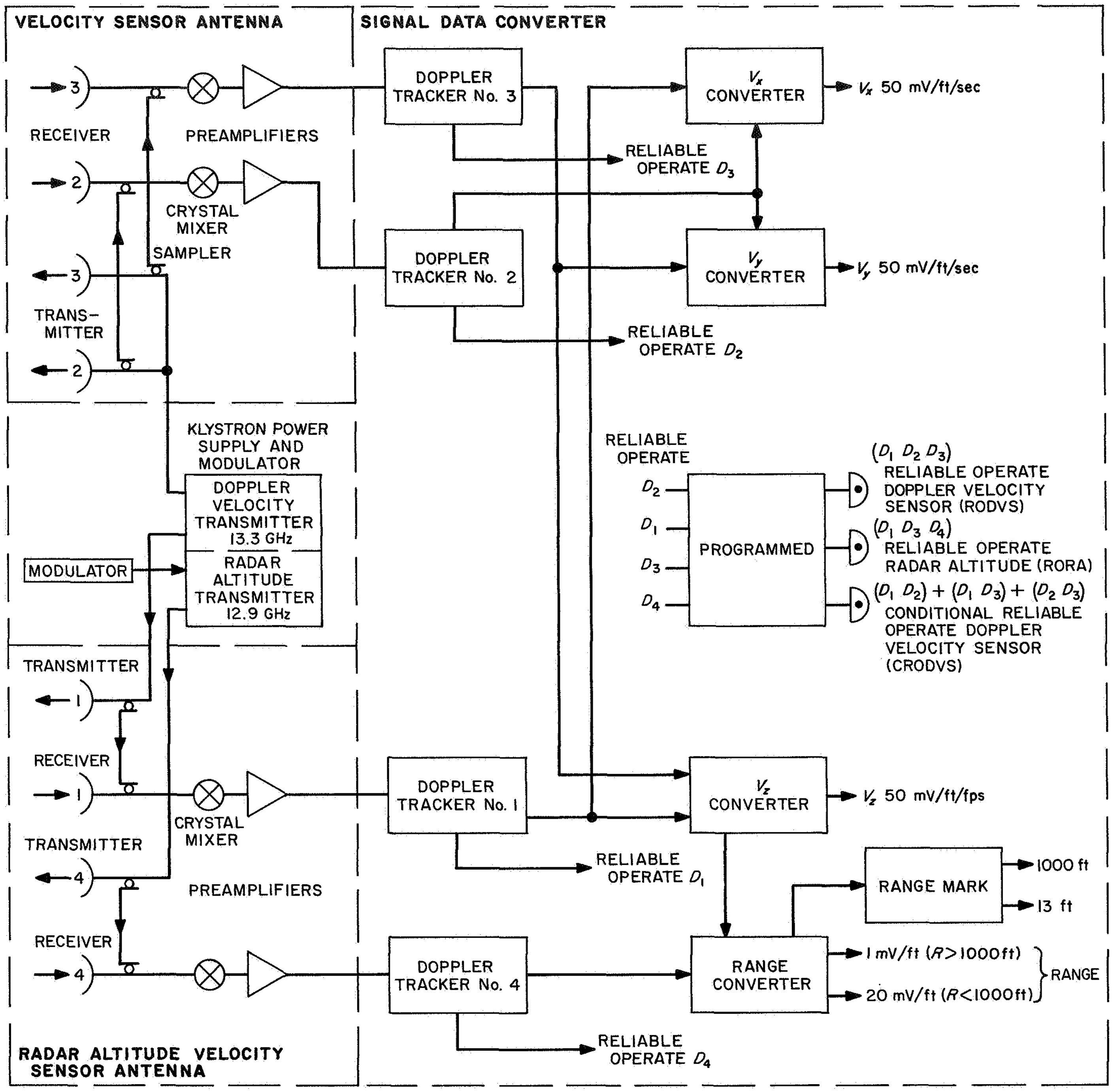


Fig. IV-38. Simplified RADVS functional block diagram

Reliability and reference circuits produce a *reliable operate doppler velocity sensor* (RODVS) signal if D_1 through D_3 lock-on signals are present beginning about 3 sec after main retro burnout. The RODVS signal is routed to the flight control electronics and to the signal processing electronics telemetry. The system is designed to produce a *conditional reliable operate doppler velocity sensor* (CRODVS) signal if only one or two, rather than all three, of the DVS beams are in lock. If the CRODVS signal occurs, the spacecraft will steer into the un-locked beam(s) to achieve lock-on of all beams and generation of RODVS. For *Surveyor IV* and later missions, the period of time that CRODVS was enabled has been extended to the *1000-ft mark* to facilitate landings at increased

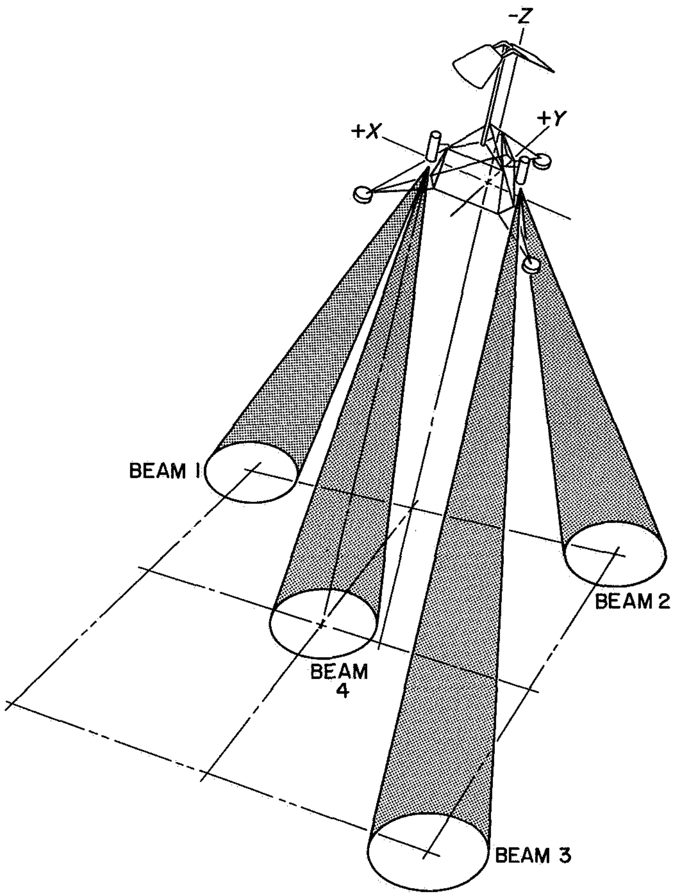


Fig. IV-39. RADVS beam orientation

approach angles. On earlier missions, CRODVS was disabled 1 sec after the three velocity trackers had achieved lock. When CRODVS is inhibited, the spacecraft switches to inertial attitude hold if beam lock is lost. Availability of CRODVS to 1000-ft altitude allows spacecraft maneuvering to reacquire lock, thus assuring greater probability of maintaining the programmed descent profile.

Cross-coupled sidelobe logic (CCSLL) is provided in the signal data converter to protect against false lock-on by a tracker of a sidelobe from another beam. The logic compares amplitude and frequencies between beams and breaks lock if the amplitude and bandwidth characteristics indicate that a cross-coupled sidelobe is being tracked. Prior to *Surveyor IV*, spacecraft had CCSLL provisions only for Beams 2 and 3. Later spacecraft have been provided with CCSLL between Beams 1, 2, and 3. Also, on the later spacecraft, the CCSLL has been disabled below 1000-ft altitude because the CCSLL is most effective at higher altitudes and may produce erroneous outputs near the lunar surface.

b. Radar altimeter description. Slant range is determined by measuring the reflection time delay between the transmitted and received signals. The transmitted signal is frequency-modulated at a changing rate so that return signals can be identified.

The RF signal is radiated, and the reflected signal is received by the altimeter/velocity sensor antenna. The received signal is mixed with two samples of transmitted energy ¾ wavelength apart, detected, and amplified by 40, 60, or 80 db in the altimeter preamp, depending on signal strength. The signals produced are difference frequencies resulting from the time lag between transmitted and received signals of a known shift rate, coupled with an additional doppler frequency shift because of the spacecraft velocity.

The altimeter tracker in the signal data converter accepts doppler shift signals and gain-state signals from the altimeter/velocity sensor antenna and converts these into a signal which is 600 kHz plus the range frequency plus the doppler frequency. This signal is routed to the altimeter converter for range dc analog signal generation.

The range mark, reliability, and reference circuits produce the 1000-ft *mark* signal and the 13-ft *mark* signal from the *range* signal generated by the altimeter converter where the doppler velocity V_z is subtracted giving the true range.

The *range mark* and *reliable operate radar altitude* (RORA) signals are routed to flight control electronics. The signals are used to rescale the *range* signal, for vernier engine shutoff and to indicate whether or not the *range* signal is reliable. The RORA signal is also routed to signal processing for transmission to DSIF.

c. RADVS performance. RADVS performance for the *Surveyor VI* mission is considered nominal. The RADVS system was turned on approximately 6.5 sec after the AMR *mark*, which initiated the automatic terminal descent sequence. Approximately 22.2 sec after RADVS primary power was applied, the radar transmitters were turned on through internal RADVS circuits.

Reflectivity of the RADVS beams is shown in Fig. IV-40 along with the preamplifier gain states. Predicted reflectivity for the DVS beams is also shown and matches the actual mission data to within 1 to 2 db. The reflectivity curve for altitude Beam 4 compares closely to previous spacecraft in signal level and gain-state position vs time to touchdown. Saturation of the IF amplifier

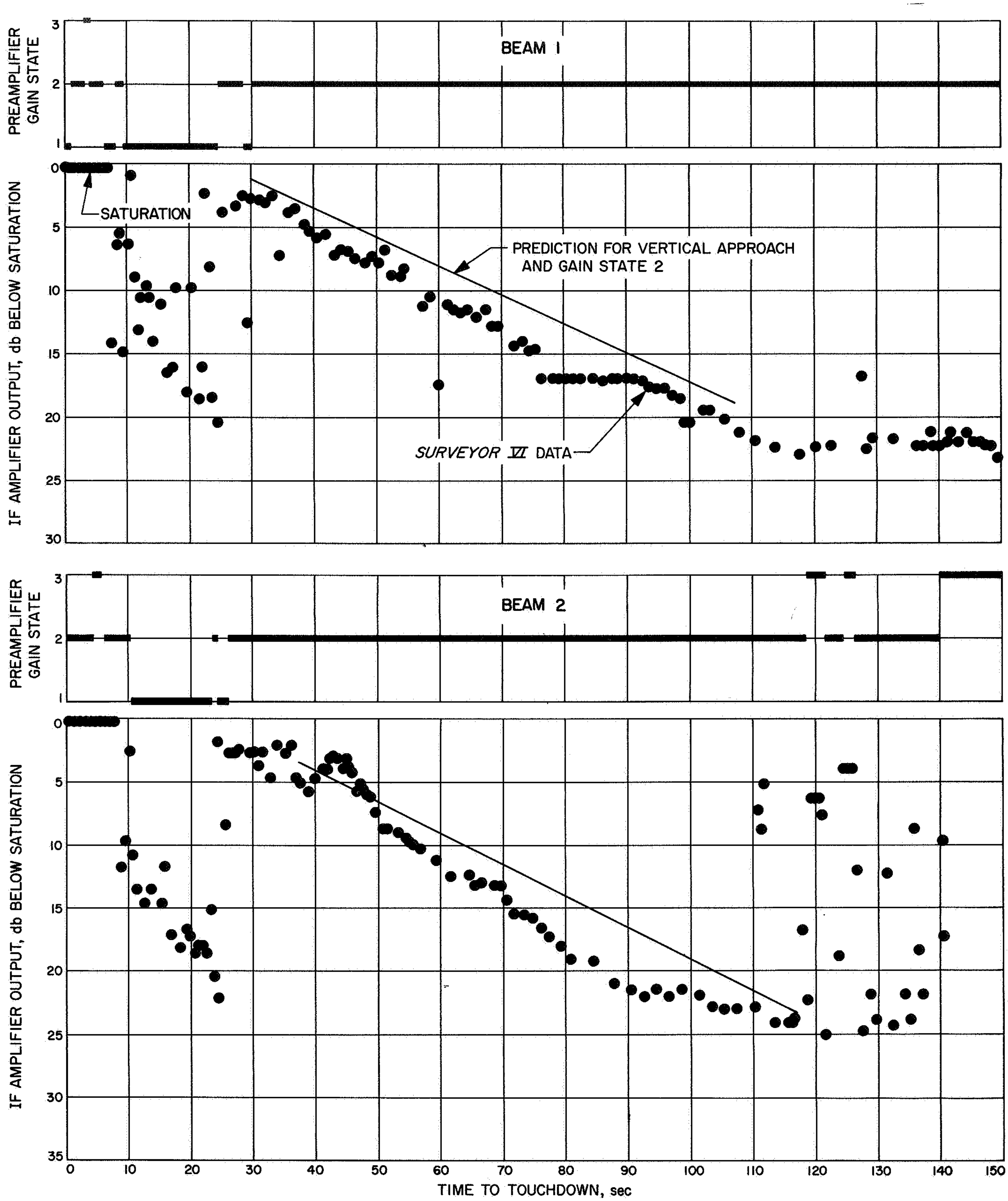


Fig. IV-40. Reflectivity of RADVS beams during terminal descent

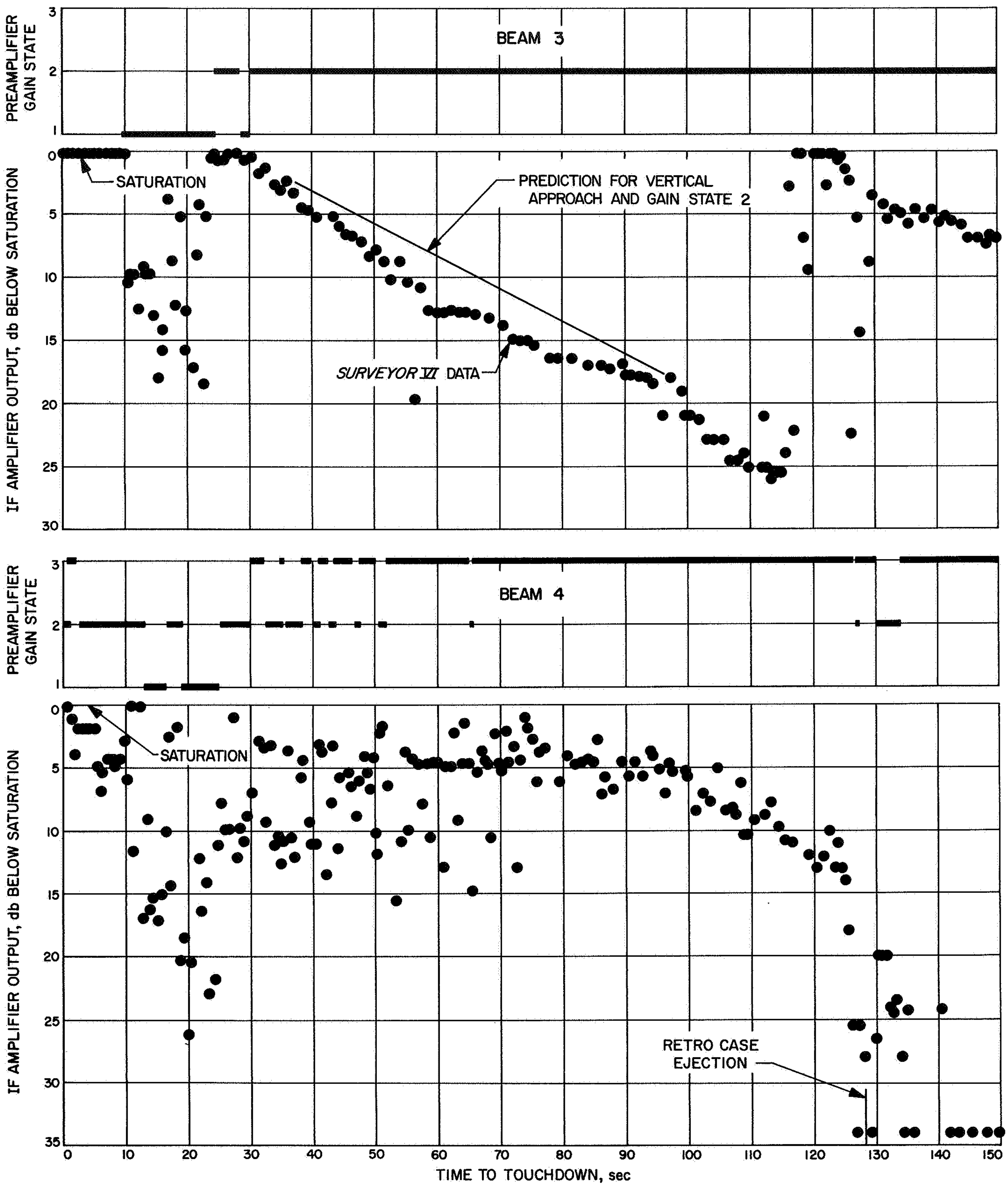


Fig. IV-40 (contd)

for the DVS beams just prior to touchdown (last 10 sec) occurred, as is expected for nominal performance.

The velocity beam trackers (Beams 1, 2, and 3) locked onto the reflected signal at about 50,000 ft slant range. The altimeter tracker locked on at a slant range of approximately 37,000 ft with a spacecraft velocity of 490 ft/sec. None of the trackers lost lock after initial signal acquisition. On previous missions various beam unlocks were noted. On the *Surveyor I* mission, Beam 3 broke lock at the approximate time of retromotor case ejection. On *Surveyor III*, Beam 4 broke lock twice, once just prior to retro case ejection and once at retro case ejection. On the missions, the trackers usually reacquired within the time required for one tracker sweep.

On the *Surveyor VI* mission, the Beam 4 preamplifier switched from high-gain state to mid-gain state prior to and at the approximate time of retro case ejection, as seen in Fig. IV-40. It is probable that Beam 4 lock would have been broken momentarily, but, at the time of case ejection, the Beam 4 tracker had not locked on. Beam 4 tracker is capable of lock-on when the spacecraft is at 40,000 ft and moving at 800 ft/sec. However, the unexplained gain-state switch just prior to retro case ejection reduced the preamp gain by approximately 20 db. This reduction of gain apparently reduced the signal level seen by the trackers below the lock threshold.

Figures IV-41 through IV-43 are plots of the velocity components along each of the three spacecraft axes during terminal descent. During spacecraft descent prior to RADVS control, the *Y*-component of velocity V_y increased to about 230 ft/sec. After initiation of RADVS control, this velocity component was removed in about 5 sec. The V_x component of velocity increased slightly to 5 ft/sec just prior to RADVS control. Figure IV-43 is a plot of spacecraft longitudinal velocity V_z vs time to touchdown. The spacecraft range vs time to touchdown is shown in Fig. IV-44. The range decreased smoothly from initial altimeter acquisition to touchdown. At approximately 15 ft altitude, the RADVS issued the 13-ft *mark* to turn the vernier engines off.

H. Telecommunications

The *Surveyor* telecommunications subsystem contains radio, signal processing, and command decoding equipment to provide (1) a method of telemetering information to the earth, (2) the capability of receiving and processing commands to the spacecraft, and (3) angle-tracking one- or two-way doppler for orbit determination.

1. Radio Subsystem

The radio subsystem utilized on the *Surveyor* spacecraft is shown schematically in Fig. IV-45. Dual receivers, transmitters, and antennas were originally meant to provide redundancy for added reliability. As presently mechanized, this is not completely true owing to switching limitations. Each receiver is permanently connected to its corresponding antenna and transmitter. The transmitters, which are capable of operation in two different power modes (100-mW low power, and 10-W high power), can each be commanded to transmit through any of the three antennas. The *Surveyor* radio system operates at S-band, 2295-MHz down-link, and 2113-MHz up-link.

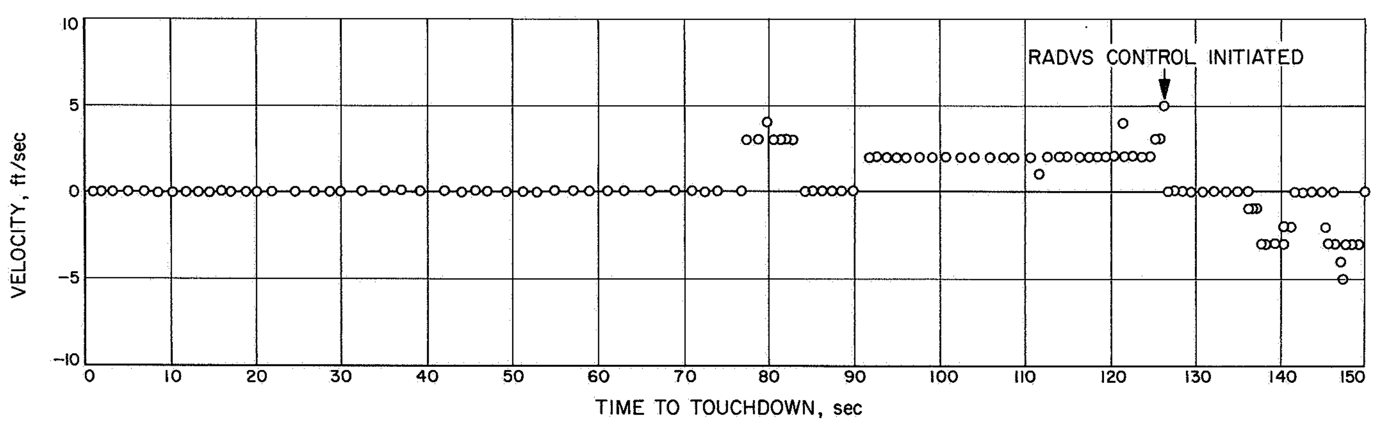


Fig. IV-41. X-component of velocity V_x during descent

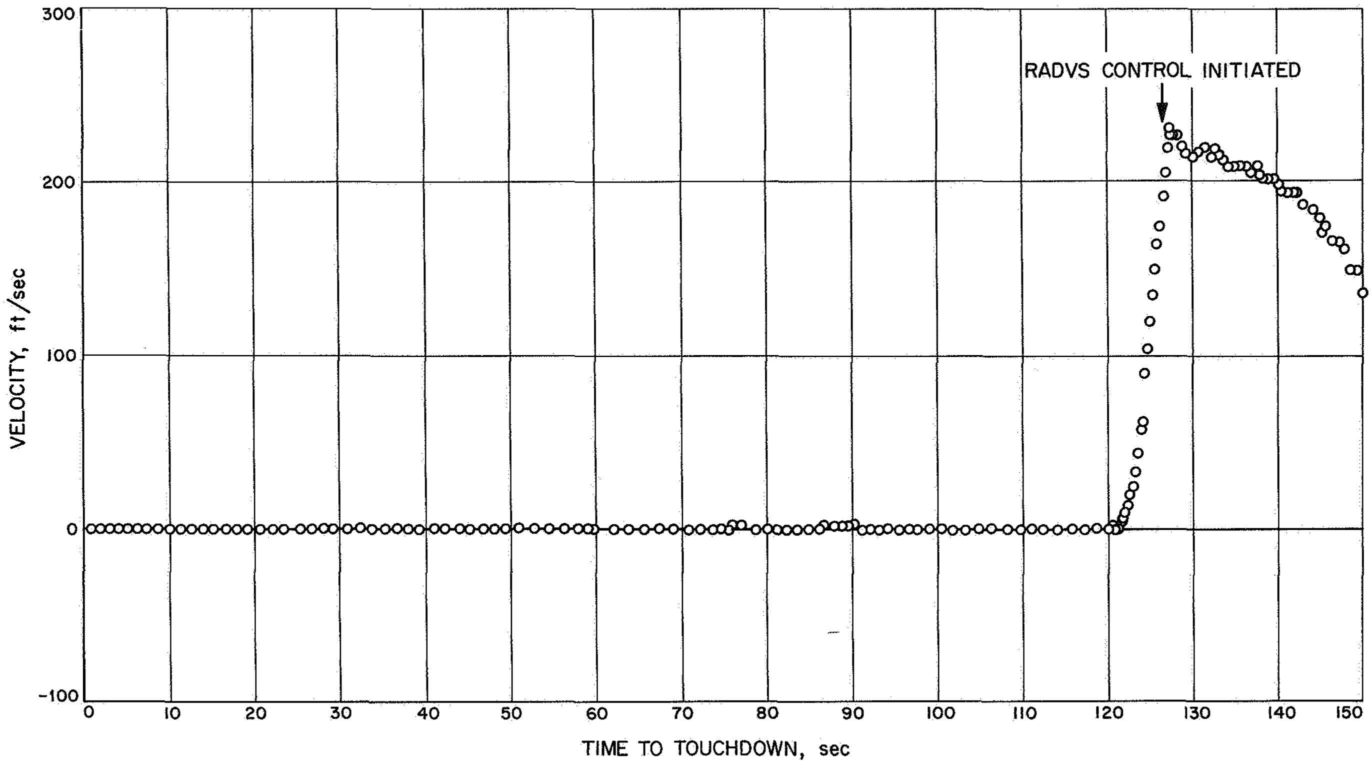


Fig. IV-42. Y-component of velocity V_y during descent

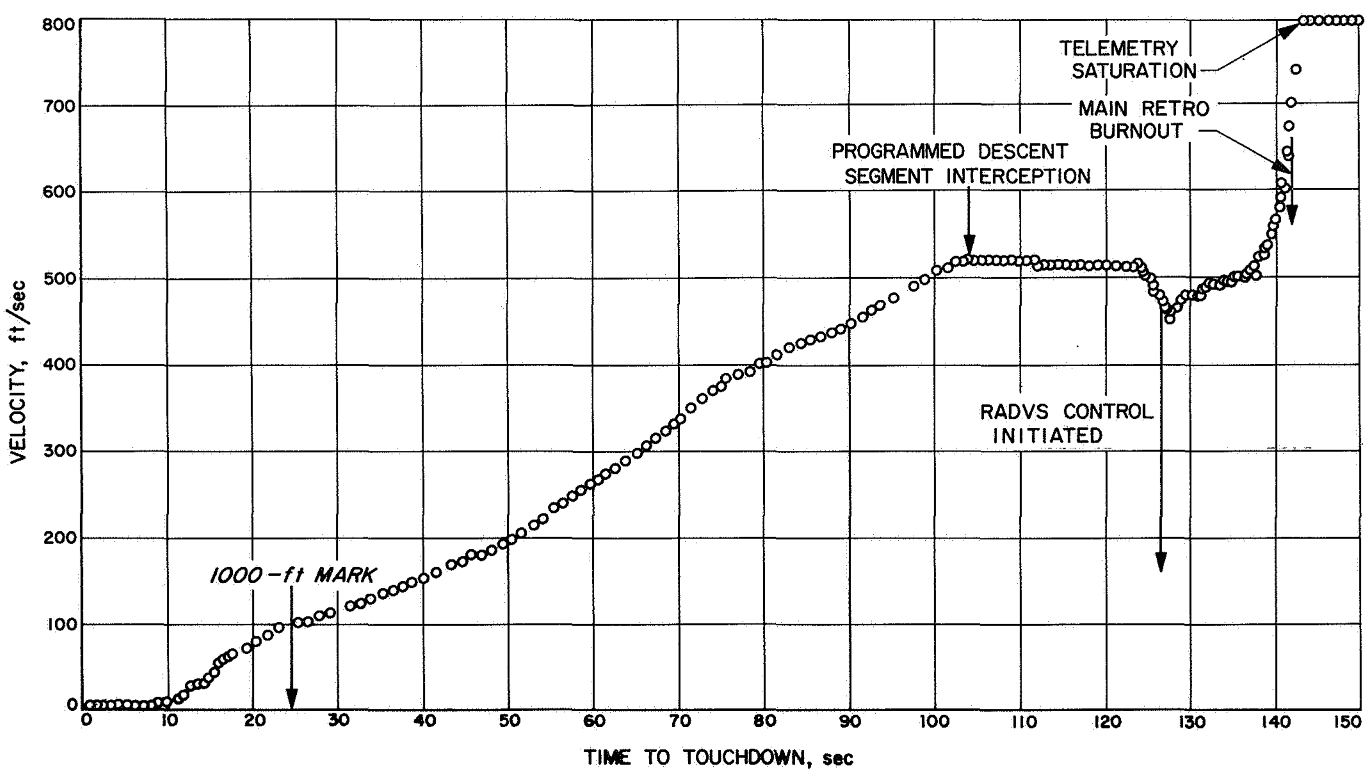
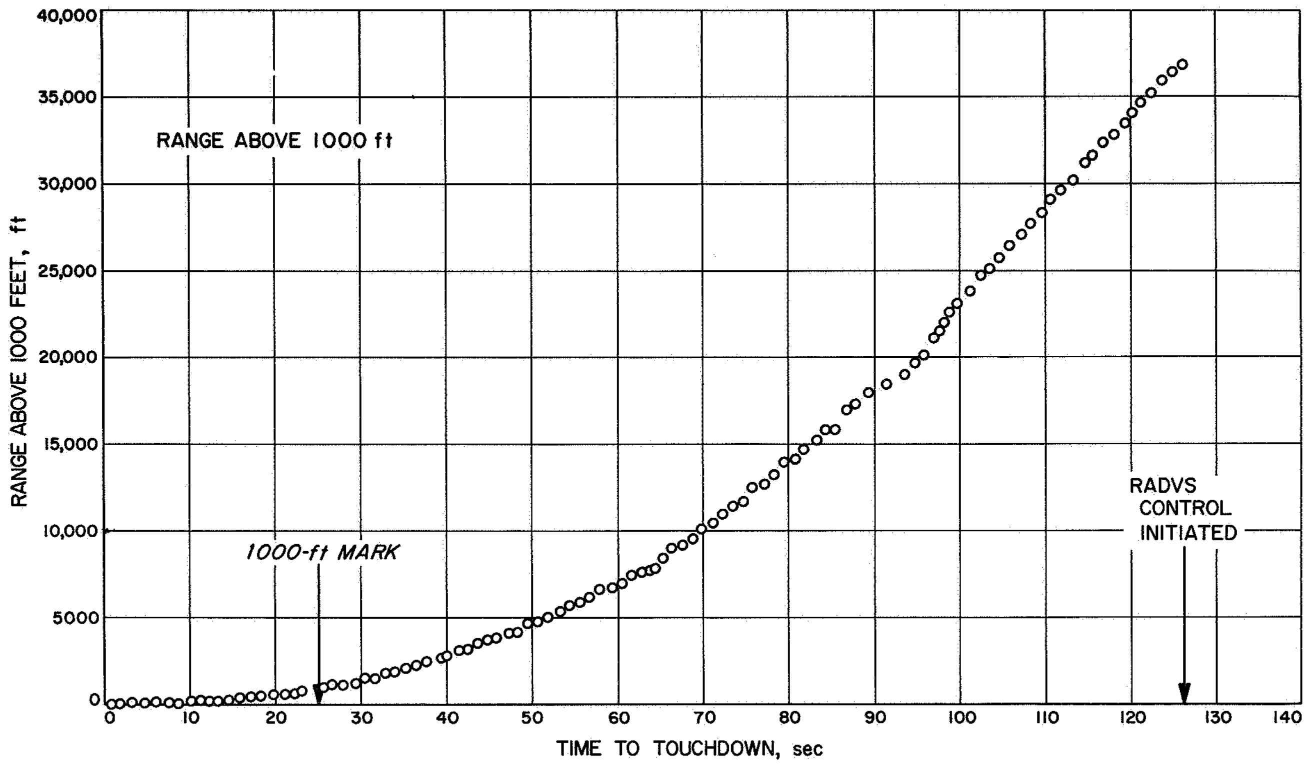


Fig. IV-43. Z-component of velocity V_z during descent



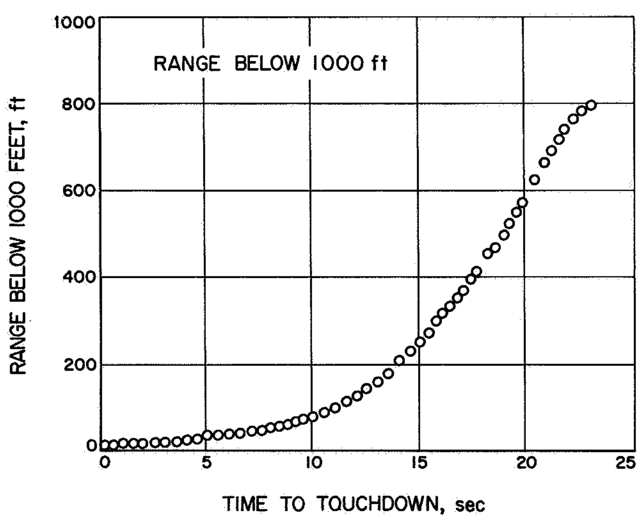


Fig. IV-44. Slant range vs time during terminal descent

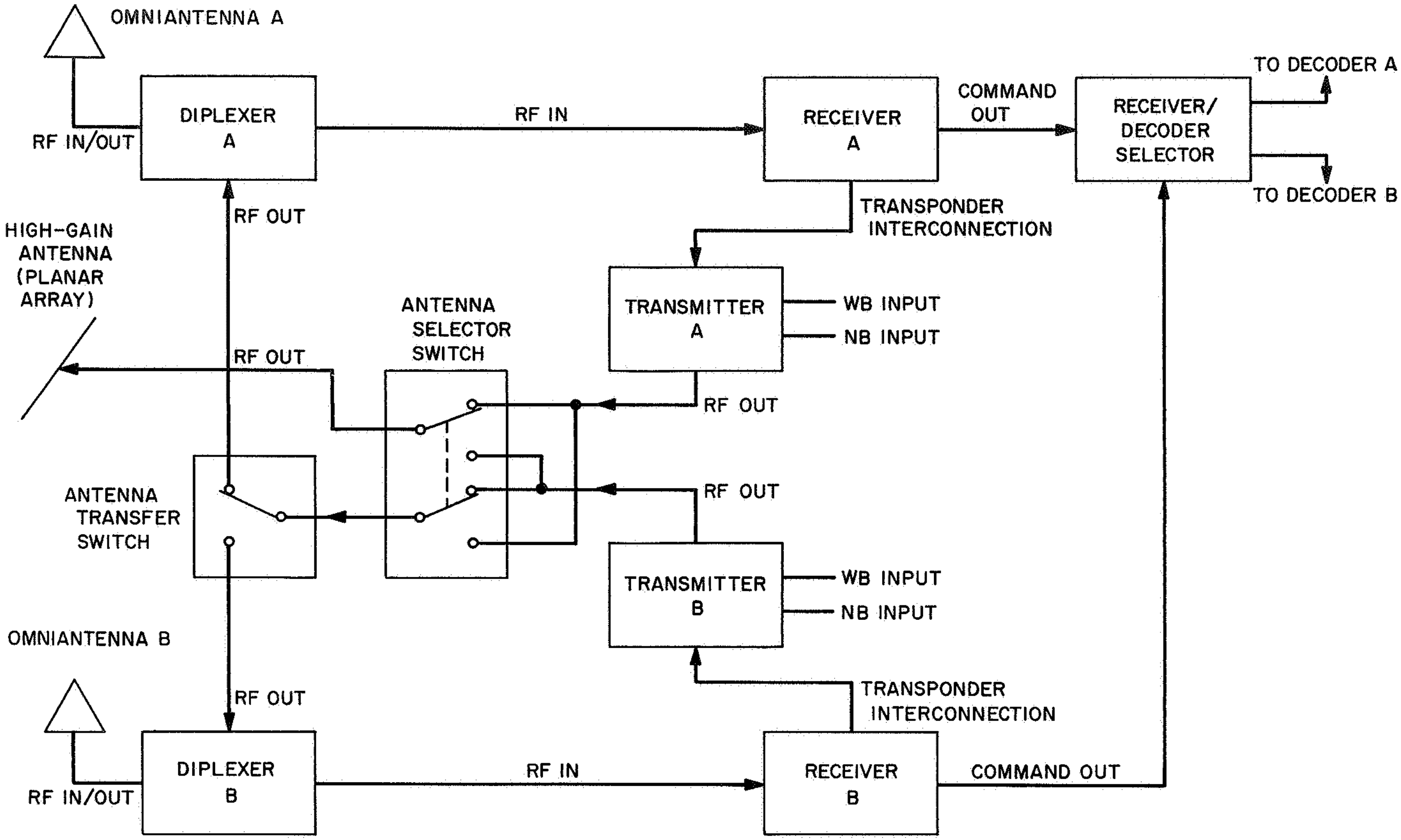


Fig. IV-45. Radio subsystem block diagram

a. Receivers. Both receivers are identical, crystal-controlled, double-conversion units, which operate continuously as they cannot be commanded off. Each unit is capable of operation in an automatic frequency control (AFC) mode, or an automatic phase control (APC) mode. The receivers provide two necessary spacecraft functions: the detection and processing of commands from the ground stations for spacecraft control (AFC and APC modes), and the phase-coherent spacecraft-to-earth signal required for doppler tracking (APC mode).

b. Transmitters. Transmitters A and B are identical units which provide the spacecraft-to-earth link for telemetry and doppler tracking information. The transmitters are commanded on, one at a time, from the ground stations. Each unit contains two crystal-controlled oscillators, wideband for TV and scientific information, narrowband for engineering data. Either unit can be commanded on at will, and, in addition, can operate from the receiver voltage-controlled oscillator (transponder mode) when coherent signals are required for two-way doppler tracking. The transmitters may be commanded to operate through any one of the spacecraft antennas as desired, and both are capable of providing either 100 mW or 10 W of output power.

c. Transponders. Two identical transponder interconnections permit each transmitter to be operated, on command, in a transponder mode. In the transponder mode, a transmitter is operated with the corresponding receiver voltage-controlled oscillator to provide coherent signals when two-way doppler tracking data is required.

d. Antennas. Three antennas are utilized on the *Surveyor* spacecraft. Two antennas are omnidirectional units which provide receive-transmit capability for the spacecraft. The third antenna is a high-gain, 27-db directional planar array used for transmission only of wideband information.

e. Radio subsystem performance. The radio subsystem performed well during the *Surveyor VI* mission, with no indications of abnormal behavior. The transmitter went to high power upon command from the *Centaur* shortly before spacecraft separation. The spacecraft was acquired by DSS 51, Johannesburg, about 35 min after launch, and the transmitter was then commanded to low-power operation.

Canopus lock was obtained about 9 hr after launch. Prior to Canopus acquisition, while the spacecraft was rolling to generate a star map in high-power and transponder (two-way) mode, DSS 61 lost receiver lock. The spacecraft roll was terminated temporarily to investigate the problem. Observations of an intermittent signal at DSS 61 resulted in transmission of the *transponder off* command. The problem was apparently caused by a deeper-than-predicted null in the Omniantenna B up-link antenna pattern, which caused the spacecraft receiver to drop phase lock, resulting in the spacecraft automatically switching to narrow-band VCXO mode. The resulting shift in frequency caused loss of DSS lock. The intermittent signal at DSS 61 was probably caused by the spacecraft transponder going in and out of phase lock with a sideband caused by the presence of command modulation in the up-link signal. This explanation seems reasonable since, after the transponder mode was commanded off, the signal was steady in the AFC mode. The spacecraft star mapping roll was resumed with transponders off. After automatic Canopus lock-on was achieved, two-way lock was reestablished and the spacecraft was returned to the low-power mode.

Plots of up-link signal levels for Receivers A and B during the Canopus acquisition roll are shown in Figs. IV-46 and IV-47. Comparison of the actual and predicted Omniantenna B antenna gains shows that loss of lock occurred while the earth vector was in a null region. The null apparently was greater than −50 db, which is much deeper than the predicted data indicates.

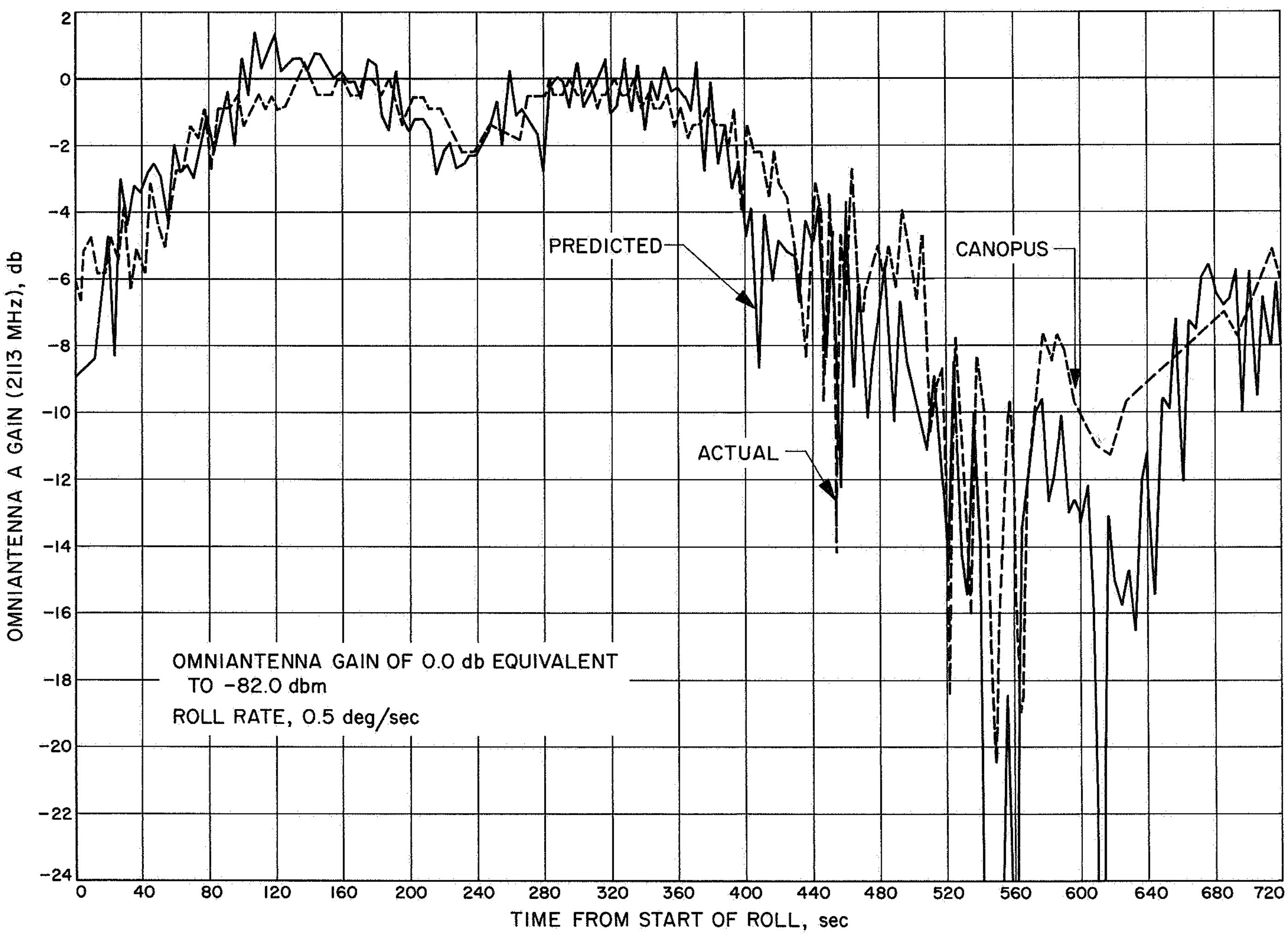


Fig. IV-46. Spacecraft Receiver A signal level during star verification/acquisition roll

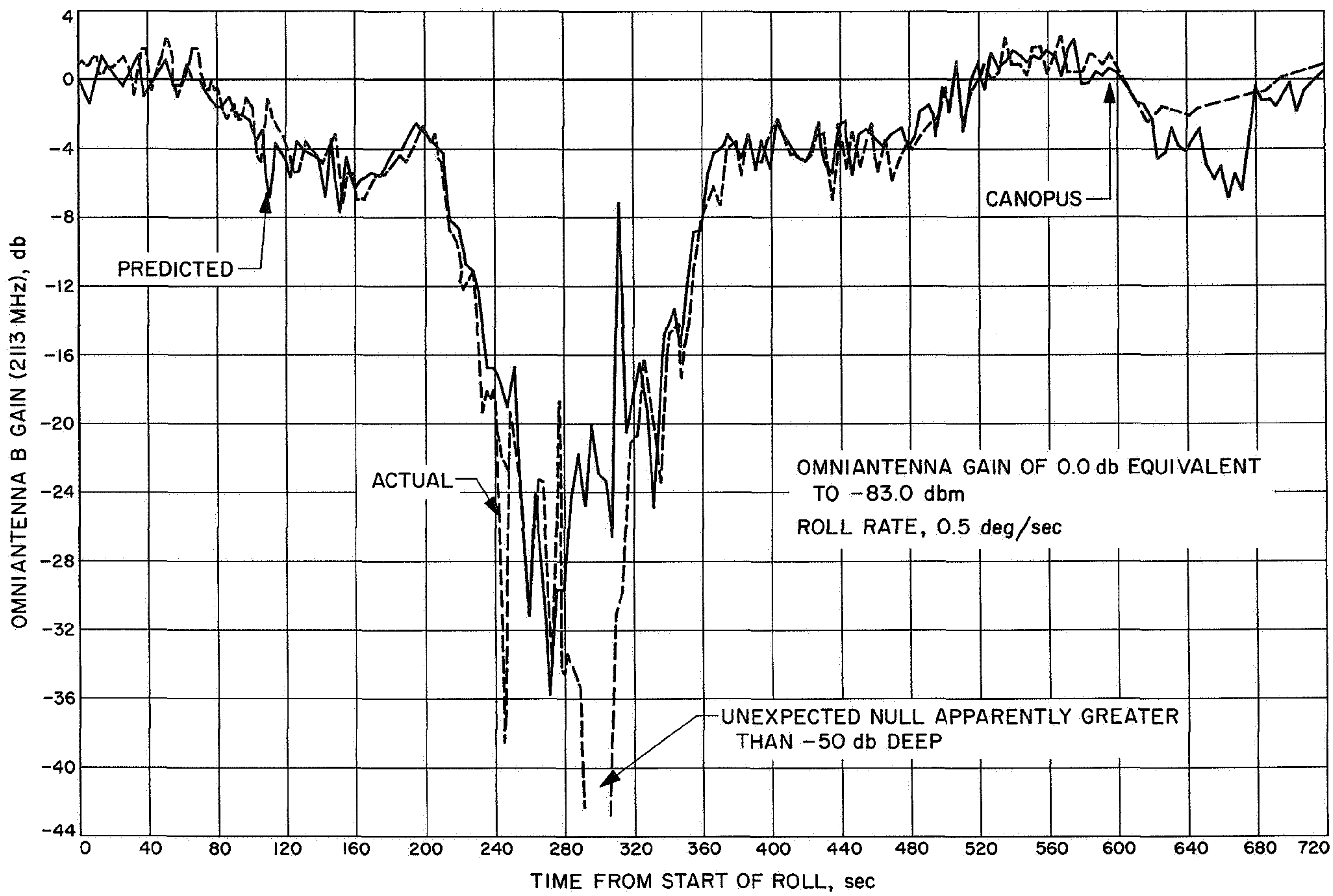


Fig. IV-47. Spacecraft Receiver B signal level during star verification/acquisition roll

However, omnidirectional antenna patterns are difficult to predict accurately based on preflight measurements, and low-gain regions are especially difficult to define.

Receiver A/Omniantenna A performance for the flight is plotted in Fig. IV-48. The signal strength to Receiver A was above the predicted levels. Prior to Canopus acquisition, the lack of spacecraft roll attitude reference was the cause of the large deviation between predicted and measured signal strength. This difference is of no concern since predicted signal strength is not expected to be valid during this period. For a period of 10 hr after Canopus acquisition, differences amounting to as much as 12 db between predicted and actual signal strengths were experienced. These differences may be due to differences between the predicted and actual antenna patterns and look angles. During later portions of the flight, the actual signal strength was only 2 to 4 db from the predicted level. Similar variations have been noted on preceding flights, and are probably due to measurement errors in the antenna patterns.

Receiver B/Omniantenna B performance for the flight is plotted in Fig. IV-49. Again, the predicted signal strength is not expected to be valid prior to Canopus lock, and deviations between predicted and measured signal strength are of no concern. After Canopus lock, the received signal strength remained 4 to 5 db below the predicted levels. However, the received signal strength remained well above the command threshold level of −114 dbm, and no difficulty in commanding the spacecraft was experienced during the entire transit to the moon.

A plot of the down-link signal strength vs the predicted signal strength is shown in Fig. IV-50. Actual down-link signal strength remained 1 to 2 db higher than the predicted levels through the entire flight.

As on previous missions, high power and one-way mode were used for terminal descent. The DSS 11 received carrier power was −124.8 dbm prior to main

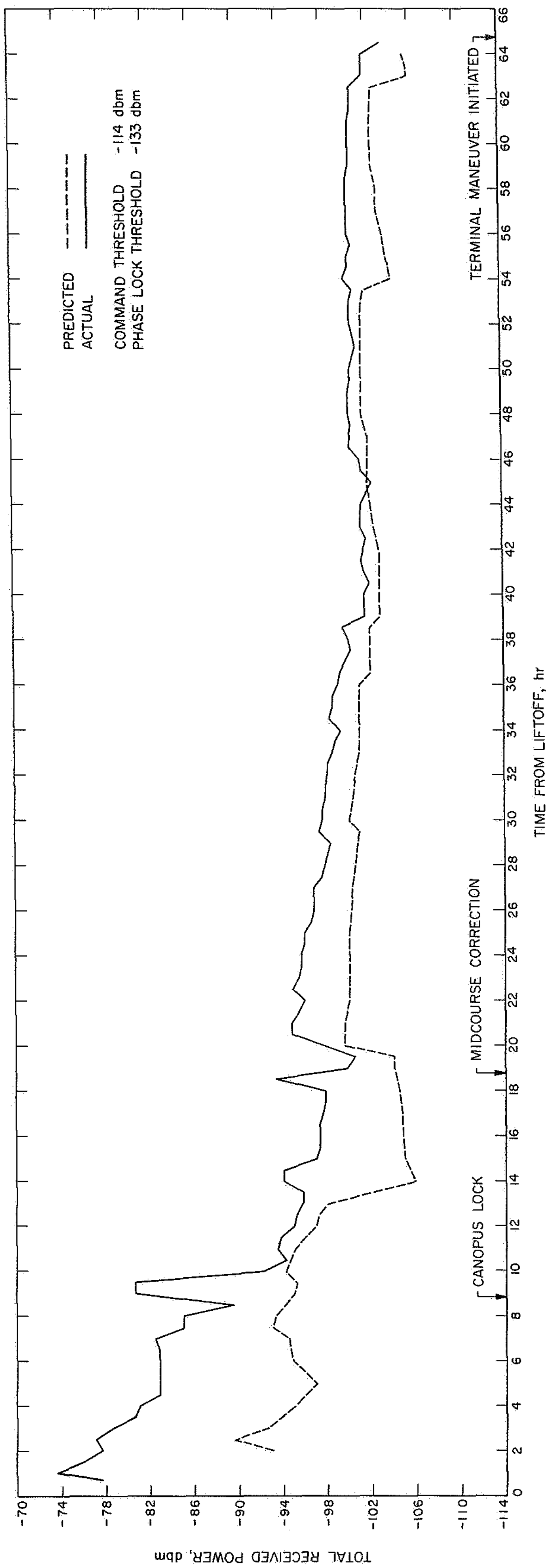


Fig. IV-48. Receiver A/Omniantenna A total received power during transit

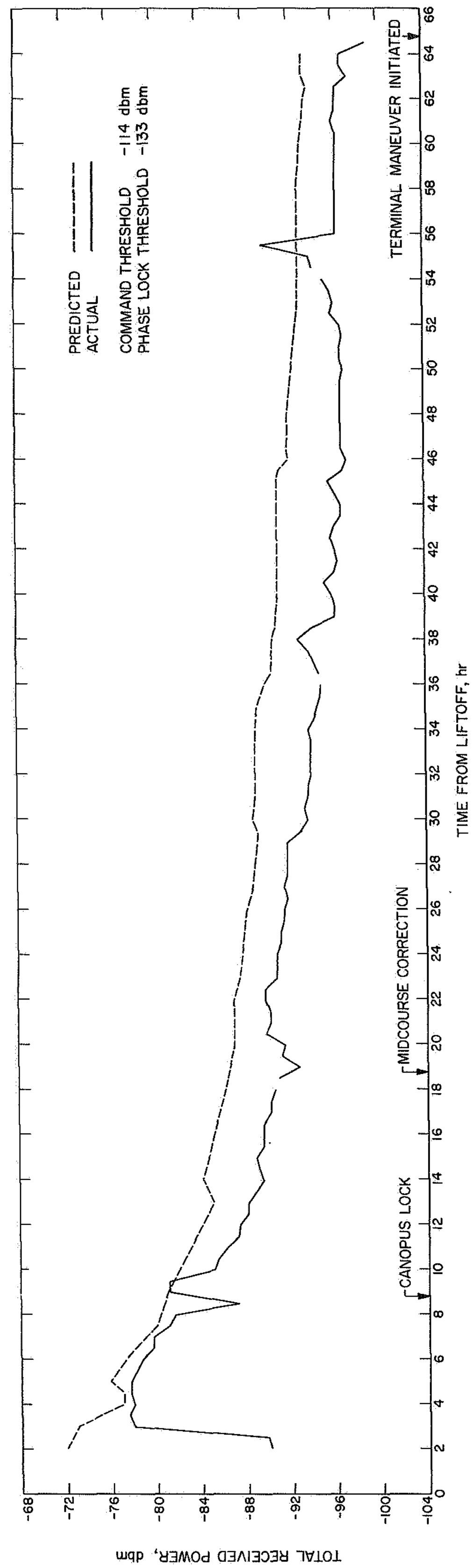


Fig. IV-49. Receiver B/Omniantenna B total received power during transit

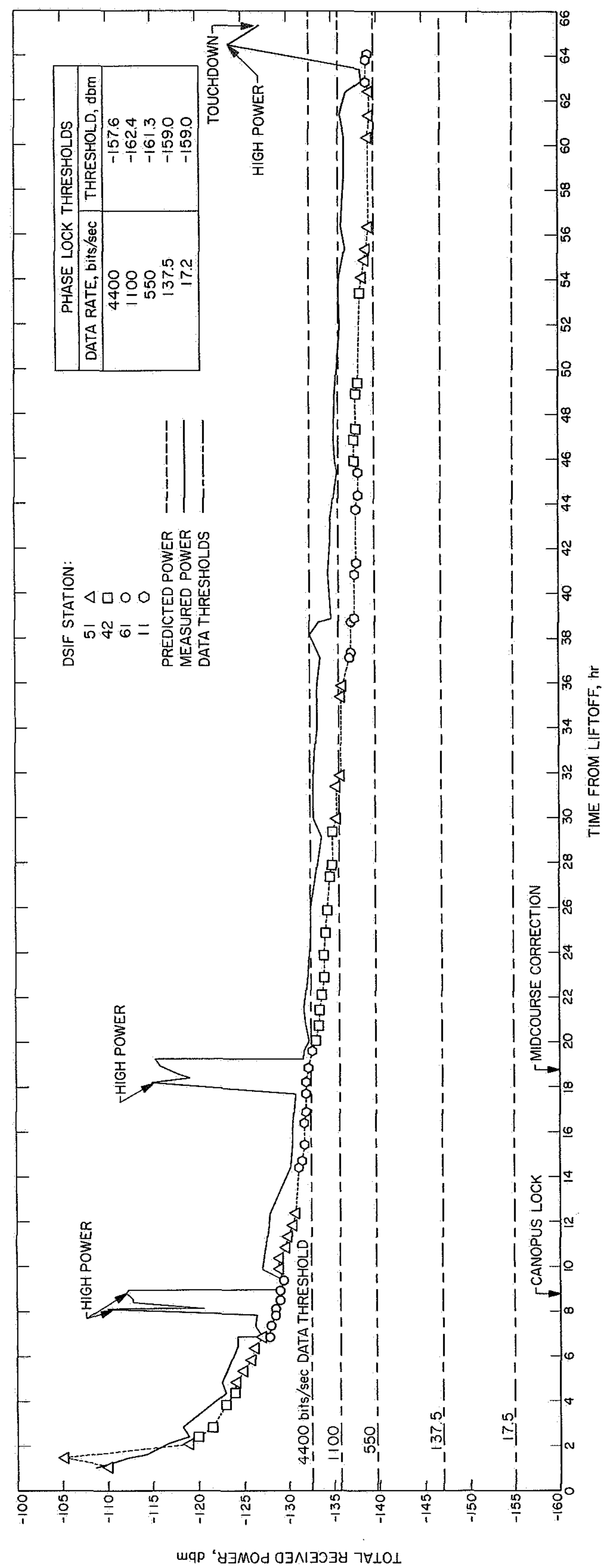


Fig. IV-50. DSIF received power during transit

retro ignition, which was in agreement with the predicted nominal value. During the main retro burn period, the signal level at DSS 11 remained steady during the main retro phase. After main retro burnout, when the spacecraft began the steering phase of descent, signal level variations of 2.7 db were noted. The minimum signal level during descent was −127.5 dbm, which agrees with predictions.

Carrier power at DSS 11 was reported at −126.9 dbm at touchdown. Receiver lock was maintained through the landing event and good PCM and strain gage data was obtained. A summary of telecommunications performance values during the transit phase is presented in Table IV-17.

During the postlanding spacecraft hop experiment, two commands were transmitted to the spacecraft at ½-sec intervals to cut off the vernier engines. Spacecraft telemetry indicates that the first command was not accepted and that it was the second command that terminated vernier thrust. Telemetry data also indicates that the signal level of both spacecraft receivers was above the command threshold value and changed very slightly during the sequence.

The DSIF system performance was investigated, and it was determined that both commands were transmitted correctly. A receiver at the DSS samples the radiated RF and processes the command information for comparison with the command subcarrier oscillator output. A positive comparison was made for both commands, with the record showing that both commands were properly modulated on the subcarrier.

The most probable cause for the command rejection, which is the first known to have occurred, is an RF multipath null that existed for so short a time that the AGC time constant did not allow the associated low signal level to be indicated in spacecraft telemetry.

Two spacecraft RF performance tests were conducted during the lunar day, on November 14 and November 21. These assessments, which exercised the subsystem in all possible transmitting and command receiving configurations, indicated that performance of the telecommunications subsystem was nominal in all respects.

2. Signal Processing Subsystem

The *Surveyor* signal processing subsystem accepts, encodes, and prepares for transmission voltages, currents, and resistance changes corresponding to various spacecraft parameters such as events, voltages, temperatures, and accelerations.

a. Signal processing functional description. The signal processor employs both pulse code modulation (PCM) and amplitude-to-frequency modulation telemetry techniques to encode spacecraft signals for frequency modulation (FM) or phase modulation (PM) of the spacecraft transmitter signals, and for recovery of these signals by the ground telemetry equipment. A simplified block diagram of the signal processor is shown in Fig. IV-51.

