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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Space Programs Summary 37-43, Vol. VI

Space Exploration Programs and Space Sciences

For the Period November 1 to December 31, 1966

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

January 31, 1967

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SPACE PROGRAMS SUMMARY 37-43, VOL. VI

Preface

The Space Programs Summary is a six-volume bimonthly publication designed to report on JPL space exploration programs and related supporting research and advanced development projects. The titles of all volumes of the Space Programs Summary are:

Vol. I. *The Lunar Program* (Confidential)

Vol. II. *The Planetary–Interplanetary Program* (Confidential)

Vol. III. *The Deep Space Network* (Unclassified)

Vol. IV. *Supporting Research and Advanced Development* (Unclassified)

Vol. V. *Supporting Research and Advanced Development* (Confidential)

Vol. VI. *Space Exploration Programs and Space Sciences* (Unclassified)

The Space Programs Summary, Vol. VI, consists of: an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of the JPL quality assurance and reliability efforts, and the environmental- and dynamic-testing facility-development activities; and a reprint of the space science instrumentation studies of Vols. I and II. This instrumentation work is conducted by the JPL Space Sciences Division and also by individuals of various colleges, universities, and other organizations. All such projects are supported by the Laboratory and are concerned with the development of instruments for use in the NASA space flight programs.

Approved by:

W. H. Pickering

W. H. Pickering, Director
Jet Propulsion Laboratory

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N67-25093

A. Introduction

The *Surveyor* flight spacecraft are designed to span the gap between the *Ranger* Project and the *Apollo* Project by making soft landings on the Moon to extend our knowledge of lunar conditions and determine the suitability of sites for proposed *Apollo* spacecraft landings. Hughes Aircraft Company (HAC), Space Systems Division, is under contract to fabricate the *Surveyor A-21* spacecraft. The launch vehicle, a combination *Atlas/Centaur*, is provided by General Dynamics/Convair. Control, command, and tracking functions for the *Surveyor* missions are performed by the *Surveyor* Mission Operations System and the JPL Deep Space Network.

Surveyor I, the first flight spacecraft, was launched from Cape Kennedy, Florida, on May 30, 1966, and soft-landed on the Moon on June 2, 1966. By June 14, when lunar sunset occurred, approximately 100,000 commands had been received by the spacecraft, and 10,338 pictures of the spacecraft and its immediate vicinity had been transmitted. During the second lunar day, 812 additional pictures were transmitted by *Surveyor I*.

Surveyor II, the second flight spacecraft, was launched from Cape Kennedy on September 20, 1966. The spacecraft performed nominally until the command was given for midcourse thrust execution. At that time, one vernier engine did not ignite. The resulting imbalance of thrust from the other two vernier engines imposed a tumbling motion on the spacecraft, from which it failed to recover. The *Surveyor II* mission was terminated on September 22 after all contact with the spacecraft had been lost.

The third *Surveyor* flight spacecraft is scheduled for launch during the second quarter of 1967.

B. *Surveyor I* Mission Operations

1. Fifth Lunar Day

At 04:26:10 GMT on October 8, engineering data were transmitted by the *Surveyor I* spacecraft to the Johannesburg Deep Space Station (DSS) after the appropriate command was sent. This reactivation of *Surveyor I* was performed primarily to conduct approximately 12 experiments originally intended for *Surveyor II*. For the

most part, these experiments were concerned with telecommunications and TV. An engineering commutator assessment was also made at that time.

After the spacecraft was reacquired and a reassessment of the engineering commutators was made, the signal strength slowly degraded. Tidbinbilla DSS visibility began at 17:00. Although the spacecraft was acquired, data lockup and a command link could not be achieved until 21:03. At that time, the spacecraft's condition was reassessed. When the main battery mode was commanded on, the spacecraft signal was lost. The signal was recovered shortly thereafter, and the spacecraft was then in the main battery mode.

On October 9, a command was sent by the Johannesburg DSS to improve the battery situation. The signal strength dropped, and spacecraft data lockup was not achieved. The command link was lost for the remainder of the Johannesburg DSS pass.

During its visibility period on October 9, the Tidbinbilla DSS obtained an uplink lock on the spacecraft which provided good data. An evaluation of the engineering commutator and the power subsystems resulted in the conclusion that the spacecraft's condition was as good as or better than it was the preceding day. When the camera power-on command was transmitted, the spacecraft signal was lost. Several attempts at signal reacquisition and camera activation were unsuccessful.

Three attempts at initiating the TV picture-taking sequence on October 10 also failed. The Johannesburg DSS lost contact with *Surveyor I* while attempting to take TV pictures during the setting of the Sun on the Moon's horizon. The Robledo DSS in Spain could not acquire the spacecraft during its visibility period, and no further efforts to operate the spacecraft were made.

2. Sixth Lunar Day

At 09:33 on November 8, one-way lock was established with *Surveyor I*. At 10:53, TV identification was obtained. On the basis of the fifth-lunar-day performance, all excessive loads were turned off before stepping the TV camera. Data Mode 7 telemetry was then received. The receiver was continuously in and out of lock, thereby forcing the spacecraft to be shut down at 11:29. The spacecraft's Transmitter B was turned back on at 11:30, but had to be shut down 3½ min later due to the requirements of the *Lunar Orbiter II* spacecraft. At 11:58, Transmitter B was turned back on, and one-way lock was achieved.

Several TV start frames were commanded, but with no response. At 13:30, Transmitter A was turned on, and the spacecraft was then acquired. The signal strength slowly degraded thereafter, and the spacecraft was switched to Transmitter B. When signal processing was turned on, the received signal level decreased. The spacecraft was then turned off to allow the battery to recharge. At 15:31, Transmitter B was turned on, and an engineering assessment was made. Transmitter A was then turned on, and the receiver was in lock. All nonessential loads were turned off to allow the battery to charge in preparation for the planned TV sequences. Transmitter A was commanded to both high and low power, but no indication of video was received. At 17:33, Transmitter B was commanded on in low power, and the receiver was in lock. Approximately 17 min later, the spacecraft was again turned off due to *Lunar Orbiter II* mission requirements.

At 19:12, Transmitter B was turned on in preparation for high-power TV picture-taking. The signal strength decreased with no bit stream obtainable. At 19:43, the spacecraft was commanded off to conserve power, but, approximately 35 min later, Transmitter B was turned back on. The receiver was locked up, but the signal strength slowly degraded. At 20:27 on November 8, the *Surveyor I* spacecraft was turned off, ending the sixth-lunar-day revival attempt.

The spacecraft's performance during the sixth lunar day was generally comparable to that during the fifth lunar day. Transmitter A was exercised for the first time since the second lunar day. Transmitter B continued to evidence the low-signal-level failure mode noted in Transmitter A during the second lunar day, i.e.: an unpredictable and occasional drop of about 17 db in signal level. It was assessed that the video channel feeding Transmitter B was inoperative and that the TV start frame command could not initiate Transmitter A high-power operation. Consequently, TV pictures could not be obtained. Two-way doppler tracking was completed, and the Mars DSS was checked and verified as being operational.

Preliminary analysis of the sixth-lunar-day signal processing of all six commutators showed that many readings from one of the commutators were substantially different than other commutator readings of the same signal. Reduced usage of this commutator will therefore be scheduled for the planned seventh-lunar-day revival of *Surveyor I*.

C. *Surveyor II* Failure Analysis

A detailed investigation was made by JPL and HAC of the failure of Vernier Engine 3 to ignite during the *Surveyor II* midcourse maneuver. An extensive series of simulation tests was conducted. The following conclusions were reached from a detailed review of the *Surveyor II* data as compared with the simulation test results:

(1) Engine 3 had approximately full fuel flow during three of the attempted firings and probably during all attempted firings.

(2) Engine 3 had little, if any, oxidizer flow during the midcourse maneuver; during post-midcourse firing attempts, oxidizer flow was less than that commanded.

(3) Engine 3 did not fire during any of the attempted firings.

(4) During two firings, Engine 1 apparently shut off 1 to 2 sec later than commanded.

(5) Engine 2 temperature rise was excessive both in the 2-sec burns after the second and third burns and in the 21.5-sec burn.

(6) Helium tank pressure readings displayed unexplainable fluctuations (suspected bad transducer).

(7) Unregulated bus current varied from 1.4 to 2.4 amp when the engines were turned on. Based on the resistance of the solenoid valves, the current reading should have been 1.7 amp.

D. Systems Engineering and Testing

1. SC-3 (Third Flight Spacecraft)

The SC-3 spacecraft completed solar-thermal-vacuum and vibration testing, and, after a final weight, balance, and alignment check, was shipped to the Eastern Test Range. This was the first flight vehicle to receive both pre- and post-vibration alignment checks.

To reflect recommendations made by the *Surveyor II* (SC-2) failure review board, the spacecraft was upgraded as follows: (1) A test access connector was added to verify vernier engine solenoid connections; (2) gas jet operation was inhibited by interlocking with the post-separation, sensing, and arming 22-v power; (3) the vernier-propulsion-system temperature data transmission was optimized by switching commutator modes; and (4) a filter was added to reduce the noise on the regulated output current telemetry.

2. SC-4 (Fourth Flight Spacecraft)

The SC-4 spacecraft was upgraded in preparation for solar-thermal-vacuum testing. During the upgrade, the changes discussed above for SC-3 were incorporated.

3. SC-5 (Fifth Flight Spacecraft)

In addition to the changes discussed above for SC-3 and -4, the harness of SC-5 was retrofitted with the wiring changes introduced by the extended payload. The SC-5 vehicle was then returned for systems testing. The structural upgrade to install the alpha-scattering experiment and TV auxiliary viewing mirrors will be performed at a later date.

4. SC-6 and -7 (Sixth and Seventh Flight Spacecraft)

The SC-6 spacecraft is nearing the completion of final assembly (Fig. 1) and is scheduled to begin systems testing in the near future. SC-7 is in the early stages of final assembly.

5. S-9 (Structural Test Model)

The S-9 vehicle was returned to HAC following the completion of a modal survey and controls vibration test program at JPL. A structural upgrade is currently being performed, including removal of the auxiliary battery, the approach TV camera, and the substructure for both units. An alpha-scattering mechanism of flight quality will be installed in the approximate area where these units were located.