The input signals to the signal processor are derived from various voltage or current pickoff points within the other subsystems as well as from standard telemetry transducing devices such as strain gages, temperature transducers, and pressure transducers. These signals generally are conditioned to standard ranges by the originating subsystem so that a minimum amount of signal conditioning is required by the signal processor.

As illustrated in Fig. IV-51, some of the signal inputs are commutated to the input of the analog-to-digital converter (ADC) while others are applied directly to subcarrier oscillators. The measurements applied directly are accelerometer and strain gage measurements which require continuous monitoring over the short intervals in which they are active.

The commutators apply the majority of telemetry input signals to the ADC, where they are converted to a digital word. Binary measurements such as switch closures or contents of a digital register already exist in digital form and are therefore routed around the ADC. In these cases, the commutator supplies an inhibit signal to the ADC and, by sampling, assembles the digital input information into 10-bit digital words. The commutators are comprised of transistor switches and logic circuits which select the sequence and number of switch closures. There are six engineering commutator configurations (or modes) used to satisfy the telemetry requirements for the different phases of the mission and one TV commutator configuration located in the TV auxiliary. The content of the telemetry modes is presented in Appendix C.

The ADC generates an 11-bit digital word for each input signal applied to it. Ten bits of the word describe the voltage level of the input signal, and the eleventh bit position is used to introduce a bit for parity checking by

Table IV-17. Telecommunications performance values[a] during transit

Mission phase	Parameter	Predicted	Observed
DSIF acquisition	Transmitter B frequency, MHz	2294.989381	2294.987892
	Receiver B frequency, MHz	2113.318564	2113.320576
Star acquisition roll (4400 bits/sec, high power)	Receiver A signal level		
	variation from nominal, db	±10	+8, −4
	minimum, dbm		−102.5
	Receiver B signal level		
	variation from nominal, db	±10	+5, −20
	minimum, dbm		−130
	Carrier level at DSS		
	variation from nominal, db	±10	+2, −7
	minimum, dbm		−153
Coast (low power)	Receiver A signal level		
	variation from nominal, db	±10[b]	+7, 0
	minimum, dbm		−103
	Receiver B signal level		
	variation from nominal, db	±10[b]	+2, −4
	minimum, dbm		−99.0
	Carrier level at DSS (550 bits/sec)		
	variation from nominal, db	±5[b]	+3.5, −1.5
	minimum, dbm		−141
Midcourse maneuvers (4400 bits/sec, high power)	Receiver A signal level total variation, db	22	17.1
	Receiver B signal level		
	total variation, db	14.2	12.1
	minimum, dbm		−99.5
	Carrier level at DSS		
	total variation, db	5.8	5.6
	minimum, dbm		−126.9
Midcourse thrusting (4400 bits/sec, high power)	Carrier level variation at DSS, db		
Terminal maneuvers (1100 bits/sec, high power)	Receiver A signal level		
	total variation, db	14.6	13.8
	minimum, dbm		−108.0
	Receiver B signal level		
	total variation, db	11.0	11.6
	minimum, dbm		−106.1
	Carrier level at DSS		
	total variation, db	5.6	5.9
	minimum, dbm		−127.2
Terminal descent (1100 bits/sec, high power)			
prior to retro ignition	Carrier signal level at DSS, dbm		−124.8
during descent	Carrier signal level at DSS		
	total variation, db	2.0	2.7
	minimum, dbm		−127.5
at touchdown	Carrier signal level at DSS, dbm		−126.9

[a]All signal levels for 85-ft DSIF antenna. Adjustment for the DSS 14 210-ft antenna is +7.5 db minimum.

[b]Not considering variations due to gyro drift checks.

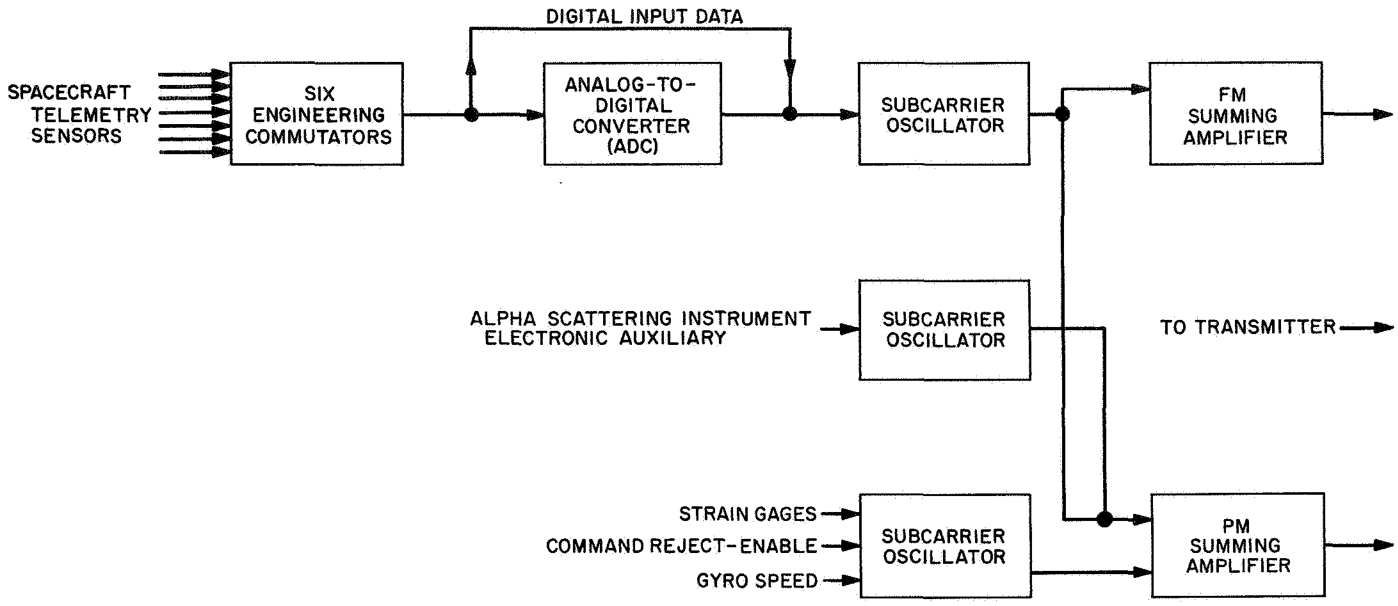


Fig. IV-51. Simplified signal processing functional block diagram

the ground telemetry equipment. The ADC also supplies commutator advance signals to the commutators at one of five different rates. These rates enable the signal processor to supply telemetry information at 4400, 1100, 550, 137.5, and 17.2 bits/sec. The bit rates and commutator modes are changed by ground commands.

The subcarrier oscillators are voltage-controlled oscillators used to provide frequency multiplexing of the telemetry information. This technique is used to greatly increase the amount of information which is transmitted on the spacecraft carrier frequency.

The summing amplifiers sum the outputs of the subcarrier oscillators and apply the composite signal to the spacecraft transmitters. Two types of summing amplifiers are employed because of the transmitter's ability to transmit either a phase-modulated or a frequency-modulated signal.

The signal processing subsystem employs a high degree of redundancy to insure against loss of vital spacecraft data. Two ADC's, two independent commutators—the engineering signal processor (ESP) and auxiliary engineering signal processor (AESP)—and a wide selection of bit rates (each with the ADC driving a different subcarrier oscillator) provide a high reliability of the signal processing subsystem in performing its function.

b. Mission performance. From launch to touchdown on the *Surveyor VI* mission, 159 commands effecting changes in the signal processor were received and properly executed by the signal processor. Table IV-18 shows representative prelaunch and in-flight values of the telemetered signal processing parameters. No signal processing anomalies were exhibited from launch to shutdown shortly after sunset.

Table IV-18. Typical signal processing parameter values

Parameter	Prelaunch	Flight
ESP reference volts, V (Mode 2)	4.871	4.871
ESP reference volts, V (Mode 4)	4.890	4.890
ESP reference return, V	0.000	0.000
ESP unbalance current, μA (Mode 2)	0.5140	0.5141
ESP unbalance current, μA (Mode 4)	0.3203	0.3203
ESP full-scale current calibration, % error from nominal	−0.491	−0.491
ESP mid-scale current calibration, % error from nominal	−0.512	−0.512
ESP zero-scale current calibration, % error from nominal	−0.380	−0.380
AESP full-scale current calibration, % error from nominal	0.0769	0.0769
AESP mid-scale current calibration, % error from nominal	0.0182	0.0182
AESP zero-scale current calibration, % error from nominal	0.347	0.347
AESP unbalance current, μA	−0.762	−0.762
AESP reference volts, V	4.943	4.943

During the second coast phase, the bit error rate became excessively high. This was due to operation of the spacecraft telemetry system at a high bit rate, for which the bit error rate was in excess of the specified limit of 3×10^{-3}.

An example of the excessive bit error rate, using the reference voltage channel, is shown in Fig. IV-52. This figure indicates a bit error rate of 15×10^{-3} prior to the point where the bit error rate could no longer be estimated owing to loss of sync information. When the spacecraft data rate was reduced to 550 bits/sec, the bit error rate dropped to zero.

3. Command Decoding Subsystem

From liftoff to touchdown on the moon, the *Surveyor VI* mission required a total of 359 earth commands. Nine of these were quantitative commands (QC's) which provided the spacecraft with the quantitative information necessary for attitude and trajectory correction maneuvers; the other 350 were direct commands (DC's) which initiated single actions such as *extend omniantennas, AMR power on, A/D clock rate 1100 bits/sec,* etc.

These commands were received, detected, and decoded by one of the four receiver/central command decoder (CCD) combinations possible in the *Surveyor* command subsystem. The selection of the combination is accomplished by stopping the command information from modulating the up-link radio carrier for ½ sec. Once the selection is made, the link must be kept locked up continuously by either sending serial command words or unaddressable command words (referred to as fill-ins) at the maximum command rate of 2 words/sec.

The command information is formed into a 24-bit Manchester-coded digital train and is transmitted in a PCM/FM/PM modulation scheme to the spacecraft. When picked up by the spacecraft omniantennas, the radio carrier wave is stripped of the command PCM information by two series FM discriminators and a Schmitt digitizer. This digital output is then decoded by the CCD for word sync, bit sync, the 5-bit address and its complement, and the 5-bit command and its complement (this latter only for DC's since the QC's contain 10 bits of information rather than 5 command bits and their complements). The CCD then compares the address with its complement and the command with its complement on a bit-by-bit basis. If the comparisons are satisfactory, the CCD then selects that one of the eight subsystem decoders (SSD's) having the decoded address bits as its address, applies power to its command matrix, and then selects that one of the 32 matrix inputs having the decoded command bits as its address to issue a 20-msec pulse which initiates the desired single action.

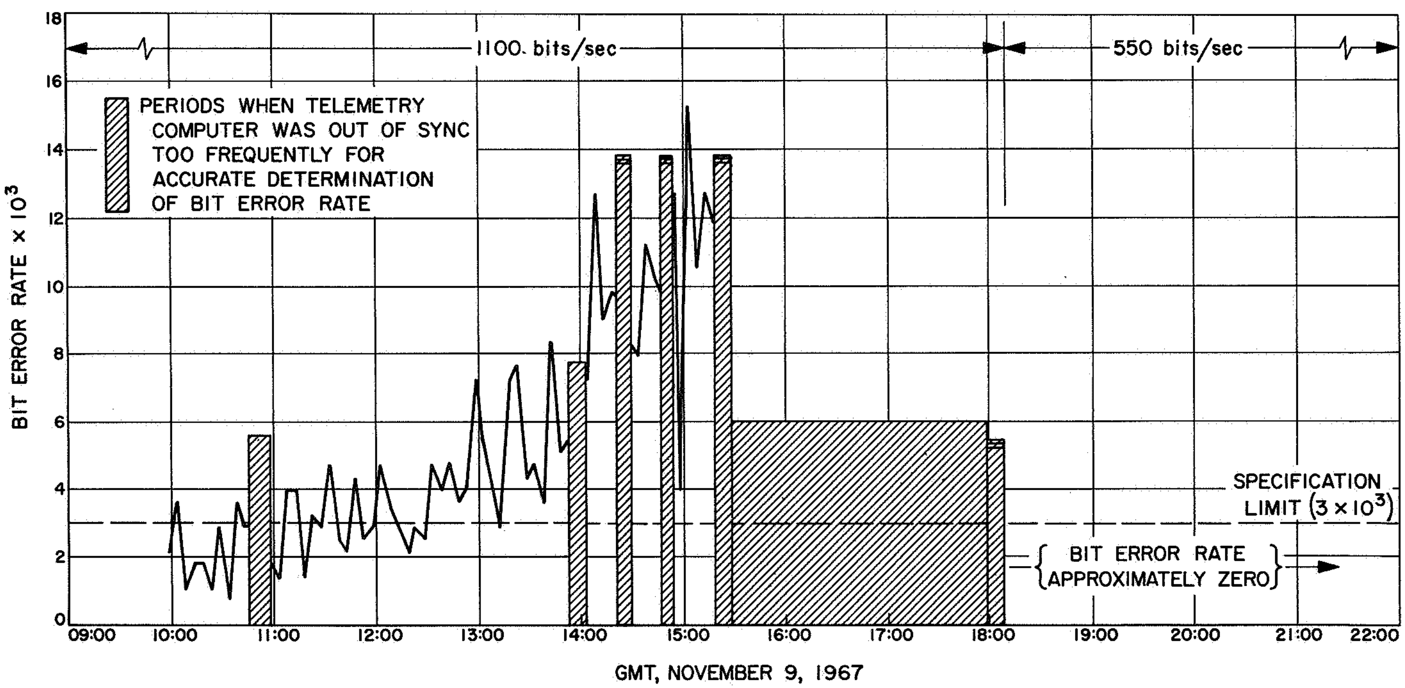


Fig. IV-52. Incidence of high bit error rate during transit (AESP reference voltage illustrated)

Those DC commands that are irreversible or extremely critical are interlocked with a unique command word. Ten of the DC's and all of the QC's are in this special category. None of these commands can be initiated if the interlock command word is not received immediately prior to the critical command.

The QC's, besides being interlocked, are also treated somewhat differently by the command subsystem. The only differences between the DC and QC are: (1) a unique address is assigned the QC words; (2) the QC word contains 10 bits of quantitative information in place of the 5 command and 5 command complement bits. Therefore, when this unique QC address is recognized, the CCD selects the flight control sensor group (FCSG) SSD and shifts the 10 bits of quantitative information into the FCSG magnitude register. Hence, the QC quantitative bits are loaded as they are decoded.

The command decoding subsystem performed properly throughout flight and through touchdown. No substandard behavior was evidenced during lunar operations. The spacecraft has responded to 162,893 commands from launch to shutdown shortly after sunset.

I. Television

The television subsystem is designed to obtain photographs of the lunar surface, lunar sky, and portions of the landed spacecraft. For missions beginning with *Surveyor III,* the subsystem has consisted of only a single television camera capable of panoramic viewing and a television auxiliary for processing commands and telemetry data.

The *Surveyor VI* television system provided the greatest quantity of pictures and the best quality pictures yet obtained. Contributing to this was the improved camera system, used for the first time on the *Surveyor VI* mission, which incorporates several design changes.

1. Survey Camera

Figure IV-53 illustrates the survey television camera, which differs in appearance from the earlier cameras because of a redesigned mirror hood assembly. The camera contains a mirror, filters, lens, shutter, vidicon, and the attendant electronic circuitry. Images can be obtained over a 360-deg panorama and over an elevation range from +40 deg above the plane normal to the camera Z-axis to −60 deg below this same plane. The camera Z-axis is inclined 16 deg from the spacecraft Z-axis. Each picture, or frame, is imaged through an optical system onto a vidicon image sensor whose electron beam scans a photoconductive surface, thus producing an electrical output proportional to the conductivity changes resulting from the varying reception of photons from the subject. The camera is designed to accommodate scene luminance levels from approximately 0.008 ft-lamberts to 2600 ft-lamberts, employing various means of exposure control. Functionally, the camera provides a resolution capability of approximately 1 mm at 4 meters and can focus from 1.23 meters to infinity.

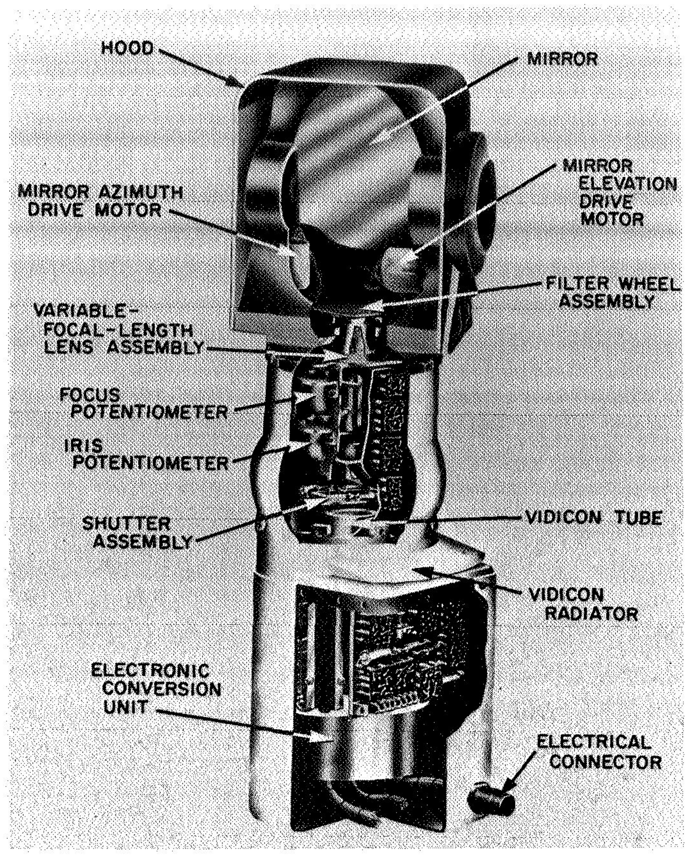


Fig. IV-53. *Surveyor VI* television camera

The mirror assembly contains a mirror supported at its minor axis by trunnions. This mirror is formed by vacuum-depositing a Kanogen surface on a beryllium blank, followed by a deposition of aluminum with an overcoat of silicon monoxide. The mirrored surface is flat over the entire surface to less than ¼ wavelength at $\lambda = 550$ mμ and exhibits an average specular reflectivity in excess of 86%. The mirror is positioned by means of two drive mechanisms, one for azimuth and the other for elevation.

The mirror assembly also contains a four-section filter wheel. As redesigned for *Surveyor VI,* three sections contain polarizing filters with polarization angles of 0, 45, and 90 deg, respectively, in a clockwise direction relative to the mirror elevation axis. The fourth section contains a neutral density filter element for nonpolarized observations. The neutral density filter transmission characteristic is selected such that the camera's response to nonpolarized light is identical to that of the polarizing filters (38%).

Camera operation is totally dependent upon reception of the proper commands from earth. Commandable operation allows each frame to be generated by shutter sequencing preceded by appropriate lens settings and mirror azimuth and elevation positioning to obtain selected views of the subject.

Figure IV-54 depicts a functional block diagram of the survey camera and television auxiliary. Commands for the camera are processed by the telecommunications command decoder, with further processing by the television auxiliary decoder. Identification signals, in analog form, from the camera are commutated by the television auxiliary, with analog-to-digital conversion being performed within the signal processing equipment of the telecommunications subsystem. The ID data in PCM form is mixed in proper time relationship with the video signal in the TV auxiliary and subsequently sent to the telecommunications system for transmission to earth.

The mirror assembly used on *Surveyor VI* is a completely redesigned version of the mirror assembly used on all previous missions. The purpose of the redesign was to improve performance and reliability. Occasional failure to respond to azimuth or elevation stepping commands was corrected by increasing motor torque output and reliability, reducing frictional and inertial loading, and providing adjustable end stops. Pointing accuracy was improved. A labyrinth seal was added to reduce

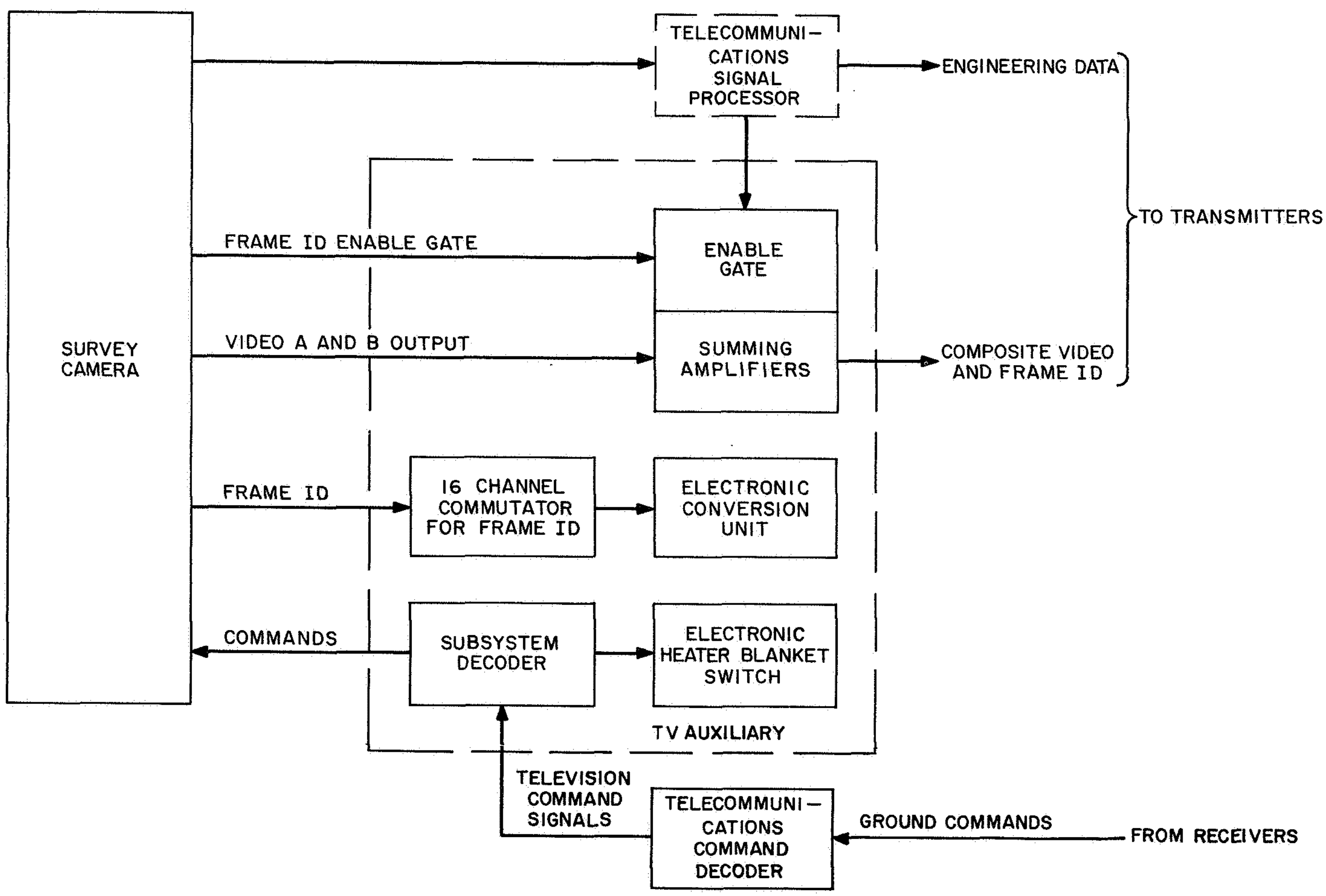


Fig. IV-54. Simplified television camera functional block diagram

dust accumulation on optical surfaces when the mirror is closed. The hood was extended at top and sides, and the front flange was painted black to reduce glare in pictures. Other design changes enabled removal of the hood or mirror assembly without requiring recalibration or requalification.

The optical formation of the image is performed by means of a variable-focal-length lens assembly placed between the vidicon image sensor and the mirror assembly. The lens is capable of providing a focal length of either 100 or 25 mm, which results in a vertical field of view of approximately 6.43 or 25.3 deg, respectively, as measured by the angular subtense of the vertical dimension in the video format. Special commanding can result in additional focal lengths between 25 and 100 mm. Because horizontal scan on the redesigned camera was increased by 10%, the horizontal field of view is, therefore, 10% greater than the vertical field of view. Additionally, the lens assembly may vary its optimum focus by means of a rotating focus cell from 1.23 meters to infinity, while an adjustable iris provides effective aperture changes from $f/22$ to $f/4$. The redesigned camera used for *Surveyor VI* provides ½-f-stop increments of adjustment compared to full-f-stop increments on the earlier camera. The aperture area increases by a factor of $\sqrt{2}$ for each ½-f-stop increment. While the most effective iris control is accomplished by means of command operation, a servo-type automatic iris is available to control the aperture area in proportion to the average scene luminance. As in the mirror assembly, potentiometers are geared to the iris, focal length, and focus elements to allow readout of these functions. A beam splitter, integral to the lens assembly, provides the necessary light sample (10% of incident light) for operation of the automatic iris.

Three modes of exposure control are afforded the camera by means of a mechanical focal plane shutter located between the lens assembly and the vidicon image sensor. In the *normal shutter* mode, upon earth command, the two shutter blades are sequentially driven by solenoids across an aperture in the shutter base plate, with appropriate delay between blades. The time interval between the blade motions determines the exposure interval, which is nominally 150 msec.

In the second shutter mode (*open shutter* mode) the blades are positioned to leave the aperture open, thereby providing continuous light energy to the image sensor. This mode of operation is useful in the imaging of scenes exhibiting low luminance levels, including some of the brighter stars. The effective shutter time in this mode is 1.2 sec, nominally.

A third exposure mode, used for extremely low luminance levels such as stellar observations, lunar surface observation under earthshine illumination conditions, and faint solar corona observations, is referred to as the *integrate* mode. This mode is implemented by opening the shutter, turning off the vidicon electron beam, and then, after any desired exposure time, turning on the vidicon electron beam. Scene luminances on the order of 0.008 ft-lamberts are easily reproduced in this mode of operation, thereby permitting photographs under earthshine conditions. Detection of sixth-magnitude stars has been accomplished using this mode of operation with an exposure time of 5 min.

The transducing process of converting light energy from the object space to an equivalent electrical signal is accomplished by the vidicon tube. A reference mark is included in each corner of the scanned format, which provides, in the video signal, an electronic dark-current level of the scanned image. A 10% wider scan was provided in the *Surveyor VI* camera to ensure inclusion of at least one reference mark under worst conditions of raster size and position change due to lunar temperatures.

In the normal, or *600-line* scan mode, the camera provides one frame every 3.6 sec. Each frame requires nominally 1 sec to be read from the vidicon and utilizes a bandwidth of approximately 220 kHz, requiring transmission over the high-gain (planar array) antenna in the high-power transmitter mode.

A second scan mode of operation provides one 200-line frame every 61.8 sec. Each 200-line frame requires 20 sec to complete the video transmission and utilizes a bandwidth of 1.2 kHz. This *200-line* mode is used for high-power transmission over either one of the omniantennas.

Integral to the spacecraft and within the viewing capability of the camera are two photometric/colorimetric reference charts (Fig. IV-55). These charts, one on Omniantenna B and the other adjacent to Footpad 3, are located such that the line of sight of the camera when viewing the chart is normal to the plane of the chart. The charts are identical, and each contains a series of 13 gray wedges arranged circumferentially around the chart. In addition, three color wedges, whose CIE chromaticity coordinates are known, are located radially from the chart center. A series of radial lines is

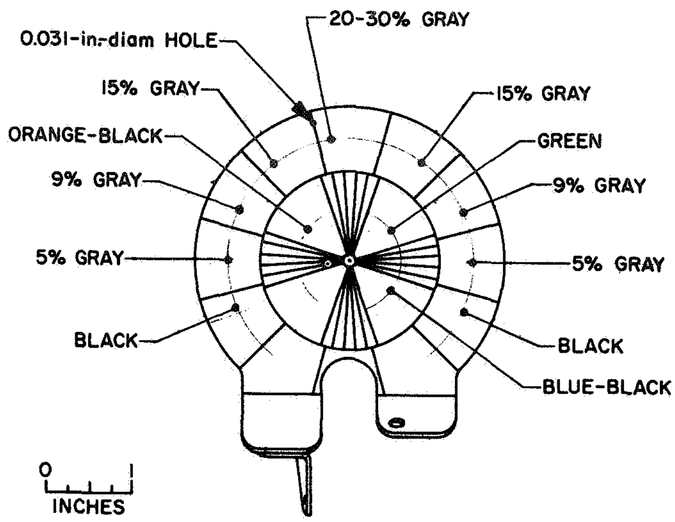


Fig. IV-55. TV photometric/colorimetric reference chart

incorporated to provide a gross estimate of camera resolution. Finally, the chart contains a centerpost which aids in determining the solar angles after lunar landing by means of the shadow information. Prior to launch, each chart is calibrated goniophotometrically to allow an estimation of postlanding camera dynamic range and to aid photometric data reduction.

The survey camera incorporates heaters to maintain proper thermal control and to provide a thermal environment in which the camera components operate. The heater elements are designed to provide a sustaining operating temperature during the lunar night if energized. These consume 36 W of power when activated. A minimum temperature of $-20°F$ should be achieved prior to camera turn-on if operation is to be performed within known functional calibrations.

Three auxiliary mirrors were installed on the *Surveyor VI* spacecraft for viewing obstructed areas of interest beneath the spacecraft. The largest of these was mounted for viewing of the area beneath Vernier Engine 3 and Crush Block 3, the next largest was mounted for viewing the area beneath Vernier Engine 2 and Crush Block 2, and the smallest was mounted for viewing the alpha scattering instrument deployment area on the lunar surface.

2. Performance Characteristics

A premission calibration was performed on the survey camera with the camera mounted on the spacecraft. Each calibration utilized the entire telecommunication system of the spacecraft, thereby including those factors of the modulator, transmitter, etc., which influence the overall image transfer characteristics. The calibration data, in FM form, was recorded on magnetic tape for playback through the ground support equipment (GSE) at Goldstone and Pasadena. Thus the final calibration data recorded on the real-time mission film and tape equipment provides a complete system calibration.

The parameters which were calibrated included light transfer characteristics, polarization response, sine wave response, erasure characteristics, automatic iris tracking, geometric linearity, and camera pointing accuracy. Data was taken in both 600- and 200-line scan modes and in normal shutter, open shutter, and integrate exposure modes. The calibration data shows that the *Surveyor VI* camera system has a 25% horizontal response at about 600 TV lines in the 600-line scan mode. The identical light transfer characteristics for the three polarizing filters and the neutral density filter provide the capability of taking a set of polarized pictures without the necessity of changing iris position. Reduction of polarimetric data is then independent of accuracy or repeatability limitations in iris control.

3. Mission Performance

Surveyor VI television camera performance was the best yet experienced on a mission in quality and quantity of pictures, as well as in the extent of operations which could be conducted unrestricted by camera problems. The camera performed perfectly throughout the mission. Picture quality was excellent primarily because the camera characteristics exhibited good dynamic range. A contributing factor was that the optical surfaces remained the cleanest for any mission because the redesigned mirror head assembly permitted the camera mirror to be fully closed during the transit phase and the postlanding hop experiment.

During the first lunar day, in addition to DSS 11 operations, the camera was operated from DSS 42, Canberra, and to a lesser extent from DSS 61, Madrid, resulting in approximately 30,000 pictures being taken. This is a much greater number of pictures than were taken on any of the previous missions.

The *Surveyor VI* pictures include views of parts of the spacecraft, views of the area beneath the spacecraft using the auxiliary mirrors, panoramic narrow- and wide-angle surveys of the lunar landscape in azimuth and elevation, photometric surveys of specially selected objects, star surveys, focus ranging sequences for mapping of the surrounding lunar surface, shadow progressions, views through different polarization filters, a solar corona sequence which lasted about 6 hr following sunset, and earthshine frames. Since it was of extreme interest to collect both spacecraft engineering data and television pictures during the first few hours following sunset, the unique camera commanding sequence first used on the *Surveyor V* mission was also utilized on this mission. This sequence permitted the camera to integrate the solar corona and earthshine images while the camera electronics were turned off. It was then only necessary to turn on the camera long enough to "read out" the stored image, thus allowing the engineering interrogation mode to be used at the rate of 4 out of every 5 min, and image integration to be done nearly 5 out of every 5 min. The image integration mode of operation afforded sufficient flexibility in camera operations to result in a very successful series of corona pictures.

Also obtained on this mission by use of the camera were: the surface erosion characteristics resulting from firing the vernier engines for the spacecraft hop experiment, views taken after the hop of the imprints made by the spacecraft upon initial landing, stereo frames taken before and after the hop experiment, and the particle accumulation and distribution on the bar magnet.

Introspection checks of the polarizing filters were conducted during three periods (shortly after landing, just after lunar noon, and just prior to sunset) to determine if delamination was a problem. Each filter was observed for at least three mirror elevations (−65, −70 and −75 deg) to determine if the reflectivity of any filter had increased, since a considerable increase in reflectivity accompanies delamination of a polarizing filter. No delamination was indicated.

Surveyor VI camera operation on the second lunar day was not attempted because of spacecraft problems.*

*When the attempt was made to contact *Surveyor VI, Surveyor V* responded and more pictures were made using the *Surveyor V* TV camera in 200-line mode in its fourth lunar day. The video amplitude was even less than it was on the second lunar day (see *Surveyor V* Mission Report for second lunar day operation) and the video polarity had reversed again, back to normal, in the 200-line mode. An attempt was made to operate again in 600-line mode but only "black level" pictures resulted with proper synchronization signals but no video information. This led to the preliminary conclusion that another resistor (R7), or its associated wiring, in the video preamplifier had opened (R9 appeared to have opened during the first lunar night).

J. Alpha Scattering Instrument

The alpha scattering technique of lunar surface chemical analysis takes advantage of the characteristic interactions of alpha particles with matter to provide information on the chemical composition. The energy spectra of the large-angle, elastically scattered alpha particles are characteristic of the nuclei doing the scattering. In addition, certain elements, when bombarded with alpha particles, produce protons, again with characteristic energy spectra. Consequently, these energy spectra and intensities of scattered alpha particles and protons can be used to determine the chemical composition of the material being exposed to the alpha particles.

The method has good resolution for the light elements found in rocks. (However, it can give only indirect information about hydrogen.) The resolution becomes poorer as the atomic weight increases (Fe, Co, and Ni cannot easily be resolved), even though the sensitivity is greater for heavy elements than for most light elements. The sensitivity for elements heavier than lithium is approximately 1 atomic percent.

The absence of an atmosphere on the moon makes practical the use, for such chemical analyses, of the relatively low-energy alpha particles from a radioactive source. $Cm^{242}(t_{1/2} = 163$ days, $T_a = 6.11$ MeV) is the nuclide used in the *Surveyor* instruments. The use of low-energy alpha particles, however, restricts the information obtained to that pertaining to the uppermost few microns of material. The method is thus one of *surface* chemical analysis. Moreover, using practical source intensities, the rate of analysis is rather slow. A relatively complete analysis requires about one day. In spite of these disadvantages, the use of a radioactive source makes possible a relatively simple and compact instrument which may be deployed directly onto the lunar surface.

1. Instrument Description

The alpha scattering instrument consists of a sensor head, which is deployed onto the lunar surface, and a digital electronics package located in a thermal compartment on the spacecraft. Associated equipment includes an electronic auxiliary, a deployment-mechanism/standard-sample assembly, and a thermally insulated electronics compartment. Figure IV-56 is a photograph of the instrument and its auxiliary hardware.

The total weight of the alpha-scattering equipment, including mechanical and electrical spacecraft-interface

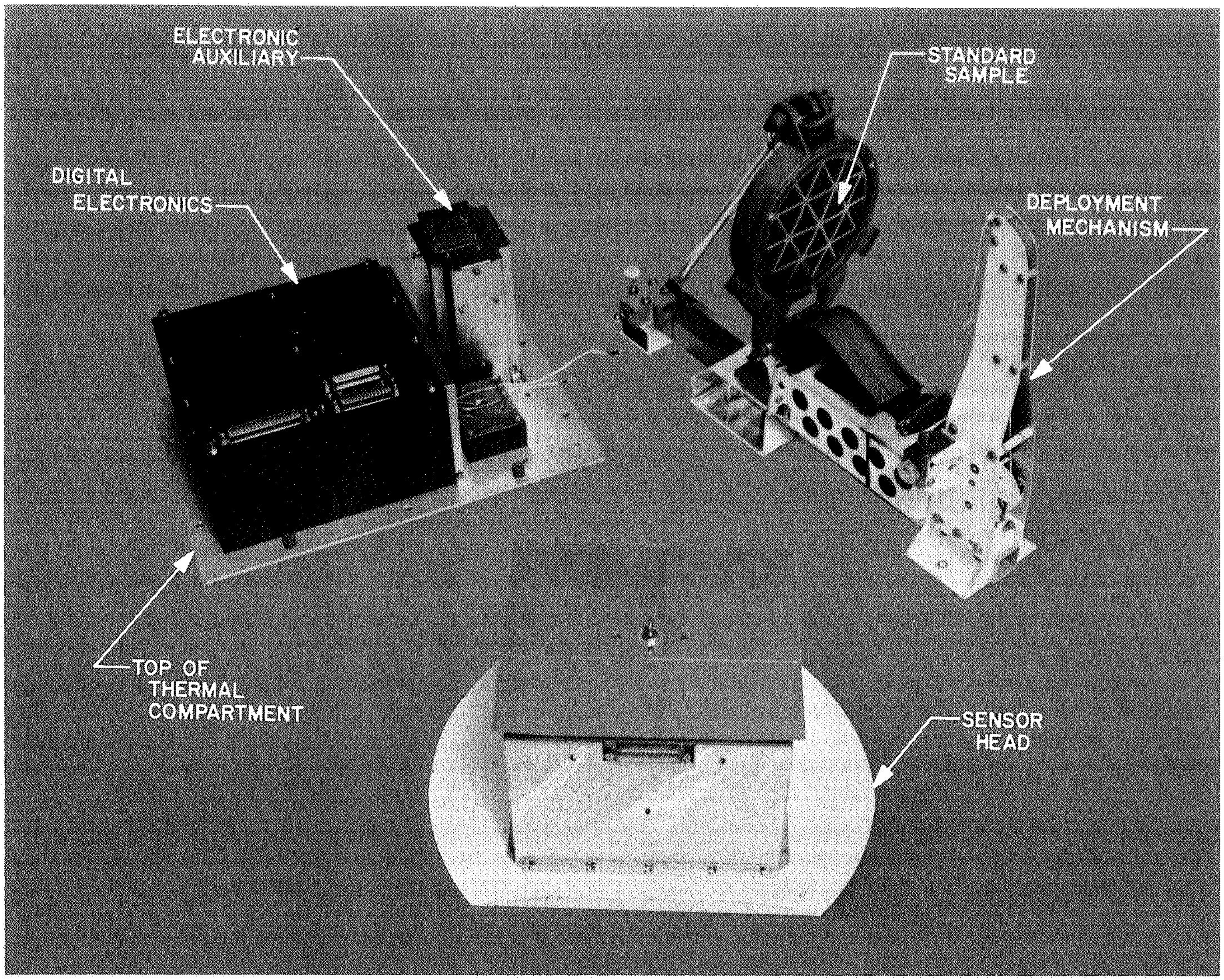


Fig. IV-56. Alpha scattering instrument components

substructure and cabling, is 29 lb. Power dissipation is 2 W, increasing to 17 W when heaters in the sensor head and electronics compartment are both active.

a. Sensor head. The sensor head is primarily a box ($6\frac{3}{4} \times 6\frac{1}{2} \times 5\frac{1}{4}$ in. high), with a 12-in.-diameter plate on the bottom. The main purpose of the plate is to minimize the probability of the box sinking appreciably into a possibly soft lunar surface. Figure IV-57 shows a bottom view of the sensor head. In the bottom of the sensor head is a sample port, $4\frac{1}{4}$ in. in diameter. Recessed $2\frac{3}{4}$ in. above this circular opening is a set of six curium-242 alpha sources, collimated in such a way that the alpha particles are directed through the opening. Across the face of each collimator is a thin, aluminum-oxide film to prevent recoils from the alpha source from reaching the sample area; a second film is mounted in front of each collimator for additional protection. Close to the alpha sources are two detectors arranged to detect alpha particles scattered back from the sample at an average angle of 174 deg from the original direction. These 0.031-in.^{2} alpha detectors are of the silicon, surface-barrier type, with evaporated-gold front surface films. Thin films are mounted on collimation masks to protect the detectors from alpha contamination and excessive light.

The sensor head also contains four lithium-drifted silicon detectors (of approximately 0.155-in.^{2} area each) designed to detect protons produced in the sample by

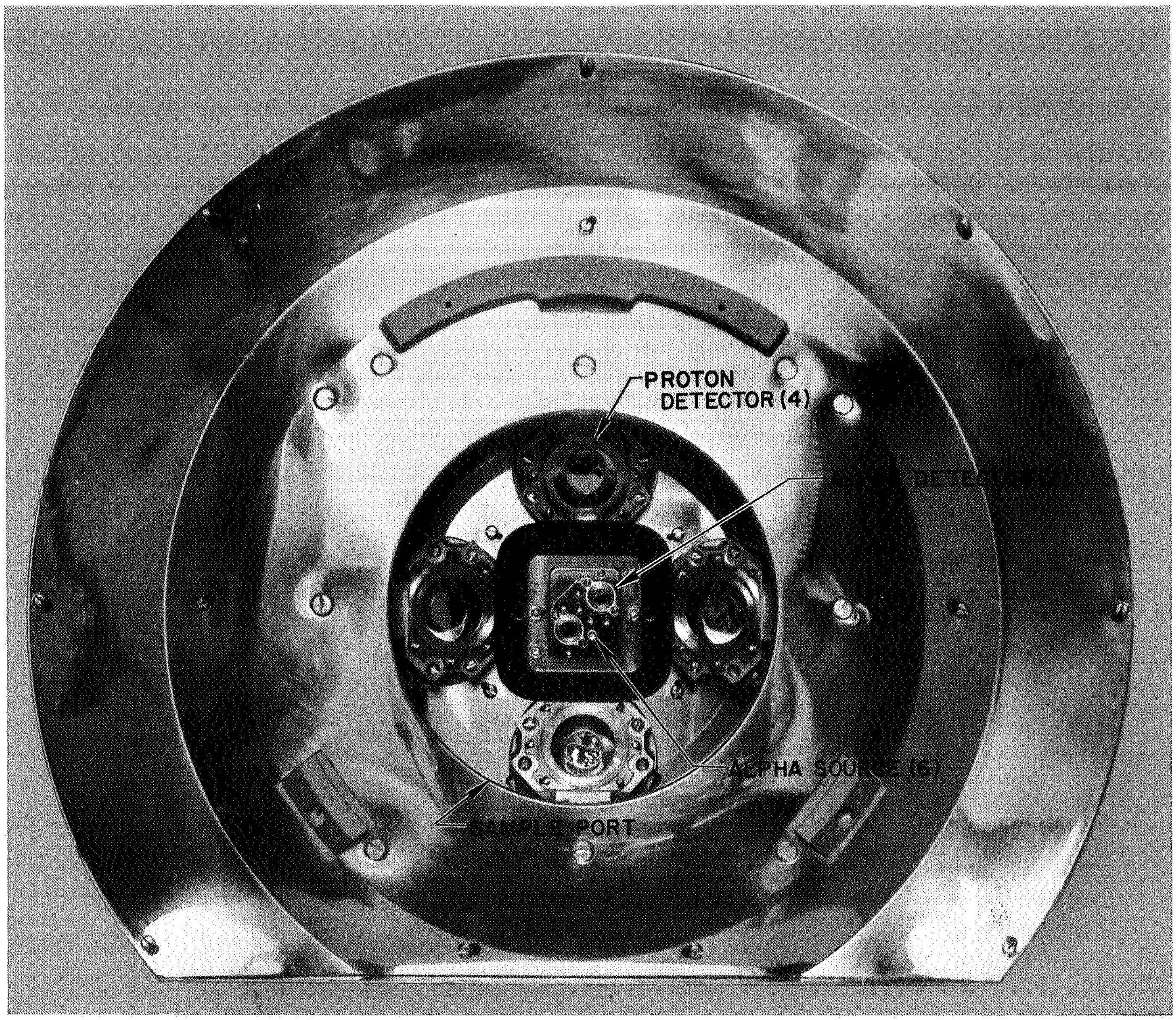


Fig. IV-57. View of bottom of alpha scattering instrument sensor head

the alpha particles. Gold foils, approximately 0.4-mil thick, prevent scattered alpha particles from reaching these detectors. Figure IV-58 is a diagrammatic side view of the sensor head, showing the configuration of sources, sample, and detectors.

Because the expected proton rates from the sample are low, and because these detectors are more sensitive to radiation from space, the proton detectors are backed by guard detectors. Most of the charged particles from space that strike a proton detector must first pass through the corresponding guard detector, whereas protons from the sample are stopped in the proton detector. The electronics associated with the guard and proton detectors are arranged so that an event registered in both will not be counted as coming from the sample. This anticoincidence arrangement reduces significantly the background in the proton mode of the instrument.

Separate, 128-channel, pulse-height analyzers are used with the alpha and proton detectors. An output pulse from a detector is amplified and converted to a time-analog signal whose duration is proportional to the energy deposited in the detector. The outputs of the two

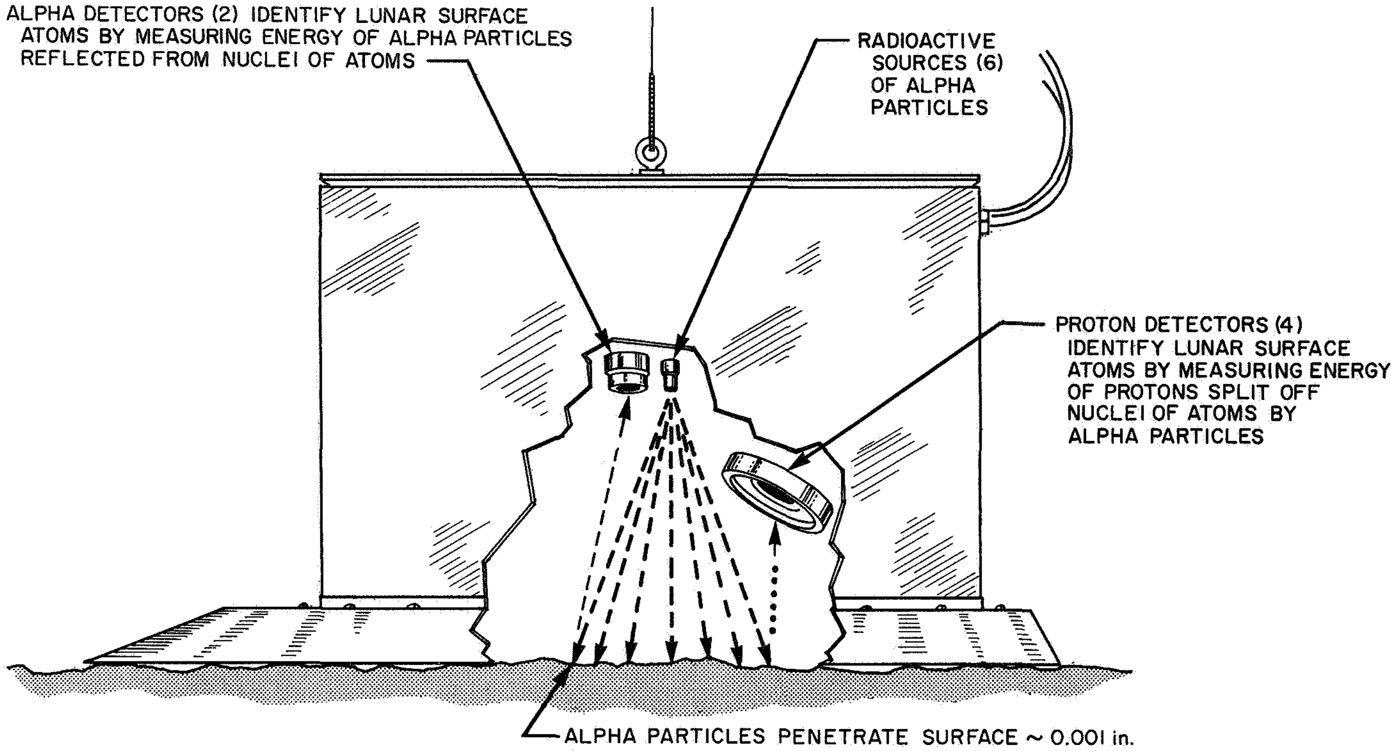


Fig. IV-58. Diagrammatic view of sensor head illustrating functional operation

alpha detectors are combined before this conversion; a separate mixer circuit is used for the four proton-detector outputs. A ratemeter circuit is used to measure the frequency of events occurring in the guard (anticoincidence) detectors, but provides no information on the energy of such events.

In addition to the curium-242 sources, detectors, and associated electronics, the sensor head contains a platinum-resistance thermometer, a 5-W heater, and an electronic pulser. The pulser is used to calibrate the electronics of the instrument by introducing electrical pulses of two known magnitudes at the detector stages of the alpha and proton systems. This calibration mode is initiated by command from earth. Small amounts of alpha-radioactive einsteinium-254 are also used to calibrate the alpha and proton systems and, in addition, serve as real-time monitors. The einsteinium is located on the gold foil facing each proton detector and on the thin films mounted in front of the alpha detectors.

The external surfaces of the sensor head have been designed for passive thermal control. A second-surface mirror on top of the sensor head is used as a radiator to cool the sensitive components inside. The 5-W heater is used at low temperatures. The operating temperature range specified for the sensor head is −40 to +122°F.

b. Digital electronics. The output of the sensor head is a signal (in time-analog form) that characterizes the energy of an event in either the scattered alpha or proton mode of the instrument. The signals from the sensor head are converted to 9-bit binary words by the digital electronics. Seven bits of each word identify which of the 128 channels represents the energy of the registered event. Two extra bits are added before transmission, one to identify the start of the word and one at the end of each word, as a parity check on transmission errors. Buffer registers provide temporary storage of the energy information for readout into the spacecraft telemetry system. The transmission rates are 2200 bits/sec for the alpha mode and 550 bits/sec for the proton mode. Measured events with energy greater than the range of the analyzers are routed to channel 126 (overflow channel).

The electronics package also contains power supplies and the logical electronic interfaces between the instrument and the spacecraft. For example, the output of an individual detector, together with its associated guard detector, can be blocked by command from earth. Also via the electronics unit, the temperature of the sensor head, as well as various monitoring voltages, can be transmitted to earth.

c. Electronic auxiliary. The required electrical interfaces between the sensor head, digital electronics and spacecraft circuits are provided by an electronic auxiliary that provides command decoding, signal processing, and power management. Basic spacecraft circuits interfacing directly with the sensor head and digital electronics are (1) the central signal processor, which provides signals at 2200 and 550 bits/sec for synchronization of instrument clocks, and (2) the engineering signal processor, which provides temperature-sensor excitation current and commutation of engineering data outputs.

The electronic auxiliary provides two data channels for the alpha scattering instrument. The separate alpha and proton channels are implemented using two subcarrier oscillators. Characteristics of these channels are defined in Table IV-19.

Table IV-19. Characteristics of alpha scattering instrument data channels

Characteristic	Alpha channel	Proton channel
Data input to electronic auxiliary	Digital (nonreturn to zero)	Digital (nonreturn to zero)
Input data rate	2200 bits/sec	550 bits/sec
Subcarrier oscillator center frequency	70,000 Hz	5400 Hz

The electronic auxiliary and the digital electronics are contained in Electronics Compartment C, which is attached to the upper part of the spaceframe. For passive control of temperatures at high sun angles the top of this compartment is painted white and the sides and bottom are insulated. A 10-W heater assembly, operated by means of the engineering signal processor, provides active thermal control at low temperatures. The operating temperature range specified for the electronic units in Compartment C is −4 to +131°F.

d. Deployment mechanism/standard sample. The deployment mechanism provides stowage of the sensor head, deployment to the background position, and to the lunar surface. The sensor head is mounted to the deployment mechanism by means of three support lugs on the bottom plate. Deployment mechanism clamps that engage these lugs are released during the deployment operation. Figure IV-59 illustrates the two-stage operation

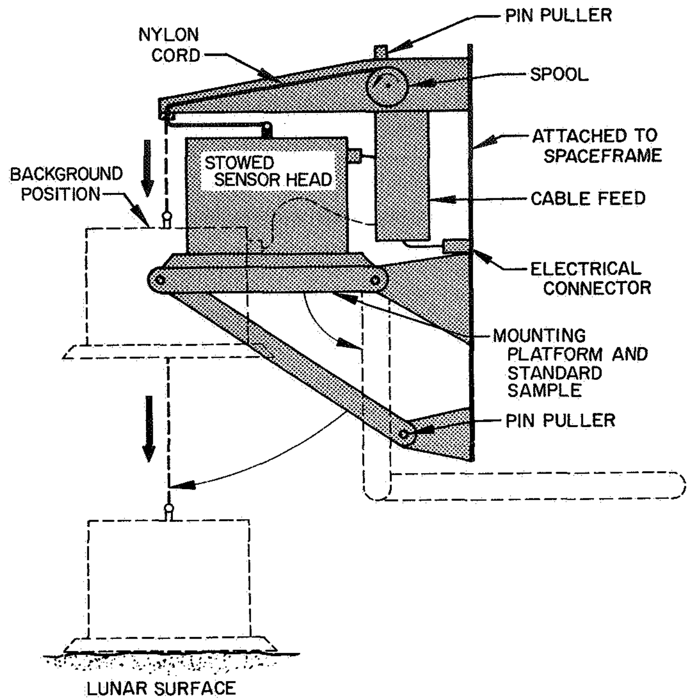


Fig. IV-59. Alpha scattering instrument deployment mechanism

of the deployment mechanism. From the stowed position, the sensor head is first released on command to a position 22 in. above the nominal lunar surface by activation of an explosive-pin-puller device. From the background position, the sensor head is then lowered directly to the lunar surface by activation of another explosive-pin-puller device. The deployment velocity is controlled by an escapement.

A sample of known composition is attached to the platform on which the sensor head is mounted in the stowed position. This standard-sample assembly covers the circular opening in the bottom of the sensor head during spacecraft transit and landing to minimize entrance of dust and light and to provide a means of assessing instrument performance shortly after the spacecraft lands on the moon. The standard sample and mounting platform move aside when the sensor head is deployed to the background position.

2. Alpha Scattering Instrument Data Handling

Two types of information relative to the alpha scattering experiment are transmitted from the spacecraft: engineering data and science data. The engineering measurements are used to monitor instrument voltages, temperatures, detector configuration, and background

Table IV-20. Engineering parameters telemetered from alpha scattering instrument

Telemetry channel	Parameter
AS-3	Sensor head temperature
AS-4	Compartment C (digital electronics temperature)
AS-5	Guard rate monitor
AS-6 (digital)	At least one alpha detector on
AS-7 (digital)	At least one proton detector on
AS-8	7-V monitor
AS-9	24-V monitor

rates in the anticoincidence detectors. The seven parameters that are monitored are listed in Table IV-20.

The 9-bit digital words that characterize the energy of each of the analyzed alpha particles or protons comprise the science data. These data leave the instrument as separate alpha and proton bit streams and modulate separate subcarrier oscillators; they are then combined with the engineering data and transmitted by the spacecraft to earth. The composite signal from the spacecraft is recorded on magnetic tapes by an FR-1400 recorder at each DSIF station. These magnetic tapes, containing the raw data, comprise the prime source of alpha scattering information for use in postmission analysis.

For purposes of monitoring the experiment in real-time, the signal is separated at the DSIF station by discriminators and bit synchronizers into 2200-bits/sec alpha data and 550-bits/sec proton data. These reconstituted bit streams are presented to an on-site computer which establishes and maintains synchronization of the 9-bit data words and assembles, within its memory, four spectra of 128 channels each. The four spectra are: alpha parity-correct, alpha parity-incorrect, proton parity-correct, and proton parity-incorrect. In this manner, data is obtained at the stations in accumulations ranging in duration from 2 min during pulser calibration to a nominal 40 min during sample and background phases. The assembled spectra are transmitted via teletype to the SFOF for display and further computer processing.

Data analysis during the mission is performed so that proper control over the experiment may be exercised. The engineering data is simply displayed and compared with prelaunch measurements and predictions to assess the performance of the instrument and the functioning of commands.

The alpha and proton science data is also used to assess the performance of the instrument and is analyzed to determine the duration of the several operational phases. This science-data analysis is conducted in the SFOF using a 7094 computer program.

3. Alpha Scattering Instrument Performance

The following table is a summary of the science-data accumulation time on the *Surveyor VI* mission in each of the operational configurations based upon spectra assembled by the on-site computers and transmitted via teletype to the SFOF.

Operational configuration	Accumulation time, min
Transit	0
Standard sample	320
Background	370
Lunar surface	1800
Upside down after spacecraft hop	780
Total	3270 (55 hr)

Surveyor VI launch, transit, and landing operations proceeded normally. Alpha scattering instrument temperatures were obtained during transit; the sensor head temperature was slightly higher than measured during the *Surveyor V* mission. There was no operation of the instrument during transit to the moon.

Analysis of the standard sample under lunar conditions verified that the alpha scattering instrument had arrived on the moon in a satisfactory condition. Sharp breakpoints in the sample spectra showed that the high quality of the curium-242 sources had been preserved. The films covering the sources and alpha detectors had survived. The electronics, calibration pulser, and einsteinium sources performed properly as evidenced by the sharpness of calibration peaks and agreement with prelaunch data. The guard detector and anticoincidence system worked as designed; guard monitor voltage agreed well with predicted values.

The deployment mechanism functioned properly to permit background and lunar surface operation.

During a period of surface operation at elevated temperatures, proton detector 3 became noisy; this detector was commanded off. Accumulations were obtained using the remaining five detectors. Subsequent operation at lower temperatures indicated that the noisy detector had recovered. Normal operation using all detectors was resumed.

At 10:32 GMT on November 17, 1967, the vernier engines were fired for the spacecraft hop experiment; subsequent television pictures showed that the spacecraft had moved 7.9 ft and that the sensor head was now upside down. A short operation showed that the instrument was functioning electronically, but that many events were being registered at around channel 103 in the alpha system. This indicated that curium-242 contamination was present in the sensor head and, therefore, at least one of the source-protective films had broken. The rates in the proton mode were also significantly higher than before the hop. Qualitatively, this was understandable in that the proton detectors in the new position of the sensor head were exposed to radiation from space.