Upon completion of the upgrade, the S-9 vehicle will undergo vibration testing, and the response of the alpha-scattering sensor head will be measured during exposure to the flight-acceptance-test environment. Then, S-9 will be upgraded to include major structural and unit changes incorporated in the SC-5 spacecraft. After this upgrade, the vehicle will be subjected to more vibration tests and worst-case antenna/solar-panel-positioner lunar landing shock tests.

6. Incorporation of Lunar Hopper Capability

A lunar "hopper" capability exists in the *Surveyor* spacecraft to provide an optional experiment for lunar

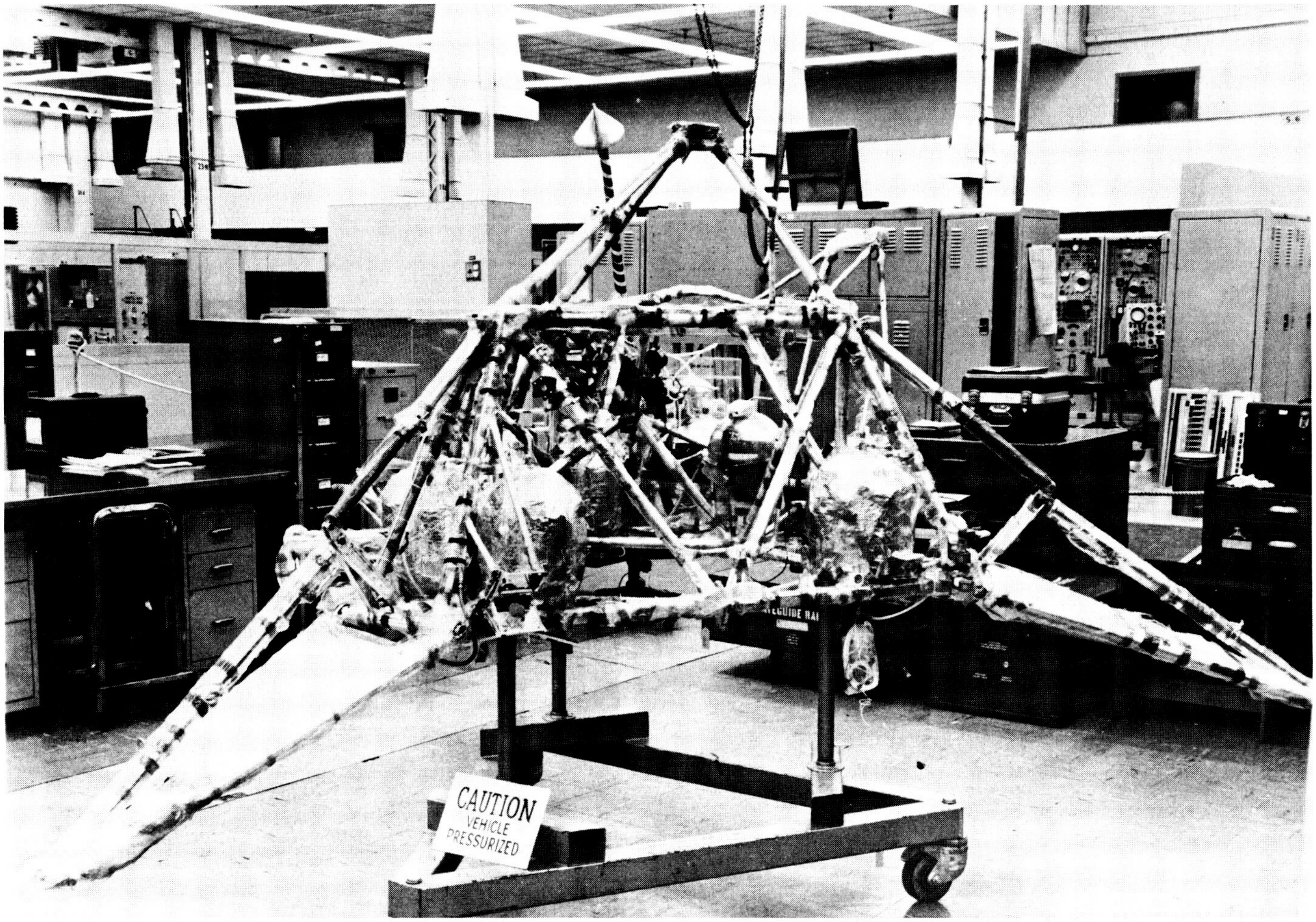

Fig. 1. SC-6 final assembly

translation after initial touchdown. A short-hop capability could provide for short translation by refiring of the vernier engines, causing a sufficient translation to provide a different view of the lunar surface. The SC-4 capability is similar to that of SC-3.

Long-hop capability would provide an optional experiment for SC-5 through -7. Inclusion of this capability would require only minor modification of existing hardware. The long translation would be accomplished by programming a certain vernier-engine thrust for a certain length of time. This long-hop capability is still being studied for its effects and cost.

E. Thermal Control

Flat auxiliary TV viewing mirrors and a surface sampler mockup were installed in the SC-3 spacecraft. Solar-thermal-vacuum testing demonstrated that neither addition significantly affects spacecraft transit thermal control. Temperatures on the units were as anticipated.

Studies were made to define the alpha-scattering-instrument sensor head thermal environment for transit and lunar surface operations. A final analysis for the SC-3 and -4 flat auxiliary TV viewing mirrors indicated that mirror temperatures will be between −110 and 260°F for an entire mission (excluding a lunar night landing situation, after which the temperature would drop to −300°F).

The JPL design for the SC-5, -6, and -7 solar panels was reviewed and compared with that for the SC-1 through -4 solar panels. No significant thermal differences were noted; however, the thermal finish for Thrust Chamber Assembly 2 was found to require modification.

F. Power

With the elimination of the auxiliary battery from SC-5 and subsequent spacecraft, the two-battery operation of *Surveyor* spacecraft during the heavy loads of terminal descent has been eliminated. However, the main battery will be required to supply increased current during that period. Expedited testing is being conducted to establish battery capability under the operating conditions of a single battery system.

G. Payload and Scientific Mechanisms

1. Alpha-Scattering Instrument

The subsystem design and integration effort for the alpha-scattering instrument continued during this reporting period. Results of a TV viewability study of the sensor head deployment area indicate a need for a subsidiary viewing mirror to be fitted to the spacecraft. Definition of the operational sequence for the experiment is continuing. Included are sequences for in-transit operation and lunar operations (stowed, background, and surface counts). In-transit operation of the instrument is a potential new requirement and is being investigated for the feasibility of its implementation.

The alpha-scattering-experiment subsystems test laboratory was reactivated, and modification of certain equipment is being accomplished to reflect the latest design changes in the alpha-scattering-instrument subsystem. Completion of assembly and modification of laboratory test equipment is scheduled to coincide with the delivery of hardware.

2. Alpha-Scattering-Instrument Auxiliary

The alpha-scattering-instrument auxiliary provides command decoding, data signal processing, power management, and squib firing control for the instrument. The circuit design and product design of the auxiliary were completed during this reporting period. A final design review was conducted, with no significant changes being required. Fabrication of the units continues on schedule, and preparations are being made for unit flight-acceptance testing.

3. Surface Sampler

A surface sampler subsystem, consisting of a mechanism, auxiliary electronics, thermal compartment installation substructure, and wiring harnesses, is being developed for incorporation on SC-3 and -4. The mechanism, controlled by the auxiliary electronics which receives and decodes ground commands, will perform picking, digging, scraping, and trenching operations on the lunar surface. The extension–retraction unit, a lightweight scissor mechanism (lazy tongs), will be used to locate a scoop shovel on the lunar terrain. This shovel will be extended by springs in the linkage joints, which

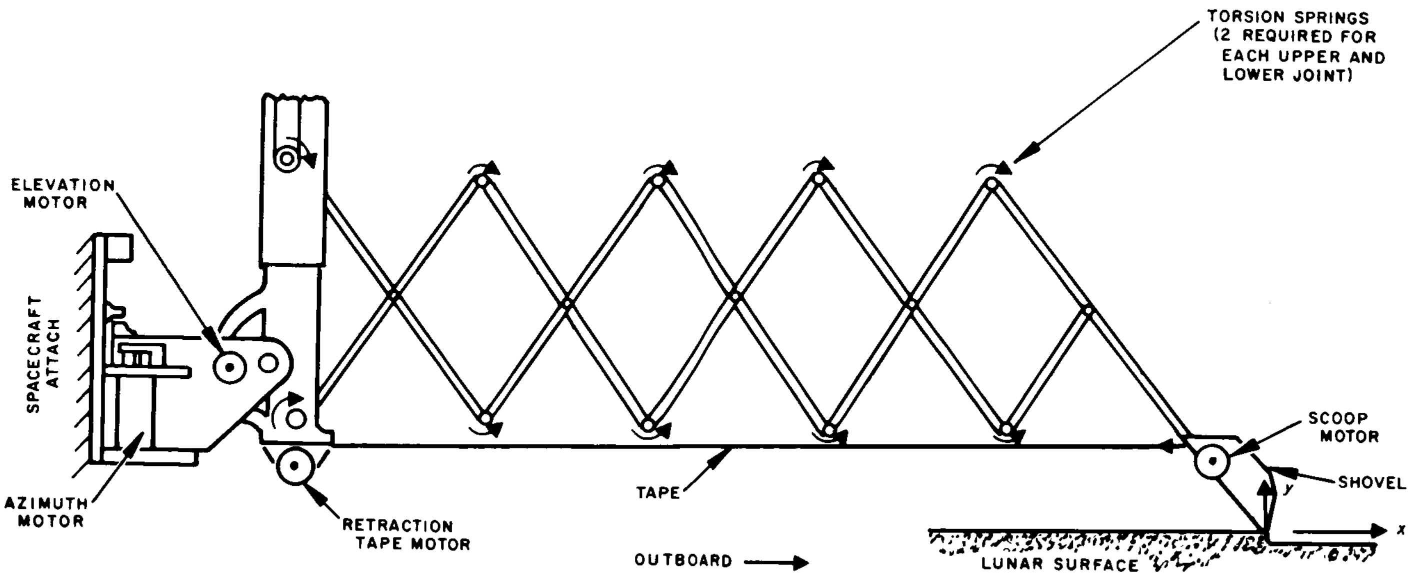

Fig. 2. Soil mechanics/surface sampler (extended position)

will push the shovel out while the tape exerts a small force in the opposite direction. After the shovel strikes the surface, the extension–retraction mechanism will be retracted by means of a motor drive which exerts a tensile force in the tape. The soil mechanics/surface sampler is shown in the extended position in Fig. 2.