The communications link from the spacecraft was excellent; a bit-error-rate of less than 10^{-6} was estimated from the parity-incorrect spectra. In general, deviations from this high-quality data reception occurred only when the spacecraft was being tracked near the earth's horizon. Both of the major alpha scattering computer programs and associated equipment gave excellent processing of the science data.

Preliminary scientific results of the *Surveyor VI* alpha scattering experiment are presented in Part II of this report.

V. Tracking and Data System

The Tracking and Data System (TDS) for the *Surveyor VI* mission included selected resources of the Air Force Eastern Test Range (AFETR), the Manned Space Flight Network (MSFN), the Deep Space Network (DSN), and the NASA Communication System (NASCOM). This section summarizes the mission preparation, flight support, and performance evaluation of each element of the TDS.

The TDS support during the near-earth phase of the *Surveyor VI* mission was considered fair. Because of nominal launch vehicle and spacecraft performance, the problems experienced by elements of the TDS did not significantly impact the overall support. Class I tracking (metric) data required for near-real-time computation of the actual transfer orbit was not provided, and initial DSS 51 acquisition information based on that orbit was not generated. All other requirements were met and in most cases exceeded. TDS support during the deep-space phase of the mission was considered excellent; all requirements were met and in most cases exceeded.

A. Air Force Eastern Test Range

The AFETR performs TDS supporting functions for *Surveyor* missions during the countdown, launch, and near-earth phases of the flight. The *Surveyor* mission requirements for launch phase tracking and telemetry coverage are classified as follows, in accordance with their relative importance to successful mission accomplishment:

> Class I requirements consist of the minimum essential needs to ensure accomplishment of first-priority flight test objectives. These are mandatory requirements which, if not met, may result in a decision not to launch.
>
> Class II requirements define the needs to accomplish all stated flight test objectives.
>
> Class III requirements define the ultimate in desired support, and would enable the range user to achieve the flight test objectives earlier in the test program.

The AFETR configuration for the *Surveyor VI* mission is presented in Table V-1. The configuration is similar to the *Surveyor V* configuration. In addition to the land stations, three Range Instrumentation Ships (RIS *Coastal Crusader*, *Sword Knot*, and *Twin Falls*) supported the mission.

Figures V-1 and V-2 illustrate the planned coverage for *Surveyor VI* on launch day and the test support positions of the Range Instrumentation Ships. AFETR preparations for *Surveyor VI* consisted of routine testing of individual facilities, followed by several Operational Readiness Tests. Most *Surveyor VI* mission requirements were met by AFETR.

Table V-1. AFETR configuration for *Surveyor VI* mission

Station		C-band tracking		Telemetry	
Designation	Location	Capability	Antenna type	Capability	Antenna type
0	Patrick AFB	beacon/skin	FPQ-6		
1	Cape Kennedy	beacon/skin	FPS-16		
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
WHI	RIS *Coastal Crusader*			VHF S-band	TAA-1 TAA5-24
UNI	RIS *Twin Falls*	beacon	FPS-16	VHF S-band	Rantec TAA5-12
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 parabola
YAN	RIS *Sword Knot*			VHF S-band	TAA-1 TAA5-24

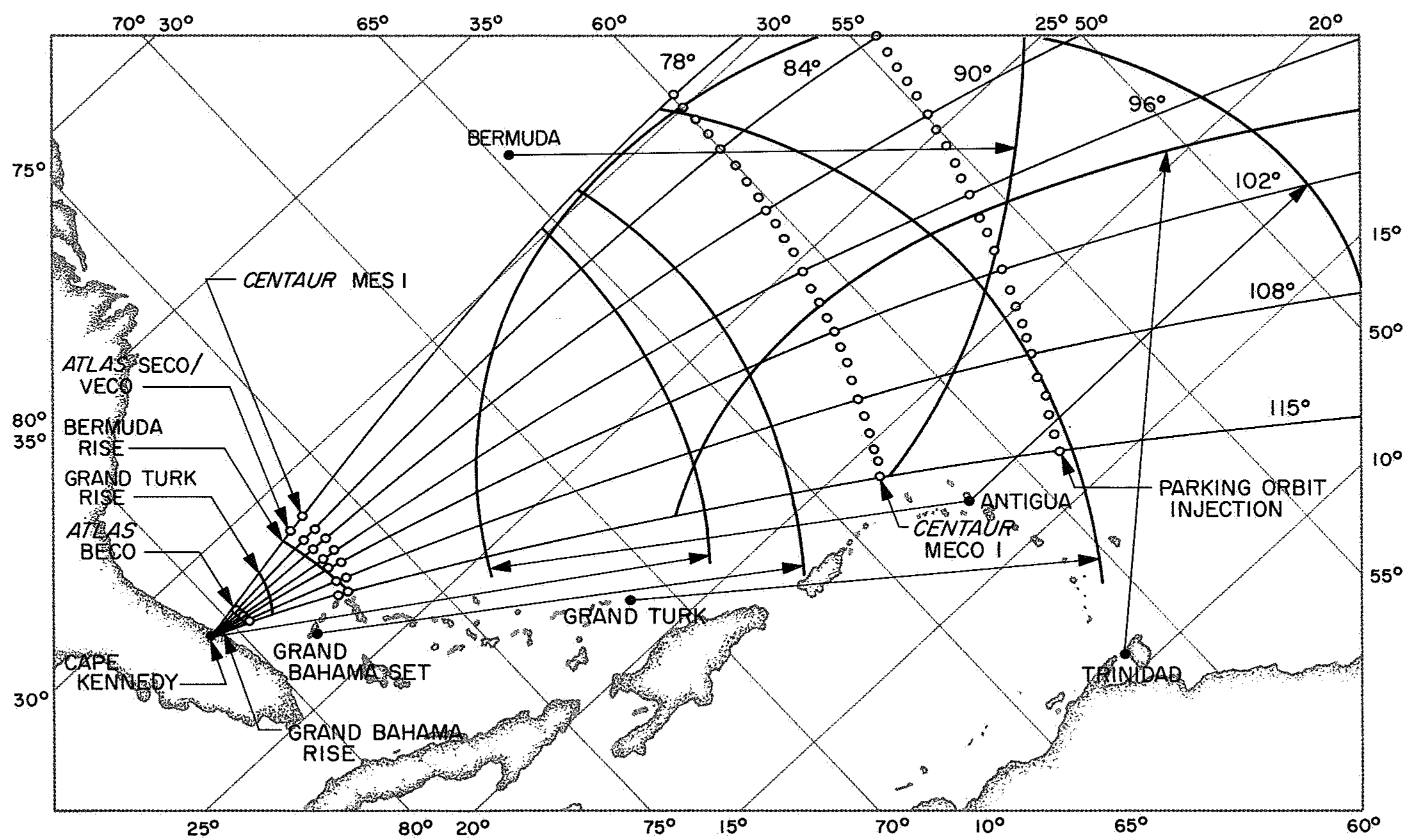


Fig. V-1. Planned up-range TDS coverage for *Surveyor VI*

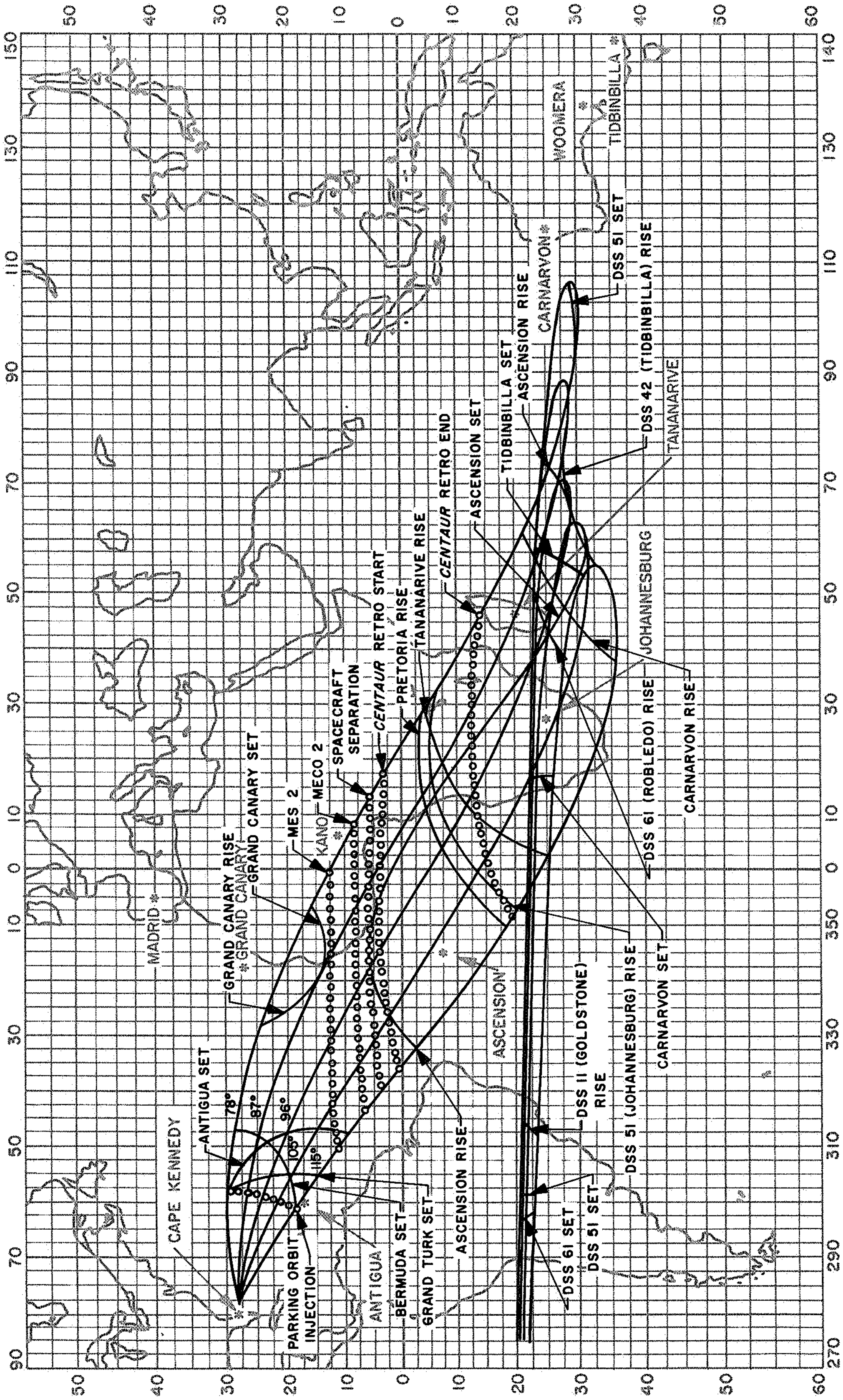


Fig. V-2. Planned near-earth TDS coverage for November 7, 1967

1. Metric Tracking Data

The AFETR tracks the C-band beacon of the *Centaur* stage to provide metric data. This data is required during intervals of time before and after separation of the spacecraft for use in calculating the *Centaur* orbit, which can be used as a close approximation of the post-separation spacecraft orbit. The *Centaur* orbit calculations are used to provide early mission status evaluations and DSN acquisition information (in-flight predicts). C-band tracking data is also required during the parking orbit phase for determining acquisition information for down-range stations and theoretical DSN acquisition information. In addition, C-band tracking data is required following the *Centaur* retro maneuver in order to determine the final *Centaur* trajectory.

Estimated and actual radar coverages are shown in Fig. V-3. Although the *Twin Falls* acquired good data during the required interval, Class I requirements were not met because the data was not returned to the Real Time Computer System (RTCS), Cape Kennedy, in time for transfer orbit computations and the generation of DSS 51 predicts. The land stations Grand Bahama and Grand Turk provided continuous coverage to $L + 614$ sec. The Grand Turk radar experienced a 40-sec dropout at $L + 488$ sec. However, the Patrick AFB radar provided redundant coverage during this period. The *Twin Falls* covered the interval from $L + 1360$ to $L + 1840$ sec and Pretoria, although experiencing intermittent track, provided continuous coverage from $L + 1877$ to $L + 5296$ sec.

The Antigua and Ascension radars did not have view on this flight azimuth.

2. *Atlas/Centaur* Telemetry (VHF)

To meet the Class I telemetry requirements, the AFETR must continuously receive and record *Atlas* telemetry (229.9-MHz link) from before liftoff until shortly after *Atlas/Centaur* separation, plus *Centaur* telemetry (225.7-MHz link) until shortly after spacecraft separation. Thereafter, *Centaur* telemetry is to be recorded as station coverage permits until completion of the *Centaur* retro maneuver.

Estimated and actual VHF telemetry coverages are shown in Fig. V-4. All requirements were met. Continuous and substantially redundant VHF telemetry data was received, beginning with the countdown and through Antigua loss of signal (LOS), at $L + 755$ sec, and from $L + 1184$ to $L + 3111$ sec. Coverage was greater than predicted. The RIS *Coastal Crusader* retransmitted launch vehicle telemetry data in real-time. Noisy data was obtained from the initial retransmission. However, a playback provided good quality data.

3. *Surveyor* Telemetry (S-Band)

The AFETR is required to receive, record, and retransmit *Surveyor* S-band (2295-MHz) telemetry in real-time from spacecraft transmitter *high power on* until 2 min after continuous view by DSIF stations begins. All primary S-band systems assigned to meet this requirement (Table V-1) were used on a limited commitment basis, since the *Centaur* vehicle is not roll-attitude-stabilized and the aspect angle cannot be predicted.

Estimated and actual S-band coverages and receiver phase-lock times are shown in Fig. V-5. Kennedy Space Center (Tel-4) and Grand Bahama provided coverage from liftoff to $L + 377$ sec. No explanation was available for the early LOS by these two stations. Unfavorable aspect angle was considered a possible cause. Antigua's coverage was obtained with a maximum elevation of 1.8 deg. With the exception of a short dropout in the RIS *Sword Knot* data, continuous coverage was provided from $L + 1220$ to $L + 6100$ sec. As indicated in Fig. V-5, however, several periods of receiver unlock occurred during this interval.

4. *Surveyor* Real-Time Telemetry Data

The AFETR retransmits *Surveyor* data (VHF or S-Band) to DSS 71 and Building AO, Cape Kennedy, for display and for retransmission to the SFOF. In addition, down-range stations monitor specific channels and report events via voice communication. For the *Surveyor VI* mission, existing hardware and software facilities were utilized to meet the real-time data requirements.

Real-time data flow, although hampered by poor RF propagation, was considered satisfactory; all requirements were met. VHF telemetry data, including spacecraft data received from the stations was transmitted in real-time to the SFOF from liftoff to spacecraft transmitter *high power on*. At *high power on*, AFETR switched, as planned, to real-time transmission of spacecraft S-band telemetry data.

In addition, all Mark Events except Nos. 22 and 23 were read out and reported by AFETR and MSFN stations. The reported Mark Event times are presented in Table V-2. Refer to Appendix A, Table A-1, for a list of the Mark Events and event times determined postflight.

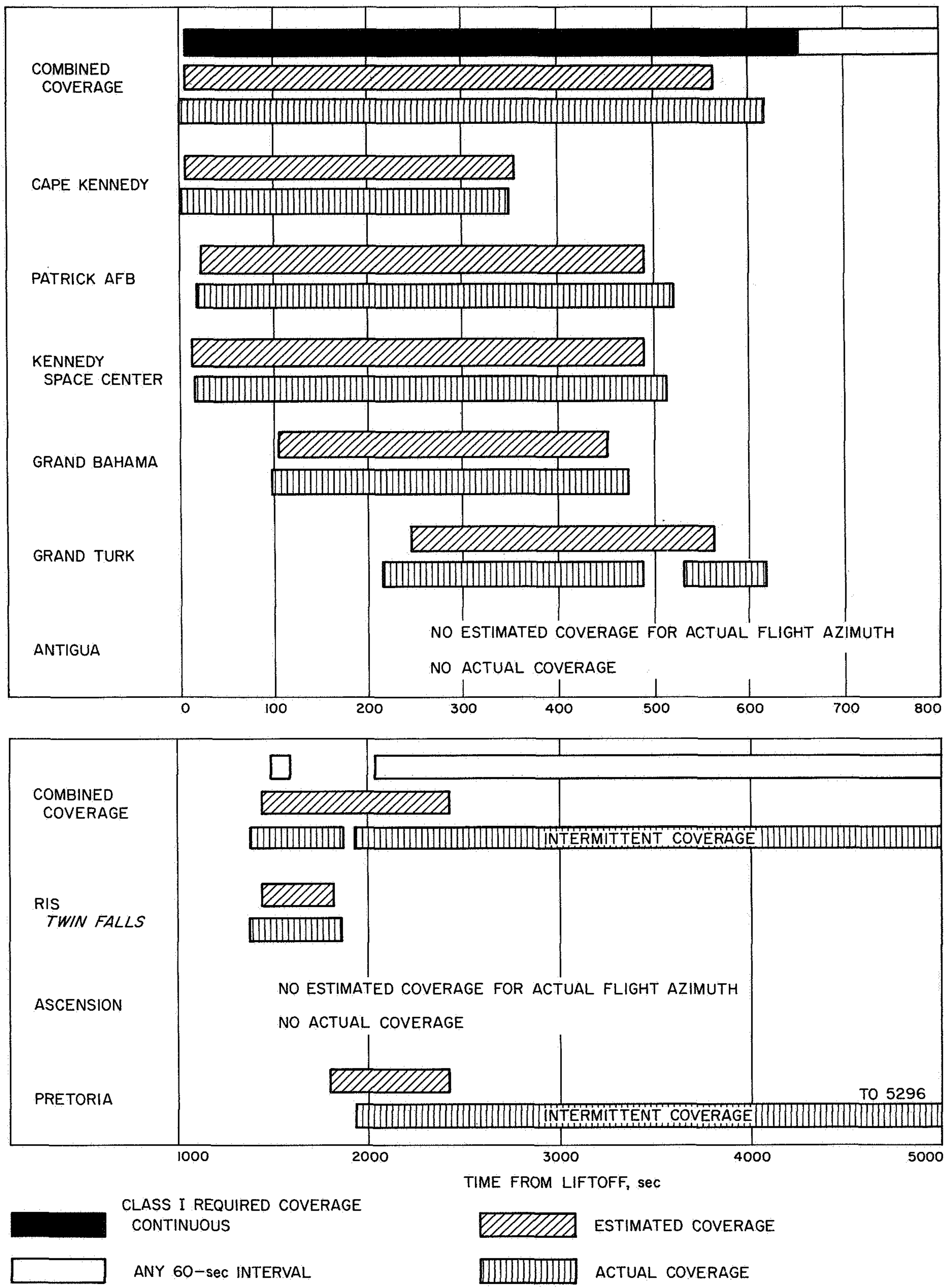


Fig. V-3. AFETR C-band radar coverage

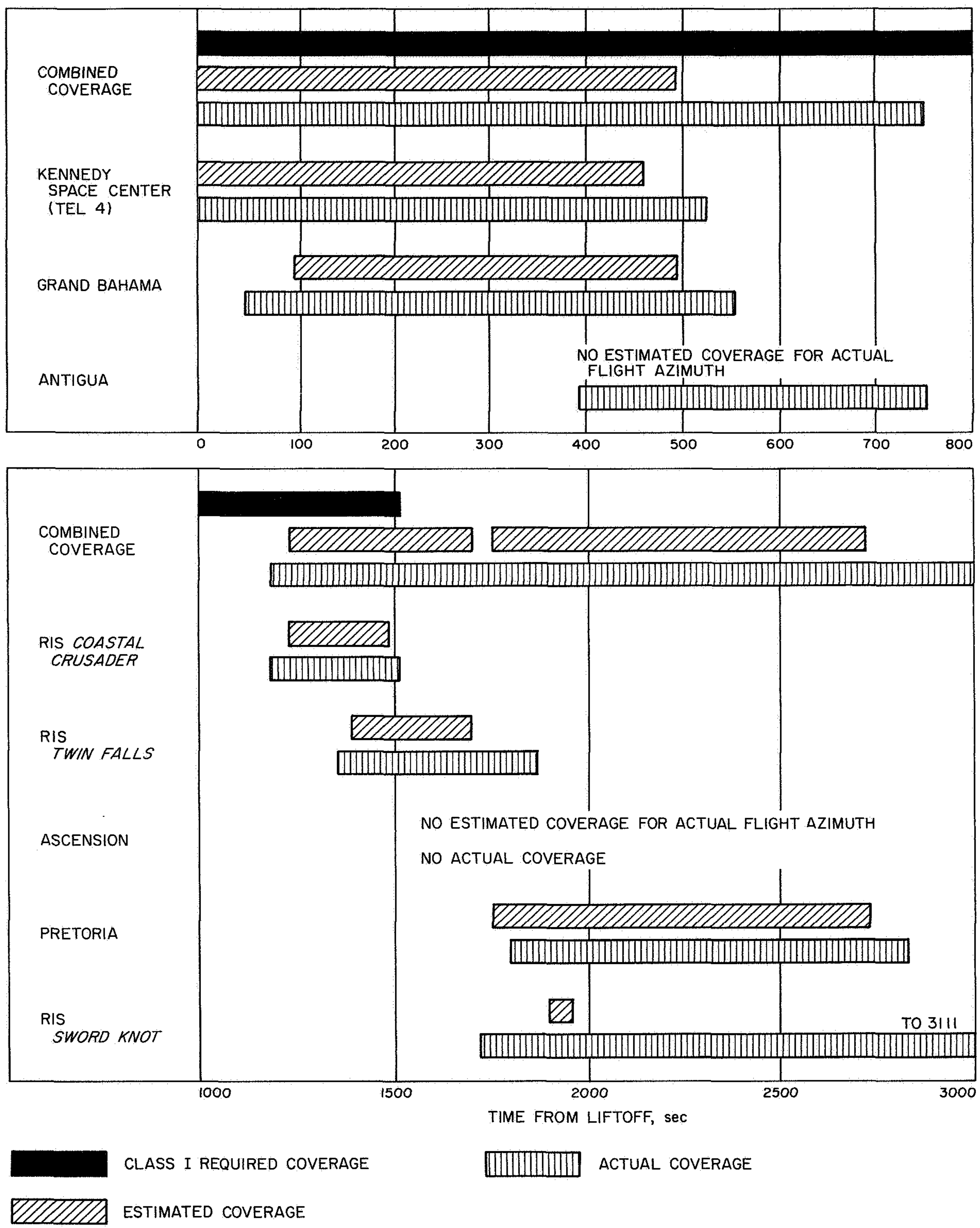


Fig. V-4. AFETR VHF telemetry coverage

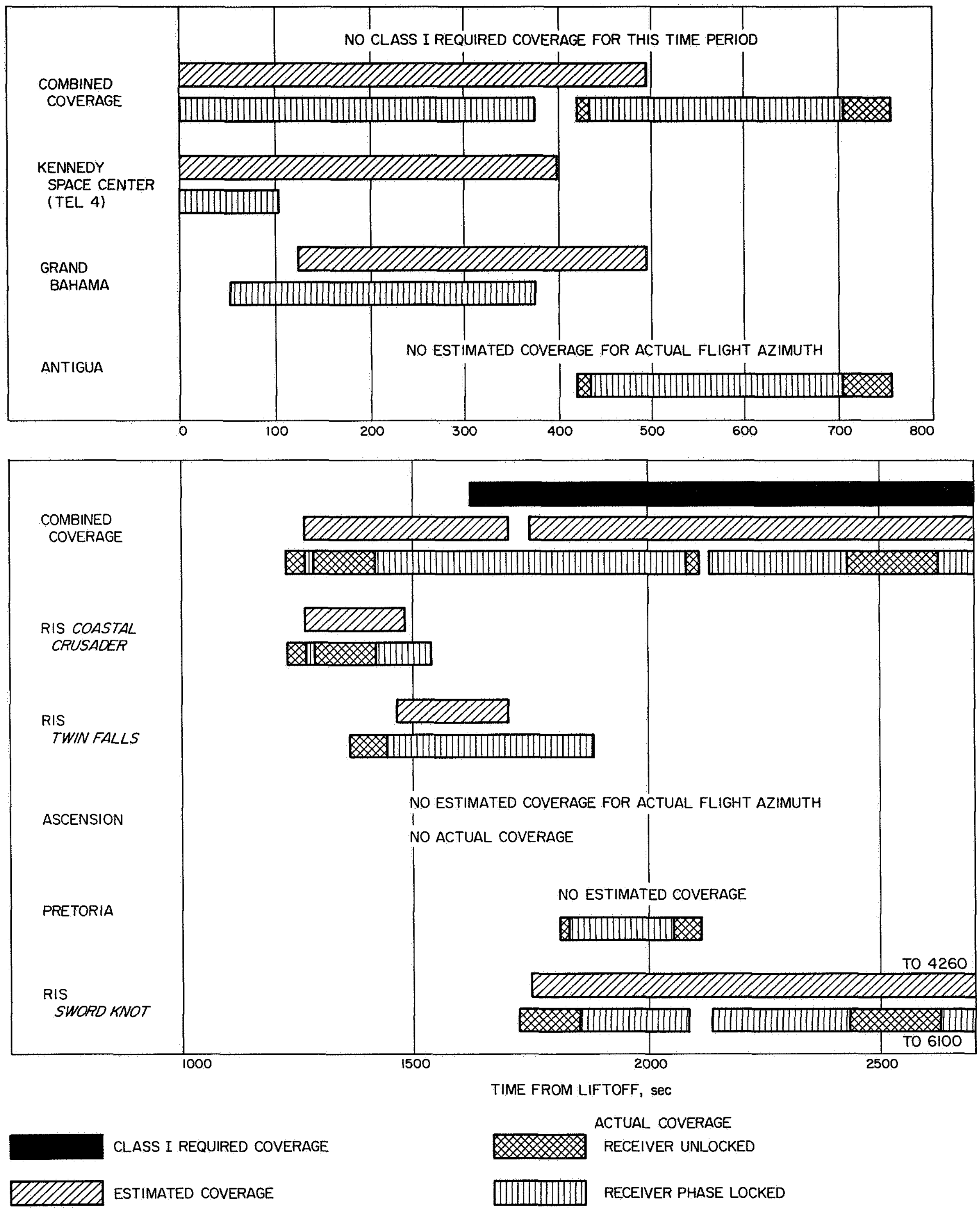


Fig. V-5. AFETR S-band telemetry coverage

Table V-2. *Atlas/Centaur* Mark Event readouts

Mark Event No.	Time of Event (GMT)	Station reporting
Liftoff	07:39:01.075	Cape Kennedy
1	07:41:34.500	Cape Kennedy
2	07:41:37.810	Cape Kennedy
3	07:42:19.410	Cape Kennedy
4	07:42:49.640	Cape Kennedy
5	07:43:06.630	Cape Kennedy
	07:43:06.500	Bermuda
6	07:43:09.310	Cape Kennedy
	07:43:09.000	Bermuda
7	07:43:19.800	Cape Kennedy
	07:43:19.800	Bermuda
8	07:48:42.100	Antigua
	07:48:42.900	Bermuda
9	07:48:44.300	Bermuda
10	07:50:09.500	Bermuda
11	07:49:58.900	Bermuda
12	08:00:57.000	Grand Canary
13	08:01:45.400	*Twin Falls*
14	08:01:35.000	*Coastal Crusader*
	08:01:45.400	*Twin Falls*
15	08:03:29.750	*Coastal Crusader*
	08:03:29.800	*Twin Falls*
16	08:03:49.000	*Coastal Crusader*
	08:03:48.900	*Twin Falls*
17	08:03:59.500	*Coastal Crusader*
	08:03:59.500	*Twin Falls*
18	08:04:20.000	*Twin Falls*
19	08:04:25.100	*Twin Falls*
20	08:04:30.000	*Twin Falls*
21	08:04:33.400	*Twin Falls*
22	Not reported	
23	Not reported	
24	08:08:30.000	*Twin Falls*
	08:08:30.1	*Sword Knot*
25	08:12:40.200	Tananarive
	08:12:40.5	Pretoria
26	08:14:20.100	Tananarive
	08:14:20.1	*Sword Knot*

5. Real Time Computer System (RTCS)

For the launch and near-earth phase of the mission, the RTCS provides trajectory computations based on tracking data and telemetered vehicle guidance data. The RTCS output includes:

(1) The interrange vector (IRV), the standard orbital parameter message (SOPM), and orbital elements and injection conditions.

(2) Predicts, look angles, and frequencies for acquisition use by down-range stations.

(3) Injection conditions mapped to lunar encounter and I-matrices defining orbit determination accuracies for early trajectory evaluation prior to the highly refined orbits generated by the Flight Path Analysis and Command (FPAC) group.

A total of six orbits were computed by the RTCS, including a parking orbit from Bermuda data, a second parking orbit from *Centaur* guidance telemetry data, a theoretical transfer orbit using Bermuda data plus nominal *Centaur* second burn data, a postretro orbit using Carnarvon data, and two actual spacecraft transfer orbits from DSS 51 data. Because of the lack of metric data from the *Twin Falls,* it was not possible to compute an actual preretro transfer orbit, and the only DSN predicts available were those based on the theoretical transfer orbit and the postretro orbit.

B. Manned Space Flight Network

The MSFN, managed by GSFC, supported the *Surveyor VI* mission by performing the following functions:

(1) Tracking of the *Centaur* beacon (C-band).

(2) Receiving and recording *Centaur*-link telemetry.

(3) Receiving, recording, and retransmitting S-band telemetry to DSS 42 in real-time.

(4) Providing real-time confirmation of certain Mark Events.

(5) Providing computing support through the use of the GSFC Data Operations Branch.

The GSFC tracking and telemetry facilities and equipment used in support of *Surveyor VI* are listed in Table V-3. GSFC also supported the Operational Readiness Test (ORT) prior to launch.

Table V-3. GSFC Network configuration

Station	Acquisition aid	VHF telemetry	C-band radar	SCAMA[a]	Radar high-speed data	Real-time readouts
Bermuda	X	X	X	X	X	X
Grand Canary	X	X	X	X		X
Tananarive	X	X	X	X		X
Carnarvon	X	X	X	X		
GSFC				X	X	

[a]Signaling, conferencing, and monitoring arrangement (voice circuits).

1. Acquisition Aids

The MSFN stations are equipped with acquisition aids to track the vehicle and provide RF inputs to the telemetry receivers from AOS to LOS. Performance recorders are used to record AGC and angle errors for postmission analysis.

The acquisition aid systems performed their required functions during the *Surveyor VI* mission.

2. Telemetry Data

The MSFN stations are also equipped to decommutate, receive, and record telemetry. The telemetry requirements placed on the MSFN were:

(1) Bermuda, Grand Canary, and Tananarive were to receive and record the *Centaur* 225.7-MHz link from AOS to LOS.

(2) Bermuda was to receive and record the *Atlas* 229.9-MHz link from AOS to LOS.

(3) *Centaur* Mark Event readouts were required from Bermuda, Grand Canary, and Tananarive in real-time or as near real-time as possible, after the vehicle was in view of the station.

(4) Bermuda was to display range safety parameters on the *Atlas* and *Centaur* links.

(5) Carnarvon was to receive, record, and retransmit S-band telemetry to DSS 42 in real-time.

(6) All stations were to provide magnetic tape recordings, strip chart recordings, and Post-Launch Instrumentation Message (PLIM) data sheets.

The telemetry systems performed all required functions. The predicted vs actual coverage is shown in Fig. V-6. See Table V-2 for Mark Events reported by the MSFN stations. Bermuda received and recorded range safety parameters and Mark Events 5 through 11. Grand Canary reported Mark Event 12, attitude rates, and propellant ullages. Although no VHF coverage was predicted for Carnarvon, the station obtained 9½ min of VHF data. Carnarvon obtained about 50 min of S-band telemetry before being released by DSS 42.

3. *Metric Tracking Data* (C-Band)

The radar requirements placed on the MSFN were:

(1) Bermuda was to provide beacon tracking of the *Centaur* from AOS to LOS and range safety support to ETR. Bermuda was the prime station for range safety from 78 to 90 deg and backup from 91 to 115 deg.

(2) Bermuda was to provide real-time transmission of high-speed and low-speed radar data to GSFC and RTCS.

(3) Bermuda was to provide magnetic tape recordings, strip chart recordings, and PLIM data sheets.

(4) Grand Canary, Tananarive, and Carnarvon were to provide metric data in a backup support role.

Bermuda provided the radar beacon tracking, magnetic tape recording (at a minimum of 10 points/sec), and real-time data transmission to GSFC and AFETR.

The predicted vs actual coverages are shown in Fig. V-7. The Bermuda FPQ-6 radar achieved 486 sec of valid autotrack data. The FPS-16 radar obtained 482 sec of data. However, 24 sec of data was lost at $L + 524$ because of obstruction by the FPQ-6 antenna. Tracking by the FPS-16 radar was inhibited, owing to phasing, from AOS until $L + 369$.

The Grand Canary radar provided 216 sec of valid data.

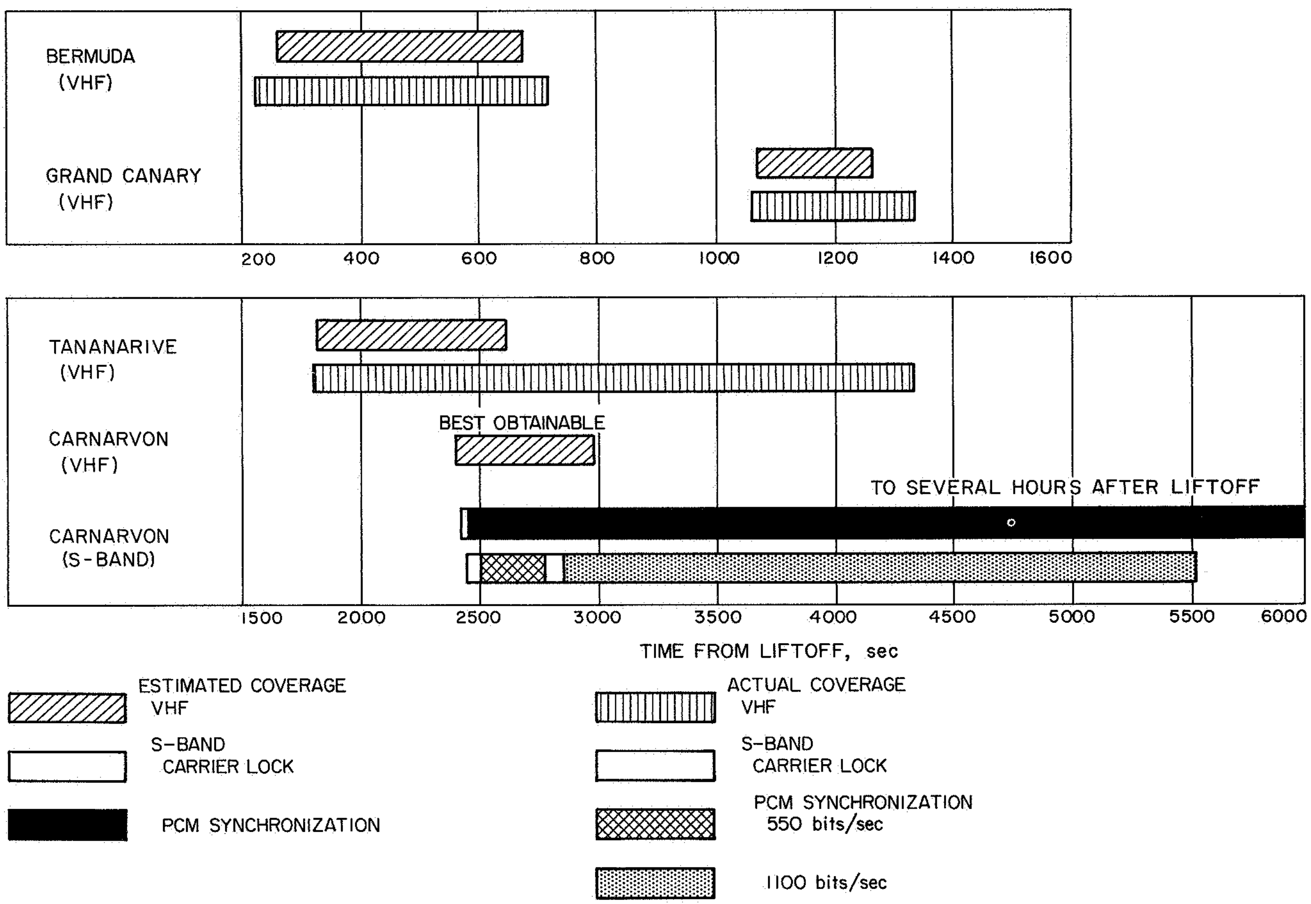


Fig. V-6. MSFN VHF and S-band telemetry coverage

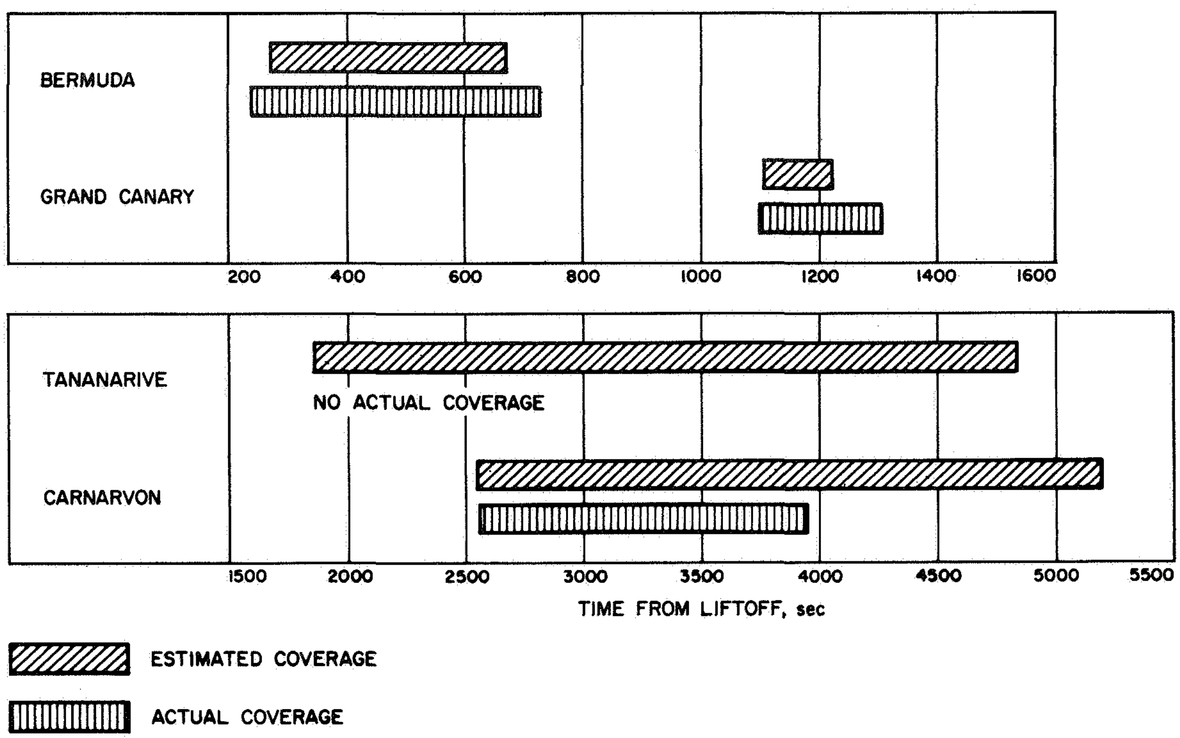


Fig. V-7. MSFN C-band radar coverage

Tananarive, which is still on an engineering test basis, did not acquire track. The reason is unknown.

Carnarvon, although experiencing several dropouts, provided data over an interval of approximately 1400 sec. The reason for the dropouts is not known, but aspect angle is suspected.

4. Computer Support, Data Handling, and Ground Communications

The GSFC Data Operations Branch (DOB) was to provide computing support for the MSFN stations during the prelaunch, launch, and orbital phases of the mission. Computer requirements included providing MSFN station view periods for mission planning purposes, MSFN station nominal pointing data and real-time acquisition messages based on flight data, and postflight reformatting of magnetic tape recordings of the radar data received from AFETR and MSFN.

The nominal pointing data for mission planning purposes was provided too late for use in preparation of draft copies of the Network Operations Plan. During flight, the Canary acquisition message arrived on site after AOS, and the first Tananarive and Carnarvon acquisition messages were invalid. Up-dated messages were transmitted, but arrived too late for Tananarive. The DOB provided all other required support.

The NASCOM Network provided 5 teletype, 10 voice, and 3 high-speed data circuits to the MSFN in support of the mission.

Special propagation forecasts were provided for all HF radio circuits effective approximately 9 hr before liftoff, and special coverage was established commencing at approximately 6 hr before liftoff. "Minimize" was placed in effect from approximately 1½ hr before liftoff until 2 hr after liftoff. High-speed transmissions for London and Canberra were paralleled with low-speed input. All circuits provided to the MSFN were operational at liftoff.

C. Deep Space Network

The DSN supports *Surveyor* missions with the integrated facilities of the Deep Space Instrumentation Facility (DSIF), the Ground Communications Facility (GCF), and the Space Flight Operations Facility (SFOF).

The DSN provides a command and telemetry link with the spacecraft upon initial acquisition of spacecraft signals by a DSIF station, enabling the DSIF station to control the spacecraft and furnish range rate data, angular tracking data, and real-time telemetry data to the SFOF. Continuous tracking and control are then provided throughout the remainder of the mission by the prime DSIF stations designated to support each *Surveyor* mission.

Following the accumulation of sufficient tracking data by the SFOF, an orbit is determined that predicts the future path of the spacecraft. This data allows the computation of a midcourse maneuver to compensate for injection errors. The DSIF, under the control of the MOS, commands the midcourse maneuver, after which engineering telemetry and tracking data is gathered and transmitted via the GCF to the SFOF, where the midcourse maneuver is evaluated and appropriate commands for the terminal maneuver are computed. After touchdown, DSIF stations command the spacecraft during lunar operations and receive engineering and scientific telemetry as well as video data.

1. The DSIF

The following Deep Space Stations (DSS) were committed as prime stations for support of the *Surveyor VI* mission:

DSS 11, Pioneer, Goldstone Deep Space Communications Complex (DSCC), Barstow, California.

DSS 42, Tidbinbilla, Australia, near Canberra.

DSS 61, Robledo, Spain, near Madrid.

In addition to the basic support provided by prime stations, the following support was provided for the *Surveyor VI* mission:

(1) DSS 71, Cape Kennedy, provided facilities for spacecraft/DSIF compatibility testing, and also received and recorded telemetry data after liftoff. In addition, DSS 71 used its Command and Data Handling Console (CDC) and Telemetry and Command Processor (TCP) computer to process AFETR telemetry data for transmission to JPL.

(2) DSS 14, Mars (Fig. V-8), Goldstone DSCC, Barstow, California, using its 210-ft-diameter antenna, provided backup tracking and command support during midcourse correction, terminal descent, and the postlanding spacecraft hop experiment. At touchdown, the baseband telemetry output of the DSS 14 prime receiver was transmitted to the SFOF.

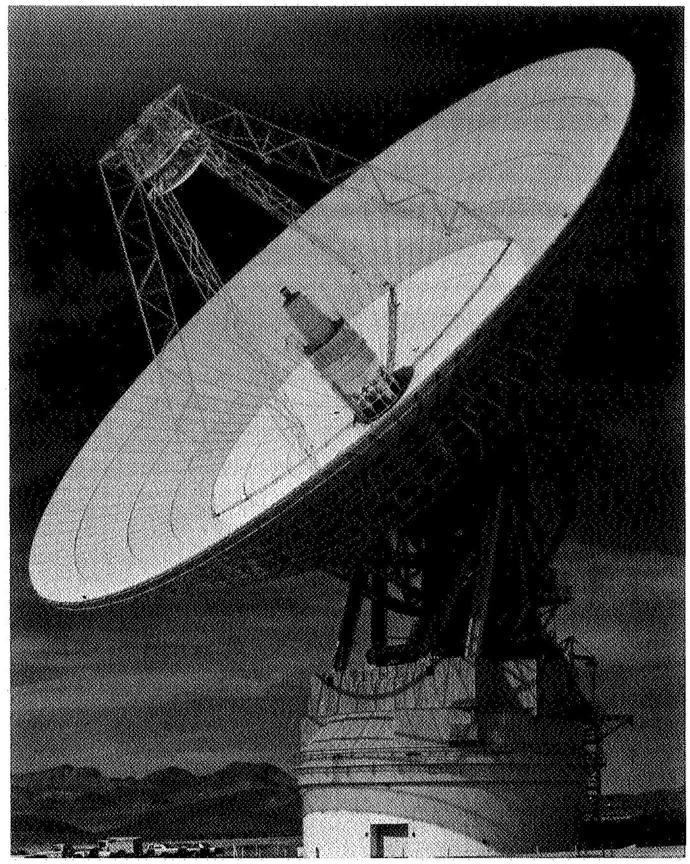


Fig. V-8. DSS 14 210-ft-diameter antenna at Barstow, California

(3) DSS 12, Echo, Goldstone DSCC, Barstow, California, provided backup telemetry coverage during the terminal descent phase.

(4) DSS 51, Johannesburg, South Africa, provided initial two-way acquisition and spacecraft commanding and tracking support during the transit phase.

For the November 6 launch date, DSS 51 was designated the initial two-way acquisition station as it had higher elevation angles and longer view periods than DSS 61.

The DSN was required to track the spacecraft and provide doppler and telemetry data as shown in Table V-4.

Data is handled by the prime DSIF stations as follows:

(1) Tracking data, consisting of antenna pointing angles and doppler (radial velocity) data, is supplied in near-real-time via teletype to the SFOF and postflight in the form of punched paper tape.

Table V-4. DSN tracking data requirements

Coverage and sampling rate	Data required
Track spacecraft from separation to first midcourse at 1-min sample rate (from initial DSIF acquisition to *L* + 1 hr, the sample rate is 1 sample per 10 sec)	Doppler (two- and three-way), antenna pointing angles, and telemetry
Track spacecraft from first midcourse to touchdown at 1-min sample rate	Doppler (two- and three-way), antenna pointing angles, and telemetry
Track spacecraft during midcourse maneuver and terminal maneuver executions at 1-sec sample rate, and transmit data at 10-sec sample rate	Doppler (two- and three-way or one-way), antenna pointing angles, and telemetry
Track spacecraft from touchdown to end of mission at 1-min sample rate during 1 hr following 10-deg elevation rise, during 1 hr centered around maximum elevation, and during 1 hr prior to 10-deg elevation set for DSS 11, 42, and 61	Doppler (two- and three-way) and telemetry

Two- and three-way doppler data is supplied full-time during transit, and also during lunar operations when requested by the *Surveyor* MOS. The two-way doppler function implies a transmit capability at the prime stations.

(2) Spacecraft telemetry data is received and recorded on magnetic tape. Baseband telemetry data is supplied to the CDC for decommutation and real-time readout. The DSIF also performs precommunication processing of the decommutated data using an on-site data processing (OSDP) computer. The data is then transmitted to the SFOF in near-real-time over high-speed data lines (HSDL).

(3) Video data is received and recorded on magnetic tape. This data is sent to the CDC and, at DSS 11 only, to the spacecraft TV Ground Data Handling System (TV-GDHS, TV-11) for photographic recording. In addition, video data from DSS 11 is sent in real-time to the SFOF for magnetic and photographic recording by the TV-GDHS, TV-1. After lunar landing, the two DSS 11 receivers are used for different functions. One provides a signal to the CDC, the other to the TV-GDHS. (Signals for the latter system are the prime *Surveyor* Project requirement during this phase of a mission.)

(4) Command transmission is another function provided by the DSIF. Approximately 280 commands are sent to the spacecraft during the nominal sequence from launch to touchdown. Confirmation of the commands sent is processed by the OSDP computer and transmitted by teletype to the SFOF.

The characteristics of the S-band tracking systems with 85-ft antennas are shown in Table V-5. The maximum doppler tracking rate depends on the loop noise bandwidth. For phase error of less than 30 deg and strong signal (greater than −100 dbm), tracking rates are as follows:

Loop noise bandwidth, Hz	Maximum tracking rate, Hz/sec
12	100
48	920
152	5000

a. DSIF preparation testing. A spacecraft compatibility test, configuration verification tests, system readiness verification (SRV) tests, and an Operational Readiness Test (ORT) are conducted for each mission to verify that all prime stations, communication lines, and the SFOF are fully prepared to meet mission responsibility.

Owing to a high level of *Surveyor V* activity immediately prior to *Surveyor VI* launch, requirements for *Surveyor VI* prelaunch testing were reduced. No special alpha scattering tests were conducted as alpha scattering operations were performed with all stations during the second lunar day of *Surveyor V*.

The *Surveyor VI* spacecraft RF compatibility test, verifying the compatibility of the spacecraft with the DSIF, was conducted in September at DSS 71.

One flight training test was conducted in two phases at DSS 11 during the last week of October. Also during October, a 2-hr data-link engineering test was performed between Carnarvon and DSS 42 to verify the unified S-band, telemetry processing equipment, and communication equipment interface configuration.

A lunar video ORT and a RF verification test involving DSS 11 and the TV-GDHS were conducted during the first week in November.

DSIF configuration tests were performed, but since data flow paths were verified by extended *Surveyor V* lunar operations, no system readiness verification tests were required.

Table V-5. Characteristics for S-band tracking systems

Antenna, tracking		Transmitter	
Type	85-ft parabolic	Frequency (nominal)	2113 MHz
Mount	Polar (HA-Dec)	Frequency channel	14b
Beamwidth ±3 db	~0.4 deg	Power	10 kW, max
Gain, receiving	53.0 db, +1.0, −0.5	Tuning range	±100 kHz
Gain, transmitting	51.0 db, +1.0, −0.5	Modulator	
Feed	Cassegrain	Phase input impedance	≥50 Ω
Polarization	LH[b] or RH circular	Input voltage	≤2.5 V peak
Max. angle tracking rate[a]	51 deg/min = 0.85 deg/sec	Frequency response (3 db)	DC to 100 kHz
Max. angular acceleration	5.0 deg/sec/sec	Sensitivity at carrier output frequency	1.0 rad peak/V peak
Tracking accuracy (1σ)	0.14 deg	Peak deviation	2.5 rad peak
Antenna, acquisition		Modulation deviation stability	±5%
Type	2 × 2-ft horn	Frequency, standard	Rubidium
Gain, receiving	21.0 db ±1.0	Stability, short-term (1σ)	1×10^{-11}
Gain, transmitting	20.0 db ±2.0	Stability, long-term (1σ)	5×10^{-11}
Beamwidth ±3 db	~16 deg	Doppler accuracy at F_{rc} (1σ)	0.2 Hz = 0.3 m/sec
Polarization	RH circular	Data transmission	TTY and HSDL
Receiver	S-band		
Typical system temperature			
With paramp	270 ±50°K		
With maser	55 ±10°K		
Loop noise bandwidth	12, 48 or 152 Hz		
threshold ($2B_{LO}$)	+0, −10%		
Strong signal ($2B_{LO}$)	120, 255, or 550 Hz		
	+0, −10%		
Frequency (nominal)	2295 MHz		
Frequency channel	14a		

[a]Both axes.

[b]Goldstone only.

At 48 hr before liftoff, DSS 11, 42, 51, 61, and 71 participated in a single mission ORT. Selected portions of the sequence of events were followed during the ORT, using both standard and nonstandard procedures. An evaluation of station and net control support during the ORT indicated the readiness of the TDS.

b. DSIF mission support through first lunar day. The DSIF stations supported the *Surveyor VI* mission with a high level of performance. Continuous tracking and telemetry coverage was provided from 29 min after liftoff on November 7 until two earth-days after lunar sunset, which occurred on November 24, 1967.

All requirements for tracking, telemetry, command, and data processing were met. High-quality angular tracking and doppler data were received throughout the *Surveyor VI* mission, with the exception of minor outages which neither adversely affected the flight or resulted in any mission objectives not being met.

A number of minor equipment anomalies and procedural problems occurred, but were readily corrected by station personnel without affecting the mission.

All of the DSIF prime stations reported "go" status during the countdown. All measured station parameters were within nominal performance specifications, and communications circuits were up. *Surveyor VI* was launched at 07:39:01 GMT on November 7 with all essential elements of the DSN functioning nominally.

Figure V-9 is a profile of the DSIF mission activity during the transit phase of the mission. This figure contains the periods each station tracked the spacecraft plotted against mission time from liftoff, and the number of commands transmitted by each DSS during each pass. (Also refer to the station view periods indicated in Fig. VII-2.)

One-way spacecraft signals were received by DSS 71 for 4 min and 49 sec after liftoff. DSS 71 also processed real-time AFETR telemetry with its CDC and TCP.

Initial two-way acquisition was achieved by DSS 51 4 min and 11 sec after spacecraft rise over the station's horizon. Less than 5 min and 39 sec elapsed from the time of receipt of the first spacecraft signals until DSS 51 was prepared to command the spacecraft. DSS 51 tracked about 14 hr during the first pass, and 12 hr during the second pass.

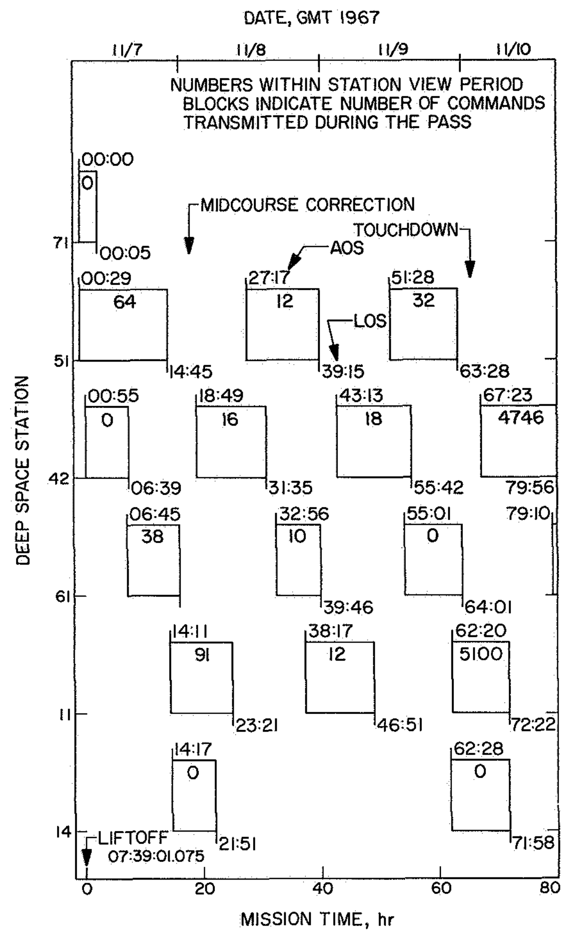


Fig. V-9. DSIF station view periods and command activity: transit phase

The signal levels received at the DSIF stations during the transit phase are shown in Fig. V-10 and correspond very closely to the predicted levels. The predicted levels prior to Canopus star acquisition are not given, as the spacecraft was not roll-stabilized during this period and the received signal level could vary over a wide range owing to variations in spacecraft antenna patterns. During star acquisition, midcourse correction, and retro maneuver, the spacecraft was in high-power mode and the received data is 16.5 db above the low-power signal level. Data-bit rate changes occurred within 2 db of the predicted bit-error-rate thresholds. The periods during which gyro drift tests were conducted are shown at the top of the figure. Drifting of the spacecraft attitude during the gyro drift checks resulted in changes of up to ±3 db in the received signal levels because of antenna pattern variations.

From the time of initial two-way acquisition by DSS 51 until approximately 40 min before spacecraft

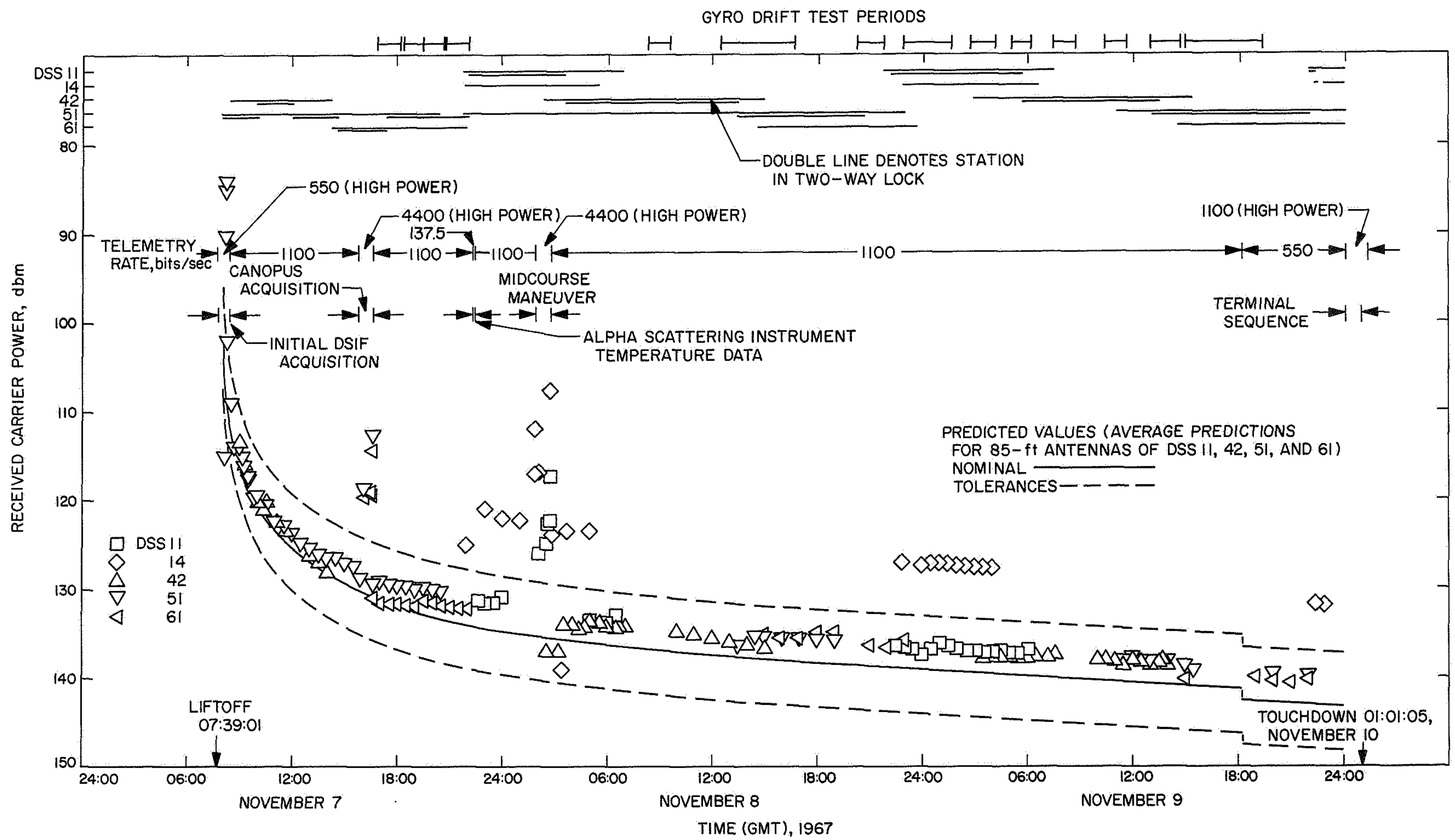


Fig. V-10. DSS received signal level during transit

main retro ignition, DSIF stations tracked *Surveyor VI* in the two-way mode and, with few exceptions, received high-quality doppler data. A loss of two-way data occurred during Canopus acquisition over DSS 61, when an unexpected loss of lock was experienced. Reacquisition of two-way communications was not attempted until completion of Canopus acquisition approximately 50 min later. The only other appreciable loss of good two-way doppler data occurred during the second pass over DSS 51 and 61. When DSS 51 lost communication with the spacecraft, an unscheduled transfer was made to DSS 61, which tracked in two-way until the spacecraft was transferred to DSS 11 during its second pass. The two-way data received during this period of approximately 1½ hr was unusable because of excessively high noise traced to a malfunctioning DSS 61 rubidium frequency standard. Replacement of the rubidium during the third pass resulted in an immediate improvement in three-way data levels at DSS 61.

Surveyor VI marked the first widespread use of doppler resolver data, at all stations except DSS 51, during the in-flight portion of a *Surveyor* mission. Unfortunately this data, which was to provide a significant decrease in target uncertainties, did not produce increased accuracy because the weighting of the doppler data was not changed.

The midcourse correction was successfully commanded by DSS 11 during the first pass, with DSS 14 supporting this phase in a background roll with its 210-ft-diameter antenna. Prior to the midcourse correction, an erroneous indication of low hydrostatic bearing clearance at the DSS 14 antenna caused an 8-min loss of backup data when tracking was interrupted to investigate the cause.

The terminal maneuvers and initiation of the retro descent were successfully commanded during the spacecraft's third pass over DSS 11. The spacecraft touchdown on the lunar surface was so gentle that none of the four receivers at the Goldstone complex (two each at DSS 11 and 14) lost one-way lock with the spacecraft.

Subsequent to the lunar landing, the DSIF provided 24-hr-per-day tracking coverage, with DSS 11, 42, and 61 providing postlanding mission support. Figure V-11 contains a summary of DSIF postlanding mission activity during the first lunar day until spacecraft shutdown approximately 41 hr after sunset. This figure contains, for each station pass: the period of tracking coverage, the number of commands transmitted, the number of TV pictures received, and the number of hours of alpha scattering instrument data received. On November 17, using DSS 11 with DSS 14 as a backup, the spacecraft was commanded to perform a liftoff and translation experiment of about 6-sec duration.

c. DSIF mission support during second lunar day. The DSN supported *Surveyor VI* second lunar day operations with DSS 11, 42, and 61, and minimal support from the Space Flight Operation Facility and Ground Communication Facility. The initial revival attempt on December 13, 1967, by DSS 11 was unsuccessful, as were subsequent attempts by DSS 11, 42, and 61 on this date. On December 14, after unsuccessful attempts by both DSS 11 and 42, DSS 61 succeeded in reviving *Surveyor VI*. However, the received signal was erratic both in strength and frequency, and was eventually lost three hours later. Some engineering data was received during this period. Subsequent interrogations by the DSN to revive *Surveyor VI* during December 15, 16, and 17 were unsuccessful.

During the attempt to revive *Surveyor VI* by DSS 11 on December 14, a nonscheduled *Surveyor V* revival attempt was initiated. Unexpectedly, *Surveyor V* responded immediately with good signals, although this was the fourth lunar day for this spacecraft. During the *Surveyor V* pass over DSS 11 on December 15, 67 TV pictures were received with images that were recognizable but of poor lighting quality, as was the case during the second lunar day. Alpha scattering instrument and TV science data received was also minimal owing to degradation of the *Surveyor V* spacecraft system. *Surveyor V* ceased transmitting signals when DSS 11 commanded the spacecraft to operate in the high-power mode for video sequences on December 17.

All subsequent efforts by the DSN to revive both *Surveyor V* and *VI* were unsuccessful, and DSN support was terminated December 21, 1967. DSN support of *Surveyor VI* second lunar day operations, as well as *Surveyor V* fourth lunar day operations, was considered excellent.

2. GCF/NASCOM

For *Surveyor* missions, the GCF transmits tracking, telemetry, and command data from the DSIF to the SFOF, and control and command functions from the SFOF to the DSIF by means of NASCOM facilities.

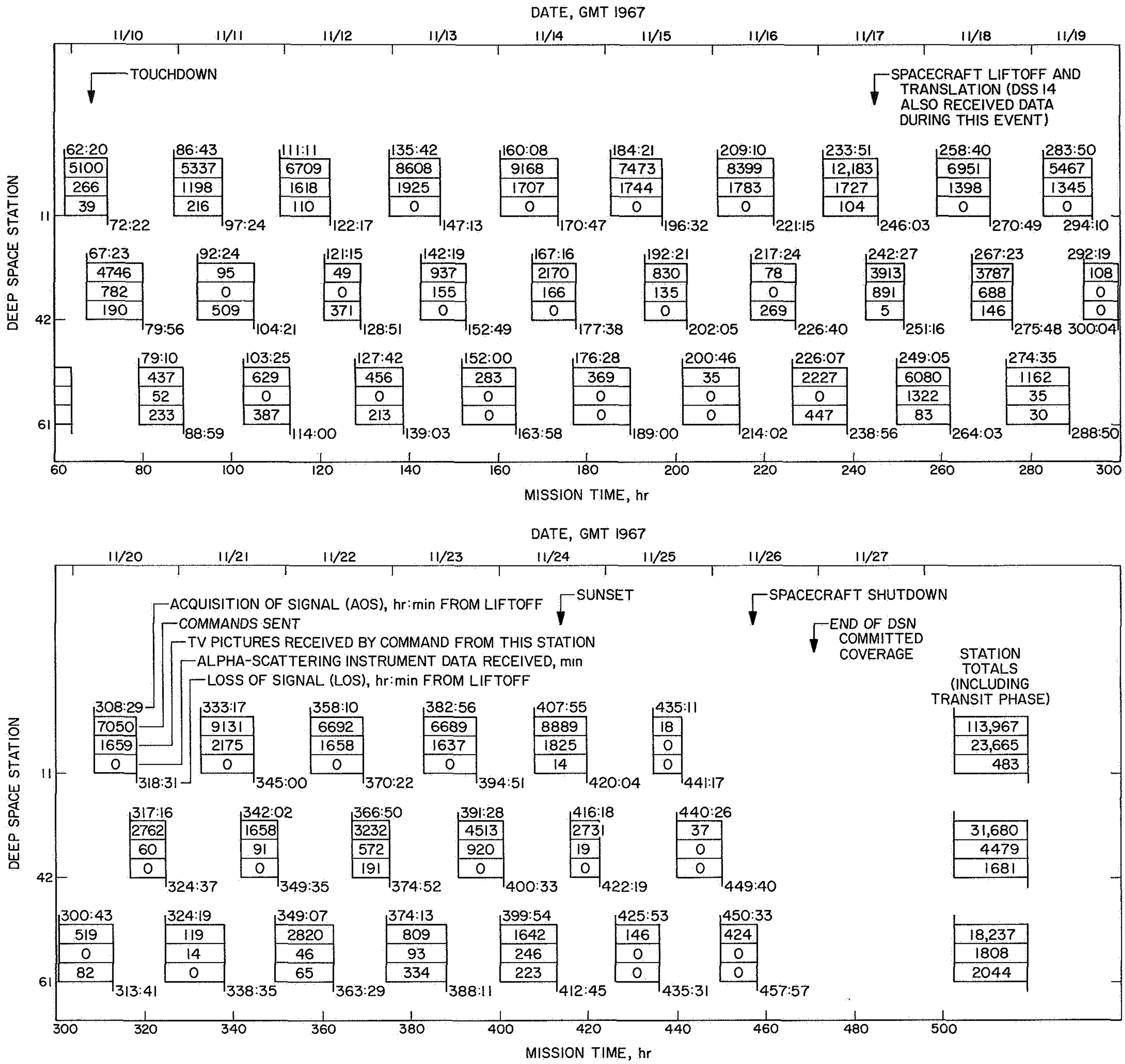


Fig. V-11. DSIF station tracking periods and reported command activity: postlanding (first lunar day)

The GCF also transmits simulated tracking data to the DSIF and video data and base-band telemetry from DSS 11, Goldstone DSCC, to the SFOF. The links involved in the system are shown in Fig. V-12.

The performance of the GCF/NASCOM facilities in support of the *Surveyor VI* mission was considered excellent, demonstrating a high degree of reliability.

a. Communication processor. On the *Surveyor VI* mission, all communication circuits between the DSIF and SFOF were automatically switched using a JPL communication processor (CP). Communication processor operations proved very reliable during critical mission phases. The JPL CP, although nonoperational during a 36-hr period prior to liftoff, provided excellent support during the mission. The only significant outage (2 min) occurred

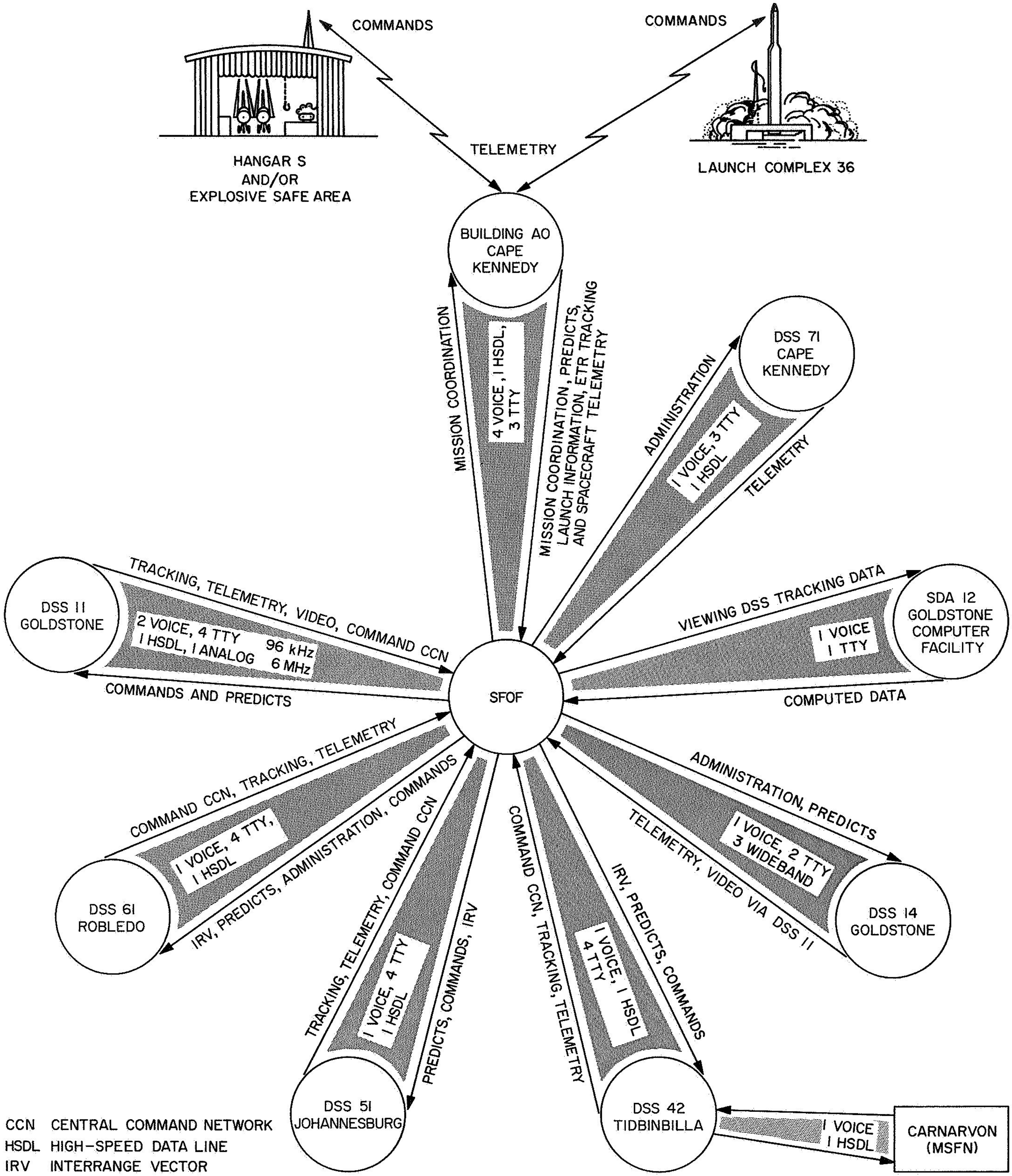


Fig. V-12. DSN/GCS communications links for *Surveyor VI* mission

during a rainstorm on November 19, when a power failure affected communications and the data processing system. A number of short-duration failures were experienced by the London, Canberra, and GSFC CP systems, but these did not cause serious loss of data.

b. Teletype circuits. Teletype (TTY) circuits (four available to each prime station) are used for transmitting tracking and telemetry data, commands, and administrative traffic. During the countdown, a major systems failure was experienced at Cocoa Beach, Florida. This affected all of the teletype as well as voice and high-speed data circuits to DSS 71, and all of the teletype and one voice circuit to Building AO. All circuits were restored about 3 hr before liftoff.

On the *Surveyor VI* mission, the teletype circuits and the CP were exceptionally reliable, the weakest circuits (to DSS 51) showing approximately 92% reliability. DSS 51 experienced intermittent outages on the high-frequency radio path circuits due to propagation problems.

c. Voice circuits. The voice circuits are shared between the DSIF and the *Surveyor* Project for administrative, control, and commanding functions.

The NASCOM voice circuits provided to the DSN for the *Surveyor VI* mission performed well except for several minor outages. All voice circuits exhibited 99 to 100% reliability except the circuit to DSS 51, which showed 38% reliability.

d. High-speed data lines (HSDL). One HSDL is provided to each prime site as well as to DSS 71 and to Building AO at Cape Kennedy for telemetry data transmission to the SFOF in real-time. The lines are also used during prelaunch testing to transmit simulation data to the stations and during the mission to back-feed various voice nets as required.

This part of the communications system performed well during both the prelaunch testing and mission phases. Dropouts on the DSS 51 HSDL were reported about 45 min before liftoff. Attempts to correct the problem were ineffective until it was discovered that a converter at Riverhead, New York, was set for the wrong channel. The circuit was restored about 15 min after liftoff, which was before DSS 51 acquisition. During the mission, all HSDL exhibited from 99 to 100% reliability except the DSS 51 circuit, which showed 94% reliability.

Both modem* types (NASCOM and Hallicrafters) were required during the *Surveyor VI* mission. NASCOM modems were used for sending high-speed data from all stations except DSS 51, and reliability was very high. DSS 51 used the Hallicrafter modem, which also performed reliably. Minor problems occurred during launch when several stations were active and modem switching was required in order for each station to perform data transfer tests.

e. Wideband microwave system. The wideband microwave link between DSS 11, Goldstone DSCC, and the SFOF consists of one 6-MHz simplex (one-way) channel for video and one 96-kHz duplex (two-way) data channel.

The major relay antennas of the microwave system were damaged by fire prior to *Surveyor VI* launch. However, the 6-MHz circuit remained quite reliable and was used to back up the 96-kHz line for base-band telemetry transmission to the SFOF. The link performed with 96% reliability, a good rating, during the *Surveyor VI* mission.

3. DSN in SFOF

The DSN supports the *Surveyor* missions by providing mission control facilities and performing special functions within the SFOF.

a. Data Processing System. The SFOF Data Processing System (DPS), under the control of the MOS, performs the following functions for *Surveyor* missions:

(1) Computation of acquisition predictions for DSIF stations (antenna pointing angles and receiver and transmitter frequencies).

(2) Orbit determinations.

(3) Midcourse maneuver computation analysis.

(4) On-line telemetry processing.

(5) Command tape generation.

(6) Simulated data generation (telemetry and tracking data for tests).

The DPS general configuration used for the *Surveyor VI* mission is shown in Fig. V-13, and consists

*A modem (modulator-demodulator) is a device for converting a digital signal to a signal which is compatible with telephone line transmission (e.g., a frequency-modulated tone).

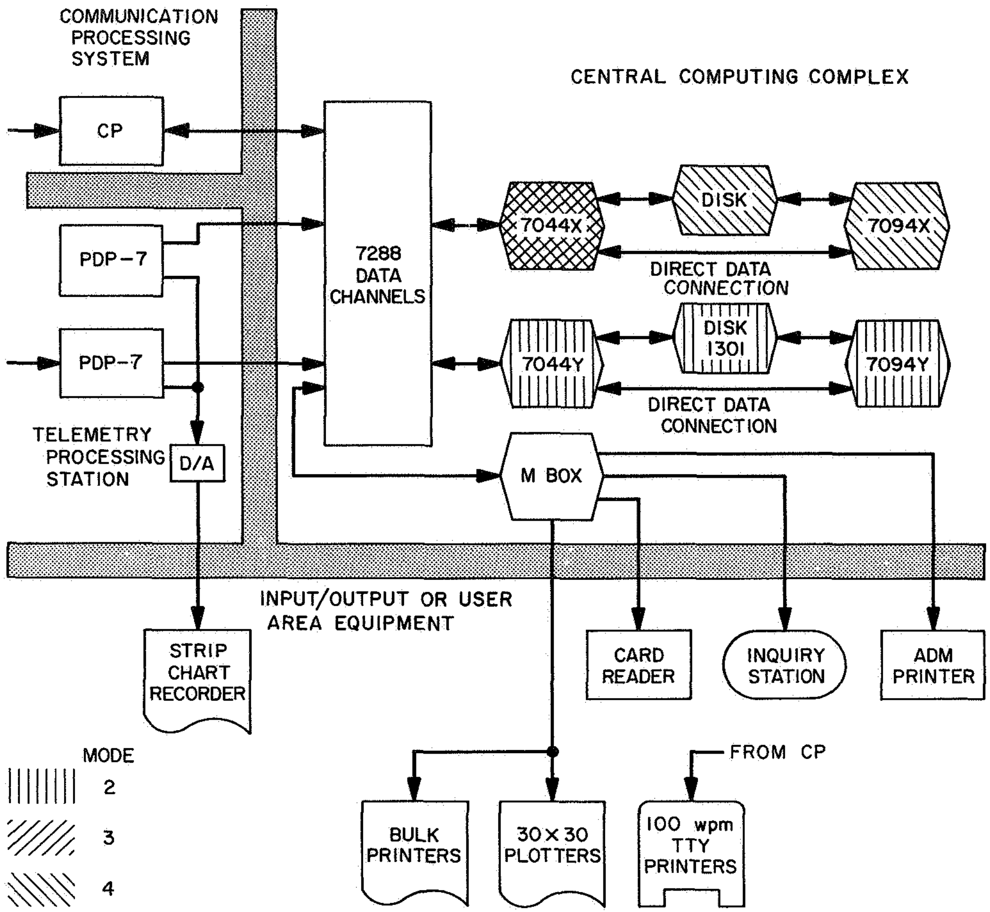


Fig. V-13. General configuration of SFOF Data Processing System

of two PDP-7 computers in the telemetry processing station (TPS), two strings of IBM 7044/7094 computers in the Central Computing Complex (CCC), and a subset of the input/output (I/O) system.

The DPS performed in an excellent manner, with only minor hardware problems which did not detract from the mission.

As implemented first on the *Surveyor V* mission, a redesigned computer configuration was utilized in conjunction with a communication processing system, which automatically switches teletype lines and allows computer sharing in support of other concurrent space program missions. Its capabilities include switching, real-time monitoring, quick access message logging and recall. Computer processor (CP) problems, although not significantly affecting the mission, did cause minor delays in providing SFOF user areas with incoming data. Considerable computer sharing was performed with *Mariner V* during *Surveyor VI* lunar operations.

The two PDP-7 computers are used to process high-speed telemetry data. This processing consists of decommutating and transferring the data to the 7044 computer via the 7288 data channels, generating a digital tape for non-real-time processing, and supplying digital-to-analog converters with discrete data parameters to drive analog recorders in both the spacecraft analysis area and the space science analysis area. The IBM 7044/7094 computer string dual configuration was used extensively for the *Surveyor VI* mission, and successfully processed all high-speed data received from the TPS and all teletype data received from the communications center, as well as all input/output requests from the user areas.

The input/output system provides the capability for entering data control parameters into the 7044/7094 computers and also for displaying computed data in the user areas via the various display devices. The input/output system performed adequately; many minor problems were reported, but were resolved in real-time with only a minor loss of data.

b. DSN Intracommunications System (DSN/ICS). The DSN/ICS provides the capability of receiving, switching, and distributing all types of information required for spaceflight operations and data analysis to designated areas or users within the SFOF. The system includes facilities for handling all voice communications, closed circuit television, teletypes, high-speed data, and data received over the microwave channels.

The DSN/ICS performed in an exceptional manner, with only minor anomalies.

The television communications subsystem experienced minor equipment problems that were resolved in real-time.

4. DSN/AFETR Interface

The DSN/AFETR interface provides real-time data transmission capability for both VHF and S-band downrange telemetry from Building AO at Cape Kennedy to the SFOF. The nominal switchover time is after the spacecraft S-band transmitter high-power turn-on. The interface with the *Surveyor* Project is at the input to the CDC, Building AO, where it is sent to DSS 71 for processing by the CDC and the telemetry and command processor (TCP) computer. The output of the CDC is transmitted to the SFOF via the GCF/NASCOM. Beginning with the *Surveyor V* mission, this configuration was established as the prime link between the SFOF and AFETR to provide an interface similar to the data transmission links with prime DSIF stations. The previously used link between Building AO and SFOF, via Bell modems, was retained as backup. An interface for Building AO and DSS 71 to receive AFETR real-time telemetry from TEL 4 at Kennedy Space Center was exercised and performed very well.

In-flight spacecraft telemetry received from the AFETR stations was relayed to the SFOF until approximately 40 min after liftoff.

Both the prime DSS 71 and the Bell modem backup systems provided good data during the mission but, owing to the high quality of the Bell modem data, the DSS 71 data was not processed. The backup system was also used during the ORT to provide simulated telemetry data from the SFOF to Building AO and performed well.

DSIF tracking data for early orbit determination was successfully back-fed to the Real Time Computer System at the AFETR. The system for transmission of real-time telemetry data from the MSFN station at Carnarvon, Australia, to the SFOF via DSS 42 was actuated for *Surveyor VI*. This system performed very well and good data was received in the SFOF from Carnarvon acquisition until DSS 42 switched over to processing their own data, a period of approximately 21 min.

VI. Mission Operations System

A. Functions and Organization

The basic functions of the Mission Operations System (MOS) are the following:

(1) Continual assessment and evaluation of mission status and performance, utilizing the tracking and telemetry data received and processed.

(2) Determination and implementation of appropriate command sequences required to maintain spacecraft control and to carry out desired spacecraft operations during transit and on the lunar surface.

The *Surveyor* command system philosophy introduces a major change in the concept of unmanned spacecraft control: virtually all in-flight and lunar operations of the spacecraft must be initiated from earth. In previous space missions, spacecraft were directed by a minimum of earth-based commands. Most in-flight functions of those spacecraft were automatically controlled by an on-board sequencer which stored preprogrammed instructions. These instructions were initiated by either an on-board timer or by single direct commands from earth. For example, during the *Ranger VIII* 67-hr mission, only 11 commands were sent to the spacecraft; whereas for a standard *Surveyor* mission, approximately 300 commands are sent to the spacecraft during the transit phase, out of a command vocabulary of 201 different direct commands. For *Suryevor VI*, about 360 commands were sent during transit and over 160,000 commands were sent following touchdown.

Throughout the space flight operations of each *Surveyor* mission, the command link between earth and spacecraft is in continuous use, transmitting either fill-in or real commands every 0.5 sec. The *Surveyor* commands are controlled from the SFOF and are transmitted to the spacecraft by a DSIF station.

The equipment utilized to perform MOS functions falls into two categories: mission-independent and mission-dependent equipment. The former is composed chiefly of the *Surveyor* TDS equipment described in Section V. It is referred to as mission-independent because it is general-purpose equipment which can be utilized by more than one NASA project when used with the appropriate project computer programs. Selected parts of this equipment have been assigned to perform the functions necessary to the *Surveyor* Project. The mission-dependent equipment (described in Section VI-B, following) consists of special equipment which has been installed at DSN facilities for specific functions peculiar to the project.

The *Surveyor* Project Manager, in his capacity as Mission Director, is in full charge of all mission operations. The Mission Director is aided by the Assistant Mission Director and a staff of mission advisors. During the mission, the MOS organization was as shown in Fig. VI-1.

Mission operations are under the immediate, primary control of the Space Flight Operations Director (SFOD) and supporting *Surveyor* personnel. Other members of the team are the TDS personnel who perform services for the *Surveyor* Project.

During space flight and lunar surface operations, all commands are issued by the SFOD or his specifically delegated authority. Three groups of specialists provide technical support to the SFOD in the flight path, spacecraft performance, and science areas.

1. Flight Path Analysis and Command Group

The Flight Path Analysis and Command (FPAC) group handles those space flight functions that relate to the location of the spacecraft. The FPAC Director maintains control of the activities of the group and makes specific recommendations for maneuvers to the SFOD in accordance with the flight plan. In making these recommendations, the FPAC director relies on five subgroups of specialists within the FPAC group.

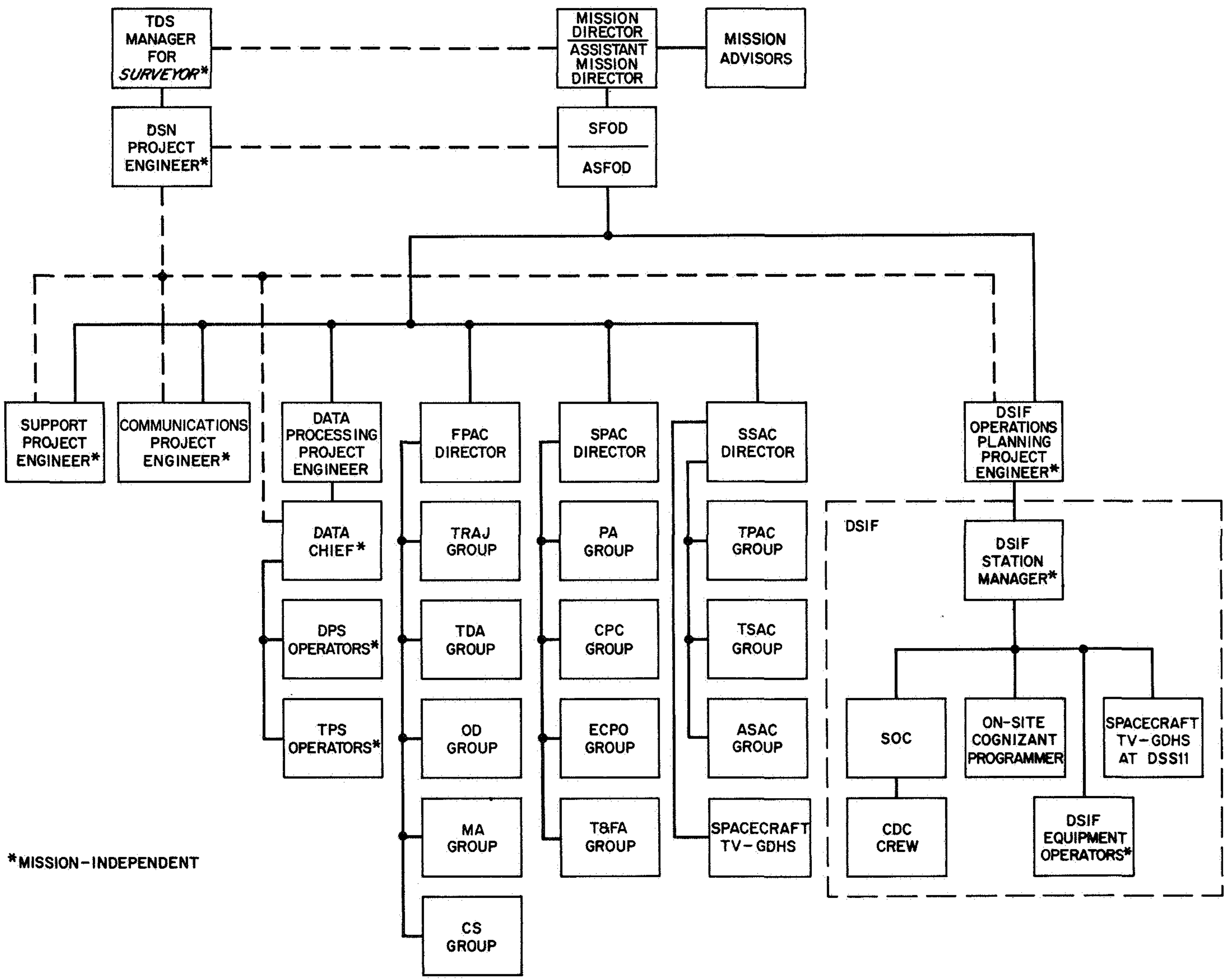


Fig. VI-1. Organization of the MOS during the *Surveyor VI* mission

(1) The Trajectory (TRAJ) group determines the nominal conditions of spacecraft injection and generates 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 are made by computer.

(2) The Tracking Data Analysis (TDA) group makes a quantitative and descriptive evaluation of tracking data received from the DSIF stations. The TDA group provides 24-hr/day monitoring of incoming tracking data. To perform these functions the TDA group takes advantage of the Data Processing System (DPS) and of computer programs generated for their use. The TDA group acts as direct liaison between the data users (the Orbit Determination group) and the DSIF and provides predicts to the DSIF.

(3) The Orbit Determination (OD) group, during mission operations, determines the actual orbit of the spacecraft by processing the tracking data received from the DSN tracking stations by way of the TDA group. Also, statistics on various parameters are generated so that maneuver situations can be evaluated. The OD group generates tracking predictions for the DSIF stations and recomputes the orbit of the spacecraft after maneuvers to determine the success of the maneuver.

(4) The Maneuver Analysis (MA) group is the subgroup of FPAC responsible for developing possible midcourse and terminal maneuvers for both standard and nonstandard missions in real-time during the actual flight. In addition, once the decision has been made as to what maneuver should be performed, the MA group generates the proper spacecraft commands to effect these maneuvers. These commands are then relayed to the Spacecraft Performance Analysis and Command group to be included with other spacecraft commands. Once the command message has been generated, the MA group must verify that the calculated commands are correct.

(5) The Computer Support (CS) group acts in a service capacity to the other FPAC subgroups, and is responsible for ensuring that all computer programs used in space operations are fully checked out before mission operations begin and that optimum use is made of the Data Processing System facilities.

2. Spacecraft Performance Analysis and Command Group

The Spacecraft Performance Analysis and Command (SPAC) group, operating under the SPAC Director, is basically responsible for the operation of the spacecraft itself. The SPAC group is divided into four subgroups:

(1) The Performance Analysis (PA) group monitors incoming engineering data telemetered from the spacecraft, determines the status of the spacecraft, and maintains spacecraft status displays throughout the mission. The PA group also determines the results of all commands sent to the spacecraft. In the event of a failure aboard the spacecraft, as indicated by telemetry data, the PA group analyzes the cause and recommends appropriate nonstandard procedures.

(2) The Command Preparation and Control (CPC) group is basically responsible for preparing command sequences to be sent to the spacecraft. In so doing they provide inputs for computer programs used in generating the sequences, verify that the commands for the spacecraft have been correctly received at the DSS, and then ascertain that the commands have been correctly transmitted to the spacecraft. If nonstandard operations become necessary, the group also generates the required command sequences. The CPC group controls the actual transmission of commands at the DSS by the *Surveyor* operations chief.

(3) The Engineering Computer Program Operations (ECPO) group includes the operators for the DPS input/output (I/O) console and related card punch, card reader, page printers, and plotters in the spacecraft performance analysis area (SPAA). The ECPO group handles all computing functions for the rest of the SPAC group, including the maintenance of an up-to-date list of parameters for each program.

(4) The Trend and Failure Analysis (T&FA) group consists of spacecraft design and analysis specialists who provide in-depth, near-real-time spacecraft performance analysis (in contrast to the PA group's real-time analysis). The group also manages the interface for the SCCF computer facility at Hughes Aircraft Company. The SCCF 1219 is used mainly for premission spacecraft ground testing but, during the mission, the T&FA group is provided two data lines to the SCCF 1219 via the TPS, which will accommodate telemetry rates up

to 4400 bits/sec, and eight incoming lines terminating at seven teleprinters and one line printer in the SPAC area. The T&FA group uses the system to process and display engineering data transmitted from the spacecraft. The group also includes draftsmen who perform wall chart plotting and maintain wall displays of spacecraft condition and performance.

3. Space Science Analysis and Command Group

The Space Science Analysis and Command (SSAC) group performs those space flight functions related to the operation of the TV camera and science instruments. SSAC is divided into the following operating subgroups:

(1) The Television Performance Analysis and Command (TPAC) group analyzes the performance of the TV equipment and is responsible for generating standard and nonstandard command sequences for the survey TV camera.

(2) The Television Science Analysis and Command (TSAC) group analyzes and interprets the TV pictures for the purpose of ensuring that the mission objectives are being met. The TSAC group is under the direction of the Project Scientist and performs the scientific analysis and evaluation of the TV pictures.

(3) The Soil Mechanics Analysis and Command (SMAC) group prepares and recommends the commands to be sent to the soil mechanics/surface sampler instrument (SM/SS), and is also responsible for operating the SM/SS and analyzing its performance. This group was not utilized on the *Surveyor VI* mission because an SM/SS instrument was not included on the spacecraft.

(4) The Alpha Scattering Analysis and Command (ASAC) group prepares and recommends the commands to be sent to the alpha scattering instrument. This group is also responsible for conducting alpha scattering instrument operations during transit and lunar phases and for analyzing its performance.

The portion of the spacecraft TV Ground Data Handling System (TV-GDHS) in the SFOF provides direct support to the SSAC group in the form of processed electrical video signals and finished photographic prints. The TV-GDHS operates as a service organization within the MOS structure. Documentation, system checkout, and quality control within the system are the responsibility of the TV-GDHS Operations Manager. During operations support the TV-GDHS Operations Manager reports to the SSAC Director.

4. Data Processing Personnel

The use of the Data Processing System (DPS) by *Surveyor* is under the direction of the Assistant Space Flight Operations Director (ASFOD) for Computer Programming. His job is to direct the use of the DPS from the viewpoint of the MOS. He communicates directly with the Data Chief, who is in direct charge of DPS personnel and equipment. Included among these personnel are the I/O console operators throughout the SFOF, as well as the equipment operators in the DPS and Telemetry Processing Station (TPS) areas.

Computer programs are the means of selecting and combining the extensive data processing capabilities of electronic computers. By means of electronic data processing, the vast quantities of mission-produced data are assembled, identified, categorized, processed and displayed in the various areas of the SFOF where the data are used. Their most significant service to the MOS is providing knowledge in real-time of the current state of the spacecraft throughout the entire mission. This service is particularly important to engineers and scientists of the technical support groups since, by use of the computer programs, they can select, organize, compare and process current-status data urgently needed to form their time-critical recommendations to the SFOD. (See Section V-C-3 for a description of the DPS.)

5. Other Personnel

The Communications Project Engineer (PE) controls the operational communications personnel and equipment within the SFOF, as well as the DSN/GCS lines to the DSIF stations throughout the world.

The Support PE is responsible for ensuring the availability of all SFOF support functions, including air conditioning and electric power; for monitoring the display of *Surveyor* information on the Mission Status Board and throughout the facility; for directing the handling, distribution, and storage of data being derived from the mission; and for ensuring that only those personnel necessary for mission operations are allowed to enter the operational areas.

The DSIF Operations Planning PE is in overall control of all DSIF stations; his post of duty is in the SFOF in Pasadena. At each station, there is a local DSIF station manager, who is in charge of all aspects of his DSIF station and its operation during a mission. The *Surveyor* personnel located at each station report to the station manager.

B. Mission-Dependent Equipment

Mission-dependent equipment consists of special hardware provided exclusively for the *Surveyor* Project to support the Mission Operations System. Most of the equipment in this category is contained in the Command and Data Handling Consoles (CDC) and spacecraft Television Ground Data Handling System (TV-GDHS), which are described below.

1. Command and Data Handling Console

a. CDC equipment. The Command and Data Handling Console comprises that mission-dependent equipment located at the participating Deep Space Stations that is used to:

(1) Generate commands for control of the *Surveyor* spacecraft by modulation of the DSS transmitter.

(2) Process and display telemetered spacecraft data and relay telemetry signals to the on-site data processor (OSDP) for transmission to the SFOF.

(3) Process, display, and record television pictures taken by the spacecraft.

The CDC consists of four major subsystems:

(1) The command subsystem generates FM digital command signals from punched tape or manual inputs for the DSS transmitter, and prints a permanent record of the command sent. The major units of the command subsystem, which can accommodate 1024 different commands, are the command generator, the command subcarrier oscillator, the punched tape reader, and the command printer. Outgoing commands are logged on magnetic tape by the DSS and are relayed to the SFOF.

(2) The FM demodulator subsystem accepts the FM intermediate-frequency signal of the DSS receiver and derives from it a baseband signal. The baseband signal consists of either video data or engineering subcarrier signals. Depending upon the type of data constituting the baseband signal, the CDC processes the data in either the TV data subsystem or the telemetry data subsystem.

(3) The TV data subsystem receives video data from the FM demodulator and processes it for real-time display at the CDC and for 35-mm photographic recording. In addition, telemetered frame-identification data is displayed and photorecorded. A long-persistence-screen TV monitor is mounted in the CDC. The operator, when requested, can thus evaluate the picture and, upon the SFOD's direction, initiate corrective commands during lunar television surveys.

(4) The telemetry data subsystem of the CDC separates the various data channels from the baseband signal coming from either the FM demodulator or the DSS receiver phase-detected output and displays the desired data to the operators. Discriminators are provided for each subcarrier channel contained in the baseband signal. In the case of time-multiplexed data, the output of each discriminator is sent to the pulse code modulation (PCM) decommutator and then relayed to both the OSDP computer for subsequent transmission to the SFOF and to meters for evaluation of spacecraft performance. In the case of continuous data transmissions, the output of the discriminator is sent to an oscillograph for recording and evaluation. Alpha scattering experiment data is demodulated like other telemetry data, but is allowed to accumulate for periods of time in a SDS 920 computer in order to form spectrum information, which is then relayed to SFOF via teletype circuits.

The CDC contains built-in test equipment to insure normal operations of its subsystems. A CDC tester, consisting of a spacecraft transponder with the necessary modulation and demodulation equipment, insures day-to-day compatibility of the CDC and DSIF stations.

b. CDC operations. Table VI-1 lists the CDC mission-dependent equipment provided for support of *Surveyor VI* at the DSIF stations. CDC's were located at DSS 11, 42, 51, 61, 71, and 72 and, during the mission, CDC operations were conducted at each of these stations except DSS 72. Table VI-2 lists the total number of commands transmitted, number of TV frames commanded, and the minutes of alpha scattering data accumulated in the telemetry and command processor (TCP) at each station during the transit phase and through the first lunar day's activities. (Refer to Figs. V-9 and V-11 for station activity during each pass, including minutes of alpha scattering instrument data received and recorded. This data is not always accumulated in full in the station TCP.)

Table VI-1. CDC mission-dependent equipment support of *Surveyor VI* at DSIF stations

DSS 11, Goldstone	Prime station with command, telemetry, TV, and alpha scattering
DSS 42, Canberra	Prime station with command, telemetry, TV, and alpha scattering
DSS 51, Johannesburg	Prime station for telemetry during cislunar phase
DSS 61, Madrid	Prime station with command, telemetry, TV, and alpha scattering
DSS 71, Cape Kennedy	Station used for spacecraft compatibility tests and prelaunch and postlaunch telemetry data processing
DSS 14, Goldstone	Station configured for command backup and telemetry reception via both the DSS 11 CDC and SFOF; also used to record the terminal descent and landing phase and the postlanding spacecraft hop experiment

Table VI-2. *Surveyor VI* reported command, TV, and alpha scattering activity before shutdown during first lunar night

Station	Commands transmitted	TV frames commanded	Alpha scattering data accumulated in station TCP, min
DSS 11	113,967	23,665	336
DSS 42	30,613	4479	1165
DSS 51	108	0	0
DSS 61	18,205	1808	1241
DSS 71	0	0	0

Following is a brief summary of CDC operations at each station before spacecraft shutdown early in the first lunar night:

(1) DSS 11 Goldstone. The Pioneer station at Goldstone participated in 18 passes. Major mission events occurring during DSS 11 operations were midcourse maneuver, terminal descent and landing, an attitude control gas jet firing experiment, and a spacecraft hop experiment. This station's activity emphasized the commanding of TV pictures, although alpha scattering data was also accumulated and engineering interrogations were performed. Of the total number of TV frames obtained, 78% was commanded from this station, but only 13% of the total alpha scattering data was accumulated by this station. Several CDC equipment problems were experienced during the lunar operations, although none had any effect upon the mission. Problems were experienced with the TV monitor, the command printer, and an intermittent switch in the command generator.

(2) DSS 42 Canberra. Canberra participated in 19 passes. This station accomplished 16% of the TV commanding and accumulated 42% of the total alpha scattering data in the TCP. For approximately 21 min prior to DSS 42 acquisition on the first pass, the Carnarvon MSFN station received data from *Surveyor VI* and sent it via a data phone link to DSS 42. The CDC at DSS 42 then processed this data for real-time transmittal to the SFOF. During the mission several CDC operational and maintenance problems occurred; however, none had any effect upon the mission. Problems were experienced with the Polaroid camera, the command printer, the tape reader, and the low-frequency oscillograph. In addition, while operating in a three-way configuration with DSS 11, the DSS 42 command generator initiated a command without operator action. The command was not transmitted to the spacecraft, however.

(3) DSS 51 Johannesburg. DSS 51 participated in 3 passes during the transit phase and was not committed for use after touchdown. Initial acquisition was accomplished in an exceptionally smooth manner 30 min after liftoff, and two-way lock was obtained 5 min later. One operator error caused a minor sequence to be transmitted prematurely, and one equipment problem occurred with the command printer.

(4) DSS 61 Madrid. DSS 61 participated in 19 passes during this mission. Primary activity during the lunar phase consisted of engineering interrogations and alpha scattering data accumulations. Only 6% of the total TV activity was commanded, yet 45% of the total alpha scattering data was accumulated by this station. There were no CDC equipment problems during this mission, although a few operational problems did occur. At one time, the wrong axis of the solar panel was stepped. Another time, the CDC had trouble receiving a signal from the station receiver. It was determined that a switch was in an incorrect position. During one television

sequence, a wrong command tape was used and the sequence had to be repeated.

(5) DSS 71 Cape Kennedy. DSS 71 support for the *Surveyor VI* mission consisted of a DSIF/Spacecraft Compatibility Test, an Operational Readiness Test, the countdown phase, and approximately the first 40 min of the launch phase. During launch, this station processed the data being received from the various AFETR tracking stations and retransmitted the data to the SFOF. As a backup, the DSS 71 receiver tracked the spacecraft from liftoff for nearly 5 min, and the data obtained was available for transmittal to SFOF if necessary. There were no CDC equipment problems during this mission at DSS 71.

(6) DSS 14 Goldstone. DSS 14 provided backup for DSS 11 during transit, touchdown, and the spacecraft hop experiment. The output of this station's receiver is sent to the SFOF and to the DSS 11 CDC for processing at either location if necessary. The station's output was used during terminal descent and the spacecraft hop experiment. The station also provided backup for command transmission using the DSS 11 CDC and the intersite microwave link.

2. Spacecraft Television Ground Data Handling System

The spacecraft Television Ground Data Handling System (TV-GDHS) was designed to record, on film, the television images received from *Surveyor* spacecraft. The principal guiding criterion was photometric and photogrammetric accuracy with negligible loss of information. This system was also designed to provide display information for the conduct of mission operations, and the production of user products, such as archival negatives, prints, enlargements, duplicate negatives, and a digital tape of the TV ID information for use in production of the ID catalogs.

The system is in two parts, TV-11 at DSS 11, Goldstone, and TV-1 at the SFOF in Pasadena. At DSS 11 is an on-site data recovery (OSDR) subsystem and an on-site film recorder (OSFR) subsystem. These subsystems are duplicated in the media conversion data recovery (MCDR) subsystem, and in the media conversion film recorder (MCFR) subsystem at the SFOF. The portion of the system used in real-time at the SFOF is comprised of the MCDR, the MCFR, the media conversion computer, the video display and drive subsystem (VDDS), and the FR-700 and HW-7600 magnetic tape recorders. (The FR-1400 was available as backup.) A film processor, the strip contact printer, and the strip contact print processor are used in near-real-time. The photographic subsystem used in non-real-time is comprised of several enlargers, a copy camera, two film processors, a film chip file, other photographic equipment, and film storage areas.

a. TV-GDHS at DSS 11 (TV-11). Data for the TV-GDHS is injected into the system at the interface between the DSS 11 receiver and the OSDR. In both the 200-line and the 600-line modes, the OSDR provides to the film recorder subsystem: (1) the baseband video signal, (2) the horizontal sync signal, (3) the vertical frame gate, (4) the resynchronized raw ID telemetry information, and (5) the time code. In 200-line mode, the OSDR supplies a 500-kHz predetection signal to the DSS 11 FR-800 and FR-1400 magnetic tape recorders and to the SFOF via the microwave communication link. In 600-line mode, the OSDR provides: (1) a 4-MHz predetected signal to the DSS 11 FR-800 and to the SFOF via the microwave communication link, and (2) a baseband video signal to the DSS 11 FR-1400 magnetic tape recorder. In addition, the DSN provided an FR-900 recorder as backup to the FR-800 recorder.

The OSFR records the following on 70-mm film: the video image, the raw ID telemetry in bit form, the "human readable" time and record number, and an internally generated electric gray scale. The film is then sent to the SFOF for development.

b. TV systems at overseas DSIF stations. In addition to the TV systems at DSS 11, Stations 61, 42, and 51 provide some TV recording capability for *Surveyor* missions. As described in Section VI-B, Stations 61, 42, and 51 all have 35-mm film recording capability in the Command and Data Consoles. They also provide FR-1400 and FR-800 recordings of video data received. In addition, DSS 61 provided an FR-900 video recorder as a backup to the FR-800 recorder. The tapes and film from these stations are sent to TV-1 for processing and for evaluation.

c. TV-GDHS at the SFOF (TV-1). The signal presented to the microwave terminal at DSS 11 is transmitted to the SFOF, where it is distributed to the MCDR and to the VDDS. The MCDR processes the signal in the same manner as the OSDR. In addition, the MCDR passes the raw ID information to the media conversion computer, which converts the data to engineering units. This converted data is used by (1) the film recorder, where it is recorded as human readable ID, (2) the wall display board in the SSAC area, (3) the disc file where the film chip index file is kept, and (4) the history tape.

The VDDS produces the signals to drive the scan converter and the signals to drive the various display monitors and the paper camera in the SSAC area. Either the VDDS or the MCDR may be used to supply the signals recorded by the FR-700 and the HW-7600 video magnetic tape recorders in the same manner as the FR-800 and the FR-1400 recorders at DSS 11. The scan converter converts the slow scan information from the spacecraft to a standard RETMA television signal for use by the SFOF closed circuit television and the public TV broadcasting stations.

The MCFR records on two different films. Both films are wet-processed off line. One of the negatives is used to make strip contact prints which are delivered to the users. Additionally, this negative is used to make a master positive for the production of a duplicate negative for the JPL Public Information Office (PIO) and a preliminary duplicate negative for the science users. The negative is then cut into chips which are entered into the chip file where they are available for use in making additional contact prints and enlargements.

Prior to the *Surveyor V* mission, a number of improvements were made to the operational system in TV-1 permitting better service to the scientific community. The modifications made to equipment as well as continued personnel training programs allow the TV-GDHS to perform the necessary pretracking countdowns much more smoothly.

d. TV-GDHS performance. The TV-GDHS supported the entire mission as a well-designed and efficient system. All products delivered were of satisfactory quality. In general, problems that occurred on the previous mission did not reappear during the *Surveyor VI* mission. The film recorders were very reliable, and most problems could be corrected during nonoperational periods. Also, logistic support (mainly film supplies) was adequate to support the entire 30,000-frame effort, the only exception being strip contact paper, a rush purchase of which was required during the mission. A substitute paper was purchased and proved to be satisfactory.

During the lunar operations phase of the *Surveyor VI* mission, the TV-1 system operated comparatively smoothly. Problems were encountered in performing 200-line countdowns during the transit phase, which caused concern over signal-to-noise ratios related to data recovery. However, this presented no difficulty when the actual spacecraft signal was received. During recalibration of the scan converter from 200-line to 600-line mode, a major failure occurred in one of the scan converter's main cable harnesses. The system was down for the remainder of the first postlanding DSS 11 pass and the performance was less than optimum for the next two passes. This problem had no effect on the quality of data transferred to film. During peak video activity periods, difficulty developed in providing adequate maintenance on nonredundant photo processing equipment. This was especially true for the strip contact printer, which at times was being used on a 100% duty cycle in support of real-time photo products. (This problem should be alleviated by additional identity copier support on the next mission.) Photo products, required in real-time by the science groups, were at no time late due to failures or malfunctions within TV-1. The long uninterrupted series of pictures, which resulted from the extensive TV operations, and the necessity of maintaining high quality control did cause some delay in the delivery of photo products.

A total of 31,518 frames were recovered by TV-1 via the DSS 11 microwave link or from FR-800 tapes from overseas stations. Of these, 30,065 frames were unique, the remainder being duplicate frames obtained during periods of station overlap. From the total number of frames processed, 558 rolls of strip contact prints, averaging 125 ft in length, were delivered. In addition, 93 rolls each of the master positives, PIO negatives, and experimenter data record (EDR) negatives, which also averaged 125 ft in length, were delivered. In addition to the above products, TV-1 delivered 2264 mosaic copies and 3719 photo enlargements during the period of November 10 to November 27, 1967.

The data recovery system of TV-1 evaluated 138 FR-800 tapes received from the DSN. These evaluations included checks to provide feedback to the DSIF so that recording quality could be maintained. Every effort was made to evaluate all tapes within 24 hr of their receipt.

Owing to a problem that developed in TV-11's film recorder, TV-1 was unable to process the film received from TV-11 using the standard process. Special handling and processing were required to guarantee the highest possible quality of the TV-11 film, but this did not permit the normal feedback of information to TV-11 during mission operations. (New procedures have been developed both in TV-1 and TV-11, which should allow developing and evaluation of TV-11 film in near-real-time so that the desired quality control information can be made available to TV-11 if this problem should occur again.) The TV-GDHS design and reliability are continually being improved, and improved performance should be demonstrated on the next mission.

C. Mission Operations Chronology

Inasmuch as mission operations functions were carried out on an essentially continuous basis throughout the *Surveyor VI* mission, only the more significant and special, or nonstandard, operations are described here. Refer also to the sequence of mission events presented in Table A-1 of Appendix A.

1. Countdown and Launch Phase

No significant problems were reported in the early phases of the countdown, which proceeded normally down to the built-in hold at $T - 5$ min. As decided earlier in the countdown, the $T - 5$ min hold was extended beyond the originally planned 10 min in order to delay launch window opening and thus improve downrange spacecraft telemetry coverage. Following an additional 17-min hold, the countdown was resumed.

At 07:33 GMT ($L - 6$ min), the FPAC Tracking Data Analysis (TDA) group started a computer run of $T - 5$-min predictions for initial DSIF acquisition, using computed center frequencies for Transmitter B and Transponder B supplied by SPAC, the actual launch azimuth of 82.995 deg, and station parameters. These $T - 5$-min predictions were completed at 07:37 GMT and sent to DSS 51 (Johannesburg) and DSS 42 (Tidbinbilla).

Liftoff ($L + 0$) occurred at 07:39:01.075 GMT on November 7, 1967, with all systems reported in a "go" condition. Launch vehicle performance appeared to be nominal, with no significant anomalies on either the *Atlas* or *Centaur*. Following the parking orbit coast period, *Centaur* second burn occurred and the spacecraft was injected into a lunar transfer trajectory which was well within established limits. The required retro maneuver was successfully performed by the *Centaur*. A description of launch vehicle performance and sequence of events from launch through injection is contained in Section III.

Following separation at 08:04:30 GMT ($L + 00$:25:29), the spacecraft executed the planned automatic sequences as follows. By using its cold-gas jets, which were enabled at separation, the flight control subsystem nulled out the small rotational rates imparted by the separation springs and initiated a roll-yaw sequence to acquire the sun. After a minus roll of 264 deg and a plus yaw of 22 deg, acquisition and lock-on to the sun by the spacecraft sun sensors were completed at 08:14:50 GMT. Concurrently with the sun acquisition sequence, the A/SPP stepping sequence was initiated to deploy the solar panel and position the roll axis. At 08:10:07 and 08:14:07 GMT, the solar panel and A/SPP roll axis, respectively, were locked in the proper transit positions. All of these operations were confirmed in real time from the spacecraft telemetry. Following sun lock-on, the spacecraft coasted with its pitch and yaw attitude controlled by tracking the sun and with its roll attitude held inertially fixed.

2. DSIF and Canopus Acquisition Phase

Both DSS 51 and DSS 42 received the $T - 5$-min predictions by $L + 5$ min, at least 25 min prior to spacecraft rise over the DSS 51 horizon. Using this data, DSS 51 achieved one-way lock with the down-link signal at 08:08:55 GMT, 30 min after liftoff and 69 sec before the predicted rise time at the station. At 08:13:25 GMT, the spacecraft acquired the up-link. DSS 51 received good two-way data at 08:14:15 GMT ($L + 00$:35:14), and initial DSIF acquisition was completed.

The first ground-controlled sequence ("Initial Spacecraft Operations") was initiated at 08:19 GMT ($L + 00$:40) and consisted of commands for (1) turning off spacecraft equipment required only until DSIF acquisition, such as transmitter high power and the solar panel deployment logic, (2) seating the solar panel and roll axis locking pins securely, (3) interrogating commutator Mode 1* at the 550-bit/sec data rate established prior to launch, (4) increasing the telemetry sampling rate to 1100 bits/sec, and (5) performing interrogation of commutator Modes 4, 2, 6, and 5, in that order. All spacecraft responses to the commands were normal.

Initial spacecraft operations were completed at 08:38 GMT, with *Surveyor VI* configured in low power, coast phase commutator on (Mode 5), and transmitting at 1100 bits/sec over Omniantenna B. Since the star error signal was fluctuating and the star intensity signal was appreciably above the no-signal level, indicating the presence of the earth in the Canopus sensor's field of view, it was recommended that the roll axis be held in

*See Section IV-A (Table IV-1) and Appendix C for data content of telemetry modes.

the inertial mode and the *cruise mode on* command not be sent to the spacecraft. (Cruise mode was established 8 hr later at the completion of the star verification/acquisition sequence.) It was also determined that there was no need to turn on Transponder A since Receiver A was already tracking the DSS 51 transmitter signal in the automatic-frequency-control (AFC) mode.

The spacecraft continued to coast normally, with its pitch–yaw attitude controlled by tracking the sun and with its roll attitude held inertially fixed. Telemetry and two-way tracking data were obtained by use of Transponder B and Transmitter B. (The spacecraft telemetry bit rate/mode profile for the complete transit portion of the mission is given in Fig. VI-2.)

The first SFOF estimate of spacecraft orbit, computed at L + 01:56, by the Orbit Determination (OD) group, was based on 22 min of DSS 51 two-way doppler and angle data and 2 min of angle data from DSS 42. The orbit solution indicated that lunar encounter would be achieved without midcourse correction. It further indicated that the correction required to land *Surveyor VI* at the prelaunch aim point was well within the nominal midcourse correction capability. These results were verified by the OD group following completion of their second and third orbit computations at L + 02:43 and L + 03:34, respectively. As additional tracking data became available, orbit solutions were periodically computed and refined.

The preliminary midcourse conference was held at 15:15 GMT (L + 07:36), shortly before the Canopus acquisition phase of the mission. Spacecraft data was provided by SPAC which showed that all subsystems were in good condition and operating properly. FPAC presented the results of computer runs to this time. Analysis of the *Surveyor VI* trajectory indicated that a midcourse correction at about L + 20:00 would be the most desirable.

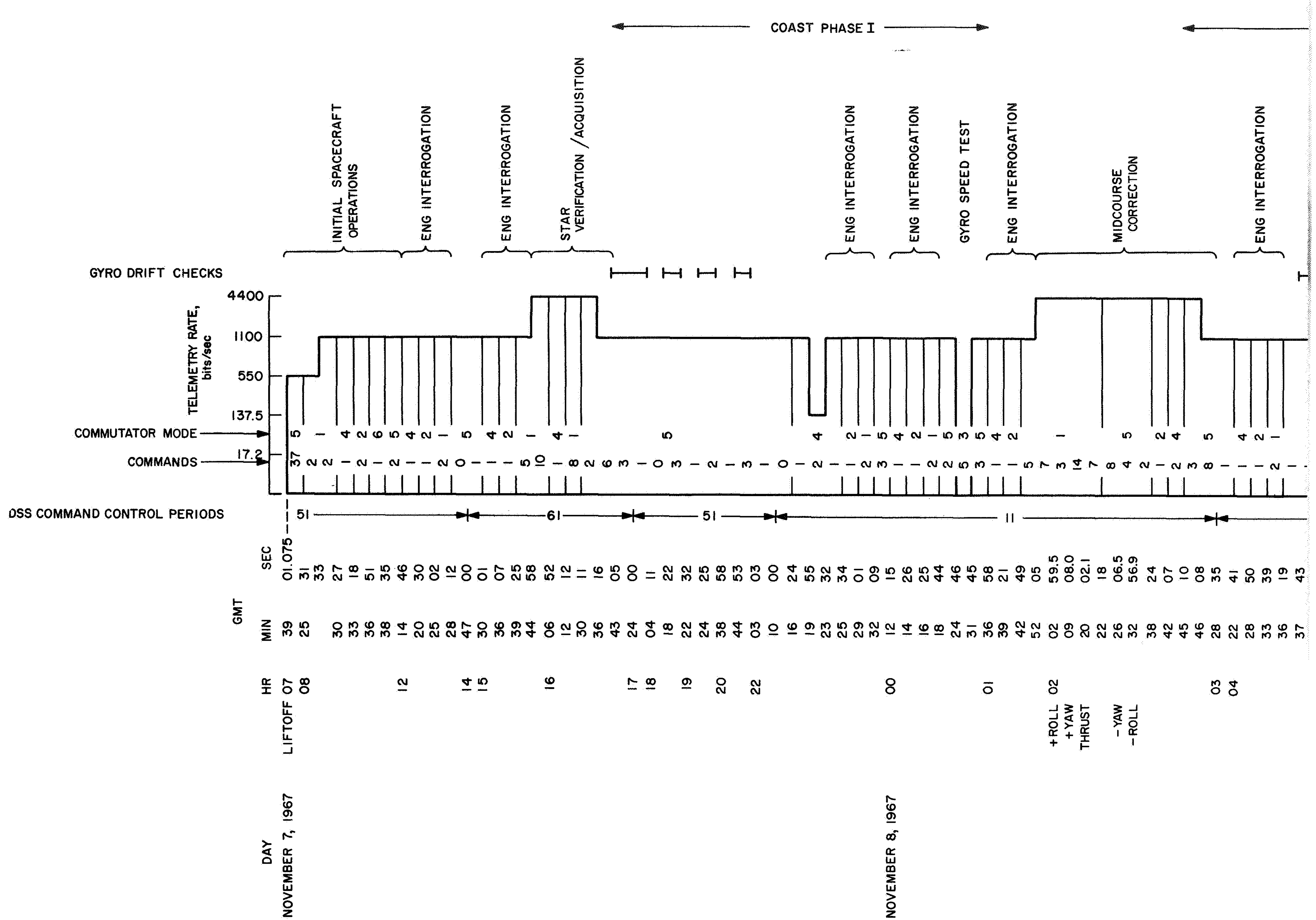
In preparation for the roll maneuver which initiates the star verification/acquisition sequence, commutator Mode 1 was selected, Transmitter B was commanded to high power, the transponder was left on, and the data rate was increased from 1100 to 4400 bits/sec. The first revolution of the sequence provides a measure of light intensity received from celestial bodies passing through the Canopus sensor's field of view. The data is used for comparison with a precomputed star map for positive identification of the star Canopus, which is usually acquired during the second revolution. Premaneuver analysis indicated that the earth vector would pass through the deep null region of both the up- and down-links of Omniantennas A and B. It was also determined that at the 4400-bit/sec data rate it would be necessary to switch antennas during the maneuver in order to map all expected celestial bodies in one revolution. The SPAC Performance Analysis (PA) group therefore recommended using Omniantenna A for the down-link during the first 180 deg of roll and Omniantenna B for the down-link for the remainder of the sequence.