Final design review of the auxiliary circuits was held, and the engineering breadboard of the auxiliary electronics was successfully tested over an extended temperature range. Two auxiliaries were fabricated and successfully tested at the unit level. A third (spare) unit is being fabricated. Also fabricated were the flight wiring harnesses that interconnect the auxiliary with the approach TV camera wiring harness and the auxiliary with the mechanism.

Installation substructure and thermal-mass mockups of the mechanism and auxiliary were fabricated and installed on SC-3 and -4. The solar-thermal-vacuum testing of SC-3 was performed with these units installed. The auxiliary mockup was replaced with an auxiliary flight unit for the vibration test of SC-3. The auxiliary successfully passed this test, and the vibration data obtained on the mechanism mockup was used to generate test levels for the vibration test of the mechanism flight unit.

The first mechanism flight unit was fabricated, assembled, and functionally tested under laboratory ambient conditions. Vibration testing of this unit, using levels resulting from the responses of the mockup on SC-3, was successfully completed. Assembly of the second unit was begun.

Mariner Venus 67 Project:

THE PLANETARY–INTERPLANETARY PROGRAM

N67-25094

A. Introduction

The primary objective of the *Mariner* Venus 67 Project is to conduct a flyby mission to Venus in 1967 to obtain scientific information which will complement and extend the results obtained by *Mariner II* relevant to determining the origin and nature of Venus and its environment. Secondary objectives are to: (1) acquire engineering experience in the conversion of a spacecraft designed for a mission to Mars (spare flight spacecraft from *Mariner* Mars 1964 Project) into one designed for a mission to Venus and in the operation of such a spacecraft, and (2) obtain information on the interplanetary environment during a period of increasing solar activity. An *Atlas/Agena D* launch vehicle will be used.

Due to the minimum length of time (18 months) between authorization of the project and the launch opportunity, techniques and hardware developed during prior projects must be utilized to the fullest extent possible. The single flight spacecraft, designated M67-2, will be a converted *Mariner* Mars 1964 Project flight spare. Portions of the proof test model and certain critical spare units from the *Mariner* Mars 1964 Project are being prepared for use as a flight support spacecraft, designated M67-1. The flight support spacecraft will serve the double function of a pseudo-proof test model and a backup spacecraft for qualifying spare subsystems.

Various changes to the *Mariner* Mars 1964 spacecraft design are necessitated by the fact that the M67-2 flight spacecraft will travel toward, rather than away from, the Sun; also, conversions must be made to accommodate the revised encounter sequencing and science payload.

The S-band radio occultation experiment, one of seven scientific experiments approved for the mission, requires the use of only the RF transmission subsystem on the spacecraft. The celestial mechanics experiment uses only the tracking doppler data derived from the RF carrier. The ultraviolet photometer, helium magnetometer, solar plasma probe, and trapped radiation detector experiments are to be accomplished using existing instrumentation with only minor modifications. Only the dual-frequency radio propagation experiment requires the incorporation of a new scientific instrument into the payload. The Principal Scientific Investigators for these experiments are given in Table 1.

Table 1. *Mariner* Venus 67 Principal Scientific Investigators

Experiment	Principal Scientific Investigator	Affiliation
S-band radio occultation	A. J. Kliore	Jet Propulsion Laboratory
Ultraviolet photometer	C. A. Barth	University of Colorado
Dual-frequency radio propagation	V. R. Eshleman	Stanford University
Helium magnetometer	E. J. Smith	Jet Propulsion Laboratory
Solar plasma probe	H. S. Bridge	Massachusetts Institute of Technology
Trapped radiation detector	J. A. Van Allen	State University of Iowa
Celestial mechanics	J. D. Anderson	Jet Propulsion Laboratory

During this reporting period, delivery of all flight-quality hardware (telecommunications and science) for the M67-1 spacecraft was successfully completed. This spacecraft has been assembled and is undergoing systems-type testing. It is anticipated that delivery of all subsystems for the M67-2 spacecraft will be completed in late-January. Assembly and checkout of the M67-2 spacecraft is generally proceeding according to original plans, with late subsystems being integrated as they become available.

B. Testing Equipment and Operations

1. Systems-Test-Complex Data System

In this reporting period, the systems-test-complex data system (SPS 37-42, Vol. VI, p. 10) began active support of spacecraft systems testing using its telemetry data processing capability. This capability includes processing and display on a line printer of the science data-automation-system (DAS) data outputs to the data encoder and to two tracks of the spacecraft tape recorder. Several versions of the systems-test support program, each with somewhat increasing capability, were used for processing the spacecraft systems-test data. The capability for non-real-time data processing is now available, which permits test data recorded on the system's log tape to be processed following a test. A real-time magnetic tape record containing DAS and science telemetry outputs in a special format for science users is now provided during all tests. An off-line program is being developed for use on computer systems other than the Control Data Corporation (CDC) 3300 to allow extraction of data from the real-time magnetic tape by experimenters or other users.

The current status of the data input subsystem is shown in Fig. 1. The modules contained in each of the six racks (from left to right) are given below:

Rack 1: low- and high-speed analog data input modules (not shown).