Omniantenna A was selected for the down-link, and at 15:50:22 GMT (L + 08:11:21), DSS 61 (Robledo) commanded the positive spacecraft roll maneuver. During the first 135 deg of roll, the earth and the star Deneb were detected by the Canopus sensor. After the spacecraft had rolled about 151 deg, a deeper-than-expected null (apparently greater than −50 db) in the Omniantenna B pattern caused Receiver B to lose phase lock with the DSS 61 transmitter. As a result, Transmitter B automatically switched from the transponder mode to the narrow-band VCXO mode of frequency control, and the associated frequency shift in turn caused the DSS 61 receivers to lose lock with the down-link. About a minute later, at 15:56:39 GMT, the command was sent to select Omniantenna B for the down-link as planned since the spacecraft had completed over 180 deg of roll. At 15:58:44 GMT, the roll maneuver was temporarily halted. An intermittent down-link signal observed in the DSS 61 receiver indicated that the spacecraft transponder was apparently going in and out of phase lock with a sideband caused by the presence of command modulation in the up-link signal. It was therefore recommended that the transponder be turned off and that the star verification/acquisition sequence be completed with the spacecraft configured in the one-way AFC mode to assure a steady signal.

The transponder was commanded off at 16:04:39 GMT, and solid ground receiver lock was established. Following a 5-min interrogation of commutator Mode 4 to check the telecommunications subsystem, the spacecraft was returned to commutator Mode 1 and commanded to resume the roll maneuver at 16:14:22 GMT. The maneuver proceeded smoothly as Canopus and the earth were detected by the Canopus sensor and identified on the map in the SPAC area. After the earth passed from the field of view, the star acquisition mode was commanded on. The spacecraft continued to roll, and when light from Canopus was sensed, *Surveyor VI* ended the

FOLDOUT FRAME 1

Foldo:



ᴊᴛ FRAME 2

FOLDOUT FRAME 3

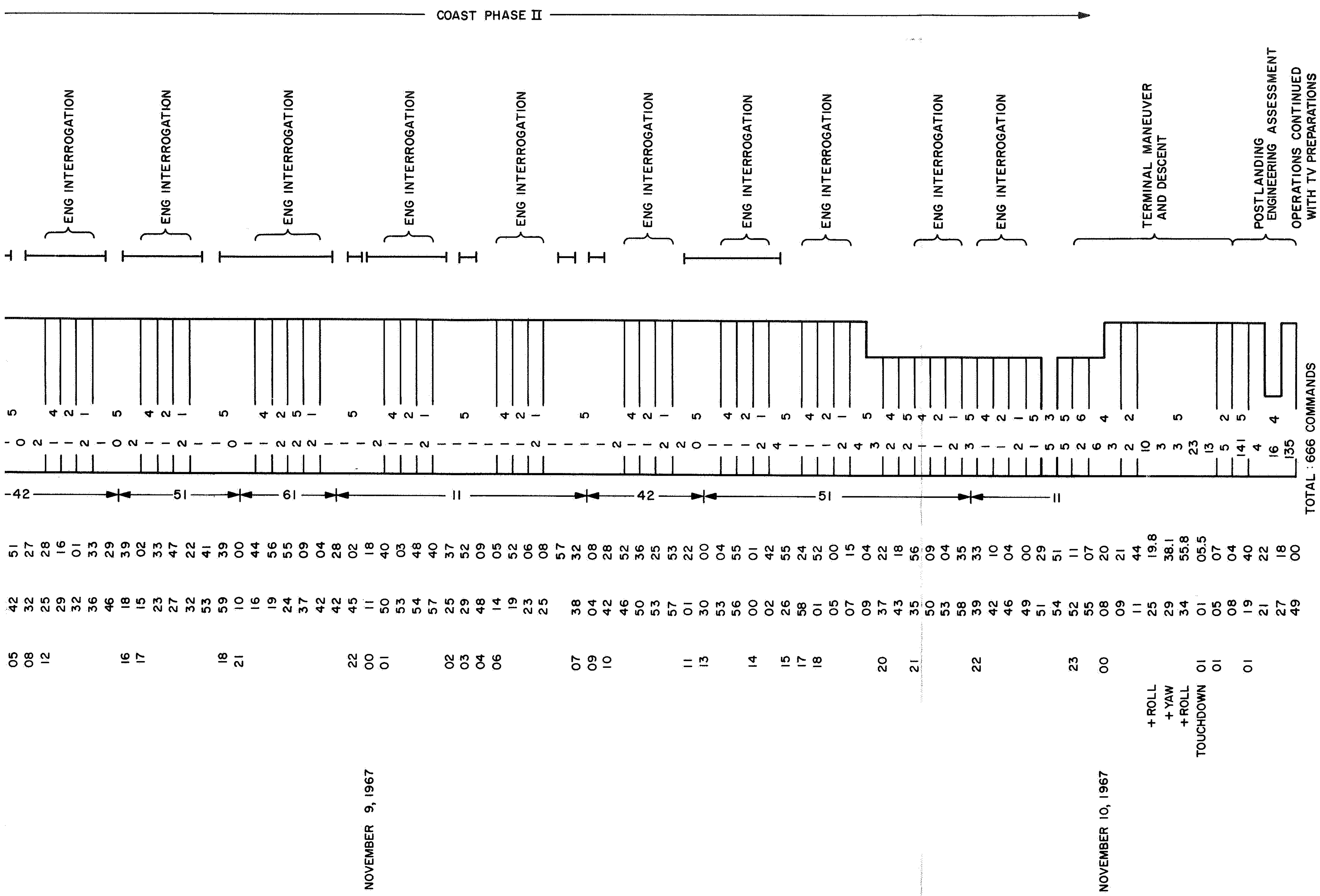


Fig. VI-2. *Surveyor VI* telemetry bit rate/mode profile during transit

maneuver by automatically locking onto the star. (In both the *Surveyor I* and *II* missions, the Canopus intensity was above the upper threshold of lock-on range, necessitating manual acquisition; *Surveyors III, IV*, and *V* locked on automatically). Canopus acquisition occurred at 16:27:49 GMT (L + 08:48:48) after the spacecraft had rolled through about 650 deg for nearly 22 min.

At the end of the star verification/acquisition sequence, commands were sent to the spacecraft to establish cruise mode of flight control, select commutator Mode 5, and turn on the transponder. After good two-way lock was confirmed at 16:33:44 GMT, the data rate was lowered to 1100 bits/sec and Transmitter B was returned to low power. The spacecraft was coasting with pitch and yaw attitudes controlled by tracking the sun and the roll attitude controlled by tracking Canopus. *Surveyor VI* was thus in a fixed attitude, a precise position from which midcourse maneuvers could be computed and executed.

3. Pre-midcourse Coast Phase

About 15 min after Canopus lock-on, the first gyro drift check was initiated by commanding the spacecraft to inertial mode. The vehicle continued to coast as before, but with its attitude held inertially so that the sun and star sensors continued to point at the sun and Canopus, respectively. During this period, the solar panel switch tripped off to protect the battery since battery voltage had built up to over 27 V. As in the *Surveyor V* mission, this occurred several times during the flight and had been predicted as a result of premission solar-thermal-vacuum tests. After the battery was allowed to discharge for a few minutes, the switch was commanded on. At 18:04:11 GMT (L + 10:25:10), the gyro drift check was terminated by commanding on cruise mode; 14 min later a second check was begun. Two other gyro drift checks were commanded during the pre-midcourse coast phase, one at 19:24:25 GMT and one at 20:44:53 GMT. All four drift checks performed during this period measured drift rate in the spacecraft's roll, pitch, and yaw axes. Just prior to termination of the first gyro drift check, Canopus passed from the field of view of the star sensor due to the magnitude and direction of yaw axis drift, which was about +1.2 deg/hr. The star was reacquired after termination of the gyro drift check.

At the final midcourse conference, held at 22:00 GMT, the decision was made to perform midcourse correction at 02:20 GMT (L + 18:41) on November 8. The following midcourse parameters were established:

(1) Use an aiming point of 1.133 deg W longitude and 0.417 deg N latitude.

(2) Maneuver in roll and yaw to position the spacecraft for the velocity correction.

(3) Fire the vernier engines for approximately 10 sec to obtain the desired velocity increment.

Following the meeting, final computation of midcourse parameters was started by FPAC. (Refer to Section VII for a discussion of the factors considered in selecting the midcourse correction plan.)

At 22:16:24 GMT, commutator Mode 4 was selected and the data rate was lowered to 137.5 bits/sec in order to obtain a more accurate measurement of alpha scattering instrument temperatures. The data rate was then returned to 1100 bits/sec, and a standard engineering interrogation was conducted which confirmed that *Surveyor VI* was in good condition. Pre-midcourse interrogation began at 00:12:15 GMT (November 8). The pre-midcourse gyro speed check was then initiated, and the gyro spin rates in roll, pitch, and yaw axes were found to be nominal at 50 Hz.

The final midcourse guidance summary computer run was completed at 00:13 GMT by FPAC and delivered to SPAC's Command Preparation and Control (CPC) group, who immediately began generation of the final midcourse command tape message. The message was sent to DSS 11 at 01:10 GMT (L + 17:31) and verified shortly thereafter.

4. Midcourse Maneuver Phase

The midcourse correction sequence was initiated at 01:36:58 GMT (L + 17:57:57) with an engineering interrogation in commutator Modes 4, 2, and 1. Telemetry readings indicated that the spacecraft was in satisfactory condition for midcourse operations. The interrogation was followed by commands to turn on Transmitter B high power and increase the telemetry sampling rate from 1100 to 4400 bits/sec. Starting at 02:03:00 GMT (L + 18:23:59), the required roll (+91.8 deg) and yaw (+127.8 deg) maneuvers were executed by the spacecraft to align the axes in the desired direction for applying midcourse thrust. The attitude maneuvers were initiated at limit cycle nulls to reduce pointing errors. The spacecraft was prepared for thrusting, including sending commands to (1) turn on propulsion strain gage power, (2) turn on inertial mode, (3) turn off cyclic loads and pressurize the vernier propulsion system, and (4) load the desired thrust time (10.25 sec) in the flight control programmer magnitude register. Midcourse velocity correction was executed at 02:20:02 GMT (L + 18:41:01) on November 8 in accordance with the planned thrust time. All engines fired simultaneously, burned for 10.24 sec for a velocity change of 10.12 m/sec, then shut down automatically. The spacecraft remained stable throughout the thrusting period.

Following midcourse velocity correction, commands were sent to turn off various power sources used during the thrusting phase, to select commutator Mode 5, and to execute attitude maneuvers in reverse order and opposite (negative) direction to reestablish cruise configuration. The sun and Canopus were reacquired automatically at the end of the yaw and roll maneuvers, confirming that the gyros had retained inertial reference during vernier engine shutdown and eliminating the need for post-midcourse star verification to insure lock-on to Canopus.

The spacecraft was returned to flight control cruise mode, and a high-power engineering interrogation was conducted in commutator Modes 2, 4, and 5 after which the 1100-bits/sec data rate and low-power transmission were established.

5. Post-midcourse Coast Phase

During *Surveyor V* terminal descent, degradation of the 1100-bits/sec data occurred at DSS 14 (Goldstone, Mars site) when the spacecraft's touchdown strain gage subcarrier oscillators (SCOs) were multiplexed with the 7.35-kHz pulse-code-modulated (PCM) SCO. DSS 14 had not been used as a prime source of real-time data until the *Surveyor V* mission. After the *Surveyor V* mission, the DSS 14 receivers were modified to a narrower tracking-loop bandwidth. Subsequent testing indicated that data degradation should not occur with the *Surveyor VI* data system in the landing configuration.

To verify the analytical and premission test results, a special test was begun at 03:12:28 GMT by commanding the *Surveyor VI* touchdown strain gage power and data channels on; the touchdown strain gage SCOs were multiplexed with the 7.35-kHz SCO (1100-bits/sec data). While DSS 11 continued to count the bit error rate, DSS 14 moved the 210-ft antenna to point slightly off the spacecraft/station vector, thereby reducing the received signal level and obtaining a PCM threshold equivalent to that at landing. Data was monitored for about 12 min at both stations; then the DSS 14 antenna was returned to its original position, touchdown strain gage power and data channels were commanded off (propulsion strain gage power was commanded off also), and the test was terminated at 03:28:35 GMT. It was determined by visual comparison of the two data streams during the test that the PCM data at DSS 14 and DSS 11 were of about equal quality. The in-flight and earlier simulation test results were also in agreement. It was therefore concluded that, with the spacecraft in the 1100-bits/sec and touchdown strain gage data multiplex mode during terminal descent and touchdown, nominal theoretical PCM threshold performance at DSS 14 could be expected using the narrower 152-Hz bandwidth.

The second coast phase was relatively uneventful. Ten gyro drift checks were performed: seven 3-axis checks and three checks of the roll axis only. The last gyro drift check of the mission (14 total) was terminated at 15:26:55 GMT, November 9, and the accumulated error was nulled. Commutator Modes 4, 2, and 1 were interrogated four times on November 8 and five times on November 9 at the 1100-bits/sec data rate, which was used from the end of midcourse phase until 18:09:04 GMT (November 9) when the rate was lowered to 550 bits/sec. The survey camera electronics temperature control and the Compartment C heater were turned on at 19:55:39 and 20:25:07 GMT, respectively. The Compartment A heater and thermal control power to vernier Fuel Tank 2 and vernier Oxidizer Tanks 2 and 3 had been turned on several hours earlier. Engineering interrogation of commutator Mode 4 was performed at 20:37:22 GMT, followed in an hour by interrogation of Modes 4, 2, and 1. The spacecraft was always returned to commutator Mode 5, the coast phase commutator, after interrogations during the coast period.

Prior to the terminal maneuver conference, a preliminary internal mission operations conference was held to generate a list of alternate maneuver plans and an outline of postlanding operations for presentation to the Mission Director. The final maneuver plan presented to and approved by the Mission Director contained the following items:

(1) Turn off the transponder prior to start of maneuvers.

(2) Use a roll–yaw–roll sequence with Omniantenna B to optimize telecommunications performance.

(3) Initiate the first of the three maneuvers 38 min prior to predicted time of retro ignition.

(4) Transmit 1100-bits/sec PCM data.

(5) Attempt to obtain strain gage data only if the received signal level is at least −128 dbm after terminal maneuvers are completed.

By 21:27 GMT (L + 61:48), SPAC had evaluated all gyro drift data and had provided drift rate estimates and predicted retro motor burn time to FPAC for use in terminal descent computations. FPAC's final terminal guidance computer run was completed at 22:36 GMT. Predicted times for automatic descent events, the delay time between *AMR mark* and vernier engine ignition, and the desired time for the AMR backup command were then delivered to the PA group, which started preparing the terminal descent command message. The final message was sent to DSS 11 and verified at 23:33 GMT (L + 63:54), about 1.5 hr before predicted retro ignition.

Heater power to the alpha scattering instrument sensor head was turned on at 22:27:27 GMT. Twelve minutes later the preterminal engineering interrogation of Modes 4, 2, and 1 was started and was followed by the gyro speed check. The VCXO check was then performed by commanding off transponder power for 1.5 min to permit determination of the frequency offset required for terminal descent. The Compartment A heater was turned off at 23:13:14 GMT. *Surveyor VI* was in good condition and ready for the terminal phase.

6. Terminal Phase

About an hour before retro ignition, Mode 6 (thrust phase commutator) interrogation was started. Then Mode 4 was selected, Transmitter B was commanded to high power, and the telemetry rate was increased from 550 to 1100 bits/sec at 00:08:20 GMT. After an interrogation in Mode 2, Mode 5 was selected. Power was turned on as required for monitoring vernier propulsion and touchdown forces. The transponder was then turned off to establish the one-way tracking mode for the terminal maneuvers.

Maneuvers were executed starting at 00:25:20 GMT (L + 64:46:19). For the first maneuver, the spacecraft was commanded to roll +81.7 deg. This was followed by commands to yaw (+111.7 deg) and to roll again (+120.5 deg). The maneuvers, which oriented the retro thrust axis in the proper direction, also resulted in optimum roll positioning (within RADVS and telecommunications constraints) for postlanding operations. The maneuvers were compensated for flight control sensor group deflections and gyro drift rates; they were also successfully initiated on limit cycle nulls to effectively reduce errors in the thrust vector direction.

In the remaining time before the automatic retro sequence, final preparations were made for terminal descent. The thrust level of the vernier engines was set at 200 lb, a delay time of 5.875 sec between the *AMR mark* event and ignition of the vernier engines was stored in the spacecraft, and commutator Mode 6 was selected. Retro sequence mode was turned on, various cyclic loads turned off, and AMR power and thrust phase power turned on. The AMR was enabled upon command at 00:56:17 GMT (L + 65:17:16), 1 min and 45 sec before main retro ignition. With the exception of the standard AMR backup command, all remaining descent events were performed automatically by the spacecraft.*

At 00:57:55.74 GMT (L + 65:18:54.66), when the spacecraft was 59.8 miles (slant range) from the moon, the *AMR mark* signal was generated by *Surveyor VI*. The AMR backup command arrived at the spacecraft 2.04 sec later. The vernier engines ignited at 00:58:01.64 GMT; 1.1 sec later the main retro motor ignited, ejecting the AMR. RADVS power came on automatically as planned, followed by the RADVS *high voltage on* signal. A few seconds later, the doppler velocity sensor beams acquired the lunar surface. The spacecraft was stable and operating properly.

Main retro motor burnout occurred at 00:58:42.35 GMT as indicated by 3.5-g switch closure. Vernier engine thrust increased to high level, and the retro case was ejected. RADVS-controlled descent began at 00:58:56.44 GMT; a few seconds later the radar altimeter beam acquired the lunar surface. With steering under control of the three doppler velocity sensor beams and altitude monitoring under control of the radar altimeter, *Surveyor VI* continued to descend, intercepting the programmed descent segment at 00:59:20 GMT. At this point, vernier thrust level increased, after which the spacecraft followed the descent contour until it reached 10-ft/sec velocity at an altitude of 50 ft. Vernier thrust level was then controlled to produce 5-ft/sec constant velocity. When the spacecraft's footpads reached an altitude of 14 ft above the moon, the vernier engines cut off, and *Surveyor VI* landed gently in Sinus Medii at 01:01:04.170 GMT (L + 65:22:03.095) on November 10, 1967.

*Times given for automatic descent events have been corrected for 1.297-sec radio transmission delay to show time of occurrence at spacecraft. Refer to Section IV-A for a more complete description of the terminal descent sequence.

7. First Lunar Day

The first quick review of the touchdown strain gage data received at the SFOF indicated that all three legs of *Surveyor VI* touched down almost simultaneously, the order of impact being Legs 1, 2, and 3 respectively. Peak landing loads in the order of impact were 1570, 1170, and 1580 lb, respectively, indicating a nominal touchdown velocity of approximately 12 ft/sec. (See Table IV-8 for final data.) The differential times between leg impacts indicated that *Surveyor VI* had landed on a nearly flat surface, and preliminary analysis of the touchdown gyro data indicated a tilt from the vertical between 2 and 3 deg.

(Based on final postflight tracking data, the estimated landing coordinates are 0.437°N latitude and 1.373°W longitude, approximately 7.2 km from the in-flight aiming point. Correlation of the terrain features pictured by *Surveyor VI* with earlier *Lunar Orbiter* photographs indicates the landed location to be 0.470°N latitude and 1.480°W longitude. Final *Surveyor VI* science results are given in Part II of the Mission Report and are repeated here only in a mission operations relationship.)

SPAC proceeded quickly to the postlanding power shutdown and assessment of essential spacecraft systems. In the next 5 min, thrust phase power, RADVS power, and all flight control power were commanded off; an interrogation of Modes 2 and 5 was initiated, and all strain gage power was commanded off. SPAC reported that the spacecraft was responding as commanded and that an estimated 107 A/hr remained in the battery. In preparation for the planned vernier static firing and/or spacecraft hop experiment, commands normally sent at this time to lock the landing legs and dump the helium pressure supply for the vernier propulsion system were not sent. Heaters for the vernier system, alpha scattering instrument, and TV system were turned on to protect against the extreme cold during the first hours of the lunar morning. After a preliminary checkout of the antenna and solar panel positioner (A/SPP), radio, signal processor, and TV camera controls, SPAC was able to confirm that these essential systems were indeed operating normally. By approximately 01:30 GMT, the spacecraft was configured for initial television operations in 200-line mode, and at 01:50 GMT, 49 min after touchdown, the first TV picture was commanded and successfully received. The initial sequence consisted of a wide-angle survey beginning with Footpad 3, then up and across the local horizon and down to Footpad 2. The sequence was designed to quickly determine if any gross structural damage had been sustained, and to provide information on the lunar location and landed attitude of the spacecraft. Between 01:50 and 02:35 GMT, a total of 24 good-quality pictures were received, revealing that *Surveyor VI* was structurally sound and had landed with close-to-nominal orientation in a relatively smooth area of Sinus Medii near a fairly prominent ridge.

Assured of a healthy spacecraft, SPAC proceeded to reconfigure for 600-line, high-resolution television operations, which require the more lengthy A/SPP stepping operations to roughly align the high-gain planar array antenna and the solar panel with the earth and the sun. This initial, or gross, positioning A/SPP sequence provides the increased signal strength for 600-line-mode TV and the necessary power via the solar panel for continuing lunar operations.

The sequence involves stepping the A/SPP to predicted sun and earth positions, which are calculated for a vertical spacecraft that has retained the roll orientation established during the terminal maneuver. The greater the deviation from the predicted spacecraft attitude and orientation, the more time required to accomplish gross positioning. During the *Surveyor V* mission, the gross positioning sequence required several hours to complete because the spacecraft had landed on the edge of a crater wall, twisting slightly in the process, and had come to rest at a 19.5-deg tilt. The time consumed in locating the earth and sun left only 10 min for high-resolution TV operations before the end of the Goldstone postlanding view period and the end of the real-time video display. The real-time display capability at the SFOF via the TV-GDHS is available only during the Goldstone view and is especially important since it permits science and television performance groups to evaluate the lunar subject in real-time and control camera performance accordingly. The results from these initial sequences are then used to direct the continuation of TV operations after transfer to DSS 42.

Owing to its favorable landing, *Surveyor VI* gross positioning was completed in a nominal 47 min, with the sun and earth acquired very close to their predicted positions. With nearly 3 hr of DSS 11 visibility remaining, the first 600-line, high-resolution picture sequence was initiated at 04:02 GMT, with excellent results. Activities for the remainder of the view period consisted of a wide-angle, 360-deg panorama, a special area survey, and engineering interrogations. The spacecraft attitude determined from this first A/SPP data indicated a tilt from the vertical of approximately 2 deg and a downhill slope of approximately 0.8 deg, and agreed favorably with the initial gross estimate based on touchdown gyro error and strain gage data.

At 05:39 GMT, television operations were momentarily halted and the spacecraft was reconfigured for alpha scattering operations. In its stowed position, the instrument was commanded on for initial calibration and the start of standard-sample data accumulations. (Refer to Section IV-J for a description of the alpha scattering instrument.) The incoming data received by teletype confirmed that the instrument was operating properly. The instrument temperatures were now safely within operating limits and the Compartment C and sensor head heaters were commanded off. After 40 min of initial operation, the instrument was turned off at 06:20 GMT and TV operations were resumed to take advantage of the desirable light and shadow effects created by the low sun angle during the early lunar morning.

At 07:20 GMT, control of the spacecraft was transferred to DSS 42. TV operations were then conducted without benefit of the real-time video display at the SFOF. As pictures were received at the Australian station, the *Surveyor* Operations Chief (SOC) at DSS 42 described the results over the voice line to the Television Science (TSAC) and Television Performance (TPAC) groups in the SFOF. With the DSS 11 pictures available for reference, the TSAC and TPAC groups monitored the voice description of the DSS 42 picture sequences and recommended camera changes as necessary.

Following completion of the initial postlanding TV sequence at 11:40 GMT, the spacecraft was reconfigured for alpha scattering operations and standard-sample data accumulations were resumed at 11:54 GMT. The A/SPP "fine positioning" sequence was conducted concurrently during this period to provide data for a more accurate determination of the spacecraft attitude. With the spacecraft attitude and orientation precisely determined, future positioning of the planar array and solar panel relative to the earth and sun can be calculated beforehand and performed more efficiently. The sequence required approximately 90 min to complete. Based on A/SPP fine-positioning data, the spacecraft tilt was determined to be less than 1 deg toward lunar south. The spacecraft $+X$ axis was located 24.96 deg clockwise from the tilt direction, placing Leg 1 roughly northwest, Leg 2 almost due south, and Leg 3 roughly northeast. (See Fig. IV-9 for an illustration of the landed attitude.)

The development of *Surveyor VI* lunar operations was moderately influenced by attempts to revive *Surveyor V* on its third lunar day. A comprehensive plan for activating *Surveyor V* and operating both spacecraft simultaneously, using a separate *Surveyor V* operations team, had been developed by Mission Control. The first revival attempt was initiated at 14:00 GMT, on November 10. With DSS 42 commanding, the *Surveyor VI* Transponder B was turned on, phase-locked, and pulled 50 kHz from its center frequency to permit the DSS 51 ground receiver to sweep the predicted *Surveyor V* center frequency without interfering with *Surveyor VI* operations. Various combinations of frequency tuning rates and turn-on command sequences were conducted during the next 3 hr without success. Two attempts were made by DSS 51 on the following day, the first with *Surveyor VI* controlled by DSS 42 and the second with *Surveyor VI* controlled by DSS 61. During the second attempt, the *Surveyor V* search frequency was swept too close to *Surveyor VI*, causing DSS 61 to lose lock with the spacecraft. During the next 45 min, DSS 61 proceeded to reestablish two-way lock but encountered some difficulty in the process. Assuming possible interference between the two spacecraft, commands were sent to place both spacecraft in full standby, followed by a sequence to bring up only *Surveyor VI*. Two-way lock was again reestablished and no further problems were encountered.

Additional *Surveyor V* revivals were attempted on November 12, 13, 14, and 19, but with *Surveyor VI* placed in full standby to allow more latitude in the search for *Surveyor V*. There were no indications, however, of any response from *Surveyor V*. Considerable effort was expended on these revival attempts since two-way doppler tracking with two spacecraft would have made it possible to obtain new information about the motions of the moon.

At 20:55 GMT, during the first DSS 61 postlanding view period, the alpha scattering instrument completed the final standard-sample data accumulation, and preparations were made to release the instrument sensor head to its "background" position to detect and calibrate natural background radiation before deployment to the lunar surface. A command tape was used to deploy the instrument to a position approximately 22 in. above the lunar surface while the operation was monitored on television. A total of 45 TV frames were received, providing a stop-motion sequence in which the instrument appeared to be slowly swinging as it deployed. Approximately 1 hr of initial background data was accumulated before transfer to DSS 11 at 22:35 GMT and the resumption of TV operations.

At the end of November 10, the general pattern of lunar operations had been established: under the direction of SSAC, TV sequences and other spacecraft activities requiring video support were alternated with alpha scattering operations, while periodic halts for engineering interrogations provided SPAC with the continuous information necessary for thermal and power management of the spacecraft. The majority of the video sequences and video-supported experiments were reserved for the Goldstone view period. Alpha scattering accumulations and other non-TV engineering activities were generally scheduled for the DSS 42 and DSS 61 view periods.

Early on November 11, two attitude control jet firing experiments were performed. The TV camera was focused on one of the jets and the surface below it, and a series of pictures were taken before, during, and after the firings to provide information on mechanical properties and movement of the lunar surface material.

Later during the day, members of the spacecraft hop operations team met with JPL and NASA Project Management to make recommendations for the vernier static firing and/or spacecraft hop experiment. Preparations for the experiment had been under way since shortly after touchdown, but owing to the inherent risk involved, Project Management had postponed the final decision until all factors had been evaluated in real-time. SPAC thermal specialists reported that, even with the present low sun angle, temperatures were beginning to rise rapidly. Temperature trend data indicated that Vernier Engine 3 would reach its upper operational limit around 09:00 GMT. Project Management then advised that the alpha scattering experiment had first priority and that neither the static firing nor spacecraft hop could be performed until adequate data had been accumulated to perform an analysis. It was desired to obtain at least 24 hr of certified noise-free surface data. Thermal and propulsion specialists were requested to continue their analyses and determine when and if thermal and propulsion systems would permit a spacecraft hop.

Following completion of 6.5 hr of background data accumulations, the alpha scattering sensor head was fully deployed to the lunar surface on November 11 at 12:07 GMT. Thermal predictions indicated, however, that the instrument would reach its maximum temperature limit late on November 12. Operation time was increased to permit accumulation of as much data as possible before the temperature limit was reached. An assessment of the transmitter, battery, and Compartment A and B temperatures disclosed that these temperatures were running 20 to 30°F below comparable *Surveyor V* readings and that there would be no need to shut down the spacecraft because of high temperatures during the lunar morning.

On November 12, SPAC thermal specialists reported that *Surveyor VI* could be properly configured thermally for a vernier static firing and/or spacecraft hop, and recommended the near-lunar-noon period as the optimum hop "window." The recommendation was based on an analysis of temperature trends and shading possibilities with the solar panel and planar array. The high solar elevation angle at lunar noon would provide optimum shading for the various flight control and propulsion system components. A second opportunity would occur late in the lunar afternoon but shading possibilities at that time would be considerably reduced. Another factor favoring the lunar noon period was that extended exposure to the lunar environment was expected to deteriorate the propulsion system (as had occurred on previous missions), since many components were not designed to withstand the temperature extremes. Moreover, it was estimated that the most important scientific data, including alpha scattering accumulations, would be obtained by the lunar noon period.

The plan was tentatively accepted by the Project Manager, but the experiment remained contingent on further observation and analysis of spacecraft trends during the next four days before lunar noon. Alpha scattering instrument temperatures were monitored throughout the remainder of November 12, and, as predicted, the instrument sensor head reached its 122°F maximum operating temperature toward the end of the day. At 23:39 GMT, the sensor head temperature reached 123°F and the instrument was turned off. Approximately 17 hr of lunar surface data had been received. The predicted first opportunity to resume operations was four days later on November 16, when a favorable combination of sun angle and A/SPP shadow patterns would provide shade.

During November 13, 14, and 15, thermal control, propulsion, and attitude determination personnel worked closely together to plot optimum shadow patterns that would allow continuous spacecraft operation and yet provide a suitable spacecraft thermal configuration, particularly with respect to the vernier engine thrust chambers, so that the spacecraft hop experiment could be achieved close to lunar noon. A variety of solar panel and planar array shadow patterns were produced using the *Surveyor* Environmental Test Laboratory one-fifth-scale model of the spacecraft and a collimated light source to simulate the sun. Polaroid pictures were taken of the recommended patterns and then compared with spacecraft TV pictures of the actual shaded areas to verify the effectiveness of the recommended A/SPP positions.

During November 14 and 15, efforts were concentrated on monitoring the spacecraft and shading its components on a priority basis. Concurrently, best-lock frequency tests, RF communications, and signal processing tests were conducted by DSS 42 and DSS 61. Propulsion specialists reported that the oxidizer system had vented a total of five times at 6- to 8-hr intervals and that the fuel system had vented twice. To preserve the propellant pressurization system, the solar panel was positioned to shade the system relief valves. The temperatures of the fuel and oxidizer tanks for Vernier Engine 3 were 192 and 178°F, respectively, and rising. These temperatures were seriously high since they could cause the tank seals to fail, dumping the propellant overboard. All three vernier engine thrust chamber temperatures appeared to be converging at approximately 240°F, 20 deg above the upper operating limit. Compartment A and B temperatures indicated shading could be delayed for another two days. An A/SPP configuration was established to shade the Engine 3 fuel and oxidizer tanks and the relief valves and also to keep the planar array directed toward the earth. Attitude specialists reported that the sun angle would permit shading the alpha scattering sensor head at approximately 09:00, November 16, and a shadow pattern was prepared accordingly.

With lunar noon predicted for 04:00 GMT, November 17, planning for the spacecraft hop intensified. Throughout November 16, temperature readings and predicted temperatures were analyzed to establish shading priorities and A/SPP positions. At 09:46 GMT, the A/SPP was stepped to shade approximately half of the alpha scattering sensor head while still maintaining shade for Oxidizer Tank 3 and the relief valves. By 12:57 GMT, the head temperature had returned to within operating limits and the instrument was turned on. At 10:00 GMT, DSS 11 conducted an antenna axial ratio test. Toward 23:30 GMT, the spacecraft hop operations team presented a review of spacecraft status, constraints, and a final recommendation for liftoff time to Project Management. Assured that all essential conditions were favorable, Project Management consented to the experiment and the final go-ahead was given.

During the next 11 hr, plans for the spacecraft hop were finalized. The flight plan called for hopping the spacecraft approximately 4 ft from its initial landing site in order to obtain comprehensive pictures of imprints made by the landing legs and pictures of any soil erosion caused by the vernier engines. The flight distance of 4 ft would be sufficient to expose a portion of the original landing site, while keeping landing impact forces and the possibility of spacecraft damage to a minimum. The telecommunications subsystem was carefully analyzed, since maintaining the command up-link throughout the maneuver was essential to success. Signal levels at the spacecraft receivers, the earth vector, and omnidirectional antenna patterns were evaluated for unfavorable conditions. The possibility of interference from the vernier engine exhaust plumes was studied. Since the firing time was short, a signal attenuation of less than 1 db was expected, with worst-case conditions increasing the attenuation to 5 db. Difficulties with maintaining the down-link signal from the spacecraft were not expected.

At 03:32 GMT on November 17, the alpha scattering experiment was terminated after accumulating a total of 30 hr of lunar surface data, of which 27 hr, 44 min was known to be noise-free. With shade for the instrument no longer required, the A/SPP was repositioned to cast shadows over the flight control sensor group, helium relief and check valves, and Vernier Engine 2. At 05:53 GMT, the A/SPP was again stepped to include Vernier Engine 3.

At 08:00 GMT, flight control power was commanded on for a liftoff dry run to obtain accurate thermal control calibrations. Gyro temperatures were brought within operating temperature range, gyro speed checks were run, and each gyro was processed. Flight control specialists verified that all elements were normal and able to support the hop. Between 08:42 and 08:53 GMT, the A/SPP was repositioned to shade Vernier Engine 1. Approximately 75% of the flight control sensor group was shadowed at the same time. At 10:02 GMT, all three engines were within safe operating limits and the A/SPP was commanded to its safe position. The spacecraft was placed in the desired telemetry configuration for the hop: transmitter high power, Transmitter A connected to Omniantenna A, Receiver A, one-way lock, Mode 6 data at 1100 bits/sec. The pitch gyro was precessed to control the direction of flight toward Leg 1. At 10:30 GMT, thrust phase power was commanded on. Vernier ignition was commanded at 10:32:01 GMT, followed by two engine shutdown commands 2 and 2.5 sec later. Telemetry indicated that the spacecraft did not respond to the first shutdown command and that the engines burned for the full 2.5 sec. Instead of a planned 4-ft hop, the spacecraft traveled a total distance of 8 ft, reaching an altitude of 10 to 15 ft. The total flight time was 6.1 sec.

After touchdown, the spacecraft responded normally to the postlanding power shutdown commands, and the solar panel and planar array were stepped without difficulty to the expected sun and earth positions. A study of the shock-absorber strain gage data showed that Leg 1 touched down first, as expected, immediately followed by Legs 2 and 3. Preliminary estimates of the shock-absorber landing loads were 3000 lb for Leg 1, 1650 lb for Leg 2, and 1700 lb for Leg 3. (Refer to Table IV-9 for final values.) The preliminary tilt estimate was approximately 3 deg, with Leg 1 pointing downhill. By 11.07 GMT, the spacecraft was reconfigured for 600-line television. Picture sequences were commanded to obtain views of the original landing site and to assess the effect of vernier engine erosion on the lunar soil and the effect of the hop on the spacecraft. The spacecraft structure appeared to be unharmed and relatively free of lunar debris. A notable, and apparently unique, exception was the photometric target mounted on the omnidirectional antenna boom, which was found to be covered with a layer of lunar soil. The alpha scattering sensor head was located approximately 3 in. outboard from the nitrogen tank but now lying upside down. The effect of the hop on the alpha scattering sensor head was not expected to be favorable, since the head cannot be retrieved or moved once it has been deployed. After completing television, the instrument was turned on and the radioactive alpha particle source was found to be damaged. The proton detector system was still operable, however, and the instrument was subsequently used to obtain periodic data on cosmic radiation.

The assessment of post-hop status disclosed that, except for the alpha scattering instrument, no damage had been sustained, subsystems were functioning normally, and temperatures were in accord with the sun angle. All indications were that the *Surveyor VI* spacecraft hop experiment had been an historic success. Owing to the relative closeness of the sun and earth to each other at this time, A/SPP fine positioning for an accurate attitude determination was postponed until November 20.

Temperature trends were watched closely during the post-lunar-noon period. Shading priorities supported overall thermal and power management needs, but continued to include the propulsion and flight control systems in preparation for a possible second hop. With the alpha scattering sensor head lying upside down on its radiator, the temperature increased rapidly, reaching 162°F at 16:47 GMT, 5 deg below its survival limit. Approximately an hour later, the shadow of the nitrogen tank reached the instrument and the temperature started downward. With some added help from the A/SPP, the temperature was reduced to within operating limits early on November 18. Periodic accumulations were resumed, now collecting cosmic radiation data. The shading of Compartment A, containing the battery, began to assume priority when the battery temperature started rising at a rate of 2 to 3°F/hr. Thermal specialists recommended an A/SPP position that provided approximately 75% shade for the compartment, while maintaining propulsion system and TV camera shading priorities. On November 19, the rise in battery temperature began to approach the upper operational limit of 115°F. It was not possible to continue shading the compartment without sacrificing earth lock. To counter the rise, all science and engineering activities were halted, and the spacecraft was placed in full standby mode beginning at 10:54 GMT. Periodic interrogations were conducted every 2 hr, primarily to assess the battery, which continued to remain on the edge of its upper limit for the next 15 hr. During this time, a "blind" transfer to DSS 61 was effected at the end of the DSS 42 view period. Normal operations were resumed the following day at 02:14 GMT, when a battery temperature of 105.7°F was reported after a final standby.

On November 19, a second, longer spacecraft hop on the order of 100 to 300 ft was proposed to Project Management for performance late in the lunar day. Also on November 19, during one of the periodic interrogations, SPAC noticed a gradual pressure decrease in the oxidizer system. On November 20, propulsion specialists performed a thorough analysis of the propulsion system; temperature and pressures were decreasing but the rates were not considered abnormal. A leak was not unexpected, but at this time the decrease in oxidizer pressure could be attributed to venting. Battery temperature was noted to be rising, and reached 114.8°F at 19:24 GMT. After transfer to DSS 42, the spacecraft was placed in standby from 20:23 to 22:25 GMT. Immediately after revival, propulsion specialists noted that oxidizer system pressure had suffered an abnormal 60-psi pressure drop during the last three hours. The pressure continued its decay to 740 psia, causing the helium pressure regulator to open, allowing helium pressure to pass into the oxidizer system and then overboard. At 05:10 GMT on November 21, with the pressure decrease continuing, Project Management cancelled plans for a second hop.

During November 21, battery temperature remained within its operating range, and operations continued on schedule. Television sequences were conducted during all three station view periods; the alpha scattering instrument remained turned off. An RF communications and signal processing test was also performed by DSS 61. The solar panel was stepped to optimize the battery charging rate, and landing gear status was analyzed in preparation for lunar night survival. Based on shock-absorber temperature predictions, a time was tentatively established to lock the landing legs. Owing to the effects of extreme cold on the gas-filled shock absorbers, the legs must be locked prior to sunset to prevent the spacecraft from sagging when the critical temperatures are reached. (During the *Surveyor V* mission, two legs sagged shortly after sunset, indicating that the squib-fired landing gear locks had apparently malfunctioned.)

With temperatures continuing to decline, the initial step in the lunar night survival plan was initiated at 20:15 GMT on November 22, when the Compartment A heater was turned on to keep the battery temperature from falling below 100°F. Effectively, the first step of the plan had begun shortly after spacecraft touchdown with initial positioning of the solar panel and the establishment of a battery charge rate that would leave a full charge in the battery at sunset. During the lunar day, the charge rate was maintained as high as possible (up to 2 A). After lunar noon, the excess solar panel current was used to raise compartment temperatures to their highest operational levels. Lunar sunset was predicted for 14:42 GMT on November 24. Landing gear shock-absorber temperatures were observed closely, and the command to lock the landing legs was sent at 16:08 GMT. Alpha scattering data accumulations were resumed at 04:52 GMT and alternated with television sequences throughout the day. The solar panel was stepped a number of times to maintain the desired battery charging rate. On November 23 and 24, preparations for lunar night consisted mainly of compartment load juggling to maintain constant temperatures and solar panel positioning to optimize the battery charge. A special engineering mechanisms auxiliary test was conducted to check to see if adequate current was available for firing the landing gear lock squibs. Test results showed that sufficient current was present in the circuit and that the legs should have been locked as commanded.

Lunar sunset (zero solar panel output) occurred at 13:53 GMT, November 24, approximately 50 min earlier than predicted due to uncertainties in determining the local lunar terrain. Phase II of the lunar night survival plan (a modification of the *Surveyor V* plan) was now put into operation and consisted of the following:

(1) Battery consumption for postsunset science activities was to be limited to approximately 10 A-hr.

(2) Alternate standby and interrogation modes were to be utilized to allow compartment temperatures to drop as rapidly as possible; Compartment A and B temperatures were to be stabilized near their lower operating limits; operations were then to be extended as far into the lunar night as possible in order to shorten the cooldown period between shutdown and sunrise on the following day to a minimum.

(3) Interrogation times were to be held to a minimum, and were to be performed every 4 hr during the first 48 hr, then at 12-hr intervals 48 to 96 hr after sunset and, finally, once every 24 hr until the battery charge remaining reached 45 A-hr.

(4) Battery was to be maintained at +20°F and Compartment B at 0°F after all Compartment A and B thermal switches had opened (effectively disconnecting the conduction paths from the interior of the compartment to the exterior).

Immediately after sunset, the TV camera was positioned to obtain pictures of the solar corona. Between 14:11 and 17:11 GMT, the corona sequences were alternated with other sunset phenomena which included a star survey and views of earthshine on the eastern horizon and on one of the footpads. At 17:18 GMT, the spacecraft was reconfigured for the alpha scattering instrument, and a final 40-min data accumulation was begun to obtain information on postsunset cosmic radiation. The instrument was finally turned off at 18:01 GMT, and the television sequences were resumed, continuing periodically until 20:06 GMT, when the camera was secured for the lunar night. The A/SPP was then stepped to place the solar panel and planar array at optimum positions for revival on the second lunar day. The solar panel was positioned so that charging current would not start until shortly after the battery reached its thawing temperature of −40°F, at which point it will receive a charge.

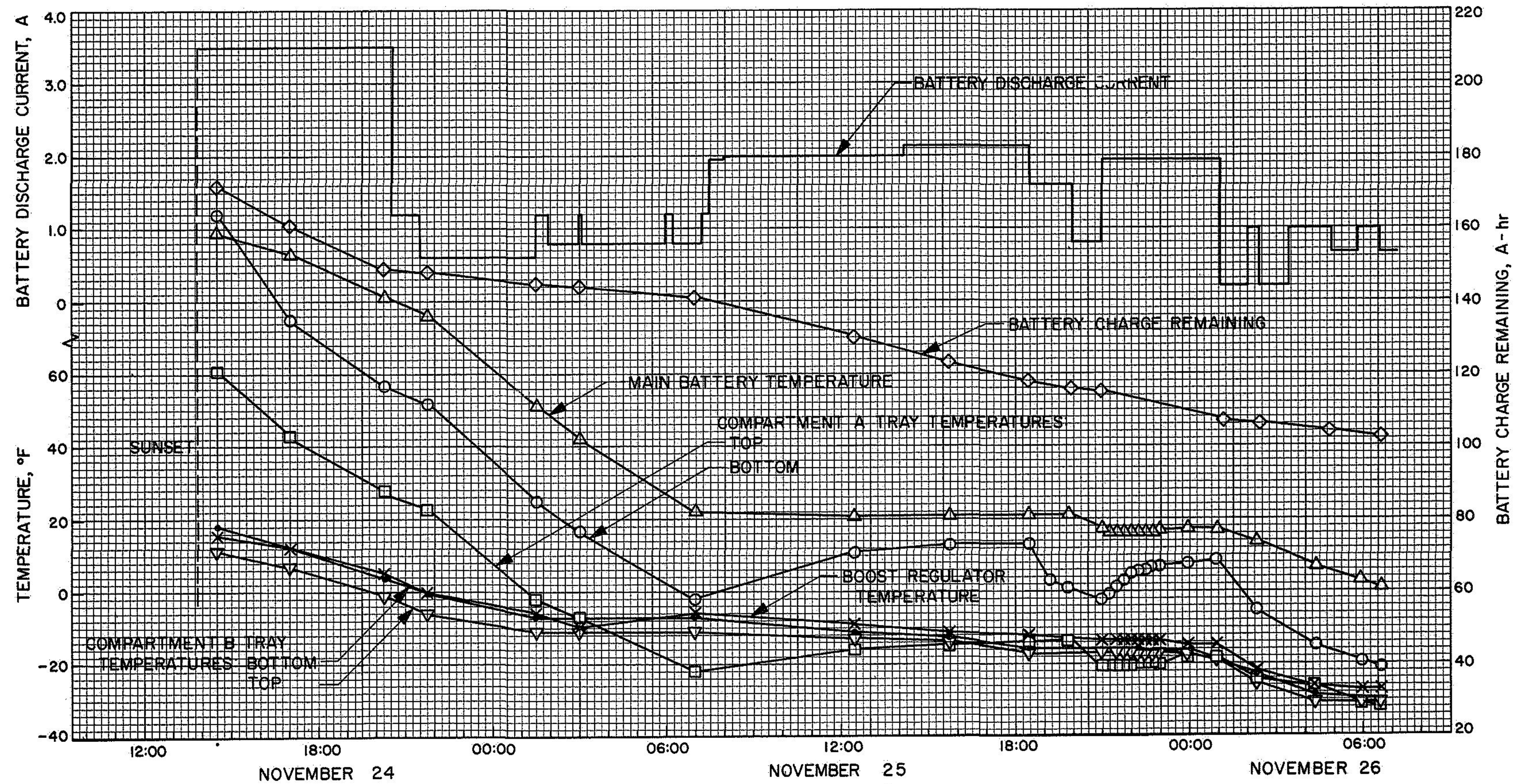


Fig. VI-3. Key spacecraft thermal and power parameters controlled during first lunar night operations through final telemetry transmission

Figure VI-3 shows several key performance parameters during lunar night operations. The data is plotted from shortly after sunset through final spacecraft data transmission.

On November 25, the spacecraft was cycled between engineering interrogations and standbys in accordance with the survival plan. At 03:00 GMT it became apparent that many of the compartment thermal switches had not opened, and that conduction paths to the outside were still intact, allowing the spacecraft compartments to cool down much more rapidly than expected. To compensate for eventual lower temperatures, the solar panel was repositioned to give the battery an additional 20-hr warmup before the solar panel was illuminated. Additional heating was turned on, but at 21:05 GMT, battery temperature had slipped below the desired 20°F to 16.2°F. It was decided not to return the spacecraft to standby but to increase heat by continuing the interrogation period until the battery temperature returned to 20°F.

At 22:25 GMT, Mission Control decided it was more desirable to conserve battery energy rather than expend it to keep the compartment warm. Power specialists had earlier estimated that the battery energy remaining was approximately 30 A-hr less than indicated in the power log. A reassessment of the calculations indicated that the battery, instead of charging during TV operations as reflected in the log, had actually been discharging. The lower battery energy level would naturally shorten the planned lunar night operations. The decision was made to let the compartments cool down rapidly, continuing periodic interrogations, until the battery reached 0°F. It was hoped that a quick reduction in temperature might shock the thermal switches into opening. During the cooldown, three interrogations and a best-lock frequency test were performed. The effect of the cooldown began to inhibit operations when difficulty with signal processing was experienced. At 07:45 GMT, November 26, a fourth interrogation was attempted, but the spacecraft could not be revived. The absence of a response was not unexpected since temperatures for both transmitters had already slipped considerably below the specified operational limit.

Recovery attempts were continued from 08:23 until 09:34 GMT, when the spacecraft was blind-commanded to its final shutdown configuration. At the time of last contact, six Compartment A thermal switches and one Compartment B switch had not yet opened. Owing to the large number of thermal switch failures, postsunset operations were limited to about 40 hr of activity instead of an expected minimum of 130 hr.

8. Second Lunar Day

Attempts to revive *Surveyor VI* on its second lunar day were initiated at 01:00 GMT December 13, approximately 80 hr after sunrise at the landing site, and the earliest time predicted for the spacecraft battery to have reached its thaw temperature of -40°F. Revival attempts were conducted periodically by DSS 11, DSS 42, and DSS 61, and continued on into December 14. A response from the spacecraft was finally achieved at 16:41 GMT on December 14 during the DSS 61 view period. The signal level was approximately -130 dbm, with Transmitter A transmitting over the planar array. The received carrier frequency was quite erratic, however, and the station could not lock on to the signal. After considerable difficulty, Transmitter A was commanded off, and at 18:24 GMT, Transmitter B/Omniantenna B were commanded on. Received signal strength was 140.5 dbm.

An engineering interrogation was attempted to find a commutator mode that would yield coherent engineering data. The station managed to lock up on commutator Mode 4 data at 17.2 bits/sec, but none of the received data was intelligible. At 19:08 GMT, Mode 5 was successfully commanded on at 137.5 bits/sec, and some good data was received. Telemetry indicated a temperature gradient in Compartment A opposite to normal with respect to the solar elevation. The reading for the top of the compartment should have been hotter than the bottom reading, but was 13°F cooler, showing that a considerable amount of energy was being dissipated by the battery and indicating a possible battery problem. While preparations were being made to position the solar panel more toward the sun, the spacecraft signal disappeared at 19:14 GMT and could not be detected again by the station. Commands to turn off commutator processing and all nonessential loads were transmitted at 19:27 GMT on all receiver/decoder combinations.

Several more attempts were made to revive *Surveyor VI* on its second lunar day, including various modified revival sequences. None was successful. And after a final attempt on December 21, 1967, all operations were terminated.

9. Fourth Lunar Day—*Surveyor V*

Surveyor V was successfully revived on its fourth lunar day at 06:45 GMT on December 14. The spacecraft responded almost immediately to a nonscheduled attempt conducted by DSS 11 between attempts to revive *Surveyor VI*. The spacecraft configuration was Transmitter B on planar array. The station reported a received signal strength of −160 dbm but was apparently not locked onto the main carrier frequency. The planar array was subsequently peaked on the earth, and signal strengths increased to −115.0 dbm on Transmitter A and 118.3 dbm on Transmitter B. High power was successfully commanded on both transmitters. Engineering commutator modes were cycled, but all received telemetry data was unintelligible except in Mode 7 (TV identification). The television data received in Mode 7 (see Appendix C) together with analysis of the down-link frequency was used to estimate Compartment A temperatures during operations. Battery power was unknown, but signal strength readings during peaking of the planar array indicated that solar panel current was being generated. A check of Receiver B/Transponder B indicated that Receiver B was not functioning. The television camera was operable but picture quality remained poor as on *Surveyor V*'s second lunar day. A check of the *Surveyor V* alpha scattering instrument revealed the same anomalies present as on the second lunar day: two alpha detectors and one of four proton detectors operating.

On December 15, after a *Surveyor VI* revival attempt, the spacecraft was configured for television, and 67 pictures were taken during the DSS 11 view. Images were recognizable but the lighting quality was poor. Assessment of horizon pictures disclosed the spacecraft attitude to be approximately the same as on the second lunar day, indicating that the landing gear had returned to normal position.

Attempts were made to obtain engineering commutator telemetry, but without success. All evaluations of spacecraft status were based on the limited data received from Mode 7 interrogations and analysis of the down-link frequency. The spacecraft was placed in two-way tracking mode whenever operations allowed. Received signal strength remained good: −120.0 dbm in low power and 103.7 dbm in high power.

On December 16, considerable difficulty was encountered during command activities, and the spacecraft configuration was changed several times in hope of improving the command link. Preparations were made to step the A/SPP in order to reposition the solar panel to a more favorable sun angle and to improve signal level. With no engineering telemetry available, the command reject/enable subcarrier oscillator was commanded on to provide verification that A/SPP commands were being processed by the spacecraft. Each stepping command, however, caused indexing between Receiver A and the inoperative Receiver B, and finally resulted in loss of the down-link signal. The down-link was shortly reacquired and operations were temporarily halted for another *Surveyor VI* revival attempt.

Surveyor V was commanded back on and A/SPP stepping was again attempted. Receiver indexing occurred with almost every command. All four A/SPP axes were commanded, but the received signal strength remained almost constant, indicating no apparent A/SPP movement.

A/SPP stepping attempts were terminated, and the spacecraft was reconfigured for television in hopes that 200-line television in low power would be sufficiently legible to verify any A/SPP movement. Three pictures of very poor quality were received. At 04:30 GMT on December 17 an attempt was made to go to high power to improve television performance, resulting in loss of the spacecraft signal. Commands were sent to return to low power, but the spacecraft remained silent. Shutdown commands were then sent on all receiver/decoder combinations.

Attempts to revive *Surveyor V* were conducted later on December 17 and then, unsuccessfully, on each day until December 21, when both *Surveyor V* and *Surveyor VI* operations were terminated.

VII. Flight Path and Events

Surveyor VI was injected first into a parking orbit and then into a near perfect lunar transfer trajectory. A midcourse maneuver of 10.06 m/sec was executed to compensate for injection errors, to adjust the spacecraft trajectory to a slightly refined aiming point, and to reduce the spacecraft main retro burnout velocity to an acceptable value. Soft landing occurred at 0.47 deg north latitude and 1.48 deg west longitude, 10.5 km from the midcourse aiming point.*

A. Prelaunch

For *Surveyor VI*, the landing site selected prior to launch for targeting of the launch vehicle ascent trajectory was the prime *Apollo* landing site located in Sinus Medii at 0.42 deg north latitude and 1.33 deg west longitude. Selection of this site was dictated by its key importance to *Apollo*. Selection of the unbraked design impact speed of 2635 m/sec was dictated by the desire to utilize the only remaining A21 main retromotor. This resulted in nonoptimum but acceptable Goldstone postlanding visibility for the *Surveyor VI* launch period.

*Landing coordinates based on correlation of *Surveyor* and *Lunar Orbiter* pictures.

B. Launch Phase

The *Surveyor VI* spacecraft was launched from AFETR Launch Pad 36B at Cape Kennedy, Florida, November 7, 1967, using an *Atlas/Centaur* (AC-14) boost vehicle. Liftoff occurred on schedule at 07:39:01.075 GMT. Two seconds after liftoff, the launch vehicle began a 13-sec programmed roll that oriented the vehicle from a pad-aligned azimuth of 115 deg to a launch azimuth of 82.995 deg. At 15 sec, a programmed pitch maneuver was initiated which was completed at *Atlas* booster engine cutoff (BECO). Following booster section jettison, the remainder of the flight to injection was guided by the *Centaur* inertial guidance system. The *Centaur* first burn injected the vehicle into a parking orbit having a 92-nm apogee and 86-nm perigee. The spacecraft was in the earth's shadow during the first 14.4 min of flight, but left the shadow during parking orbit coast and remained in sunlight during the remainder of the flight. After a 12.9-min coast, the *Centaur* burned a second time to inject the *Surveyor VI* spacecraft into the desired lunar transfer trajectory. All event times for the launch phase were close to nominal. The launch phase sequence is discussed in greater detail in Section III, and actual event times for all phases of the mission are summarized in Table A-1 of Appendix A.

C. Pre-midcourse Cruise Phase

Separation of *Surveyor VI* from *Centaur* occurred at 08:04:30.0 GMT on November 7, 1967, at an approximate geocentric latitude and longitude of 6 and 4 deg, respectively. Automatic sun acquisition began about 1 min after separation and was completed after a maneuver of −264 deg roll and +22 deg yaw.

Predictions indicated *Surveyor VI* would rise over the DSS 51 horizon at L + 00:31:03. DSS 51 received good one-way data at predicted rise − 69 sec and good two-way data at rise + 4 min, 11 sec. This initial acquisition was entirely nominal and comparatively rapid (previous *Surveyor* initial acquisitions have averaged over 6 min).

A plot of the *Centaur* and *Surveyor VI* trajectories, projected on the earth's equatorial plane, is provided in Fig. VII-1. The earth track traced by *Surveyor VI* appears in Fig. VII-2. Specific events such as sun and Canopus acquisition and rise and set times for DSIF stations are also noted.

Prior to Canopus acquisition, the spacecraft was rolled to generate a star map. Then Canopus was acquired at L + 08:49 with a net roll of +292.5 deg. In normal cruise mode, the spacecraft $-Z$ axis is aligned to the sun and the $-X$ axis to the projection of Canopus on the spacecraft X–Y plane.

From the time of two-way acquisition by DSS 51 until approximately 40 min before retro ignition, the DSN tracked the *Surveyor VI* spacecraft in the two-way mode and, with minor exceptions, returned extremely high-quality two-way doppler data. All four DSIF stations supporting *Surveyor VI* (DSS 11, 42, 51, and 61) provided usable data during the pre-midcourse phase of the flight. Two-way doppler was obtained during the midcourse maneuver. One-way doppler was obtained during terminal descent and touchdown. The only appreciable loss of two-way data during the pre-midcourse cruise phase occurred while performing the Canopus acquisition roll maneuver over DSS 61, which was initiated in two-way mode. When an unexpected loss of lock occurred during the roll, the spacecraft was commanded to one-way mode and reacquisition of two-way was not attempted until approximately 50 min later, after completion of Canopus acquisition.

Surveyor VI marked the first wide-scale use of doppler resolver data (from all stations except DSS 51) during the in-flight portion of a *Surveyor* mission. The result of the doppler resolver use was a reduction in the standard deviation of the doppler data (for 60-sec sample rate) by a factor of 4, from a previous level of approximately 0.008 Hz to about 0.002 Hz.

As shown in Fig. VII-3, the current best estimate of the uncorrected, unbraked impact point is in the central area of Sinus Medii at selenographic coordinates of 3.21 deg south latitude and 0.66 deg east longitude. The target point was 0.42 deg north latitude and 1.33 deg west longitude. The two points are approximately 125 km (78 miles) apart on the surface of the moon.

A few selected pre-midcourse orbit computations mapped to the moon are also shown in Fig. VII-3. The numerical data for these selected pre-midcourse computations is presented in Table VII-1. Included are the results of the orbit determination from the Real Time Computer System (RTCS), Cape Kennedy, which were obtained at L + 01:40 and L + 02:10. The computations of the spacecraft orbit by the RTCS were based on DSS 51 data, since the *Centaur* C-band data from the RIS *Twin Falls* was garbled in transmission to the RTCS, and no other C-band tracking data was provided after injection into the transfer orbit.

The customary initial transfer orbit estimate based on AFETR data was not computed at the SFOF because no data was available, as mentioned above. The first SFOF estimate of the spacecraft orbit (PROR XA) was computed at L + 01:56, based on approximately 22 min of two-way doppler and angle data from DSS 51 and 2 min of angle data from DSS 42. When mapped to the moon, this orbit solution indicated that the midcourse correction required to achieve encounter at the prelaunch aim point was well within the nominal correction capability. These results were further verified by the second (ICEV YA) and third (PREL XA) orbit computations completed at L + 02:43 and L + 03:34, respectively.

When sufficient two-way doppler data had been received to compute an orbit using doppler data only, the angle data was deleted. This was first accomplished in the PREL XA orbit computation, which utilized approximately 2½ hr of two-way doppler data from DSS 42 and DSS 51. Removing the angle data from the solution, mapped to the moon, resulted in changes of approximately −1 km in $\mathbf{B \cdot TT}$ and +7 km in $\mathbf{B \cdot RT}$, in the conventional **R-S-T** coordinate system.*

*Kizner, W. A., *A Method of Describing Miss Distances for Lunar and Interplanetary Trajectories*, External Publication 674, Jet Propulsion Laboratory, Pasadena, August 1, 1959.

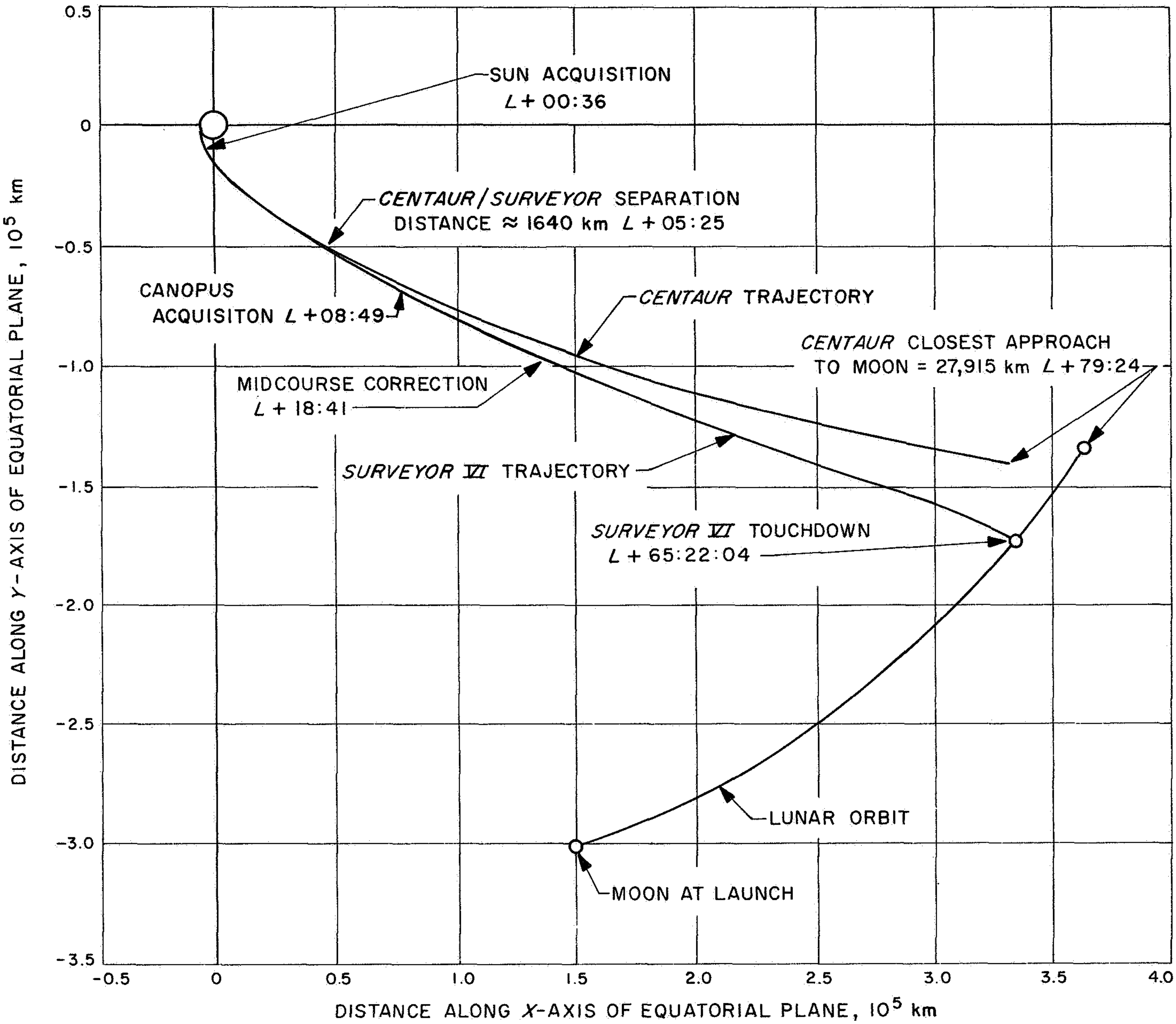


Fig. VII-1. *Surveyor* and *Centaur* trajectories in earth's equatorial plane

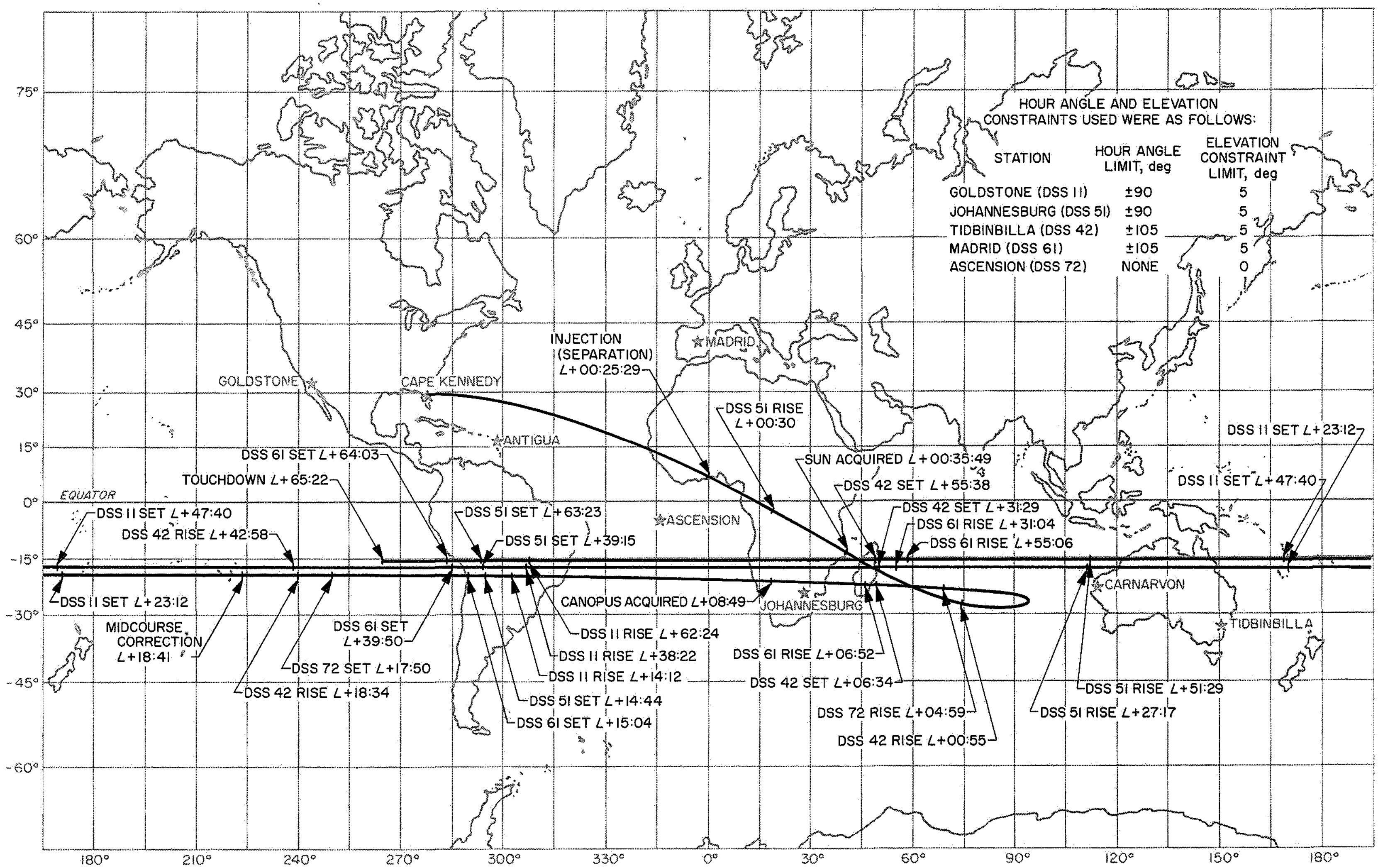


Fig. VII-2. *Surveyor VI* earth track

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table VII-1. *Surveyor VI* encounter conditions based on selected pre-midcourse orbit determinations

Orbit identification	Time computation completed (from liftoff)	Target statistics										Unbraked impact conditions			Data used
		B, km	B • TT, km	B • RT, km	TL, hr	SMAA (1σ), km	SMIA (1σ), km	THETA, deg	σ_T IMPACT (1σ), sec	PHIP 99, deg	SVFIX R (1σ), m/sec	Lat, deg	Long, deg	GMT Nov 10	
RTCS 1	01:40	1994.33	1889.64	637.68	64.24							−7.279	4.129	00:26:42.1	DSS 51 angular and two-way doppler
PROR XA	01:56	1740.71	1709.72	326.99	64.63	117.73	20.85	105.9	95.41	4.513	0.6700	−0.9879	359.65	00:36:57.370	DSS 51 angular and two-way doppler, and DSS 42 angular
RTCS 2	02:10	1817.59	1768.38	420.07	64.35							−2.878	0.987	00:35:30.1	DSS 51 angular and two-way doppler
ICEV YA	02:43	1805.80	1753.65	430.84	64.60	65.92	6.36	99.56	19.69	0.9785	0.6184	−3.059	0.6650	00:35:53.985	DSS 51 angular and two-way doppler, and DSS 42 angular
PREL XA	03:34	1808.22	1754.53	437.39	64.60	22.18	6.54	99.18	3.749	0.3484	0.6169	−3.187	0.6913	00:35:42.703	DSS 42, 51 two-way doppler
DACO YD	09:13	1807.62	1754.06	436.78	64.60	5.050	3.31	101.34	0.8168	0.0963	0.6169	−3.174	0.6806	00:35:42.886	DSS 42, 51, and 61 two-way doppler
LAPM XC[a]	16:02	1807.42	1753.30	438.99	64.60	11.07	4.09	95.28	1.8401	0.1860	0.6169	−3.216	0.6646	00:35:42.987	DSS 11, 42, 51, and 61 two-way doppler
PRCL XB[b]	20:38	1807.17	1753.15	438.55	64.60	11.76	3.99	98.07	1.7711	0.1992	0.6169	−3.207	0.6610	00:35:43.051	DSS 11, 42, 51, and 61 two-way doppler

[a]Orbit used for midcourse computations.
[b]Current best estimate.

SMAA semimajor axis of 1σ dispersion ellipse.
SMIA semiminor axis of 1σ dispersion ellipse.
THETA orientation angle of 1σ dispersion ellipse measured counterclockwise from TT axis in B-plane.
σ_T IMPACT 1σ uncertainty in predicted unbraked impact time.
PHIP 99 99% velocity vector pointing error.
SVFIX R 1σ uncertainty in magnitude of velocity vector at unbraked impact.
TL flight time from injection.

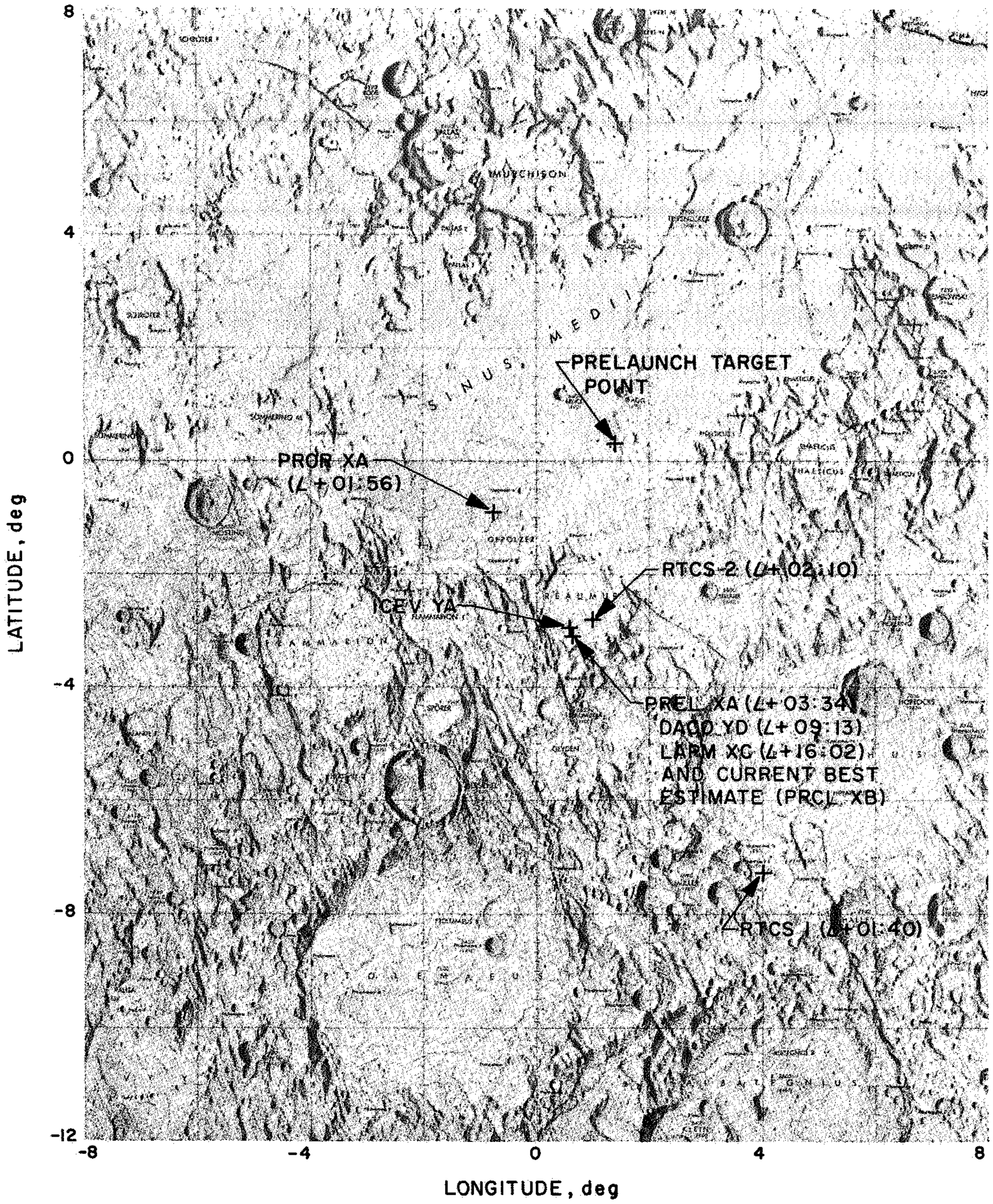


Fig. VII-3. *Surveyor VI* pre-midcourse unbraked impact locations

During the data consistency (DACO) orbit computation period, nine orbit solutions were computed. These solutions included various combinations of two-way doppler data from DSS 42, 51, and 61. During this period, the first data from DSS 61 was received and found to be consistent with data from DSS 42 and 51. Angle data was not included in any of the DACO orbit solutions. As the computers became available, additional orbit solutions were computed to update and evaluate the orbit determination program (ODP) data file.

The LAPM orbit solutions were the first computations to utilize data from DSS 11, and verified the consistency of DSS 11 data with the other data. After updating the ODP data file, the pre-midcourse orbit solution (LAPM XC), on which the midcourse maneuver was based, was computed. This solution utilized all the two-way doppler data available up to 3 hr before the midcourse correction. When mapped to the moon, this solution indicated an unbraked impact point at 3.216 deg south latitude and 0.664 deg east longitude.

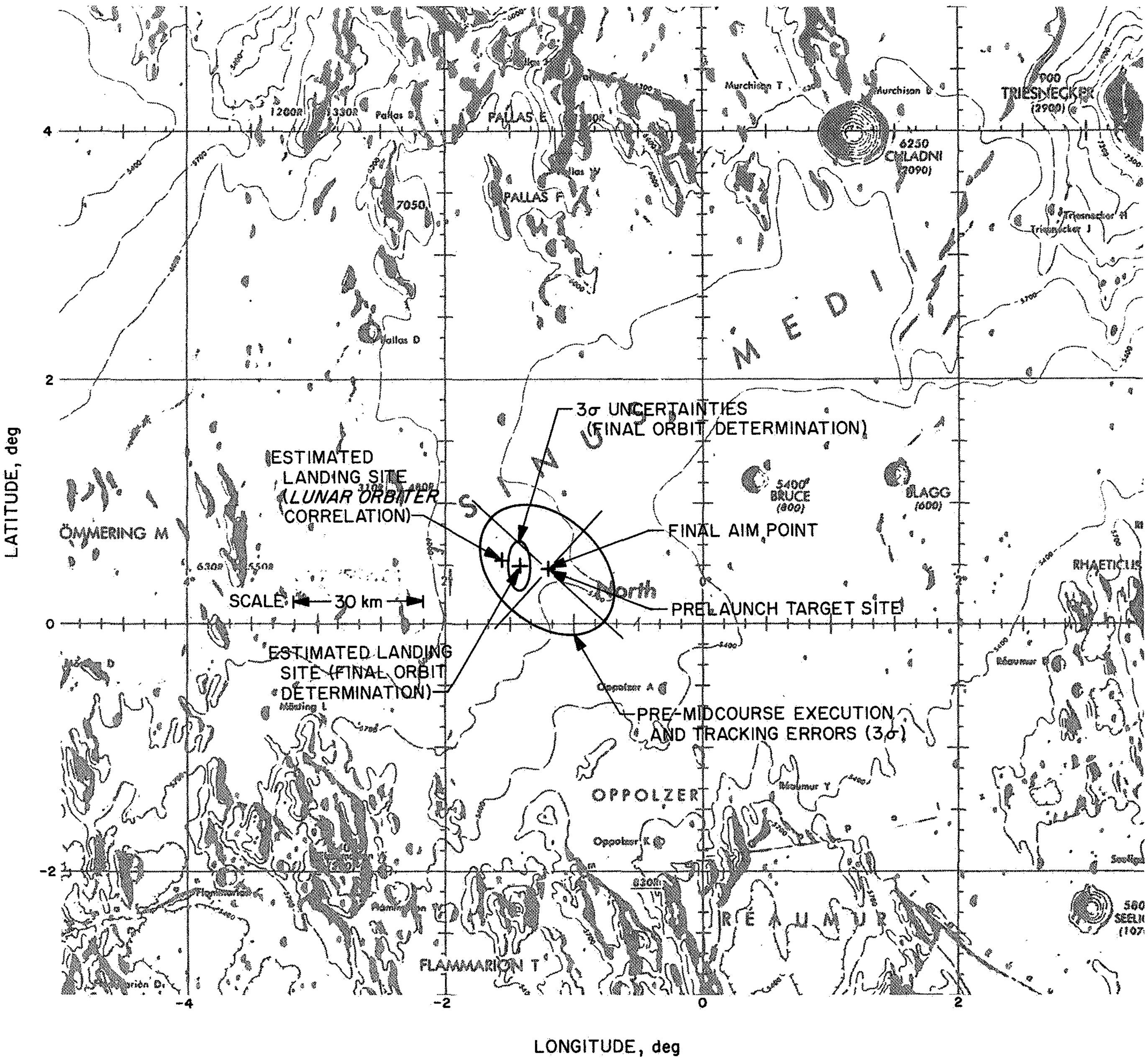


Fig. VII-4. *Surveyor VI* landing location

For the current best estimate of the spacecraft pre-maneuver orbit (PRCL XB), all usable data from DSS 11, 42, 51, and 61 taken from initial DSIF acquisition to start of midcourse maneuver was utilized. This did not include data taken at elevation angles below 17 deg.

D. Midcourse Maneuver Phase

The final aim point was selected in flight based upon consideration of three closely spaced sites and their respective dispersion ellipses. Approximately the same probability of successful landing (~0.45) was associated with all three sites; hence, the final site was selected because it allowed the highest probability of landing within the area covered by *Lunar Orbiter* high-resolution photographs. The *Surveyor VI* midcourse correction, computed to enable the spacecraft to soft-land at the desired landing site of 0.417 deg north latitude, 1.133 deg west longitude, was 10.06 m/sec. This correction was executed upon ground command at 02:20 GMT on November 8, 1967. The resulting soft-landing site is estimated (based on correlation of *Surveyor* and *Lunar Orbiter* pictures) to be at 0.470 deg north latitude, 1.480 deg west longitude, well within the 3σ dispersions predicted prior to the correction. Figure VII-4 shows the prelaunch target site, the final aim point, the estimated soft-landing site, and the associated dispersions. The small ellipse is centered on the current best estimate of the landing site based on orbit determination (OD), and represents the 3σ OD uncertainties in touchdown location. The 99% (3σ) midcourse dispersions are shown as an ellipse on the surface with a semimajor axis of 18.5 km and a semiminor axis of 12.6 km. The semimajor axis of the midcourse dispersion ellipse is oriented 41.9 deg south of east.

The maximum midcourse correction capability, as a function of the unbraked impact speed, is shown in Fig. VII-5. Typical 3σ *Centaur* injection guidance dispersions and the effective lunar radius are also shown. The midcourse capability contours are in the conventional R-S-T coordinate system.

The maneuver execution time of 18.28 hr after injection allowed 4 hr and 29 min of pre-midcourse visibility and 4 hr and 31 min of post-midcourse visibility of the spacecraft from Goldstone.

The midcourse correction of 10.06 m/sec was selected to correct to the final aim point and to adjust other parameters. The velocity component in the critical plane, required to correct "miss only," was only 1.18 m/sec. The velocity component normal to the critical plane is referred to as the noncritical component since it does not affect miss to the first order. The noncritical velocity component was +10.0 m/sec. Figure VII-6 presents the variation in flight time, main retro burnout velocity, vernier propellant margin, and landing site dispersion with the noncritical velocity component U_s. A value of +10.0 m/sec was selected for U_s, which reduced the burnout velocity to 480 ft/sec, to allow adequate attitude control damping during the steering phase after initiation of RADVS control, without increasing the landing site dispersions significantly. The selected value of U_s delayed the time of touchdown by 23 min. If the maneuver had been to correct "miss-plus-flight-time," the required noncritical component would have been +0.42 m/sec, giving a total correction of 1.25 m/sec.

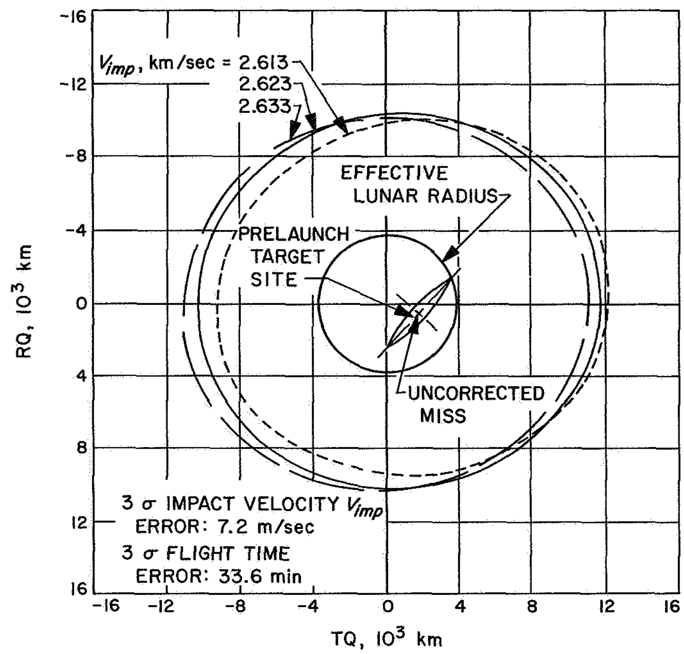


Fig. VII-5. Midcourse correction capability contours (for a correction 20 hr after injection)

Since the final aim point was changed from the prelaunch target point, the above required correction does not properly evaluate the performance of the *Centaur* guidance system. Correcting 20 hr after injection to the prelaunch target point gives a miss-only requirement of 1.25 m/sec and a miss-plus-flight-time requirement of 1.32 m/sec, both remarkably small values.

The predicted results of the selected midcourse correction and other alternatives considered are given in

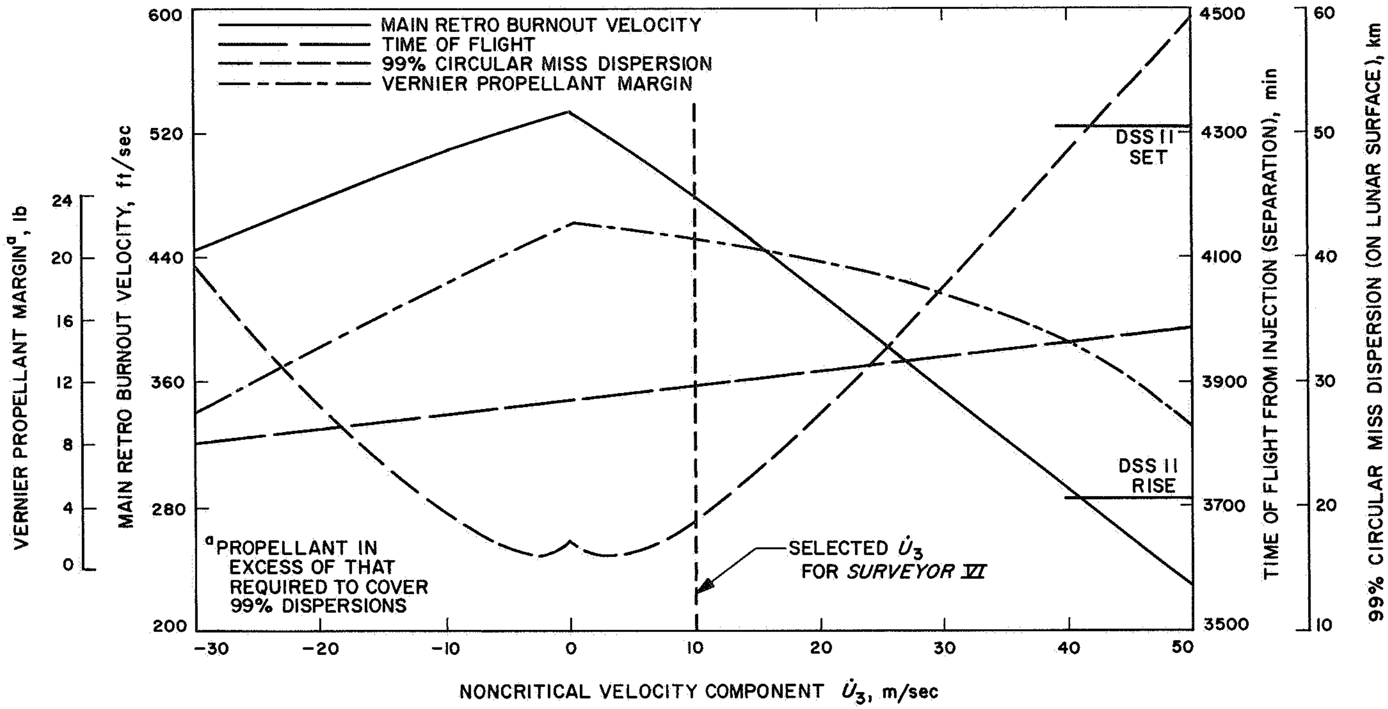


Fig. VII-6. Effect of noncritical component of midcourse velocity correction on terminal descent parameters

Table VII-2. The possibility of not performing a midcourse correction was eliminated because the touchdown point would have been outside the *Lunar Orbiter* high-resolution picture area. Execution of the midcourse correction during the second Goldstone pass, approximately 42.75 hr after injection, would have reduced expected landing site dispersions by one third owing to greater tracking accuracy. However, an evaluation by Project Science indicated no appreciable increase in soft-landing probability would result from reducing the landing site dispersions. In fact, some of the individual sites considered exhibited a higher soft-landing probability for greater dispersions. Also, by conducting the midcourse correction early, more time would be available to correct problems which might develop as a result of the maneuver. The possibility of performing a velocity correction along the spacecraft–sun direction was investigated and eliminated because neither the desired site nor the *Lunar Orbiter* high-resolution picture area could be reached.

An analysis was made of eight pairs of pre-midcourse attitude rotations, any pair of which would have correctly oriented the spacecraft for the midcourse velocity correction. A combination of +91.79 deg roll and +127.75 deg yaw using Omniantenna B was chosen because it maximized the probability of mission success by providing maximum sun-lock time and continuous high antenna gain without requiring antenna switching. The selected attitude rotations were compensated to achieve alignment of the actual thrust axis (determined prior to launch) with the desired midcourse vector. Also, the attitude rotations were initiated at limit cycle nulls and were compensated for estimated gyro drifts for the first time in an attempt to further minimize pointing errors.

E. Post-midcourse Cruise Phase

Surveyor VI two-way doppler data from DSS 11, 42, and 61 was characterized by standard deviations as low as 0.001 Hz owing to the high effectiveness of the resolver data utilized during this mission. In Fig. VII-7 is an example of residuals from the DSS 42 post-midcourse data using the resolver correction. Sample DSS 51 data with no resolver correction is also shown for comparison.

The results of a few selected post-midcourse orbit computations mapped to the moon are presented in Table VII-3. The first post-midcourse orbit computation period was completed approximately 10 hr after the midcourse correction. For the final orbit computation during this period (1POM XG), approximately 48 min of DSS 11 and 8 hr of DSS 42 two-way doppler data were used. When the 1POM XG solution was mapped to target, it indicated an unbraked impact point of 0.08 deg north latitude and 358.98 deg east longitude.

Table VII-2. Midcourse maneuver alternatives

Parameters	Selected midcourse	Alternate considerations: No midcourse	Alternate considerations: Midcourse during second Goldstone pass
Midcourse parameters			
Velocity magnitude, m/sec	10.06		10.25
Critical component, m/sec	1.18		2.25
Noncritical component, m/sec	10.0		10.0
Propellant required, lb	8.37		8.53
First rotation, roll, deg	91.79		
Second rotation, yaw, deg	127.75		
Landing site dispersions (mechanization plus tracking), 3σ			
Semimajor axis, km	18.5		7.1
Semiminor axis, km	12.6		6.8
Ellipse inclination, deg (positive TQ toward RQ)	41.9		6.3
Terminal parameters			
Aim point			
North latitude, deg	0.417	−3.215	0.417
East longitude, deg	358.867	0.65	358.867
Incidence angle, deg	25.0	26.8	25.0
Unbraked impact speed, m/sec	2631.1	2635.1	2631.4
Burnout velocity, ft/sec	482.0	542.2	482.0
Burnout altitude, ft	37,005.4		
Propellant margin, lb	21.5	22.2	21.4

The necessity of having data from at least three tracking stations was emphasized when DSS 51 data was first utilized during the second post-midcourse orbit computation period. The final solution obtained during this period (2POM YB) indicated an unbraked impact point at 0.3736 deg north latitude and 358.93 deg east longitude. The DSS 51 data was consistent with that of DSS 11 and DSS 42. Using data from the third station, the impact parameters stabilized and never varied more than ±0.05 deg during the remaining orbit computations.

The only major losses of good two-way doppler data during the post-midcourse cruise phase occurred on the second passes over DSS 51 and DSS 61. A problem was encountered with the data transmission lines from DSS 51 at 19:38 GMT on November 8 and, at 20:45, an unscheduled transfer was made to DSS 61, which stayed in two-way lock until the spacecraft was transferred to DSS 11 at 22:19. The two-way doppler data taken at DSS 61 during this time (approximately 1½ hr) was unusable because of excessively high noise; the problem was traced to the rubidium frequency standard. When a rubidium change was made during the third pass, an immediate improvement to reasonable levels was observed in the three-way data at DSS 61.

Because of the problems encountered with the data transmission lines from DSS 51, the possibility of using DSS 61 during the time just prior to DSS 11 terminal phase view period was considered. However, in view of the poor quality of DSS 61 data, it was decided to remain with DSS 51 and assume that any data lost could be replayed in time for the final orbit computations. This proved to be a good decision, and data from DSS 11 and DSS 51 was used for the final orbit computations during the last 3 hr of flight. The final terminal phase computations were based on the 5POM WD orbit solution.

Both pre- and post-midcourse injection and terminal conditions are presented in Table VII-4. It should be noted that the post-midcourse terminal conditions presented in Table VII-4 do not match the conditions shown in Table VII-3 very closely. This is due to use of estimated gas jet perturbing forces in the orbit determination program to give a better fit of the data presented in Table VII-3. The trajectory program used to generate the data of Table VII-4 cannot simulate these perturbations.

A terminal attitude maneuver consisting of +81.62 deg roll, +111.67 deg yaw, and +120.55 deg roll was selected using Omniantenna B. The selection of this combination was based on an assessment of all possible maneuvers to provide satisfactory telecommunication performance during the terminal maneuver and terminal descent phases.

Roll attitude of the spacecraft at main retro ignition was dictated by a determination to: (1) maximize margins against RADVS beam rejects, (2) refrain from roll attitudes for which RADVS scintillation problems were possible, and (3) obtain a favorable touchdown orientation for postlanding operations. The final roll maneuver was selected so that at touchdown the projection of the sun in the spacecraft X-Y-plane would be 60 deg from the $-X$ axis toward the $-Y$ axis.

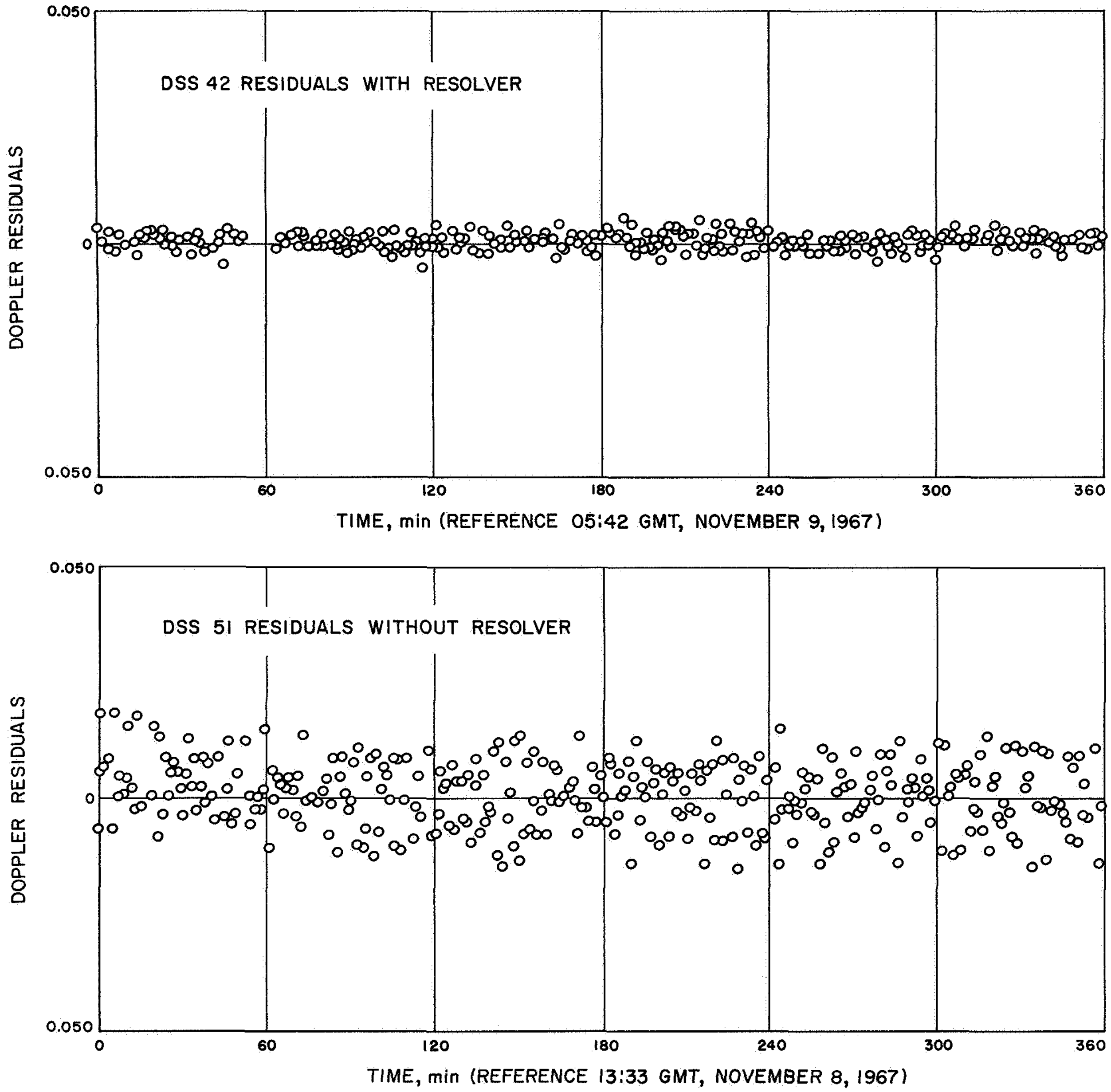


Fig. VII-7. Two-way doppler residuals with and without resolver

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table VII-3. *Surveyor VI* encounter conditions based on selected post-midcourse orbit determinations

Orbit identification	Time computation completed (from liftoff)	Target statistics										Unbraked impact conditions			Data used
		B, km	B • TT, km	B • RT, km	TL, hr	SMAA (1σ), km	SMIA (1σ), km	THETA, deg	σ_T IMPACT (1σ), sec	PHIP 99, deg	SVFIX R (1σ), m/sec	Lat, deg	Long, deg	GMT November 10	
1POM XG	30:29	1721.95	1700.10	273.47	46.29	383.96	33.02	108.7	136.29	9.602	0.7838	0.08287	358.98	00:58:25.977	DSS 11 and 51 two-way doppler
2POM YB	35:00	1717.44	1697.88	258.46	46.30	9.366	7.657	22.03	4.9679	0.3370	0.6184	0.3736	358.93	00:58:29.823	DSS 11, 42, and 51 two-way doppler
3POM WD	50:26	1713.40	1693.68	259.21	46.30	8.602	7.509	25.79	4.6370	0.3127	0.6183	0.3650	358.84	00:58:31.631	DSS 11, 42, and 51 two-way doppler
4POM WG	60:09	1711.64	1692.13	257.70	46.30	14.61	8.569	116.41	3.4048	0.1901	0.6218	0.3947	358.81	00:58:32.785	DSS 11, 42, and 51 two-way doppler
5POM WD[a]	61:45	1711.70	1692.20	257.59	46.30	14.58	8.235	116.46	2.8789	0.1869	0.6216	0.3967	358.82	00:58:32.883	DSS 11, 42, and 51 two-way doppler
FINAL WE[b]	64:42	1711.52	1691.85	258.71	5.748	2.446	0.7096	84.35	0.50256	0.0360	0.6183	0.3757	358.81	00:58:32.972	DSS 11 and 51 two-way doppler
POST 2[c]	Postlanding	1711.04	1691.62	257.11	46.30	9.059	3.637	93.20	1.130	0.1320	0.6189	0.4071	358.80	00:58:32.923	DSS 11, 42, and 51 two-way doppler

[a]Orbit used for terminal maneuver computations.

[b]Orbit used for AMR backup calculations.

[c]Current best estimate.

Table VII-4. Injection and terminal conditions for pre- and post-midcourse trajectories

Coordinate system	Pre-midcourse injection conditions, Nov 7, 1967, 08:03:19.099 GMT					
Inertial cartesian	X = −6257.0723 km	Y = 1587.3009 km	Z = 1136.6406 km	DX = −3.6172231 km/sec	DY = −9.1323663 km/sec	DZ = −4.8987852 km/sec
Inertial spherical	RAD = 6554.5732 km	DEC = 9.9862512 deg	RA = 165.76542 deg	VI = 10.976453 km/sec	PTI = 2.0465176 deg	AZI = 117.37074 deg
Earth-fixed spherical	RAD = 6554.5731 km	LAT = 9.9862512 deg	LON = 359.09665 deg	VE = 10.560919 km/sec	PTE = 2.1270771 deg	AZE = 118.54574 deg
Orbital elements	C3 = −1.1428793 km^2/sec^2	ECC = 0.98123076	INC = 29.002829 deg	TA = 4.1322006 deg	LAN = 4.2844097 deg	APF = 154.91131 deg
Coordinate system	**Pre-midcourse encounter conditions, Nov 10, 1967, 00:35:43.133 GMT**					
Selenocentric	RAD = 1736.3996 km	LAT = −3.2074397 deg	LON = 0.66183209 deg	VP = 2.6339533 km/sec	PTP = −63.315617 deg	AZP = 105.21276 deg
Miss parameter earth equator	BTQ = 1684.5006 km	BRQ = 654.56369 km	B = 1807.2067 km			
Miss parameter moon equator	BTT = 1753.1903 km	BRT = 438.54789 km	B = 1807.2080 km			
Coordinate system	**Post-midcourse injection conditions, Nov 8, 1967, 02:45:00.000 GMT**					
Inertial cartesian	X = 139648.61 km	Y = −99993.393 km	Z = −60984.704 km	DX = 1.5841255 km/sec	DY = −0.67765081 km/sec	DZ = −0.43832036 km/sec
Inertial spherical	RAD = 182262.30 km	DEC = −19.548128 deg	RA = 324.39598 deg	VI = 1.7778608 km/sec	PTI = 76.984607 deg	AZI = 68.012942 deg
Earth-fixed spherical	RAD = 182262.30 km	LAT = −19.548128 deg	LON = 236.53901 deg	VE = 12.277166 km/sec	PTE = 8.1109323 deg	AZE = 270.70668 deg
Orbital elements	C3 = −1.2131410 km^2/sec^2	ECC = 0.97945704	INC = 29.094202 deg	TA = 161.10795 deg	LAN = 4.0445096 deg	APF = 155.41019 deg
Coordinate system	**Post-midcourse encounter conditions, Nov 10, 1967, 00:58:32.010 GMT**					
Selenocentric	RAD = 1735.9997 km	LAT = 0.43056573 deg	LON = 358.86649 deg	VP = 2.6294101 km/sec	PTP = −65.019809 deg	AZP = 100.80734 deg
Miss parameter earth equator	BTQ = 1649.3791 km	BRQ = 465.02953 km	B = 1713.6813 km			
Miss parameter moon equator	BTT = 1694.5024 km	BRT = 255.67541 km	B = 1713.6827 km			

The selected attitude rotations were compensated for a flight control sensor group deflection of 0.34 deg and for measured gyro drift rates which would have resulted in a 0.38-deg offset in the retro thrust direction. Also, it was attempted to initiate the rotations at limit cycle nulls in an effort to further reduce errors in the thrust direction.

The terminal attitude maneuver was initiated approximately 33 min prior to retro ignition with a 2- to 3-min delay between rotations.

A number of times were computed for transmitting backup commands for the terminal phase including: (1) the backup AMR *mark*, (2) the emergency AMR *mark*, and (3) the start of the emergency terminal descent command tape. The computation of these times was based on the final estimate of the unbraked impact time obtained from the FINAL WE orbit computation (see Table VII-3) generated 40 min prior to main retro ignition.

The strategy employed to generate the backup AMR *mark* command is based on a predicted probability of 0.999 that the AMR will work. The 1σ orbit determination and manual execution uncertainties associated with the AMR backup command were determined to be 0.62 sec RSS. Considering this value and the predicted usable vernier propellant, a delay of 1 sec (which is the minimum allowable or "red-line" value) after normal AMR *mark* was specified for arrival of the backup command at the spacecraft. Fixed delays required for operator execution, command generation, radio propagation, and command decoding totaled 2.34 sec and were compensated for by executing the command early. The backup AMR *mark* should have arrived at the spacecraft 1.28 sec after AMR *mark*, which was predicted to occur at 00:57:56.06 GMT. Based on the actual data, the backup *mark* arrived 2.04 $\pm$ 0.15 sec after AMR *mark*, which it is estimated occurred at 00:57:55.74 GMT.

The emergency AMR *mark* would have been employed if *AMR power on* and *AMR enable* were not confirmed, indicating that the AMR was inoperative. The emergency terminal descent command tape contains commands beginning with *vernier ignition*, and would have been started if the spacecraft flight control timer were known to be inoperative. The approach taken to determine the times required to employ these backups is to predict a new main retro burnout altitude centered with respect to the programmed descent contour, predicted nominal burnout velocity, and the maximum burnout velocity condition for which the spacecraft has capability to soft-land with the available vernier propellants. In general, the new burnout altitude is greater than the nominal value, and this higher burnout altitude gives rise to the desire for earlier-than-nominal backup command times. For *Surveyor VI*, the computed emergency AMR *mark* time was 00:57:52 GMT, approximately 4 sec earlier than predicted AMR *mark*. The time computed for start of the emergency terminal descent command tape was 00:57:57 GMT, about 1 sec after predicted AMR *mark*.

The terminal descent sequence is described in Section IV-A. Table A-1 of Appendix A gives terminal phase event times. Initial touchdown occurred at 01:01:05.467 GMT on November 10, 1967, at a mission time of L + 65:22:04.392.

F. Landing Site

Based upon final postflight orbit determination using data obtained in flight, the landing site of *Surveyor VI* was computed to be at 0.437 deg north latitude and 1.370 deg west longitude. This corresponds to a miss of 7.2 km. The 3σ uncertainty ellipse about the computed touchdown location is defined by a semimajor axis of 6 km and a semiminor axis of 3 km, with the major axis in approximately a north–south orientation. However, correlation of *Surveyor VI* and *Lunar Orbiter* pictures indicates that *Surveyor VI* is located at 0.470 deg north and 1.480 deg west, corresponding to a miss of 10.5 km. These touchdown locations are illustrated in Fig. VII-4.

Appendix A

Surveyor VI Mission Events

Table A-1. Mission flight events

Event	Mark No.	Mission time (predicted)[a]	Mission time (actual)	GMT (actual)
Liftoff to DSIF acquisition				
		Nov 7, 1967 (Day 311)		
Liftoff (2-in. rise)		L + 00:00:00.00	L + 00:00:00.00	07:39:01.075
Initiate roll program		L + 00:00:02.00	L + 00:00:02	07:39:03
Terminate roll, initiate pitch program		L + 00:00:15.00	L + 00:00:15	07:39:16
Mach 1		L + 00:01:05	L + 00:01:04.15	07:40:05.23
Maximum dynamic pressure		L + 00:01:24	L + 00:01:22	07:40:23
Atlas booster engine cutoff (BECO)	1	L + 00:02:33.89	L + 00:02:33.26	07:41:34.34
Jettison booster package	2	L + 00:02:36.99	L + 00:02:36.34	07:41:37.42
Admit guidance steering		L + 00:02:41.89	L + 00:02:41.4	07:41:42.5
Jettison *Centaur* insulation panels	3	L + 00:03:18.89	L + 00:03:17.98	07:42:19.06
Start *Centaur* boost pumps		L + 00:03:35.89	L + 00:03:34.86	07:42:35.94
Jettison nose fairing	4	L + 00:03:48.89	L + 00:03:47.77	07:42:48.85
Atlas sustainer engine cutoff (SECO)	5	L + 00:04:09.30	L + 00:04:05.86	07:43:06.94
Atlas/*Centaur* separation	6	L + 00:04:11.30	L + 00:04:07.75	07:43:08.83
Prestart *Centaur* main engines (chilldown)		L + 00:04:12.80	L + 00:04:09	07:43:10
Centaur main engines start (MES 1)	7	L + 00:04:20.80	L + 00:04:17.32	07:43:18.40
Centaur main engines cutoff (MECO 1)	8	L + 00:09:38.07	L + 00:09:41.21	07:48:42.29
100-lb-thrust on	9	L + 00:09:38.07	L + 00:09:41.23	07:48:42.31
Vehicle destruct system safed by ground command			L + 00:10:10.08	07:49:11.16
100-lb-thrust off	10	L + 00:10:54.07	L + 00:10:56.76	07:49:57.84
6-lb-thrust on	11	L + 00:10:54.07	L + 00:10:56.78	07:49:57.86
100-lb-thrust on	12	L + 00:21:52.64	L + 00:21:52.90	08:00:53.98
Start *Centaur* boost pumps		L + 00:22:04.64	L + 00:22:04.18	08:01:05.26
Prestart *Centaur* main engines (chilldown)		L + 00:22:15.64	L + 00:22:16[b]	08:01:17
Centaur C2 main engine start (MES 2)	13	L + 00:22:32.64	L + 00:22:32.90	08:01:33.98
Centaur C1 main engine start	14	L + 00:22:32.64	L + 00:22:32.90	08:01:33.98
Centaur main engines cutoff (MECO 2)	15	L + 00:24:26.42	L + 00:24:28.57	08:03:29.65
Extend *Surveyor* landing legs command	16	L + 00:24:46.64	L + 00:24:46.7	08:03:47.8
Extend *Surveyor* omniantennas command	17	L + 00:24:57.14	L + 00:24:57.3	08:03:58.4
Surveyor transmitter high power on command	18	L + 00:25:17.64	L + 00:25:17.3	08:04:18.4
Surveyor/*Centaur* electrical disconnect	19	L + 00:25:23.14	L + 00:25:22.9	08:04:24.0
Surveyor/*Centaur* separation	20	L + 00:25:28.64	L + 00:25:28.9	08:04:30.0
Start *Surveyor* solar panel stepping			L + 00:25:30	08:04:31
Start *Centaur* 180-deg turn	21	L + 00:25:33.64	L + 00:25:34.0	08:04:35.1
Start *Centaur* lateral thrust	22	L + 00:26:13.64	L + 00:26:14.0	08:05:15.1
Start *Surveyor* sun acquisition roll			L + 00:26:17	08:05:18

[a]The predicted times were computed postflight utilizing actual launch azimuth, tanked propellant weights, and atmospheric data which depend on day and time of liftoff.

[b]MES 2 minus 17 sec; actual flight data not available.