Rack 2: a diagnostic test module (a computer-controlled stimulus generator being developed to test all other modules in the system, including the computer system input/output communication channels), with a DAS data input module located below.

Rack 3: two events/status data input modules and a power supply for interface circuitry.

Rack 4: a counters module, a monitoring oscilloscope, and a junction box module, covered by a large panel which terminates all of the operational-support-equipment interface circuits.

Rack 5: two additional DAS data input modules and a telemetry data input module.

Rack 6: a telemetry data input module and the data input subsystem controller (the interface element between the CDC 3300 computer system and the data input subsystem; also contains the real-time clock buffer, which acquires systems-test-complex time with 1-msec resolution for input to the computer).

The telemetry data input and DAS data input modules are basically similar, except for differences necessary for interfacing with data and sync lines of different level and format.

2. Structural Test Model

After the first composite vibration test of the structural test model (SPS 37-41, Vol. VI, p. 10), the solar panel dampers were found to have deteriorated, and gross leakage of the silicone oil from the dampers was also noted. The dampers have since undergone modification to correct these deficiencies. During this reporting period, a second composite vibration test of the structural test model was conducted to: (1) obtain supplemental

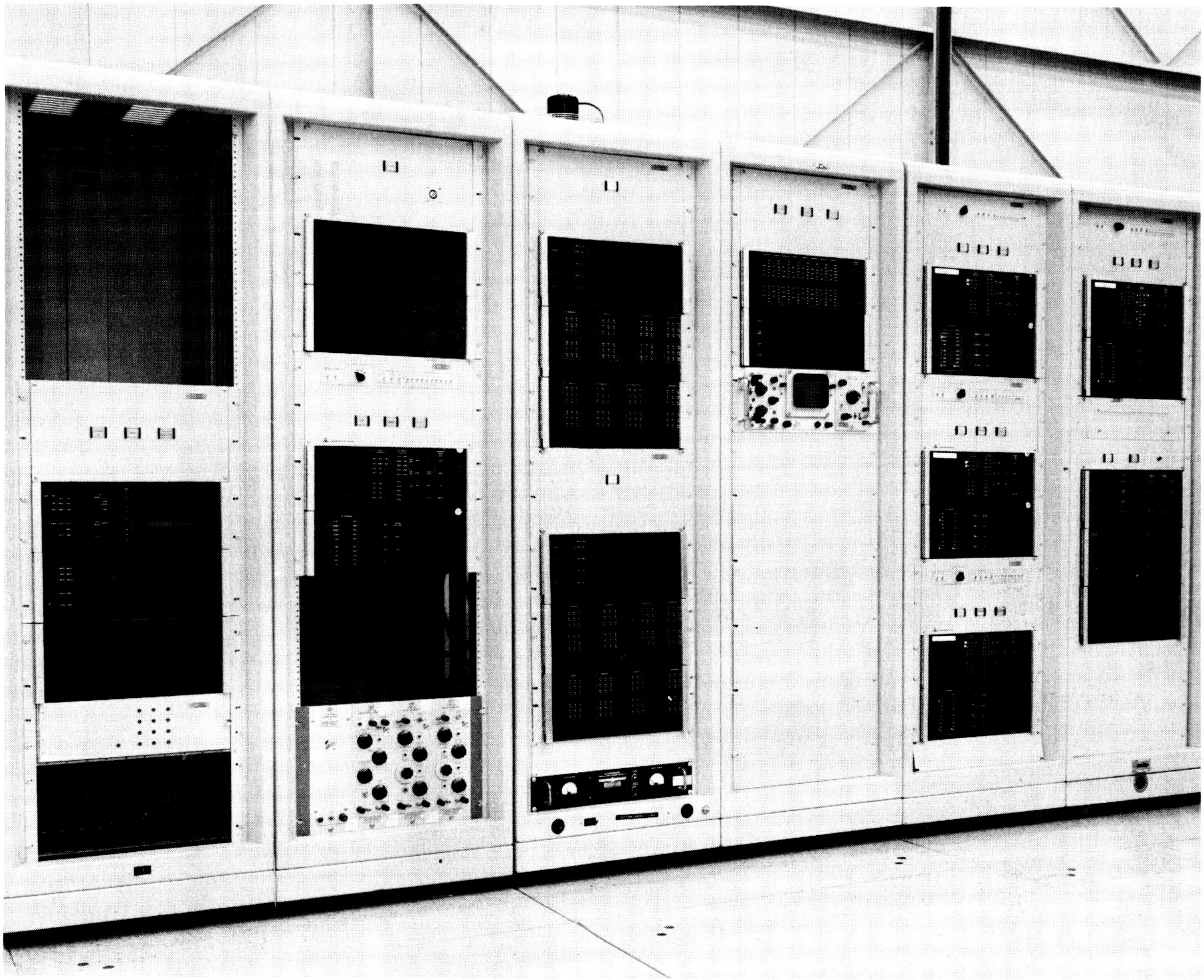

Fig. 1. Data input subsystem

dynamic response data required by recent design changes in the basic spacecraft structure; (2) qualify structural hardware either not included in, or modified since, the previous tests; and (3) verify the complex-wave vibration control equipment operational procedure to be used for the flight-spacecraft vibration testing.