Table A-1 (contd)

Event	Mark No.	Mission time (predicted)	Mission time (actual)	GMT (actual)
Liftoff to DSIF acquisition (contd)				
			Nov 7, 1967 (Day 311)	
Cut off *Centaur* lateral thrust	23	*L* + 00:26:33.64	*L* + 00:26:33.9	08:05:35.0
Start *Centaur* tank blowdown (retro)	24	*L* + 00:29:28.64	*L* + 00:29:28.7	08:08:29.8
Surveyor solar panel locked for transit; start A/SPP roll axis stepping			*L* + 00:31:06	08:10:07
Cut off *Centaur* blowdown, 100-lb-thrust on	25	*L* + 00:33:38.64	*L* + 00:33:38.3	08:12:39.4
Complete *Surveyor* roll; begin sun acquisition yaw turn			*L* + 00:35:05	08:14:06
Surveyor A/SPP roll axis locked for transit			*L* + 00:35:06	08:14:07
Initial DSIF acquisition (two-way lock) completed			*L* + 00:35:14	08:14:15
Cut off *Centaur* electrical power; 100-lb-thrust off	26	*L* + 00:35:18.64	*L* + 00:35:18.3	08:14:19.4
Surveyor primary sun sensor lock-on			*L* + 00:35:49	08:14:50

Event	Mission time (actual)	GMT (actual)
Initial spacecraft operations to Canopus acquisition		
Initial spacecraft operations		
1. Command Transmitter B from high to low power	*L* + 00:40:40	08:19:41
2. Command off A/D isolation amplifier, solar panel deployment logic, and Transmitter A high power	*L* + 00:43:39	08:22:40
3. Command step solar panel minus, then plus, to seat locking pin	*L* + 00:44:19	08:23:20
4. Command step roll axis plus, then minus, to seat locking pin	*L* + 00:45:27	08:24:28
5. Command on telemetry mode 1 for interrogation at 550-bits/sec data rate	*L* + 00:46:30	08:25:31
6. Command on telemetry modes 4, 2, 6, and 5 for interrogation at 1100-bits/sec data rate	*L* + 00:51:26 to *L* + 00:59:34	08:30:27 to 08:38:35
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	*L* + 04:35:45 to *L* + 04:49:11	12:14:46 to 12:28:12
Command on telemetry modes 4, 2, and 1 for interrogation at 1100-bits/sec data rate	*L* + 07:51:00 to *L* + 08:00:24	15:30:01 to 15:39:25
Star verification/acquisition		
1. Command Transmitter B from low to high power	*L* + 08:05:20	15:44:21
2. Command on 4400-bits/sec data rate	*L* + 08:05:58	15:44:59
3. Command on cruise mode manual delay mode; then command maneuver direction (positive)	*L* + 08:06:51	15:45:52
4. Command select Omniantenna A	*L* + 08:07:30	15:46:31
5. Command execution of positive roll (light sources successively detected by Canopus sensor: earth and Deneb)	*L* + 08:11:21	15:50:22
6. Command select Omniantenna B	*L* + 08:17:38	15:56:39
7. Command on sun acquisition mode to stop roll	*L* + 08:19:43	15:58:44
8. Command off Transponder B power	*L* + 08:25:38	16:04:39
9. Command on telemetry modes 4 and 1 for interrogation at 4400-bits/sec data rate	*L* + 08:27:51 and *L* + 08:33:11	16:06:52 and 16:12:12
10. Command on cruise mode and manual delay mode; then command roll direction (positive)	*L* + 08:34:00	16:13:01
11. Command execution of positive roll (light sources successively detected by Canopus sensor: Canopus, earth, and Deneb)	*L* + 08:35:21	16:14:22

Table A-1 (contd)

Event	Mission time (actual)	GMT (actual)
Initial spacecraft operations to Canopus acquisition (contd)		
	Nov. 7, 1967 (Day 311)	
12. Command on star acquisition mode	L + 08:46:27	16:25:28
13. Automatic Canopus acquisition	L + 08:48:48	16:27:49
14. Command on cruise mode	L + 08:50:11	16:29:12
15. Command on telemetry mode 5 for interrogation at 4400-bits/sec data rate	L + 08:51:10	16:30:11
16. Command on Transponder B power	L + 08:53:06	16:32:07
17. Command on 1100-bits/sec data rate	L + 08:57:17	16:36:18
18. Command Transmitter B from high to low power	L + 08:58:28	16:37:29
Pre-midcourse coast phase		
Command on inertial mode (start first gyro drift check, all axes)	L + 09:04:04	16:43:05
Command on solar panel switch, then off	L + 09:13:57 and L + 09:18:41	16:52:58 and 16:57:42
Command on cruise mode (end first gyro drift check)	L + 10:25:10	18:04:11
Command on inertial mode (start second gyro drift check, all axes)	L + 10:39:21	18:18:22
Command on solar panel switch	L + 10:40:31 and L + 11:41:45	18:19:32 and 19:20:46
Command on cruise mode (end second gyro drift check)	L + 11:43:31	19:22:32
Command on inertial mode (start third gyro drift check, all axes)	L + 11:45:24	19:24:25
Command on solar panel switch	L + 12:44:31	20:23:32
Command on cruise mode (end third gyro drift check)	L + 12:59:57	20:38:58
Command on inertial mode (start fourth gyro drift check, all axes)	L + 13:05:52	20:44:53
Command on solar panel switch	L + 14:21:28	22:00:29
Command on cruise mode (end fourth gyro drift check)	L + 14:24:02	22:03:03
Command on telemetry mode 4 for interrogation at 1100-bits/sec data rate	L + 14:37:23	22:16:24
Command on 137.5-bits/sec data rate	L + 14:41:00	22:20:01
Command on 1100-bits/sec data rate	L + 14:44:31	22:23:32
Command on telemetry modes 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 14:46:33 to L + 14:53:08	22:25:34 to 22:32:09
Command on solar panel switch	L + 15:09:03	22:48:04
Pre-midcourse interrogation and gyro speed check	Nov 8, 1967 (Day 312)	
1. Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 16:33:14 to L + 16:39:43	00:12:15 to 00:18:44
2. Command on gyro speed signal processor and check angular rate of gyro spin in roll, pitch, and yaw axes; command off gyro speed signal processor and return to telemetry mode 5	L + 16:45:45 to L + 16:52:44	00:24:46 to 00:31:45
Midcourse phase		
Command on telemetry modes 4, 2, and 1 for interrogation at 1100-bits/sec data rate	L + 17:57:57 to L + 18:03:48	01:36:58 to 01:42:49
Command Transmitter B from low to high power	L + 18:12:18	01:51:19
Command on 4400-bits/sec data rate	L + 18:13:04	01:52:05

Table A-1 (contd)

Event	Mission time (actual)	GMT (actual)
Midcourse phase (contd)		
	Nov 8, 1967 (Day 312)	
Pre-midcourse attitude maneuvers		
1. Command on cruise mode; then command maneuver direction (positive) and store roll magnitude of 183.8 sec (91.9 deg)	L + 18:17:47	01:56:48
2. Command roll execution (automatic end of roll at 02:06:04 GMT after +91.82-deg maneuver)	L + 18:23:59	02:03:00
3. Store yaw magnitude of 254.6 sec (127.3 deg)	L + 18:28:03	02:07:04
4. Command yaw execution (automatic end of yaw at 02:13:23 GMT after +127.35-deg maneuver)	L + 18:30:07	02:09:08
Command on propulsion strain gage power and inertial mode	L + 18:35:13	02:14:14
Command off thermal control power to the AMR and to vernier lines, Vernier Fuel Tank 2, and Vernier Oxidizer Tanks 2 and 3; then unlock Vernier Engine 1 roll actuator and pressurize the vernier system (helium)	L + 18:36:56	02:15:57
Midcourse correction		
1. Command on flight control thrust phase power	L + 18:38:16	02:17:17
2. Store vernier engines burn magnitude of 10.25 sec	L + 18:38:41	02:17:42
3. Command execution of midcourse velocity correction (automatic shutdown of engines at 02:20:13 GMT after 10.24-sec burn, a velocity correction of 10.12 m/sec)	L + 18:41:01	02:20:02
4. Command standard emergency terminate midcourse velocity correction	L + 18:41:14	02:20:15
5. Command off flight control thrust phase power, propulsion strain gage power, and touchdown strain gage power	L + 18:41:41	02:20:42
Command on telemetry mode 5 for interrogation at 4400-bits/sec data rate	L + 18:43:17	02:22:18
Command on vernier lines and AMR temperature controls	L + 18:44:19	02:23:20
Post-midcourse attitude maneuvers		
1. Store yaw magnitude of 254.6 sec (127.3 deg)	L + 18:45:53	02:24:54
2. Command yaw execution, returning spacecraft to sun lock-on (automatic end of yaw at 02:30:22 GMT after −127.37-deg maneuver)	L + 18:47:06	02:26:07
3. Command on sun acquisition mode and store roll magnitude of 183.8 sec (91.9 deg)	L + 18:52:43	02:31:44
4. Command roll execution, returning spacecraft to Canopus lock-on (automatic end of roll at 02:36:01 GMT after −91.97-deg maneuver)	L + 18:53:56	02:32:57
Command on cruise mode	L + 18:58:38	02:37:39
Command on telemetry modes 2, 4, and 5 for interrogation at 4400-bits/sec data rate	L + 18:59:23 to L + 19:06:09	02:38:24 to 02:45:10
Command on 1100-bits/sec data rate	L + 19:07:08	02:46:09
Command Transmitter B from high to low power	L + 19:08:08	02:47:09
Post-midcourse coast phase		
Command on touchdown strain gage power and data channels, then off; command off propulsion strain gage power (to test simultaneous transmission of engineering and strain gage data)	L + 19:33:27 to L + 19:49:34	03:12:28 to 03:28:35
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 20:43:45 to L + 20:57:18	04:22:46 to 04:36:19
Command on inertial mode (start fifth gyro drift check, all axes)	L + 20:58:42	04:37:43
Command on cruise mode (end fifth gyro drift check)	L + 22:03:50	05:42:51
Command on sun acquisition mode (start sixth gyro drift check, roll axis only)	L + 24:53:26	08:32:27

Table A-1 (contd)

Event	Mission time (actual)	GMT (actual)
Post-midcourse coast phase (contd)		
	Nov 8, 1967 (Day 312)	
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 28:46:27 to L + 28:57:32	12:25:28 to 12:36:33
Command on cruise mode (end sixth gyro drift check)	L + 29:07:28	12:46:29
Command on inertial mode (start seventh gyro drift check, all axes)	L + 32:39:38	16:18:39
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 33:36:01 to L + 33:53:21	17:15:02 to 17:32:22
Command on cruise mode (end seventh gyro drift check)	L + 34:14:40	17:53:41
Command on sun acquisition mode (start eighth gyro drift check, roll axis only)	L + 35:20:38	18:59:39
Command on telemetry modes 4, 2, and 5 for interrogation at 1100-bits/sec data rate	L + 37:37:43 to L + 37:45:54	21:16:44 to 21:24:55
Command on telemetry modes 1 and 5 for interrogation at 1100-bits/sec data rate	L + 37:58:13 and L + 38:03:03	21:37:14 and 21:42:04
Command on cruise mode (end eighth gyro drift check)	L + 38:03:27	21:42:28
Command on inertial mode (start ninth gyro drift check, all axes)	L + 39:06:00	22:45:01
	Nov 9, 1967 (Day 313)	
Command on cruise mode (end ninth gyro drift check)	L + 40:32:17	00:11:18
Command on inertial mode (start tenth gyro drift check, all axes)	L + 41:32:24	01:11:25
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 42:11:39 to L + 42:18:39	01:50:40 to 01:57:40
Command on cruise mode (end tenth gyro drift check)	L + 42:46:36	02:25:37
Command on inertial mode (start eleventh gyro drift check, all axes)	L + 43:50:51	03:29:52
Command on cruise mode (end eleventh gyro drift check)	L + 45:09:08	04:48:09
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 46:35:04 to L + 46:46:07	06:14:05 to 06:25:08
Command on inertial mode (start twelfth gyro drift check, all axes)	L + 46:46:56	06:25:57
Command on cruise mode (end twelfth gyro drift check)	L + 47:59:31	07:38:32
Command on inertial mode (start thirteenth gyro drift check, all axes)	L + 49:25:07	09:04:08
Command on cruise mode (end thirteenth gyro drift check)	L + 51:03:27	10:42:28
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 51:07:51 to L + 51:18:52	10:46:52 to 10:57:53
Command on sun acquisition mode (start fourteenth gyro drift check, roll axis only)	L + 51:22:21	11:01:22
Command on Compartment A heater	L + 51:24:55	11:03:56
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 54:14:03 to L + 54:23:41	13:53:04 to 14:02:42
Command on thermal control power to Vernier Fuel Tank 2 and Vernier Oxidizer Tanks 2 and 3	L + 55:14:16	14:53:17
Command on cruise mode (end fourteenth gyro drift check)	L + 55:47:54	15:26:55
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 1100-bits/sec data rate	L + 58:19:23 to L + 58:28:14	17:58:24 to 18:07:15
Command on 550-bits/sec data rate	L + 58:30:03	18:09:04
Command on survey camera electronics temperature control	L + 60:16:38	19:55:39

Table A-1 (contd)

Event	Mission time (actual)	GMT (actual)
Post-midcourse coast phase (contd)		
	Nov 9, 1967 (Day 313)	
Command on Compartment C heater	L + 60:46:06	20:25:07
Command on telemetry modes 4 and 5 for interrogation at 550-bits/sec data rate	L + 60:58:21 and L + 61:04:17	20:37:22 and 20:43:18
Command on telemetry modes 4, 2, 1, and 5 for interrogation at 550-bits/sec data rate	L + 61:56:55 to L + 62:19:34	21:35:56 to 21:58:35
Command on alpha scattering instrument sensor head heater power	L + 62:48:26	22:27:27
Preterminal interrogation, gyro speed check, and VCXO check		
1. Command on telemetry modes 4, 2, 1, and 5 for interrogation at 550-bits/sec data rate	L + 63:00:32 to L + 63:09:59	22:39:33 to 22:49:00
2. Command on gyro speed signal processor and check angular rate of gyro spin in roll, pitch, and yaw axes; command off gyro speed signal processor and return to telemetry mode 5	L + 63:12:28 to L + 63:15:50	22:51:29 to 22:54:51
3. Command off transponder power to perform VCXO check, then command Transponder B power back on	L + 63:17:42 to L + 63:19:03	22:56:43 to 22:58:04
Command off Compartment A heater	L + 63:34:13	23:13:14
Terminal phase		
Command on telemetry modes 6 and 4 for interrogation at 550-bits/sec data rate	L + 64:13:10 and L + 64:16:06	23:52:11 and 23:55:07
	Nov 10, 1967 (Day 314)	
Command Transmitter B from low to high power	L + 64:28:31	00:07:32
Command on 1100-bits/sec data rate	L + 64:29:19	00:08:20
Command off summing amplifiers and command on phase-summing amplifier B	L + 64:30:03	00:09:04
Command on telemetry modes 2 and 5 for interrogation at 1100-bits/sec data rate	L + 64:30:20 and L + 64:32:43	00:09:21 and 00:11:44
Command on propulsion strain gage power, touchdown strain gage power, touchdown strain gage data channels, and survey camera vidicon temperature control	L + 64:35:33	00:14:34
Command off transponder power (one-way tracking)	L + 64:37:58	00:16:59
Terminal attitude maneuvers		
1. Command on cruise mode, then command maneuver direction (positive) and store roll magnitude of 163.4 sec (81.7 deg)	L + 64:39:37	00:18:38
2. Command roll execution (automatic end of roll at 00:28:04 GMT after +81.82-deg maneuver)	L + 64:46:19	00:25:20
3. Store yaw magnitude of 223.4 sec (111.7 deg)	L + 64:49:30	00:28:31
4. Command yaw execution (automatic end of yaw at 00:33:22 GMT after +111.71-deg maneuver)	L + 64:50:37	00:29:38
5. Store roll magnitude of 241.0 sec (120.5 deg)	L + 64:54:58	00:33:59
6. Command roll execution (automatic end of roll at 00:38:57 GMT after +120.55-deg maneuver)	L + 64:55:55	00:34:56
Command on phase-modulated presumming amplifier	L + 65:01:02	00:40:03
Command reset nominal thrust bias to 200-lb vernier thrust level	L + 65:01:59	00:41:00
Store retro sequence delay magnitude of 5.875 sec (between *AMR mark* and vernier engine ignition)	L + 65:02:41	00:41:42
Command on telemetry mode 6 (thrust phase commutator)	L + 65:09:57	00:48:58
Command on retro sequence mode	L + 65:13:04	00:52:05

Table A-1 (contd)

Event	GMT (predicted)	Mission time (actual)	GMT (actual)
Terminal phase (contd)			
		Nov 10, 1967 (Day 314)	
Command off thermal control power to vernier lines, Vernier Fuel Tank 2, and Vernier Oxidizer Tanks 2 and 3; command off temperature controls to survey camera vidicon, survey camera electronics, and AMR; and command off Compartment C heater and alpha scattering instrument heater power		L + 65:13:26	00:52:27
Command on AMR power		L + 65:14:16	00:53:17
Command on thrust phase power		L + 65:15:16	00:54:17
Command AMR enable		L + 65:17:16	00:56:17
Terminal descent[c]	**GMT (predicted)**	**Mission time (actual)**	**GMT (actual)**
1. *AMR mark* at 59.8-mile slant range from moon: start delay magnitude countdown	00:57:56.06	L + 65:18:54.66	00:57:55.74
2. Standard *emergency AMR mark* command received at spacecraft	00:57:57.34	L + 65:18:56.70	00:57:57.78
3. Vernier engine ignition	00:58:01.96	L + 65:19:00.56	00:58:01.64
4. Main retro motor ignition and AMR ejection	00:58:03.06	L + 65:19:01.66	00:58:02.74
5. RADVS power on for warmup		L + 65:19:03.42	00:58:04.50
6. RADVS high voltage on		L + 65:19:25.42	00:58:26.50
7. *Reliable operate doppler velocity sensor* (RODVS) signal, following acquisition of lunar surface by RADVS beams 1, 2, and 3		L + 65:19:31.72	00:58:32.80
8. Main retro burnout (3.5-*g* level sensed by inertia switch)	00:58:42.90	L + 65:19:41.18	00:58:42.26
9. Vernier engines to high thrust level (200 lb)		L + 65:19:50.92	00:58:52.00
10. *Retro case eject* signal		L + 65:19:53.26	00:58:54.34
11. Retro case ejection		L + 65:19:53.57	00:58:54.65
12. Start RADVS-controlled descent (retro sequence mode off)	00:58:57.05	L + 65:19:55.36	00:58:56.44
13. *Reliable operate radar altimeter* (RORA) signal following acquisition of lunar surface by RADVS beam 4		L + 65:19:56.92	00:58:58.00
14. Programmed descent segment intercept and vernier thrust level increase	00:59:26.56	L + 65:20:18.92	00:59:20.00
15. *1000-ft mark:* change RADVS flight-control-loop parameters	01:00:42.05	L + 65:21:38.16	01:00:39.24
16. *10-ft/sec mark:* increase vernier thrust level	01:00:59.43	L + 65:21:55.26	01:00:56.34
17. *14-ft mark:* cut off vernier engines	01:01:04.98	L + 65:22:01.76	01:01:02.84
18. Touchdown (Footpad 1)	01:01:06.76	L + 65:22:03.095	01:01:04.170[d]

[c]Events are automatic except for the *emergency AMR mark* command, which is sent from the earth as standard procedure. Actual times of all terminal descent events have been corrected for 1.297-sec radio transmission delay to show time of occurrence at spacecraft.

[d]Touchdown time of 01:01:05.467 GMT is used elsewhere in this document and indicates time that DSS 14 received and recorded Footpad 1 touchdown data.

Table A-2. Lunar operations[a]

<table>
<tr><th rowspan="2">Date, 1967, and sun elevation at 00:00 GMT</th><th rowspan="2">Engineering (SPAC or DSIF)</th><th colspan="2">Science (SSAC)</th></tr>
<tr><th>Television</th><th>Alpha scattering instrument</th></tr>
<tr><td colspan="4">First lunar day[b]</td></tr>
<tr><td>Nov 10, ~3 deg</td><td>01:01 to 01:44
Postlanding power shutdown, engineering assessment (including unlocking roll axis and solar panel), and spacecraft preparations for 200-line-mode television

02:55 to 03:44
Solar panel and planar array gross positioning for sun and earth acquisition

12:04 to 13:27
Solar panel and planar array fine positioning for attitude determination

14:00
Surveyor V revival attempt initiated by DSS 51

Surveyor VI maintained in engineering mode by DSS 42, then DSS 61, using large frequency offset in station transmitter

18:50 to 19:20
DSS 61 transmitter turned off to conduct Receiver 1 search for Surveyor V signal while maintaining Receiver 2 lock with Surveyor VI

19:35
A/SPP roll axis stepped to increase solar panel power output and raise battery charging rate

20:42
Modulation interrupt test (DSS 61)</td><td>01:50 to 02:34
200-line pictures (24) of Footpad 3, helium tank, lunar surface, horizon, and Footpad 2

04:05 to 05:25
360-deg W/A panorama

06:45 to 09:42
Special areas (2), auxiliary mirrors, and N/A magnets

Polarimetric survey: Footpad 2

N/A panorama: segment 3

10:29 to 11:40
N/A panorama: segments 2 and 1

Planar array and solar panel shadows

Polarimetric survey

N/A panorama: segment 4

Alpha scattering instrument

21:00 to 21:30
Alpha scattering instrument deployment support, including 45 pictures during 3 min of "swing" for views of lunar surface area to which instrument would eventually be lowered

22:47 to 23:58
Polarimetric survey

Special area and auxiliary mirrors</td><td>05:39
Instrument turned on

05:41 to 06:20
Data accumulation and calibration from standard sample in stowed position

09:52 to 10:07
Data accumulation

11:54 to 13:54
Data accumulation and calibration

14:24 to 14:47
15:21 to 15:45
Data accumulation

16:10 to 17:00
Data accumulation and calibration

17:30 to 18:30
19:54 to 20:55
Final data accumulation from standard sample in stowed position

21:17
Instrument deployed to background position

21:37 to 22:30
Data accumulation and calibration in background position</td></tr>
<tr><td>Nov 11, 15.10 deg</td><td>03:22
Attitude control jet firing experiment with coincident television

03:46
Second attitude control jet firing experiment

03:57 to 04:07
Special test to calibrate vernier engine thrust command levels

13:28
Surveyor V revival attempt initiated by DSS 51

Surveyor VI maintained in engineering mode by DSS 42, then DSS 61, using large frequency offset in station transmitter</td><td>00:00 to 02:38
Polarimetric survey completed

Special area and auxiliary mirrors

N/A panorama: segments 3, 5, and 2

N/A magnets

03:22 to 03:30
Attitude control jet firing experiment support, including 3 pictures before, during, and after first firing

04:12 to 04:43
360-deg W/A panorama

Star survey: Sirius, Canopus, and Capella</td><td>05:06 to 08:42
Data accumulation and calibration

08:56 to 10:14
10:47 to 12:07
Final data accumulation in background position

12:07
Instrument deployed to lunar surface

12:08 to 13:10
Data accumulation from lunar surface</td></tr>
</table>

[a]All times given are GMT. Routine engineering interrogations are not included in the table. DSIF two-way doppler tracking was performed throughout lunar operations as schedule permitted. All television sequences are 600-line mode unless otherwise noted. Wide angle and narrow angle are abbreviated W/A and N/A, respectively.

[b]*Surveyor V* third lunar day; all revival attempts were unsuccessful.

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 11 (contd)	16:30 and 17:08 A/SPP roll axis stepped to decrease solar panel power output and lower battery charging rate 19:22 Solar panel stepped down to decrease power output and lower battery charging rate 19:30 and 19:35 Loss of DSS 61 receiver lock with *Surveyor VI*; lock reestablished both times 20:26 to 20:29 *Surveyor V* revival attempt terminated by sending spacecraft shutdown commands, which simultaneously shut down *Surveyor VI*; *Surveyor VI* then commanded back on	23:19 to 23:59 Filter interrogation Polarimetric survey	14:14 to 14:44 14:59 to 19:44 Data accumulation and calibration 20:26 to 20:29 Instrument turned off for *Surveyor VI* shutdown 20:32 to 23:00 Data accumulation and calibration 23:01 Instrument turned off (about 8.5 hr) for television operations
Nov 12, 27.26 deg	03:49 Planar array stepped up 19:58 *Surveyor VI* placed in standby mode 20:00 to 22:08 *Surveyor V* revival attempt by DSS 61 22:26 Surveyor VI standby ended 23:20 Solar panel stepped down to decrease power output and lower battery charging rate	00:00 to 03:37 Polarimetric survey completed Focus ranging: azimuths 0, +36, +72, and −72 deg 04:03 to 07:38 360-deg W/A panorama N/A panorama: segment 4 Special area and auxiliary mirrors N/A panorama: segment 3	07:48 Instrument turned on 07:52 to 08:22 Data accumulation (noisy) 08:25 to 08:59 Calibration to isolate noise source 09:02 to 10:32 Data accumulation (noisy) 10:33 Proton Detector 3 turned off (determined to be noise source) 10:33 to 13:54 Data accumulation 14:06 to 16:00 Data accumulation and calibration 16:15 to 19:46 Data accumulation 19:57 to 22:38 Instrument turned off for spacecraft standby 23:39 Instrument turned off (sensor head reached 122°F upper operational limit); to be turned on at 79-deg sun elevation

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 13, 39.42 deg	14:16 Solar panel stepped down to decrease power output and lower battery charging rate 16:30 Best-lock frequency test (DSS 61) 20:15 *Surveyor VI* placed in standby mode 20:32 to 22:15 *Surveyor V* revival attempt by DSS 61 22:19 *Surveyor VI* standby ended	00:02 to 05:28 Polarimetric survey Special area, magnets, and auxiliary mirrors Focus ranging: azimuths −36, +90, and +108 deg N/A panorama: segments 5 and 2 05:44 to 07:58 N/A panorama: segment 1 360-deg W/A panorama Star survey: Alpha Camelopardalis N/A panorama: segments 1 (with supplement on stop) and 3 08:29 to 09:28 N/A panorama: segments 4 and 5 (with supplement on stop) Auxiliary mirror-surface focus range Earth view test 12:09 to 13:11 Special area and auxiliary mirrors	
Nov 14, 51.56 deg	03:47, 04:13, and 04:21 A/SPP roll axis, planar array (up), and solar panel (down) stepped to shade oxidizer and fuel relief/check valves for pressure and fuel maintenance (nylon valve seats sensitive to extended periods of high temperature) and to partially shade camera 14:23 to 15:25 RF communications test (DSS 42) 15:30 to 16:40 Telecommunications signal processing test (DSS 42) 17:00 Best-lock frequency test (DSS 61) 20:34 Planar array stepped down to further shade oxidizer and fuel relief/check valves 21:00 *Surveyor VI* placed in standby mode 21:05 to 23:26 *Surveyor V* revival attempt by DSS 61 23:35 *Surveyor VI* standby ended	00:33 to 03:32 Iris interrogation Polarimetric survey Special area and magnets 04:33 to 07:13 Auxiliary mirrors 360-deg W/A panorama N/A panorama: segments 1 (with supplement), 2, 3, and 4 07:37 to 09:39 N/A panorama: segment 5 (with supplement) Star survey (negative results): Jupiter and Vega Selected polarimetric Pictures of oxidizer and fuel relief/check valves to confirm shading 13:06 to 13:59 Special area and auxiliary mirrors	

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 15, 63.71 deg	02:57 Planar array stepped up to shade oxidizer and fuel relief/check valves 12:14 Best-lock frequency test (DSS 42)	00:53 to 02:44 Filter interrogation Polarimetric survey 360-deg W/A panorama 03:24 to 10:33 Special area and auxiliary mirrors Auxiliary mirror-focus range (2) 360-deg N/A panorama with supplements in segments 1 and 5 Star survey: Jupiter and Vega Selected polarimetric and polarimetric of chart on Footpad 2 Pictures of oxidizer and fuel relief/check valves Magnets 13:03 to 13:36 Special area	
Nov 16, 75.85 deg	02:56 Solar panel stepped up to shade Oxidizer Tank 3 and Fuel Tank 3 while maintaining shade on oxidizer and fuel relief/check valves 09:46 A/SPP roll axis stepped to shade alpha scattering instrument 50% and Oxidizer Tank 3 20% while maintaining shade on relief valves 10:00 DSS 11 antenna axial ratio test 18:11 Solar panel stepped up to further shade Oxidizer Tank 3 and lower temperature on solar panel	01:25 to 02:46 Pictures of oxidizer and fuel relief/check valves Polarimetric survey Earth picture 03:10 to 07:51 Special area, magnets, and auxiliary mirrors 360-deg W/A panorama 360-deg N/A panorama with supplements in segments 1 and 5 Selected polarimetric Earth picture 08:18 to 09:15 Iris calibration interrogation Alpha scattering survey 12:00 Earth picture (approximate)	12:57 Instrument turned on 12:59 to 16:47 Data accumulation and calibration 16:49 Proton Detector 3 turned back on after cooling to noise-free level 16:50 to 17:44 17:59 to 23:59 Data accumulation and calibration
Nov 17, 87.99 deg lunar noon at 04:00	01:07 and 01:30 Solar panel (down) and A/SPP roll axis stepped to shade Vernier Engines 1 and 3, relief valves, camera, and alpha scattering instrument 03:37 and 03:44 A/SPP roll axis and solar panel (up) stepped to shade flight control sensor group (FCSG), helium relief/check valves, and Vernier Engine 2 05:53 A/SPP roll axis stepped to include Vernier Engine 3 among subsystems being shaded	04:28 to 05:40 Video test on omniantenna and planar array Selected polarimetric in segments 3 and 4 06:00 to 07:50 Selected polarimetric in segments 4 (completed), 5, and 2 09:00 to 09:40 Selected polarimetric in segment 2 completed Selected special area Auxiliary mirrors	00:00 to 01:30 01:46 to 03:30 Final data accumulation from lunar surface 03:32 Instrument turned off

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 17 (contd)	08:00 to 08:35 Dry run of spacecraft liftoff and translation (hop) experiment, including calibration of FCSG active thermal control subsystem to fix time for gyro preheating, and gyro speed checks and precession of each gyro 08:42 and 08:53 A/SPP roll axis and solar panel (down) stepped to shade Vernier Engine 1, leaving Vernier Engines 2 and 3 illuminated and FCSG projected area 25% illuminated 09:45 Flight control power turned on to preheat gyros 10:04 to 10:17 A/SPP roll axis, planar array (down), and solar panel (down) stepped to safe positions for hop experiment, leaving Vernier Engine 2 partially shaded 10:23 to 10:30 Spacecraft configured for hop experiment: telemetry mode 6, Transmitter A in high power (with transponder off), Receiver A, Omniantenna A, touchdown and strain gage power on, touchdown strain gage data channels on, 1100-bits/sec data rate, and nominal thrust (150 lb) bias Pitch gyro precessed +7.1 deg to control direction of lateral movement (toward Leg 1) during hop experiment Thrust phase power turned on		
	10:32 Hop experiment: vernier engines on for 2.5 sec (flight time of 6.078 sec); 10-ft vertical and 8-ft lateral spacecraft movement with 7-deg counterclockwise rotation about roll axis	10:32 Camera mirror remained fully closed during hop experiment to prevent contamination from exhaust and particles	
	10:33 Spacecraft reconfigured for lunar operations 10:46 to 10:54 Solar panel and planar array both stepped up to reacquire sun and earth 12:59 A/SPP roll axis stepped to improve signal strength 20:41 A/SPP roll axis stepped to shade Compartment A 75%, and alpha scattering instrument, relief/check valves, and camera 70%	11:07 to 12:45 360-deg W/A panorama N/A panorama: segment 4 Special area 13:23 to 13:33 Auxiliary mirrors 14:37 to 14:41 Auxiliary mirrors 15:21 to 15:43 N/A panorama: segment 3 using polarized Filter 2	11:41 and 11:43 Television pictures revealed instrument was upside down (radiator in contact with hot lunar surface and cavity exposed to solar and cosmic rays) about 3 in. outboard from nitrogen tank 12:48 Instrument turned on 12:50 to 12:52 Data accumulation from solar protons and cosmic rays

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 17 (contd)		16:57 to 17:49 N/A panorama: segment 3 using polarized Filters 3 and 4 18:35 to 20:30 N/A panorama: segment 4 using polarized Filters 2, 3, and 4 21:00 to 23:07 N/A panorama: segment 5 using polarized Filters 2, 3, and 4	Data indicated one broken curium-242 source-protective film in alpha system and no damage to alpha and proton detectors 12:53 Instrument turned off until temperature dropped below upper operational limit 17:52 Nitrogen tank shadow reached and began cooling instrument
Nov 18, 79.87 deg	06:25 A/SPP roll axis stepped to unshade alpha scattering instrument for television operations 07:03 A/SPP roll axis stepped to resume shading established at 20:41 on Nov 17 14:40 A/SPP roll axis stepped to maintain proper shading 15:30 Planar array stepped down then slightly up to improve signal strength 18:21 and 19:02 A/SPP roll axis and solar panel (up) stepped to improve signal strength and maintain proper shading 19:32 to 19:50 A/SPP roll axis and planar array (down) stepped to improve signal strength 20:20 Best-lock frequency test (DSS 61)	02:50 to 06:11 Polarimetric survey 360-deg W/A panorama Special area, magnets, and auxiliary mirrors 06:33 to 07:00 Pictures of inside of overturned alpha scattering instrument 07:48 to 11:37 N/A panorama: segments 1 and 2 Solar corona calibration Polarimetric survey of Footpad 3 area Star survey: Venus N/A panorama: segment 3 12:31 to 13:44 N/A panorama: segments 3 (completed), 4, and 5 Shadow progression A sequence 16:06 to 16:55 Special area and auxiliary mirrors 17:59 to 18:05 Shadow progression A sequence 20:03 to 20:10 Shadow progression A sequence 21:30 to 23:10 Shadow progression A sequences (2)	00:40 Instrument turned on 00:53 to 02:16 Data accumulation and calibration 02:19 Instrument turned off 14:00 Instrument turned on 14:04 to 15:40 Data accumulation and calibration 17:03 to 17:53 18:59 to 19:29 Data accumulation 19:30 Instrument turned off
Nov 19, 67.73 deg	03:03 Best-lock frequency test (DSS 61) 07:44 A/SPP roll axis stepped to partially shade Compartment A	01:00 to 02:42 Shadow progression A sequences (2) 03:37 to 07:09 Polarimetric survey	

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 19 (contd)	10:04 to 13:02 13:49 *Surveyor VI* placed in standby mode to permit battery cooling 15:22 to 15:50 *Surveyor V* revival attempt by DSS 42 15:52 *Surveyor VI* standby ended 16:26 to 18:16 18:37 to 20:19 20:39 to 22:18 22:38 *Surveyor VI* placed in standby mode to permit battery cooling	Special area, magnets, and auxiliary mirrors 360-deg W/A panorama N/A panorama: segments 1 and 2 08:15 to 09:46 N/A panorama: segment 3 Star survey: Venus and Achernar	
Nov 20, 55.59 deg	00:17 *Surveyor VI* standby ended 00:32 to 02:14 *Surveyor VI* placed in standby mode to permit battery cooling 02:31 Solar panel (down) and A/SPP roll axis stepped to shade relief/check valves and partially shade camera 12:46 to 12:59 Planar array (up), A/SPP roll axis, and planar array (down) stepped to maintain proper alignment with earth for optimum signal strength 13:38 Solar panel stepped up to lower battery charging rate 13:43 to 13:57 A/SPP roll axis, planar array (up), and solar panel (up) stepped to improve automatic gain control level for television operations and to lower battery charging rate 16:20 to 16:35 Best-lock frequency test (DSS 42) 17:55 to 19:15 Solar panel and planar array fine positioning for post-hop experiment attitude determination 20:23 to 22:25 *Surveyor VI* placed in standby mode to permit battery cooling	04:30 to 08:03 Polarimetric survey 360-deg W/A panorama N/A panorama: segments 3 and 4 with polarimetric survey 09:17 to 12:21 Star survey: Rigil Kentaurus, Achernar, and Vega N/A panorama: segment 1, and segment 2 with polarimetric survey Shadow progression sequence N/A panorama: segment 5 Special area and auxiliary mirrors 14:15 to 16:10 Shadow progression sequences (2) 16:45 to 17:45 Shadow progression sequence 19:25 to 19:31 Shadow progression sequence 23:33 to 23:44 Shadow progression sequence	02:44 Instrument turned on 02:55 to 04:17 Data accumulation and calibration 04:21 Instrument turned off

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
	First lunar day (contd)		
Nov 21 43.45 deg	05:10 Decision reached to not perform a second hop experiment due to propulsion subsystem limitations 09:46 A/SPP roll axis stepped to raise battery charging rate 13:37 Solar panel stepped up to maintain desired battery charging rate 21:47 to 22:25 RF communications test (DSS 61) 22:25 to 23:14 Telecommunications signal processing test (DSS 61)	05:30 to 09:31 360-deg W/A panorama Shadow progression sequence Focus ranging: azimuths −90, −108, −126, and −144 deg Shadow progression sequence Polarimetric survey and selected polarimetric survey 10:25 to 13:25 Shadow progression sequence Polarimetric survey completed Special area and auxiliary mirrors N/A panorama: segments 3, 4, 5, and 2 14:35 to 15:07 Star survey: Saturn Shadow progression sequence 15:29 to 16:26 Selected polarimetric survey 17:28 to 19:35 Shadow progression sequences (2) 21:19 to 21:42 Shadow progression sequence	
Nov 22, 31.30 deg	00:40 to 01:50 02:35 to 04:00 Optimum antenna pointing for lunar surface operations 04:12 to 04:18 A/SPP roll axis and planar array (up) stepped to maintain proper alignment with earth for optimum signal strength 04:25 to 04:49 A/SPP elevation axis unlocked; elevation axis, roll axis, and planar array (up) stepped to complete planar array alignment with earth 05:01 Solar panel stepped up to maintain desired battery charging rate 06:05 to 06:24 Solar panel stepped down to maintain desired battery charging rate 16:08 Landing gear locked (gear temperatures had dropped to required 0 to 125° F range) 19:06 Compartment A heater turned on to maintain about 100°F in compartment (initial step in lunar-night planning for thermal control)	02:00 to 02:25 Shadow progression sequence 06:44 to 14:31 Polarimetric survey 360-deg W/A panorama N/A panorama: segments 3 (and polarimetric) and 4 (and polarimetric) Focus ranging: azimuths −72 and −54 deg Shadow progression sequence Focus ranging: azimuths −36 and +108 deg Auxiliary mirror Selected polarimetric Shadow progression sequence Special area 15:32 to 16:00 Special area completed 16:15 to 18:11 Star survey: Rigil Kentaurus, Achernar, and Vega Shadow progression sequence N/A panorama: segments 5 and 2	04:52 Instrument turned on 04:58 to 06:03 Data accumulation and calibration 06:05 Instrument turned off 18:26 Instrument turned on 18:28 to 18:59 Data accumulation and calibration 19:23 to 20:58 21:19 to 21:39 23:18 to 23:38 23:57 to 23:59 Data accumulation

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 22 (contd)	20:23 Solar panel stepped up to maintain desired battery charging rate 23:40 Best-lock frequency test (DSS 61)	19:09 to 19:14 21:04 to 21:09 23:00 to 23:05 Shadow progression sequences (3)	
Nov 23, 19.15 deg	00:15 Compartment A heater off; Compartment B heater on 02:03 Spacecraft engineering mechanisms auxiliary (EMA) test: confirmed locked landing gear by checking squib firing circuit to ensure absence of a short at squib 04:55 Compartment B heater off; Compartment A heater on 07:01 Solar panel stepped up to maintain desired battery charging rate 07:07 AMR heater on 09:35 Solar panel stepped up to maintain desired battery charging rate 16:12 AMR heater off; TV electronics and video heater on 18:46 Compartment A heater off; Compartment B heater on 20:45 Compartment B heater off; Compartment A heater on	01:13 to 01:18 03:08 to 03:13 05:04 to 05:09 Shadow progression sequences (3) 07:21 to 09:20 Polarimetric survey 360-deg W/A panorama N/A panorama: segment 3 (and polarimetric) 09:45 to 11:49 N/A panorama: segment 3 (and polarimetric) completed; segment 4 (and polarimetric) Focus ranging: azimuth, +90 deg 12:20 to 13:59 Focus ranging: azimuths +72, 0, and −18 deg 14:23 to 15:21 Shadow progression A and B sequences Special area and auxiliary mirrors Star survey: Achernar, Rigil Kentaurus, and Vega 16:21 to 18:20 Star survey completed: Achernar, Rigil Kentaurus, and Vega N/A panorama: segment 5 Shadow progression A sequence N/A panorama: segment 2 (and polarimetric) Filter interrogation 18:26 to 18:36 18:56 to 19:01 21:06 to 22:04 23:05 to 23:33 Shadow progression sequences	00:00 to 01:05 Data accumulation and calibration 01:26 to 03:01 03:21 to 04:56 Data accumulation 05:04 Instrument turned off
Nov 24, 7 deg	01:54, 02:09, and 04:13 Compartment C heater on, off, and on 10:09 Solar panel stepped up to maintain desired battery charging rate	01:31 to 01:38 03:28 to 03:36 Shadow progression sequences (2) 06:07 to 07:14 Shadow progression A sequence Special area and auxiliary mirrors	02:11 Instrument turned on 02:14 to 03:22 Data accumulation and calibration 03:41 to 05:57 Data accumulation

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 24 (contd)		07:57 to 09:56 360-deg W/A panorama N/A panorama: segments 3 (and polarimetric) and 4 10:20 to 12:07 N/A panorama: segments 5 and 2 360-deg W/A panorama Shadow progression B sequence N/A horizon scan	
lunar sunset at 13:53		12:33 to 13:52 360-deg W/A panorama and N/A horizon scan 360-deg W/A panorama and N/A horizon scan (sunset)	
lunar night	14:04 Compartment heater off 16:01 to 17:53 TV electronics and video heaters on at intervals 18:02 Compartment C heater and alpha scattering instrument sensor head heater off 19:46 and 20:37 TV video heater and electronics heater off 20:23 to 20:44 A/SPP roll axis, elevation axis, planar array (down), and solar panel (down) stepped to optimum attitude for spacecraft revival on second lunar day 21:34 *Surveyor VI* placed in standby mode	14:11 to 17:11 Solar corona W/A eastern horizon Solar corona Star survey: Antares Solar corona Footpad 2 in earthshine Solar corona 18:08 to 18:38 Star survey: Vega Solar corona 18:54 to 19:28 Solar corona Footpad 2 in earthshine 19:53 to 20:04 Solar corona, 20-min exposure (negative results) 20:06 Camera mirror closed and camera secured for lunar night	17:18 to 17:58 Final data accumulation from solar protons and cosmic rays 18:01 Instrument turned off for lunar night
Nov 25 lunar night	01:32 *Surveyor VI* standby ended 01:47 to 03:01 *Surveyor VI* placed in standby mode, leaving mode 3 and 5 commutators on to provide proper loads for compartment temperature stabilization 03:08 Solar panel stepped down to allow additional 20-hr spacecraft warmup prior to first illumination on solar panel on second lunar day 03:11 to 06:02 06:14 to 07:03 *Surveyor VI* placed in standby mode		

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 25 (contd)	07:08 Compartment A heater on 14:10 to 18:24 *Surveyor VI* placed in standby mode 18:25 Compartment A heater off 18:25 to 19:59 *Surveyor VI* placed in standby mode 20:00 Spacecraft transmitter turned off and AMR heater on to balance loads in attempt to open thermal switches 20:01 to 21:01 *Surveyor VI* placed in standby mode 21:05 Compartment A heater on 21:06 *Surveyor VI* placed in standby mode 22:55 Decision reached to conserve battery energy and therefore to discontinue heater reconfiguring until battery temperature dropped to 0°F or until Compartment B temperature decreased to near −40°F		
Nov 26 lunar night	01:10 *Surveyor VI* standby ended and all heaters turned off 01:19 to 02:00 02:10 to 03:30 *Surveyor VI* placed in shutdown mode 03:32 *Surveyor VI* turned on with good spacecraft signal but no data due to effect of lunar night temperature on signal processing equipment; data transmission resumed following spacecraft reconfiguration 04:00 to 04:52 05:00 to 05:40 *Surveyor VI* placed in shutdown mode 05:41 to 05:52 Final engineering interrogation (battery 1.1°F, 20.5 V, 11.9 psia, with 102 A-hr charge remaining) and estimate of thermal condition (6 Compartment A and 1 Compartment B switches stuck closed) 06:12 to 06:40 Best-lock frequency test (DSS 61)		

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
First lunar day (contd)			
Nov 26 (contd)	06:41 *Surveyor VI* placed in standby mode, leaving all signal processing on 07:45 Unsuccessful attempt to turn on Transmitters A and B due to low temperatures of −35 and −32°F 08:23 to 09:34 Unsuccessful *Surveyor VI* revival attempt 09:35 *Surveyor VI* shut down for lunar night		
Second lunar day[c]			
Dec 13, 44.2 deg	Five unsuccessful *Surveyor VI* revival attempts		
Dec 14	Two unsuccessful *Surveyor VI* revival attempts		
Surveyor VI, 56.3 deg *Surveyor V*, 80.8 deg *Surveyor V* lunar noon at 18:10	06:45 *Surveyor V* signal acquisition followed by loss of signal 07:18 Good DSS 11 receiver lock on Transmitter B (Transponder B/Receiver B not working) Engineering interrogation: incoherent PCM data but good telemetry mode 7 data (TV ID and temperatures) Planar array stepped to raise signal strength and Transmitter A selected to provide optimum signal transmission Periodic *Surveyor V* operations: (1) signal strength measurements; (2) mode 7 interrogations to obtain TV temperatures; (3) ground station receiver VCO frequency readings to estimate Compartment A temperatures; (4) best-lock frequency tests; (5) two-way doppler tracking via Transponder A as schedule permitted 14:00 *Surveyor V* placed in standby mode	Camera turned on and checked: 600-line pictures contained no usable images; 200-line pictures degraded but usable 200-line pictures (67) of horizon, surface, footpads, and spacecraft components	Instrument turned on for brief check: two alpha detectors and one proton detector still operating properly Limited data accumulation and calibration from lunar surface (of engineering value only) Instrument turned off
	Unsuccessful *Surveyor VI* revival attempt		

[c]*Surveyor V* fourth lunar night.

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
Second lunar day (contd)			
Dec 14 (contd)	16:41 *Surveyor VI* revived; erratic signal strength and frequency Engineering interrogation: incoherent telemetry data (sunlight not yet on solar panel for battery charging) 18:25 Good DSS 61 receiver lock following commands to select Transmitter B and Omniantenna B; telemetry data still incoherent 19:08 Commands to select telemetry mode 5 at 137.5-bits/sec data rate resulted in 6 min of good data 19:14 Loss of spacecraft signal just before initiation of A/SPP stepping to permit using solar panel for low-power operations, thereby decreasing amount of energy being dissipated by battery 19:27 *Surveyor VI* placed in shutdown mode, followed by brief and unsuccessful search for spacecraft signal		
	22:00 *Surveyor V* standby ended A/SPP stepped to partially shade Compartment A and camera with solar panel while maintaining proper planar array alignment with earth for optimum signal strength 23:47 *Surveyor V* placed in standby mode		
Dec 15 *Surveyor VI*, 68.4 deg	00:00 to 01:40 Unsuccessful *Surveyor VI* revival attempt		
Surveyor V, 87.1 deg	01:47 *Surveyor V* standby ended Engineering interrogation: incoherent PCM data but good telemetry mode 7 data	200-line pictures (33)	Instrument turned on

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
Second lunar day (contd)			
Dec 15 (contd)	Periodic *Surveyor V* operations: (1) signal strength measurements once each hour; (2) mode 7 interrogation once each hour to obtain TV temperatures; (3) ground station receiver VCO frequency readings once each hour to estimate Compartment A temperatures; (4) best-lock frequency test once each hour; (5) two-way doppler tracking via Transponder A as schedule permitted 17:44 *Surveyor V* placed in standby mode		Limited data accumulation and calibration from lunar surface (of engineering value only) Instrument turned off
	17:55 to 21:55 Unsuccessful *Surveyor VI* revival attempt (first sunlight on solar panel at 15:00)		
	22:04 *Surveyor V* standby ended A/SPP stepping attempts; unchanging signal strength indicated no movement occurred		
Dec 16 *Surveyor VI*, 80.5 deg *Surveyor V*, 75 deg *Surveyor VI*, lunar noon at 18:48	Periodic *Surveyor V* operations: (1) signal strength measurements; (2) mode 7 interrogation to obtain TV temperatures; (3) ground station receiver VCO frequency readings to estimate Compartment A temperatures; (4) two-day doppler tracking via Transponder A as schedule permitted 10:39 Command reject/enable SCO turned on to verify processing of A/SPP stepping commands, indicated receiver indexing with each stepping command Camera and mode 7 commutator turned off to reduce power load Transponders turned off to prevent recurrence of occasional signal loss as receivers indexed (Receiver B/Transponder B nonoperational throughout fourth lunar day) A/SPP stepping command, followed by loss of signal; signal reacquired and stepping discontinued		

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC) Television	Science (SSAC) Alpha scattering instrument
Second lunar day (contd)			
Dec 16 (contd)	18:00 *Surveyor V* placed in shutdown (rather than standby) mode during *Surveyor VI* revival attempt to remove power loads (too-high battery voltage and temperature caused by inability to position solar panel for power-output reduction and Compartment A shading) 18:10 to 21:20 Unsuccessful *Surveyor VI* revival attempt (estimated 14-A-hr charge—sufficient to support one transmitter and one commutator—available by 20:00 after 29 hr of sunlight on solar panel) 21:30 *Surveyor V* turned on 21:35 Command *reject/enable* SCO turned on and A/SPP commanding resumed (indexing continued); although commands entered decoders (no *reject* signals), unchanging signal strength indicated no A/SPP movement occurred		
Dec 17 *Surveyor VI*, 87.3 deg *Surveyor V*, 62.9 deg	02:40 *Surveyor V* solar panel stepping apparently successful as 0.3-db signal strength transient was observed with each effective command; unchanging signal strength during planar array stepping attempts 02:53 A/SPP commanding halted for television operations 03:35 Loss of spacecraft signal following Transmitter A high-power commands to check ability for high-power television support 03:50 Signal reacquisition following Transmitter A low power commands 04:30 Loss of spacecraft signal following second high-power configuration attempt; low-power commands sent but signal reacquisition attempts unsuccessful 04:58 *Surveyor V* placed in shutdown mode	03:00 200-line-mode operations in low power to obtain shadow pictures for verification of A/SPP movement; pictures (3 of omniantenna) were of poor quality due to low power and were unusable for verification purposes	
	05:35 to 13:10 Unsuccessful *Surveyor VI* revival attempt 13:22 to 16:06 Unsuccessful *Surveyor V* revival attempt		

Table A-2 (contd)

Date, 1967, and sun elevation at 00:00 GMT	Engineering (SPAC or DSIF)	Science (SSAC)	
		Television	Alpha scattering instrument
Second lunar day (contd)			
Dec 17 (contd)	16:40 to 22:00 Unsuccessful *Surveyor VI* revival attempt 22:14 *Surveyor V* revival attempt initiated		
Dec 18 *Surveyor VI,* 75.2 deg *Surveyor V,* 50.8 deg	01:00 Unsuccessful *Surveyor V* revival attempt terminated 01:10 to 03:35 Unsuccessful *Surveyor VI* revival attempt Unsuccessful *Surveyor V* and *VI* revival attempts		
Dec 19 *Surveyor VI,* 63.1 deg *Surveyor V,* 38.7 deg	Unsuccessful *Surveyor V* and *VI* revival attempts		
Dec 20 *Surveyor VI,* 51.0 deg *Surveyor V,* 26.6 deg	Unsuccessful *Surveyor V* and *VI* revival attempts		
Dec 21 *Surveyor VI,* 38.9 deg *Surveyor V,* 14.5 deg	Unsuccessful *Surveyor V* and *VI* revival attempts 21:00 Mission terminated		

Appendix B

Surveyor VI Spacecraft Configuration

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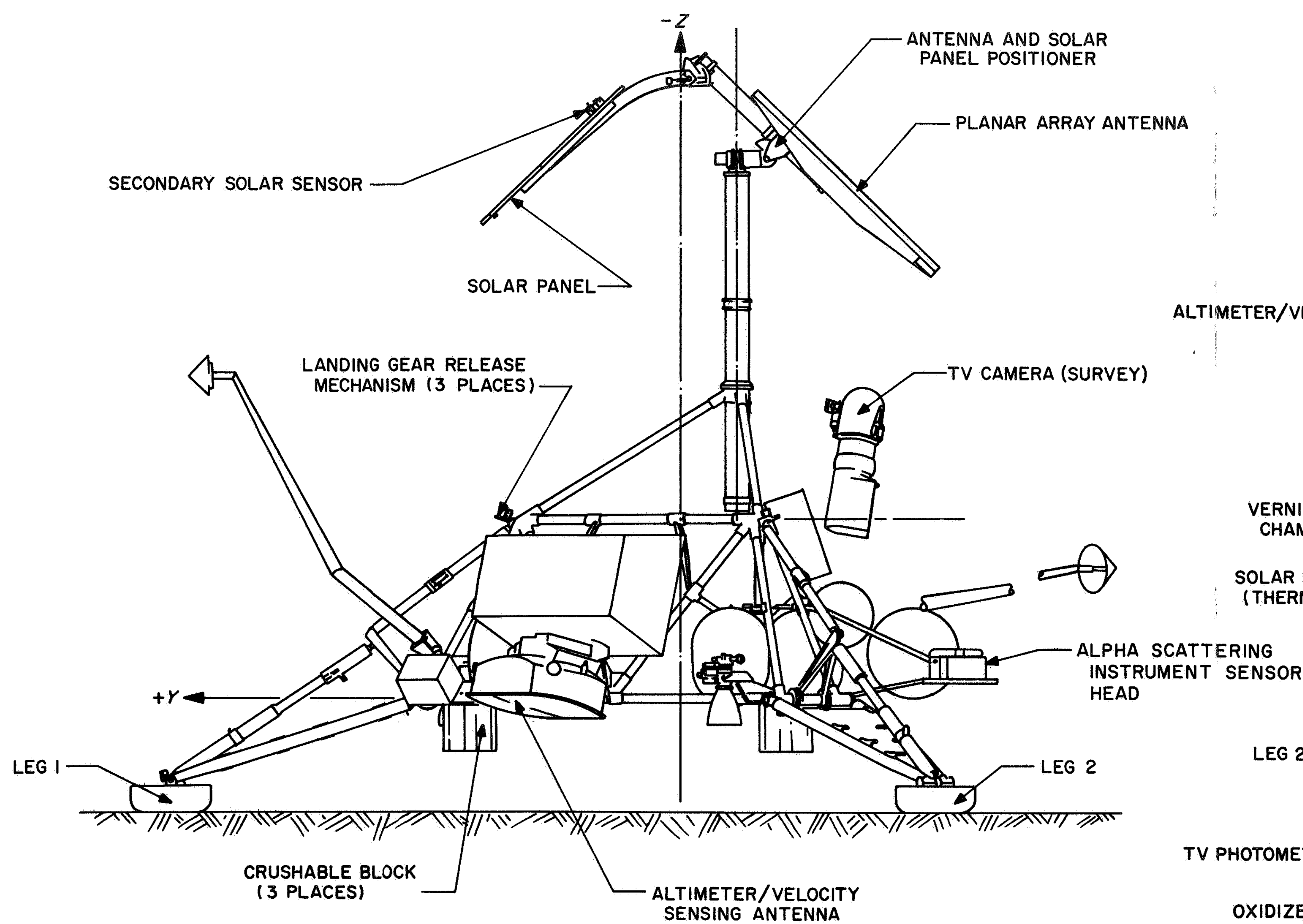
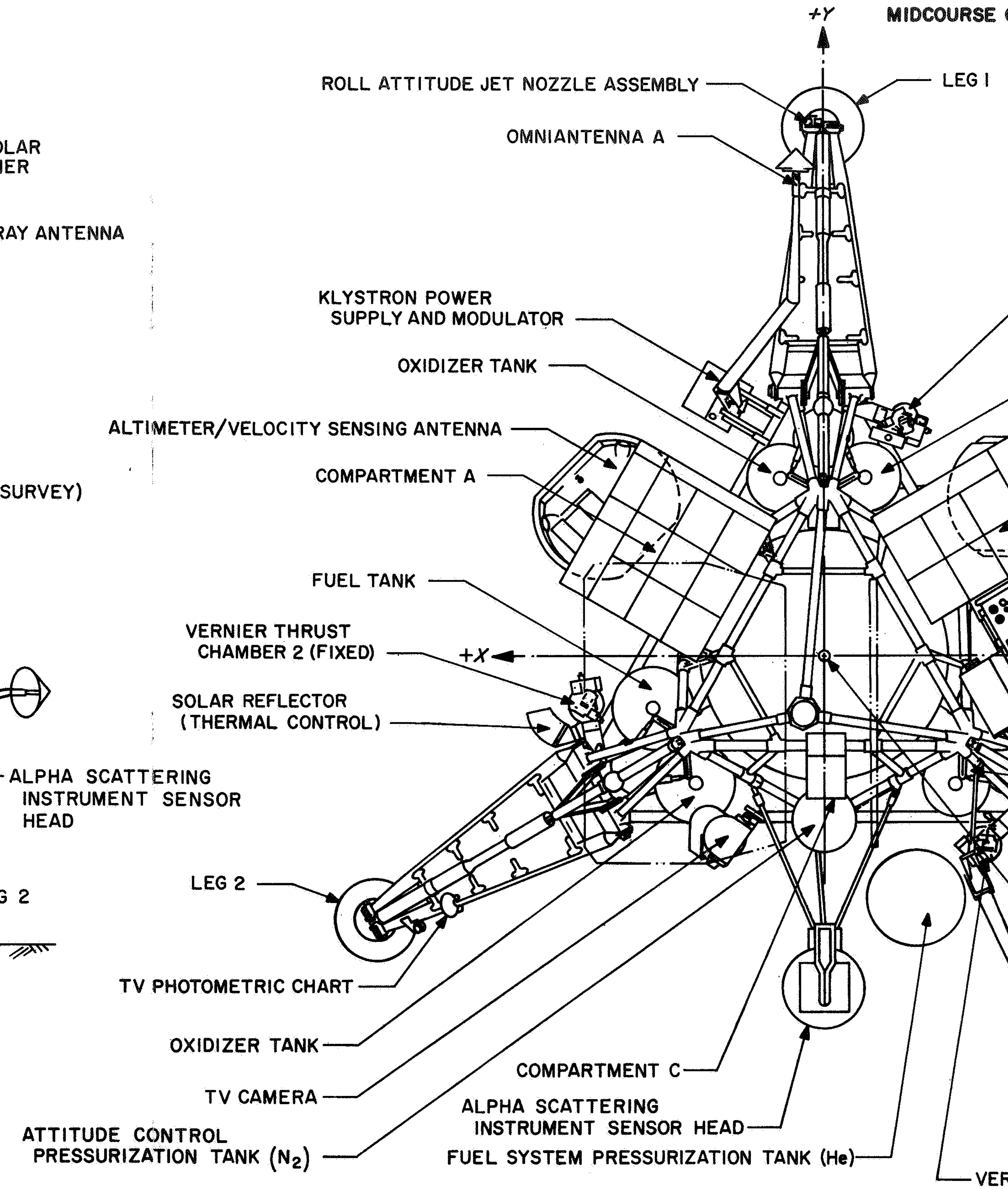
FOLDOUT FRAME

FOLDOUT FRAME 2

POSTLANDING CONFIGURATION



MIDCOURSE CO

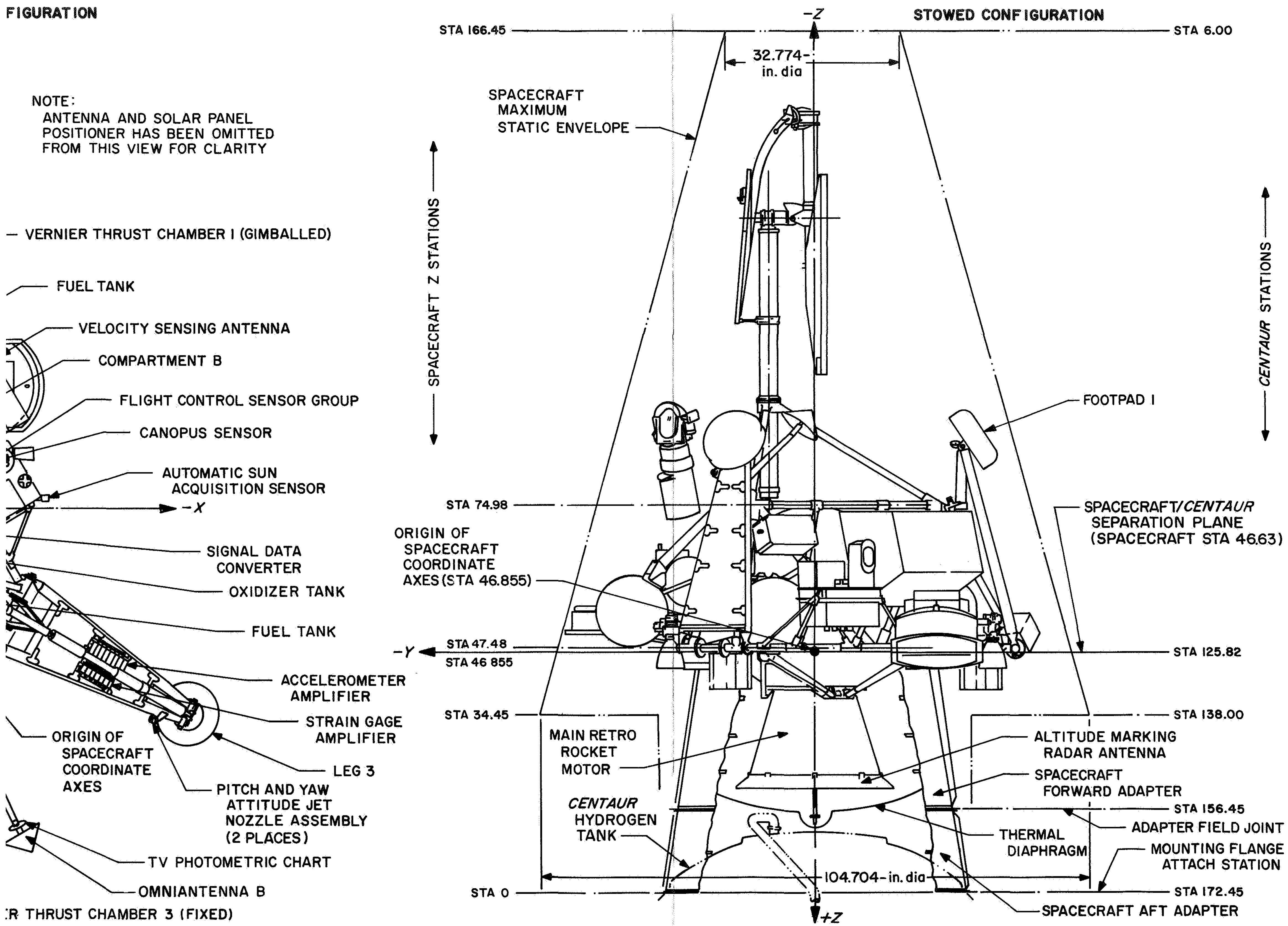


FOLDOUT FRAME 3

FOLDOUT FRAME 4

FIGURATION

STOWED CONFIGURATION



Appendix C

Surveyor VI Spacecraft Content of Telemetry Modes

ALPHA SCATTERING INSTRUMENT

CHAN	NAME	MODE 1 WORDS	MODE 2 WORDS	MODE 3 WORDS	MODE 4 WORDS	MODE C (5) WORDS	MODE T (6) WORDS
AS-3	SENSOR HEAD TEMP.	—	—	—	60	—	—
AS-4	ELECTRONICS TEMP.	—	—	—	70	—	—
AS-5	GUARD EVENT MONITOR	—	—	—	91	—	—
AS-8	+7 VDC MONITOR	—	—	—	37	—	—
AS-9	+24 VDC MONITOR	—	—	—	41	—	—
RADIO AND COMMAND DECODING (DATA LINK)							
D-1	OMNI A TRANSMITTER POWER	—	—	—	1	87	—
D-3	OMNI B TRANSMITTER POWER	—	—	—	11	117	—
D-7	STATIC PHASE ERROR A	66	—	—	3,23,43,63,83	17	—
D-8	STATIC PHASE ERROR B	98	—	—	13,33,53,73,93	27	—
D-9	RECEIVER A AGC	28,78	81	—	81	31	81
D-10	RECEIVER B AGC	38,88	55	—	17	33	79
D-13	TRANSMITTER A TEMPERATURE	—	34	—	2	6	—
D-14	TRANSMITTER B TEMPERATURE	—	44	—	12	30	30
D-16	RECEIVER A AFC	—	61	—	61	37	—
D-17	RECEIVER B AFC	—	15	—	15,35,55,75,95	41	—
ELECTRICAL POWER							
EP-1	29V NON-ESSENTIAL VOLTAGE	90	90	—	27	59	—
EP-2	UNREGULATED BUS VOLTAGE	48	48	48	7,57	69	69
EP-3	MAIN BATT MANIFOLD PRESS.	—	—	—	51	63	—
EP-4	UNREGULATED OUTPUT CURRENT	22	22	—	22	9	9
EP-6	BATTERY CHARGE CURRENT	—	—	—	49	45	—
EP-7	BOOST REG DIFFERENCE CURRENT	—	69	—	69	15	—
EP-8	MAIN BATTERY TEMPERATURE	44	—	—	4	40	40
EP-9	BATTERY DISCHARGE CURRENT	72	72	—	72	29	29
EP-10	SOLAR CELL ARRAY VOLTAGE	—	—	—	47	67	—
EP-11	SOLAR CELL ARRAY CURRENT	59	59	—	59	25	—
EP-12	SOLAR CELL ARRAY TEMPERATURE	—	—	—	8	36	—
EP-13	BOOST REGULATOR TEMPERATURE	24	—	—	10	20	20
EP-14	REGULATED OUTPUT CURRENT	42	42	—	42	39	39
EP-17	RADAR AND SQUIB CURRENT	—	89	—	89	49	49
EP-21	COMPARTMENT A HEATER CURRENT	—	—	—	9	—	—
EP-22	COMPARTMENT B HEATER CURRENT	—	—	—	29	—	—
EP-30	BOOST REG. PREREG. VOLTAGE	64	21	—	21	93	—
EP-34	BATTERY CHARGE REG. TEMP.	—	—	—	78	46	—
EP-35	SOLAR CELL SHORT CKT. CURRENT	—	—	—	—	35	—
EP-36	IRRADIATED SOLAR CELL S.C. CURRENT	—	—	—	—	65	—
EP-37	SOLAR CELL SECTION 2 CURRENT	79	79	—	79	—	—
EP-38	SOLAR CELL OPEN CKT. VOLTAGE	—	—	—	—	55	—
EP-40	FLIGHT CONTROL UNREGULATED CURRENT	32	32	—	32	19	19
FLIGHT CONTROL							
FC-4	NITROGEN GAS PRESSURE	—	—	—	—	89	—
FC-5	PRIM. SUN SENSOR PITCH ERROR	10,60	—	—	—	71	—
FC-6	PRIM. SUN SENSOR YAW ERROR	20,70	—	—	—	77	—
FC-7	SECONDARY SUN SENSOR CELL A	—	27	—	—	99	—
FC-8	SECONDARY SUN SENSOR CELL B	—	35	—	—	101	—
FC-9	SECONDARY SUN SENSOR CELL C	—	47	—	—	105	—
FC-10	SECONDARY SUN SENSOR CELL D	—	37	—	—	107	—
FC-12	CANOPUS ERROR	7,27,47,67,87	—	—	—	109	—
FC-14	STAR INTENSITY SIGNAL	9,29,49,69,89	—	—	—	115	—
FC-15	ACCELERATION ERROR	11,31,51,71,91	11	11	—	—	6,36,66,96
FC-16	PITCH GYRO ERROR	12,62	12,73	12	—	1	1,31,61,91
FC-17	YAW GYRO ERROR	18,68	18,77	18	—	3	3,33,63,93
FC-25	THRUST CMD TO VERNIER ENG 1	15,35,55,75,95	9,63	9,19,29,39	—	—	17,47,77,107
FC-26	THRUST CMD TO VERNIER ENG 2	16,36,56,76,96	1,49	1,21,31,41	—	—	15,45,75,105
FC-27	THRUST CMD TO VERNIER ENG 3	3,23,43,63,83	2,52	2,22,32,42	—	—	16,46,76,106
FC-32	RETRO ACCELEROMETER	—	7,57	24	—	—	8,38,68,98
FC-35	RADAR ALTIMETER RANGE SIGNAL	—	74	44	—	—	18,48,78,108
FC-39	DOPPLER VELOCITY Vx	—	51	38	—	—	24,54,84,114
FC-40	DOPPLER VELOCITY Vy	—	43,93	34	—	—	25,55,85,115
FC-41	DOPPLER VELOCITY Vz	—	3,53	3,23,33,43	—	—	26,56,86,116
FC-43	ROLL ACTIVATOR SIGNAL	1,21,41,61,81	41,91	7,27,37,47	—	—	27,57,87,117
FC-44	FLIGHT CONTROL E/U TEMP 1	—	70	—	—	—	—
FC-45	FLIGHT CONTROL E/U TEMP 2	—	24	—	—	18	—
FC-46	ROLL GYRO TEMPERATURE	—	60	—	—	68	—
FC-47	CANOPUS SENSOR TEMPERATURE	—	—	—	—	48	—
FC-48	NITROGEN GAS TANK TEMP.	—	—	—	18	82	—
FC-49	ROLL PRECESSION COMMAND	17,37,57,77,97	17	17	—	5	5,35,65,95
FC-50	PITCH PRECESSION COMMAND	—	10	—	—	7	58
FC-51	YAW PRECESSION COMMAND	—	87	—	—	11	109
FC-53	SEC. SUN SENSOR CENTER CELL	—	29	—	—	97	—
FC-54	PITCH GYRO TEMPERATURE	—	20	—	—	110	110
FC-55	YAW GYRO TEMPERATURE	—	80	—	—	98	—
FC-70	ATTITUDE GAS JET 2 TEMP	—	—	—	—	102	—
FC-71	ROLL ACTUATOR TEMPERATURE	—	—	—	—	104	—
FC-77	FLT. CONT'L REFERENCE RETURN	—	23	—	—	—	99
MECHANISMS							
M-3	SOLAR PANEL POSITION	—	—	—	71	57	—
M-4	POLAR AXIS POSITION	—	98	—	98	61	—
M-6	ELEVATION AXIS POSITION	—	—	—	77	79	—
M-7	ROLL AXIS POSITION	—	—	—	87	91	—
M-8	PLANAR ARRAY TEMPERATURE	—	—	—	14	38	—
M-10	SOLAR PANEL STEP. MOTOR TEMP.	—	—	—	58	44	—
M-12	ELEV. AXIS STEP. MOTOR TEMP	—	—	—	64	106	—
SPARE	(VOLTAGE)	—	—	—	5,25,45,65,85	—	—
SPARE	(HIGH-ACC. TEMP)	—	—	—	80,90	—	—
CHAN	NAME	MODE 1 WORDS	MODE 2 WORDS	MODE 3 WORDS	MODE 4 WORDS	MODE C (5) WORDS	MODE T (6) WORDS

PROPULSION

CHAN	NAME	MODE 1 WORDS	MODE 2 WORDS	MODE 3 WORDS	MODE 4 WORDS	MODE C (5) WORDS	MODE T (6) WORDS
P-1	HELIUM PRESSURE	84	25	—	—	21	21
P-2	OXIDIZER PRESSURE 3	30	30	—	—	51	51
P-3	UPPER RETRO CASE TEMPERATURE	—	68	—	—	52	—
P-4	VERNIER LINES 2 TEMPERATURE	2	—	—	—	80	80
P-5	VERNIER FUEL TANK 2 TEMP.	—	66	—	—	12	—
P-6	VERNIER OXIDIZER TANK 3 TEMP.	—	86	—	—	56	—
P-7	VERNIER ENGINE 1 TEMPERATURE	26	—	—	—	70	70
P-8	VERNIER LINES 1 TEMPERATURE	52	—	—	—	90	90
P-9	VERNIER LINES 3 TEMPERATURE	82	—	—	—	100	100
P-10	VERNIER ENGINE 2 TEMPERATURE	80	—	—	—	50	50
P-11	VERNIER ENGINE 3 TEMPERATURE	46	—	—	—	60	60
P-12	LOWER RETRO CASE TEMPERATURE	—	88	—	—	28	—
P-13	VERNIER FUEL TANK 1 TEMP.	—	56	—	—	4	—
P-14	VERNIER FUEL TANK 3 TEMP.	—	76	—	—	54	—
P-15	VERNIER OXIDIZER TANK 1 TEMP.	—	82	—	—	92	—
P-16	VERNIER OXIDIZER TANK 2 TEMP.	—	84	—	—	2	—
P-17	HELIUM TANK TEMP.	—	—	—	20	86	—
P-18	VERNIER ENGINE 1 STRAIN GAGE	13,33,53,73,93	13	13	—	—	7,37,67,97
P-19	VERNIER ENGINE 2 STRAIN GAGE	14,34,54,74,94	14	14	—	—	11,41,71,101
P-20	VERNIER ENGINE 3 STRAIN GAGE	8,58	8	8	—	—	28,88
P-22	RETRO NOZZLE TEMPERATURE	—	—	—	—	42	—
P-23	VERNIER FUEL LINE #1 TEMP	—	78	—	—	—	—
P-24	VERNIER FUEL LINE #2 TEMP	—	—	—	—	84	—
P-25	VERNIER FUEL LINE #3 TEMP	—	58	—	—	—	—
P-26	VERNIER FUEL LINE PRESSURE	86	—	—	—	—	—
RADAR							
R-2	RADVS-R AMPLITUDE	—	26	26	—	—	4,64
R-3	RADVS-D_1 AMPLITUDE	—	28	28	—	—	44,104
R-4	RADVS-D_2 AMPLITUDE	—	36	36	—	—	14,74
R-5	RADVS-D_3 AMPLITUDE	—	46	46	—	—	34,94
R-6	AMR ANTENNA TEMPERATURE	—	54	—	—	—	—
R-7	AMR ELECTRONICS TEMPERATURE	—	64	—	—	26	—
R-8	RADVS KLY. UNIT TEMP.	—	38	—	—	10	10
R-9	RADVS SIG. DATA CONVERT. TEMP.	—	62	—	—	22	—
R-10	DOPPLER RADAR SENSOR TEMP.	—	94	—	—	76	—
R-12	AMR PWR OUTPUT (MAG CURRENT)	—	—	—	—	—	59
R-13	ALTMETER RADAR SENSOR TEMP.	—	96	—	—	24	—
R-14	ALTITUDE MARKING RADAR AGC	—	33	—	—	—	43
R-29	AMR LATE GATE SIGNAL	—	71	—	—	—	89
SIGNAL PROCESSING							
S-1	REFERENCE VOLTAGE	—	67	—	67	—	—
S-2	REFERENCE RETURN	—	97	—	97	—	—
S-5	COMMUTATOR UNBAL. CURR. (ESP)	—	31	—	31	—	—
S-7	COMMUTATOR UNBAL. CURR. (AESP)	—	—	—	—	47	—
S-8	AESP REFERENCE VOLTAGE	—	—	—	—	81	—
EP-18	FULL SCALE CURR. CALIB. (ESP)	92	92	—	92	—	—
EP-19	MID-SCALE CURR. CALIB. (ESP)	39	39	—	39	—	—
EP-20	ZERO SCALE CURR. CALIB. (ESP)	19	19	—	19	—	—
EP-27	FULL SCALE CURR. CALIB. (AESP)	—	—	—	—	75	—
EP-28	MID-SCALE CURR. CALIB. (AESP)	—	—	—	—	85	—
EP-29	ZERO SCALE CURR. CALIB. (AESP)	—	—	—	—	95	—
TELEVISION							
TV-16	SURV. CAM. ELECTRONICS TEMP.	—	—	—	16	62	—
TV-17	SURV. CAM. MIRROR ASSY. TEMP.	—	—	—	82	116	—
STRUCTURES (VEHICLE)							
V-5	LANDING GEAR 1 DEFLECTION	—	75	—	—	—	—
V-6	LANDING GEAR 2 DEFLECTION	—	95	—	—	—	—
V-7	LANDING GEAR 3 DEFLECTION	—	83	—	—	—	—
V-15	COMP. A TEMP. THERMAL TRAY TOP	—	—	—	24	66	—
V-16	COMP. A TEMP. LOWER SUPPORT	—	—	—	26	78	—
V-18	COMP. A TEMP. CANISTER	—	—	—	30	64	—
V-19	COMP. A TEMP. SW. 5 IN CONT. RING	—	—	—	34	—	—
V-20	COMP. A TEMP. SW. 5 IN FACE RAD.	—	—	—	36	32	—
V-21	COMP. B TEMP. TRAY TOP CENTER	—	—	—	44	58	—
V-22	COMP. B TEMP. LOWER SUPPORT	—	—	—	46	74	—
V-23	COMP. B TEMP. CANISTER	—	—	—	48	—	—
V-24	COMP. B TEMP. SW 4 IN FACE RAD.	—	—	—	52	34	—
V-25	COMP. A TEMP. SW 8 IN FACE RAD.	—	—	—	38	94	—
V-26	COMP. B TEMP. SW 4 IN CONT. RING	—	—	—	54	—	—
V-27	UPPER SPACEFRAME TEMP. NO. 1	—	—	—	56	—	—
V-28	SPACEFRAME TEMP. UNDER COMP. A	—	—	—	62	14	—
V-29	WIRE HARN. TEMP. THERM TUNNEL	—	—	—	66	—	—
V-30	SHOCK ABSORB 1 TEMPERATURE	—	—	—	68	16	—
V-32	SHOCK ABSORB 2 TEMPERATURE	—	—	—	—	8	—
V-33	SHOCK ABSORB 3 TEMPERATURE	—	—	—	76	96	—
V-35	UPPER SPACEFRAME TEMP. NO. 2	—	—	—	74	—	—
V-36	SPACEFRAME TEMP. UNDER COMP. 3	—	—	—	96	—	—
V-37	RETRO ATTACH POINT 1 TEMP.	—	—	—	84	—	—
V-38	RETRO ATTACH POINT 2 TEMP.	—	—	—	—	72	—
V-39	RETRO ATTACH POINT 3 TEMP.	—	—	—	88	—	—
V-44	CRUSHABLE BLOCK TEMP.	—	—	—	—	88	—
V-45	COMP. B TEMP. SW 1 IN FACE RAD.	—	—	—	—	108	—
V-46	COMP. B TEMP. SW 5 IN FACE RAD.	—	—	—	94	112	—
V-47	COMP. A TEMP. SW 2 IN FACE RAD.	—	—	—	86	114	—
V-49	COMP. A TEMP. SW 3 IN FACE RAD.	—	—	—	28	—	—
CHAN	NAME	MODE 1 WORDS	MODE 2 WORDS	MODE 3 WORDS	MODE 4 WORDS	MODE C (5) WORDS	MODE T (6) WORDS

* COMPUTER ID NO. FOR PCM ANALOG SIGNALS/DIGITAL WORDS

COMMUTATED BINARY SIGNALS

RADIO AND COMMAND DECODING (DATA LINK)

CHAN	NAME	DIGITAL WORD (BIT)
D-2	TRANSMITTER A FILAMENT ON	6,10(5)
D-4	TRANSMITTER B FILAMENT ON	6,10(6)
D-5	TRANSPONDER A PHASE LOCKED	6,10(3)
D-6	TRANSPONDER B PHASE LOCKED	6,10(4)
D-11	MESSAGE REJECT	5,6,10,12(2)
D-12	CENTRAL DECODER A ON	6,10(8)
D-15	MESSAGE ENABLE	5,6,10,12(1)
D-18	CENTRAL DECODER B ON	6,10(7)
ELECTRICAL POWER		
EP-33	EMA RADVS POWER SWITCH ON	11(8),5(4)
EP-39	BCR TO UNREG. BUS	10(9)
FLIGHT CONTROL		
FC-13	CANOPUS LOCK-ON	1(6)
FC-18	MAGNITUDE REGISTER READOUT	2
FC-19	PROGRAMMER ROLL SIGNAL	3(1)
FC-21	PROGRAMMER PITCH SIGNAL	3(3)
FC-23	PROGRAMMER YAW SIGNAL	3(5)
FC-28	VERNIER IGNITION SIGNAL	3(10),4(3)
FC-29	RETRO IGNITION SIGNAL	4(4)
FC-30	RETRO BURNOUT SIGNAL	4(5)
FC-31	RETRO EJECT SIGNAL	4(6)
FC-33	RELIABLE OPERATION-RADAR ALT	9(9)
FC-34	RELIABLE OPERATION-DOPP. RADAR	9(10)
FC-36	TEN FOOT PER SECOND MARK	4(9)
FC-37	ONE THOUSAND FOOT MARK	4(8)
FC-38	THIRTEEN FOOT MARK	4(10)
FC-42	START RADVS CONTR'LD DESCENT	4(7)
FC-52	NOMINAL ENGINE THRUST BIAS	9(8)
FC-56	MANUAL DELAY MODE ON	3(2)
FC-57	GAS JET AMPL. INHIBITED	1(8)
FC-58	RATE MODE ON	9(1)
FC-59	SUN MODE ON	9(2)
FC-60	STAR MODE ON	9(3)
FC-61	CRUISE MODE ON	9(4)
FC-62	RETRO SEQ. MODE ON	4(1)
FC-63	INERTIA SWITCH SIGNAL	9(5)
R-28	RADVS ON SIGNAL	3(6)
FC-65	M/C VELOCITY CORRECTION SIG.	3(9)
FC-66	THRUST PHASE POWER ON	3(4)
FC-67	PROG. POS. ANGLE MANEUVER SIG.	3(7)
FC-69	PRIM SUN SENSOR CNTR. CELL L-O	1(5)
FC-73	MAN. L-O (SUN OR STAR)	9(7)
FC-74	PITCH GYRO HTR. CONTROL OFF	1(9)
FC-75	ROLL GYRO HTR. CONTROL OFF	1(10)
FC-76	ACQUISITION SUN SENSOR ILLUM.	1(7)
FC-78	VERNIER ENGINE HIGH THRUST	9(6)
FC-79	YAW GYRO HEATER CONTROLLER OFF	3(8)
MECHANISMS		
M-1	OMNI ANTENNA A EXTENDED	7,11(9)
M-2	OMNI ANTENNA B EXTENDED	7,11(10)
M-9	VEHICLE ELECTRICAL DISCONNECT	1(4)
M-11	SOLAR PANEL LOCK IN TRANS. POS.	7,11(3)
M-13	ROLL AXIS LOCK IN TRANS. POS.	7,11(4)
M-14	SOLAR PANEL LAUNCH LOCK UNLOCKED	7,11(2)
M-15	SOLAR PANEL DEPLOY. LOGIC ON	7,11(7)
RADAR		
R-1	ALTITUDE MARKING RADAR ON	5,12(7)
R-11	AMR ENABLED	5,12(8)
R-15	RADVS-D1 TRACKER LOCK	8,13(1)
R-16	RADVS-D2 TRACKER LOCK	8,13(10)
R-17	RADVS-D3 TRACKER LOCK	5,12(3)
R-18	RADVS-R TRACKER LOCK	5,12(6)
R-19	RADVS-D1 GAIN 1 (65 TO 40 db)	8,13(2)
R-20	RADVS-D1 GAIN 2 (90 TO 65 db)	8,13,(3)
R-21	RADVS-D2 GAIN 1 (65 TO 40 db)	8,13(4)
R-22	RADVS-D2 GAIN 2 (90 TO 65 db)	8,13(5)
R-23	RADVS-D3 GAIN 1 (65 TO 40 db)	8,13(6)
R-24	RADVS-D3 GAIN 2 (90 TO 65 db)	8,13(7)
R-25	RADVS-R GAIN 1 (60 TO 40 db)	8,13(8)
R-26	RADVS-R GAIN 2 (80 TO 60 db)	8,13(9)
FC-64	ALTITUDE MARKING RADAR SIG.	4(2)
SIGNAL PROCESSING		
S-3	COMUT. SEL. F/F R500 (ESP MODE 2-3)	5,6(10)
S-4	COMUT. SEL. F/F R501 (ESP MODE 3-4)	5,6(9)
S-6	COMUT. SEL. F/F R500 (AESP MODE C)	10(10)
TELEVISION		
TV-18	VID. SURV. HEATER OFF	11(5), 7(8)
CHAN	NAME	DIGITAL WORD (BIT)

ALPHA SCATTERING INSTRUMENT

CHAN	NAME	DIGITAL WORD (BIT)
AS-6	ALPHA CMD.MEM. VERIF.	7(6)
AS-7	PROTON CMD. MEM. VERIF.	7(5)
STRUCTURES (VEHICLE)		
V-1	LANDING GEAR 1 EXTENDED	1(1)
V-2	LANDING GEAR 2 EXTENDED	1(2)
V-3	LANDING GEAR 3 EXTENDED	1(3)
V-4	RETRO NOT EJECTED	5,12(5)
SPARE	(ESP)	7(1)
SPARE	(AESP)	11(1,6),12(49,10)
CHAN	NAME	DIGITAL WORD (BIT)

DIGITAL WORD (DW) SUMMARY

BIT	DW1	DW3	DW4	DW5	DW6	DW7
1	V-1	FC-19	FC-62	D-15	D-15	
2	V-2	FC-56	FC-64	D-11	D-11	M-14
3	V-3	FC-21	FC-28	R-17	D-5	M-11
4	M-9	FC-66	FC-29	EP 33	D-6	M-13
5	FC-69	FC-23	FC-30	V-4	D-2	AS-7
6	FC-13	R-28	FC-31	R-18	D-4	AS-6
7	FC-76	FC-67	FC-42	R-1	D-18	M-15
8	FC-57	FC-79	FC-37	R-11	D-12	TV-18
9	FC-74	FC-65	FC-36	S-4	S-4	M-1
10	FC-75	FC-28	FC-38	S-3	S-3	M-2

BIT	DW8	DW9	DW10	DW11	DW12	DW13
1	R-15	FC-58	D-15		D-15	R-15
2	R-19	FC-59	D-11	M-14	D-11	R-19
3	R-20	FC-60	D-5	M-11	R-17	R-20
4	R-21	FC-61	D-6	M-13		R-21
5	R-22	FC-63	D-2	TV-18	V-4	R-22
6	R-23	FC-78	D-4		R-18	R-23
7	R-24	FC-73	D-18	M-15	R-1	R-24
8	R-25	FC-52	D-12	EP-33	R-11	R-25
9	R-26	FC-33	EP-39	M-1		R-26
10	R-16	FC-34	S-6	M-2		R-16

COMMUTATED DIGITAL WORDS

DIGITAL WORD	COMMUTATOR WORDS					
	MODE 1	MODE 2	MODE 3	MODE 4	MODE C (5)	MODE T (6)
1	40	40	---	40	23	23
2	50	50	---	---	53	53
3	5, 25, 45, 65, 85	65	---	---	13, 43, 73, 103	13
4	---	5, 45, 85	5, 10, 15, 20, 25, 30, 35, 40, 45	---	---	2,12,22,32,42,52,62 72,82,92,102,112
5	---	6	6	---	---	---
6	6	---	---	6	---	---
7	---	---	---	50	---	---
8	---	16	16	---	---	---
9	4	4	4	---	83	83
10	---	---	---	---	111	111
11	---	---	---	---	113	113
12	---	---	---	---	---	73
13	---	---	---	---	---	103
SYNC	(100)	(100)	(50)	(100)	118,120	118,120
SYNC COMP	(99)	(99)	(49)	(99)	119	119

TV COMMUTATOR

CHAN	NAME	MODE 7 WORDS
TV-1	CAMERA NUMBER	2
TV-2	MIRROR AZIMUTH	3,11
TV-3	MIRROR ELEVATION	7
TV-4	LENS FOCAL LENGTH SETTING	12
TV-5	LENS FOCAL SETTING	5
TV-6	LENS IRIS SETTING	13
TV-7	LENS IRIS SERVO ON	15
TV-8	FILTER WHEEL POSITION	10
TV-9	VIDICON FACEPLATE TEMP.	8
TV-10	CAMERA ELECTRONICS TEMP.	4
TV-11	SHUTTER MODE (OPEN OR NORMAL)	9
TV-12	CALIBRATE VOLTAGE	6,14
TV-13	MULTIPLE STEP FOCUS ON	16
SYNC	(BARKER WORD)	1

SUBCARRIER SUMMARY

CHAN	NAME	SCO FREQ. (KC)	CARRIER MOD. MODE *
D-11,15	COMMAND REJECT/ENABLE	2.3	PM
FC-1	ROLL GYRO SPEED		
FC-2	PITCH GYRO SPEED	5.4	PM
FC-3	YAW GYRO SPEED		
AS-1	ALPHA PARTICLE COUNT (2200 BPS)	70	PM
AS-2	PROTON COUNT (550 BPS)	5.4	
V-12	LEG 1 SHOCK ABSORBER STRAIN GAGE	1.7	
V-13	LEG 2 SHOCK ABSORBER STRAIN GAGE	1.3	PM
V-14	LEG 3 SHOCK ABSORBER STRAIN GAGE	.96	
TV-14	COMPOSITE VIDEO	CARRIER	FM

COMMUTATED PCM DATA

BIT RATE (BPS)	SCO CENTER FREQ. (KC)	CARRIER MODULATION
4400	33	FM OR PM
1100	7.35	FM OR PM
550	3.9	FM OR PM
550 (LOW MOD INDEX)	3.9	PM
137.5	.96	PM
17.2	.56	PM
4400 (NORMAL MODE TVID)	CARRIER	FM

* BRACKETED SCO'S TURNED ON BY A SINGLE COMMAND

Appendix D

Surveyor VI Spacecraft Temperature Histories

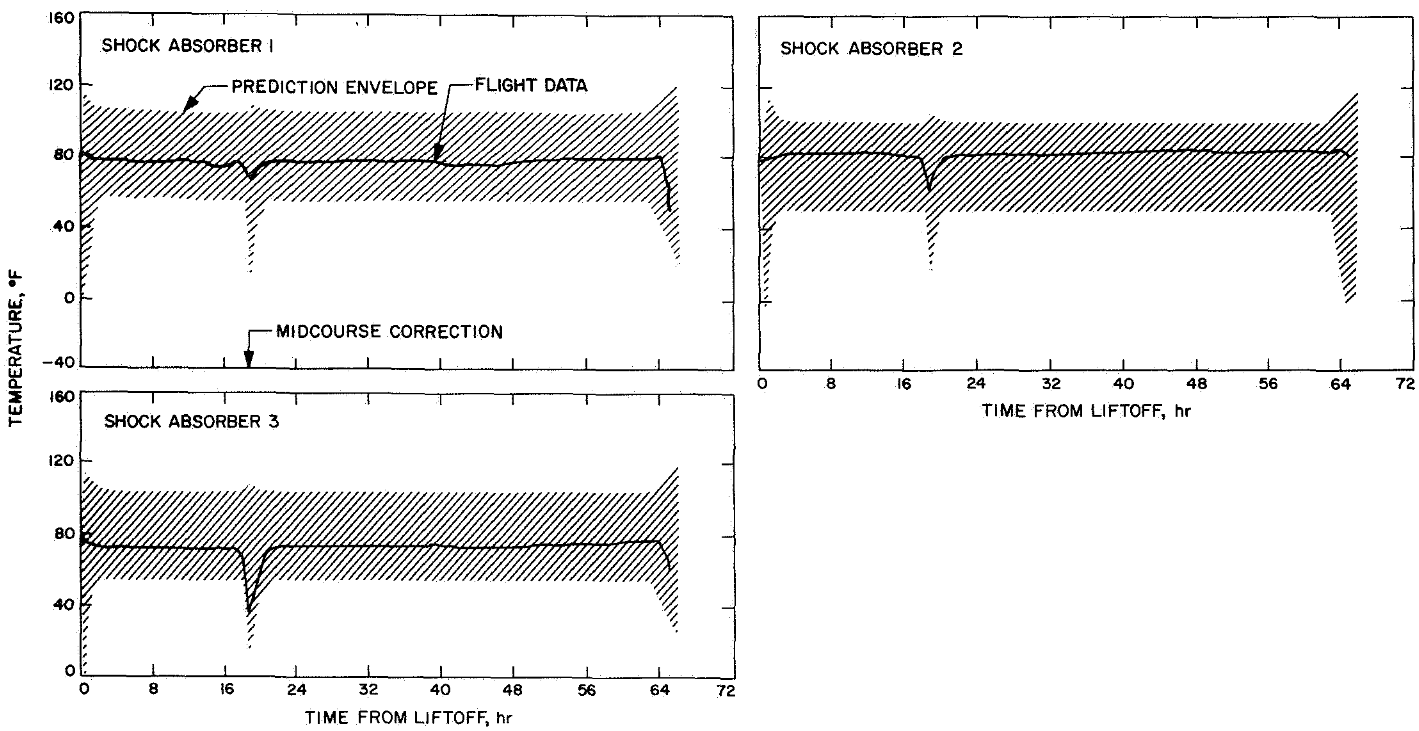


Fig. D-1. Landing gear transit temperatures

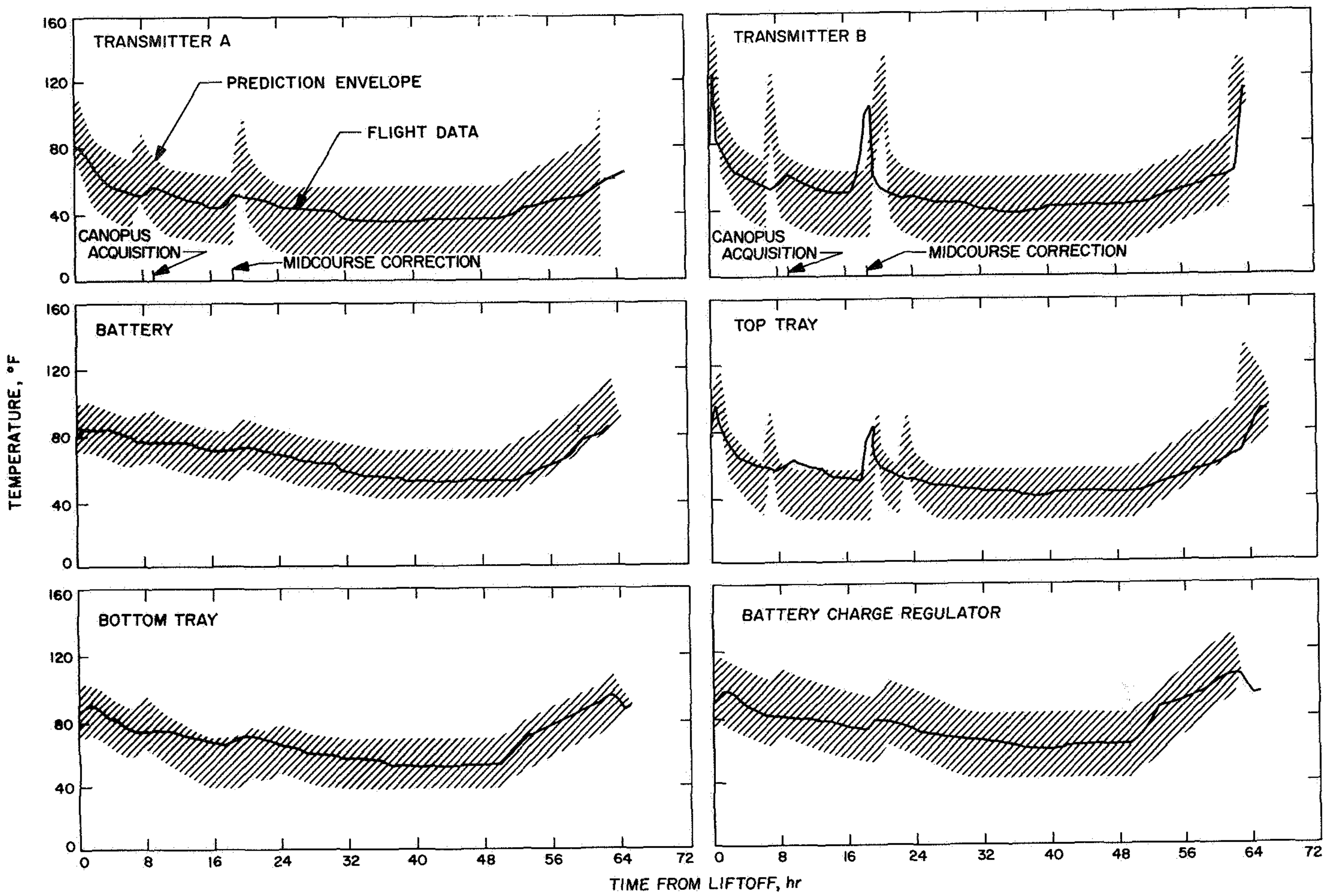


Fig. D-2. Compartment A transit temperatures

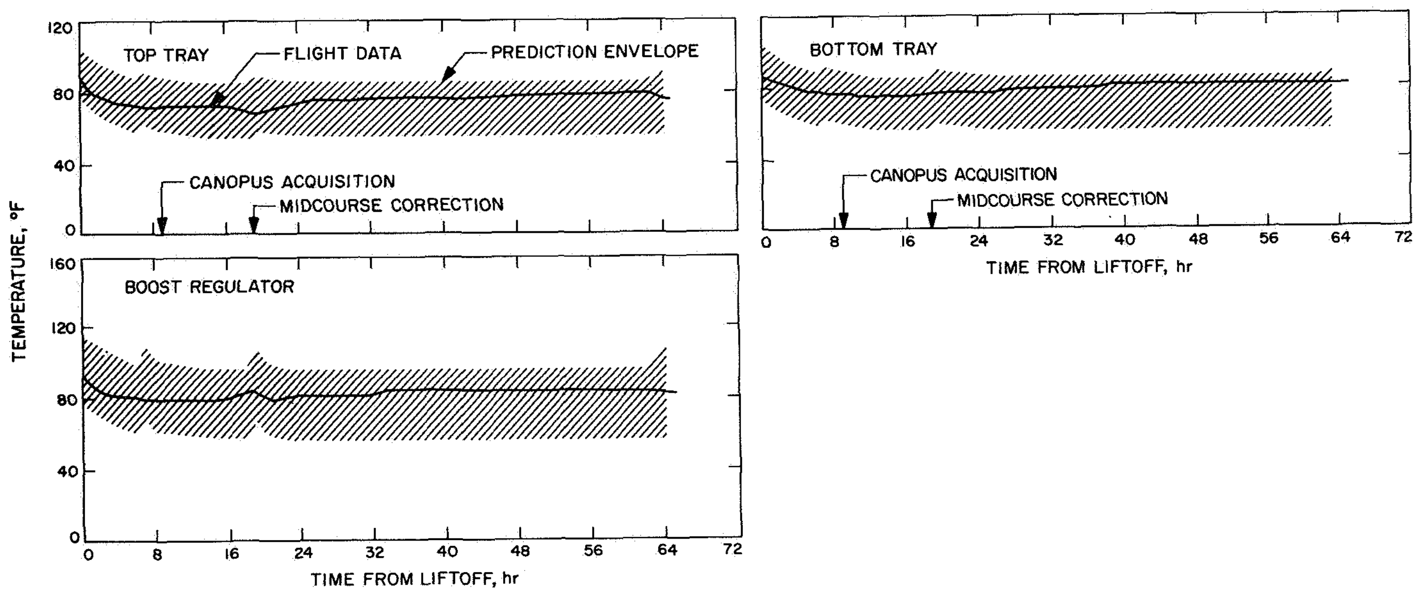


Fig. D-3. Compartment B transit temperatures

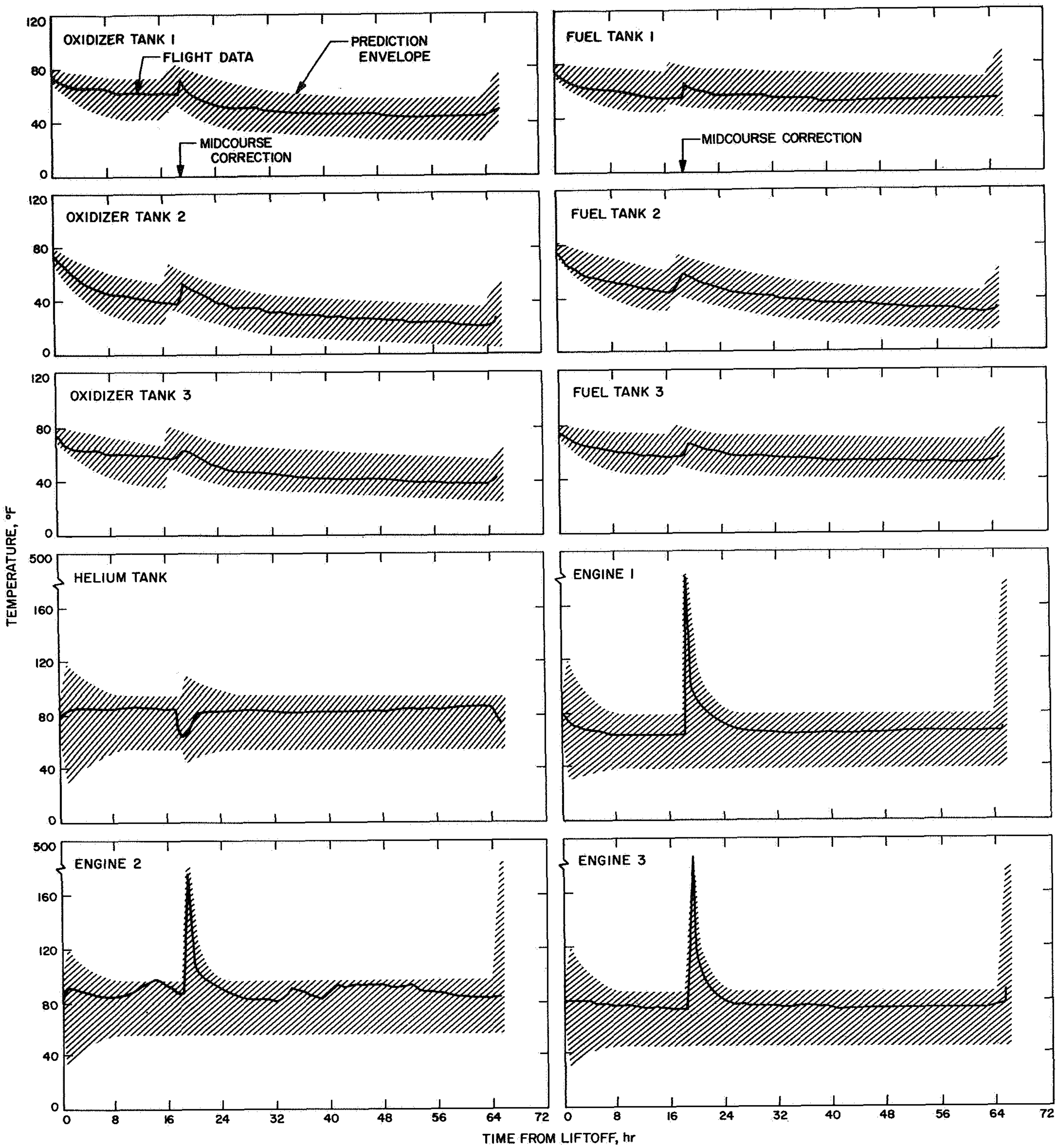


Fig. D-4. Vernier propulsion system transit temperatures

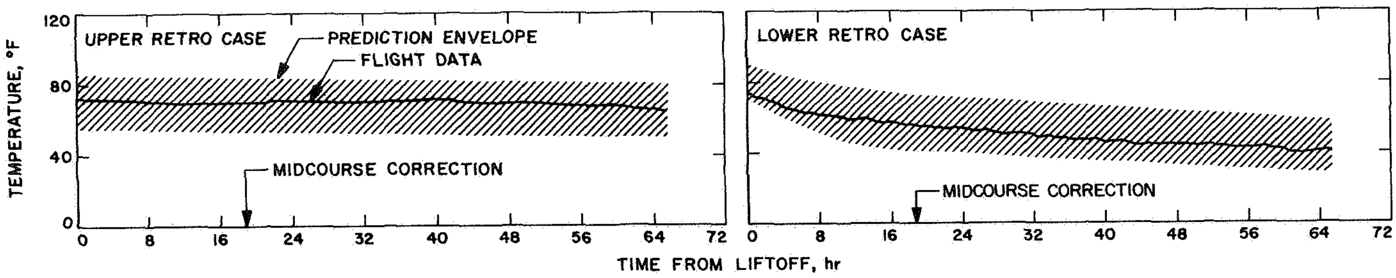


Fig. D-5. Main retromotor transit temperatures

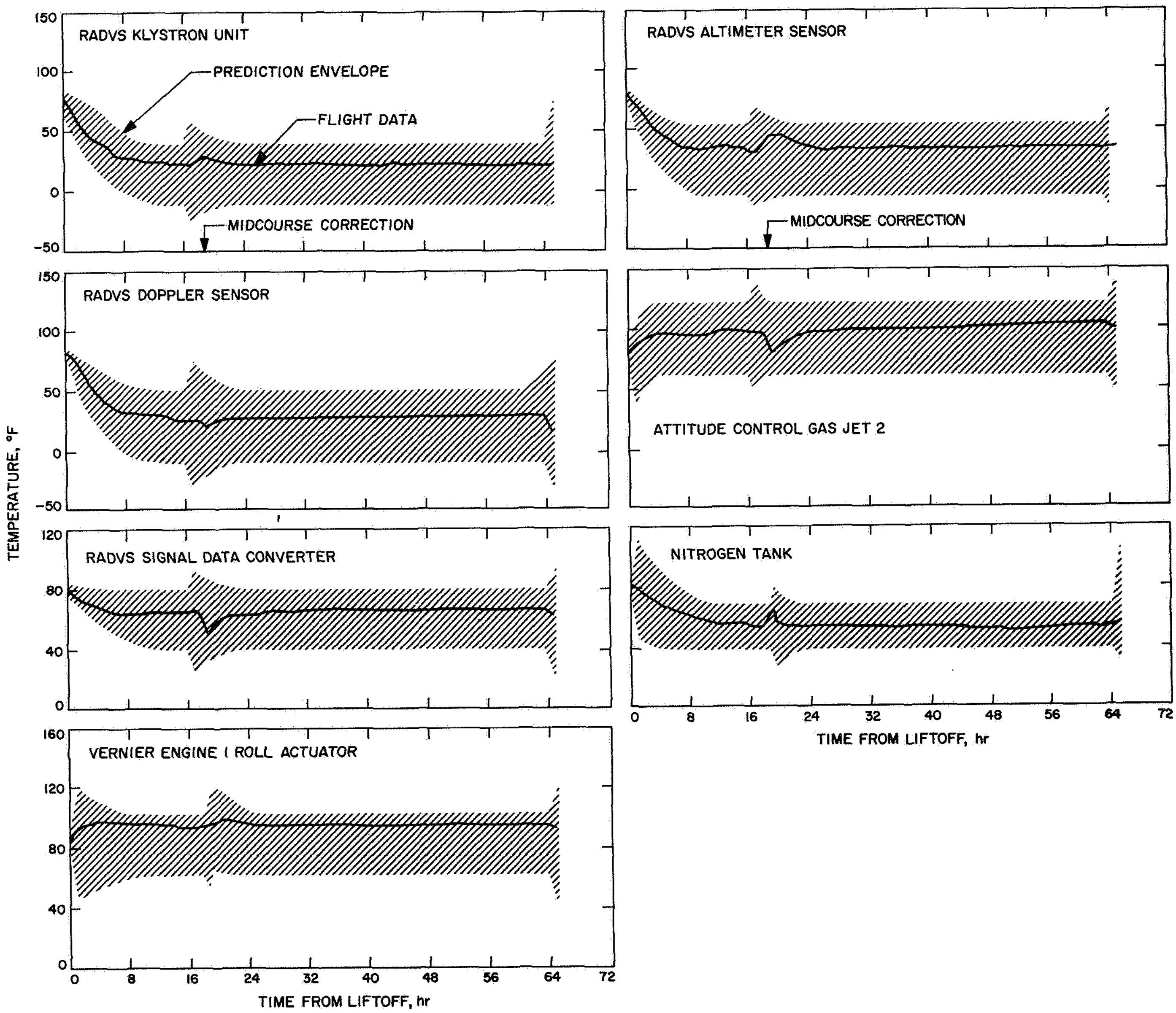


Fig. D-6. Radar and flight control transit temperatures

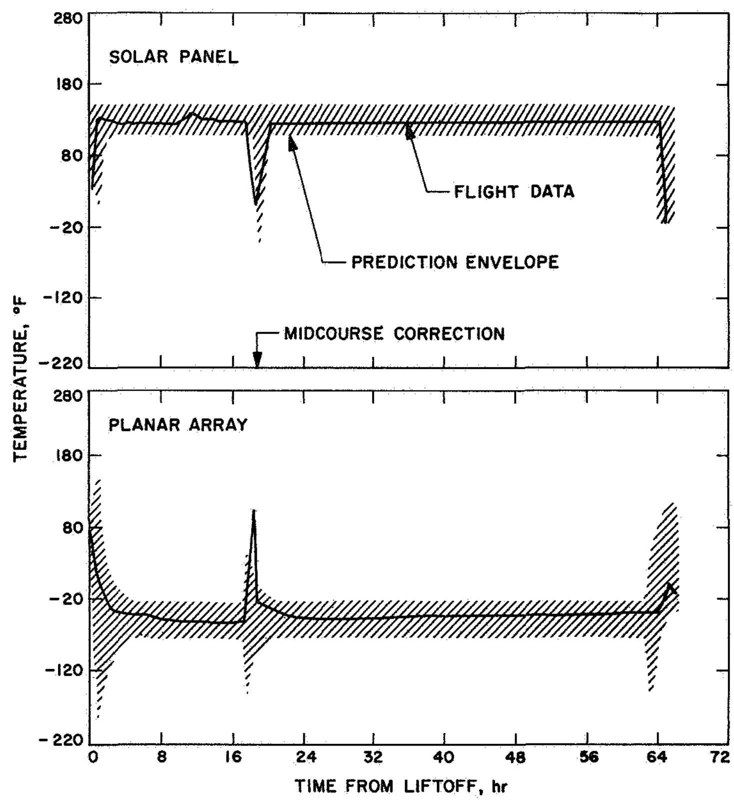


Fig. D-7. Solar panel and planar array transit temperatures

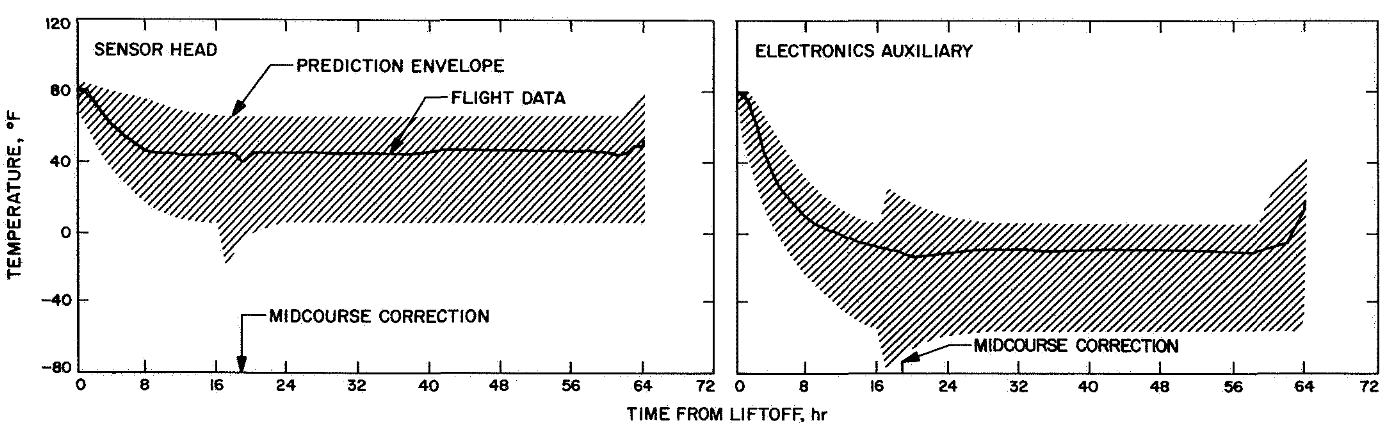


Fig. D-8. Alpha scattering instrument transit temperatures

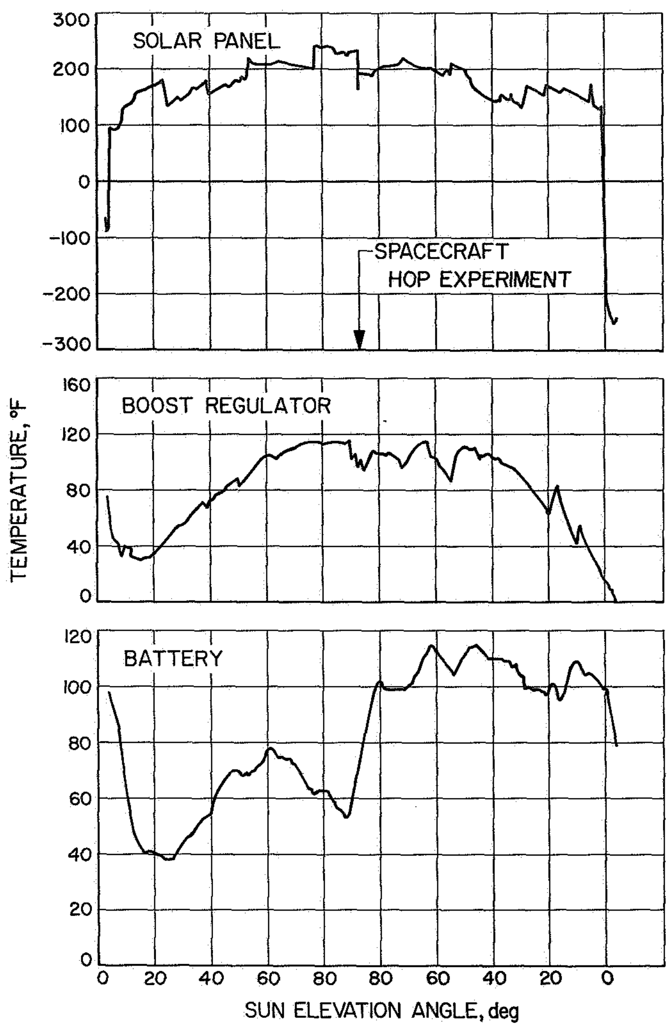


Fig. D-9. Electrical power subsystem postlanding temperatures, first lunar day

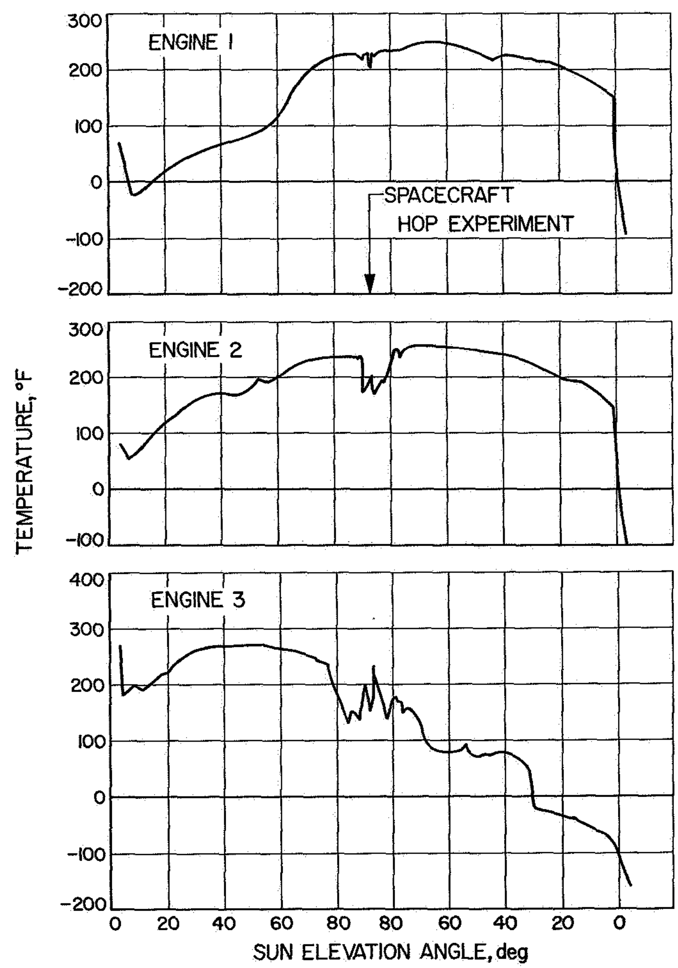


Fig. D-10. Vernier propulsion system postlanding temperature, first lunar day

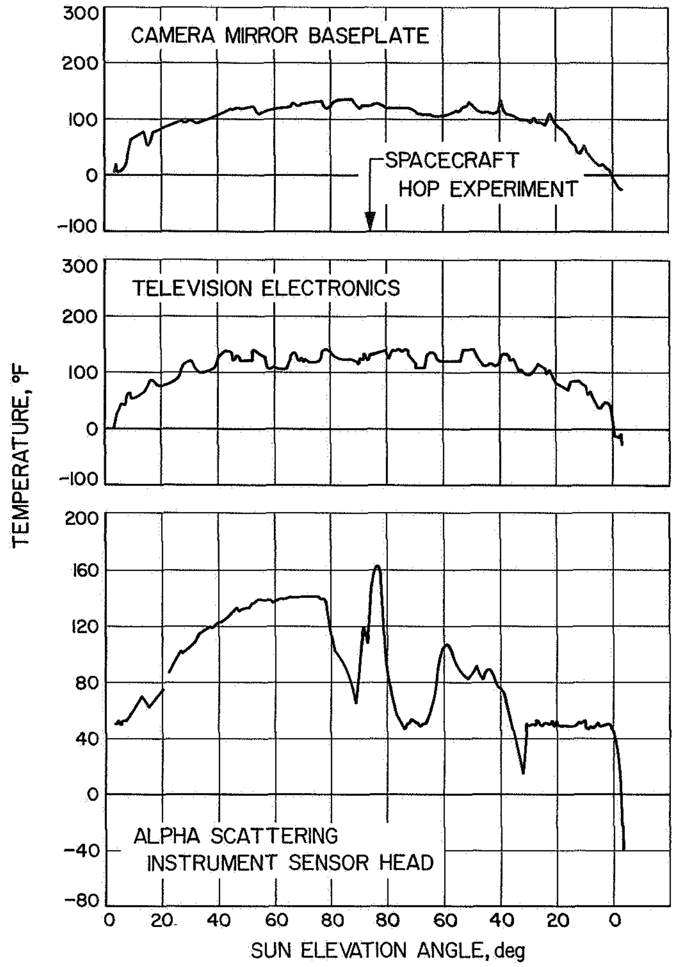


Fig. D-11. Television camera and alpha scattering instrument postlanding temperatures, first lunar day

Glossary

ABC	auxiliary battery control
AC	*Atlas/Centaur*
A/D	analog-to-digital
ADC	analog-to-digital converter
AESP	auxiliary engineering signal processor
AFC	automatic frequency control
AFETR	Air Force Eastern Test Range
AGE	aerospace ground equipment
AOS	acquisition of signal
APC	automatic phase control
A/SPP	antenna and solar panel positioner
BCD	binary coded digital
BECO	booster engine cutoff
BR	boost regulator
CCC	Central Computing Complex
CCN	Control Command Network
CDC	command and data (handling) console
CDS	computer data system
CP	Command Preparation (Group)
CRT	Composite Readiness Test
CSP	central signal processor
CSTS	Combined Systems Test Stand
DC	direct command
DOD	Department of Defense
DPS	Data Processing System
DSCC	Deep Space Communications Complex
DSIF	Deep Space Instrumentation Facility
DSS	Deep Space Station
DVS	doppler velocity sensor
ECPO	Engineering Computer Program Operations (Group)
EM	electromagnetic
EMA	engineering mechanism auxiliary
ESF	Explosive Safe Facility
ESP	engineering signal processor
FC	flight control
FCSG	Flight Control Sensor Group
FPAC	Flight Path Analysis and Command
FRB	Failure Review Board
FRT	fine resolution tracking
GCF	Ground Communications Facility
GSE	ground support equipment
GSFC	Goddard Space Flight Center
HSDL	high-speed data line
ICS	Intracommunications System
IF	intermediate frequency
I/O	input/output
IRIG	Inter-Range Instrumentation Group
IRV	interrange vector
J-FACT	Joint Flight Acceptance Composite Test
KPSM	klystron power supply modulator
KSC	Kennedy Space Center
LOS	loss of signal
MAG	Maneuver Analysis Group
MCDR	media conversion data recovery (subsystem)
MCFR	media conversion film recorder (subsystem)
MECO	main engine cutoff
MES	main engine start
MSFN	Manned Space Flight Network
NASCOM	NASA World-Wide Communication Network
OCR	optimum charge regulator
ODG	Orbit Determination Group
ORT	Operational Readiness Test
OSCP	on-site computer program
OSDP	on-site data processing

Glossary (contd)

OSDR	on-site data recovery (subsystem)
OSFR	on-site film recorder (subsystem)
OTC	overload trip circuit
OVCS	operational voice communication system
PA	Performance Analysis (Group)
PAM	pulse-amplitude modulation
PCM	pulse code modulation
PLIM	postlaunch instrumentation message
PU	propellant utilization
PUVEP	propellant utilization valve electronics package
PVT	Performance Verification Tests
QC	quantitative command
RA	radar altimeter
RADVS	radar altimeter doppler velocity sensor
RATAC	radar target acquisition (system)
RETMA	Radio Electronics Television Manufacturing Association
RFI	radio frequency interference
RIS	range instrumentation ship
RODVS	reliable operate doppler velocity sensor
RORA	reliable operate radar altimeter
RTCS	Real Time Computer System, Cape Kennedy
SCAMA	signaling, conferencing, and monitoring arrangement (voice circuits)
SCAT	Spacecraft Analysis Team
SDC	signal data converter
SECO	sustainer engine cutoff
SFOD	Space Flight Operations Director
SFOF	Space Flight Operations Facility
SOCP	*Surveyor* on-site computer program
SOPM	standard orbital parameter message
SOV	solenoid-operated valves
SRT	System Readiness Test
SPAC	Spacecraft Performance Analysis and Command (Group)
SSAC	Space Science Analysis and Command
SSD	subsystem decoder
SSE	Standard Sequence of Events
STEA	system test equipment assembly
STV	solar-thermal-vacuum
TCP	telemetry and command processor
TDA	Tracking Data Analysis (Group)
TDM	time division multiplexer
TDS	Tracking and Data System
T&DA	tracking and data acquisition
TelPAC	Television Performance Analysis and Command (Group)
TPS	Telemetry Processing Station
TSAC	Television Science Analysis and Command (Group)
TTY	teletype
TV-GDHS	TV Ground Data Handling System
VCXO	voltage-controlled crystal oscillator
VECO	vernier engine cutoff
VPS	vernier propulsion system

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