The modified structural test model successfully withstood the qualification dynamic loads. Low-frequency structural and complex-wave qualification-level excitations were introduced along two lateral axes and one axial axis of the spacecraft. Analysis of the reduced test data indicates no major changes in the dynamic response characteristics of the modified test model. The duration of the forced vibration excitation during the testing was 60 min.

3. Attitude-Control Subsystem

The attitude-control subsystem has performed nominally during all planned tests. The gyro control assembly was subjected to an unplanned test wherein the gyros were given input rates in excess of those specified. A complete retest of the assembly failed to reveal any degradation or change in performance characteristics. The attitude-control gas system was also inadvertently

subjected to a rather severe test when it was almost completely drained of nitrogen. Retesting revealed that the pressure regulator's characteristic pressure drop had changed slightly, but was still within specification.

During qualification testing, a spacecraft reflectance test was performed to verify the compatibility of the spacecraft and the optical sensors. The testing was performed by mounting the temperature-control-model spacecraft in various attitudes within a 28-in.-diameter column of sunlight and measuring the incident and reflected light at the various sensor locations. The results indicate that each sensor has a sufficient margin of safety to spacecraft reflected light.

One continuing problem has been the apparent loss of torque gain (output torque versus input current) of the jet vane actuators. These units are spares from the *Mariner* Mars 1964 Project, with the only modification being a substitution of the shaft seal lubricant. Several possible causes for the degradation in torque gain are still being investigated.

4. Planet Sensor

The planet sensor for the *Mariner* Venus 67 spacecraft (Fig. 2) will sense the illuminated limb of Venus and initiate the spacecraft encounter sequence. The output of the sensor, backed up by a timer on the spacecraft, will provide a start signal approximately 1 hr before encounter with the planet. The design of the narrow-angle Mars gate from the *Mariner* Mars 1964 Project was utilized with as little modification as necessary. The sensor's location is in an area near the Canopus sensor. The planet sensor, weighing 257 g, measures 8 by 6.25 in.

In early-August 1966, a prototype planet sensor was assembled and successfully passed extensive type-approval tests. This unit became the type-approval unit. Units for the M67-1 and M67-2 spacecraft were also fabricated, tested, and delivered in late-August and December. During the interim period, an intensive test program was conducted wherein the cadmium sulfide cells and spare welded modules were subjected to a wide range of long-term extremes in environmental conditions. The results were encouraging in establishing long-term reliability.

A measured survey of stray-light sources from various locations on the spacecraft established a probable 200-ft-c maximum intensity, no closer than 9 deg from the mean planet sensor optical axis. Measured performance of the sensor indicated the unit could reject in excess of 30,000 ft-c at 8.5 deg off axis, thereby assuring proper operation even if the Sun shade should not deploy.

5. Sun Shade

During boost, a deployable thermal Sun shade will be stowed within the *Agena* adapter. It will be attached to each solar panel by lanyard which will release the shade

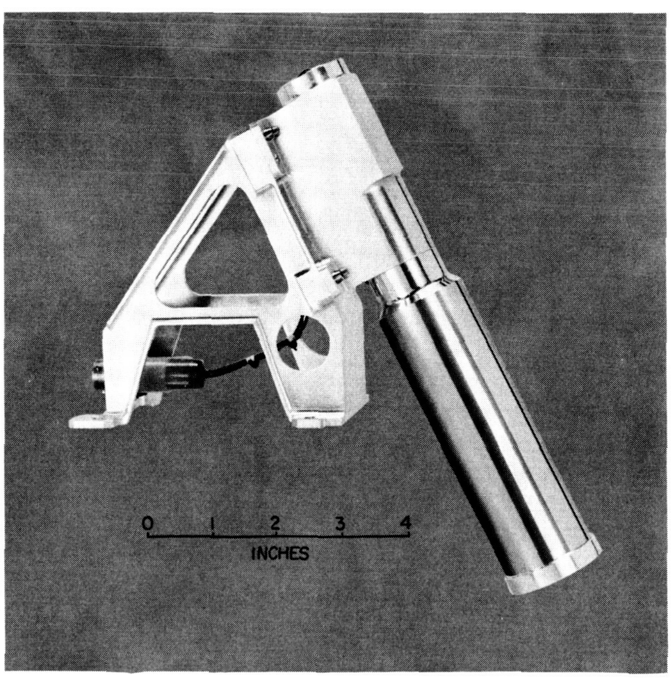

Fig. 2. Planet sensor

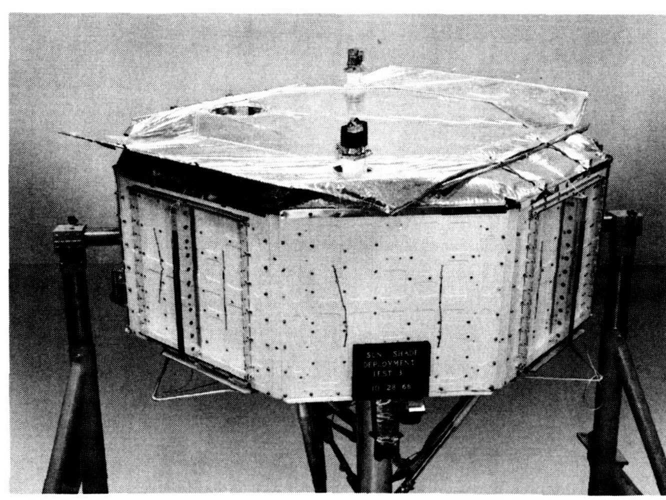

Fig. 3. Sun shade deployment test setup

at panel deployment. To determine the effects of a premature Sun shade deployment while the spacecraft is still attached to the *Agena* stage, a series of tests was conducted with the structural test model and an *Agena* adapter. The results indicate that, in the event of premature deployment of the Sun shade, the spacecraft will successfully separate from the *Agena*, the tumbling rates will be acceptable, and no detrimental effects to the Sun shade will occur.

Additional tests were conducted to test nominal Sun shade deployment and to investigate deployment failure modes. As shown in Fig. 3, the spacecraft was mounted on a modified solar panel stand and could rotate 360 deg about a horizontal axis. It was rotated in 90-deg intervals to four positions to study gravity effects on the deployment mechanics. The tests demonstrated adequate Sun shade deployment mechanization.

6. Space Simulator Radiometers

During space simulator tests of the temperature-control model (SPS 37-41, Vol. VI, p. 10), the readings from the two instruments used to measure the solar intensity differed by approximately 15%. To resolve this anomaly and to increase confidence that the flight spacecraft will be tested at the correct solar intensity, a radiometer comparison test was conducted in the JPL 10-ft Space Simulator. The results showed that the cone radiometer gave readings that were substantially correct; therefore, this radiometer will be used to set the intensity levels for the flight spacecraft.

N67-25095

Mariner Mars 1969 Project

THE PLANETARY–INTERPLANETARY PROGRAM

A. Introduction

The primary objective of the *Mariner* Mars 1969 Project is to conduct two flyby missions to Mars in 1969 to make exploratory investigations of the planet which will set the basis for future experiments – particularly those relevant to the search for extraterrestrial life. The secondary objective is to develop the technology needed for succeeding Mars missions.

The spacecraft design concept will be based on that of the successful *Mariner IV* spacecraft developed under the *Mariner* Mars 1964 Project. However, considerable modifications will be made to meet the 1969 mission requirements and to enhance mission reliability.

The launch vehicle will be the *Atlas/Centaur SLV-3C*. This vehicle, developed under contract for and direction by the Lewis Research Center by General Dynamics/Convair, has a single- or double-burn capability in its second stage and a considerably increased performance rating over the *Atlas D/Agena D* used in the *Mariner IV* mission.

Mariner Mars 1969 missions will be supported by the Eastern Test Range launch facilities at Cape Kennedy, the tracking and data acquisition facilities of the Deep Space Network, and other NASA facilities.

The six planetary-science experiments selected by NASA for the *Mariner* Mars 1969 missions are the following: TV, infrared spectrometer, ultraviolet airglow spectrometer, infrared radiometer, S-band occultation, and celestial mechanics. Additionally, a planetary-approach-guidance engineering experiment will be incorporated to test the feasibility and flight performance of onboard optical sensors and associated data processing techniques necessary for optical approach guidance. The Scientific Investigators for the planetary-science experiments are listed in Table 1.

During this reporting period, the initial formal issue of the spacecraft design book, containing system and subsystem functional requirements and defining the system design, was completed. The system preliminary design review, including examination of the design for the

Table 1. *Mariner* Mars 1969 Scientific Investigators

Experiment	Scientific Investigator	Affiliation
TV	R. B. Leighton[a]	California Institute of Technology
	B. C. Murray	California Institute of Technology
	R. P. Sharp	California Institute of Technology
	N. H. Horowitz	California Institute of Technology
	J. D. Allen	Jet Propulsion Laboratory
	A. G. Herriman	Jet Propulsion Laboratory
	L. R. Malling	Jet Propulsion Laboratory
	R. K. Sloan	Jet Propulsion Laboratory
	N. Davies	Rand Corporation
	C. Leovy	Rand Corporation
Infrared spectrometer	G. C. Pimentel[a]	University of California, Berkeley
	K. C. Herr	University of California, Berkeley
Ultraviolet airglow spectrometer	C. A. Barth[a]	University of Colorado
	W. G. Fastie	Johns Hopkins University
Infrared radiometer	G. Neugebauer[a]	California Institute of Technology
	G. Munch	California Institute of Technology
	S. C. Chase	Santa Barbara Research Center
S-band occultation	A. J. Kliore[a]	Jet Propulsion Laboratory
	D. L. Cain	Jet Propulsion Laboratory
	G. S. Levy	Jet Propulsion Laboratory
Celestial mechanics	J. D. Anderson[a]	Jet Propulsion Laboratory

[a]Principal Scientific Investigator.

mission, the spacecraft, other systems, and the major interfaces, was conducted November 29 and 30. A quarterly project review was conducted December 13. Proposals were received for the propulsion subsystem, the television optics and vidicons, and the spacecraft power simulators to be used in scientific instrument testing; letter contracts were issued for the data automation system, the infrared radiometer, the television electronics and integration, and the ultraviolet airglow spectrometer; and definitive Phase I contracts were executed for the attitude-control and approach guidance subsystems.

B. Systems

1. Orbit Determination

It has been shown that the major factors affecting the capability to achieve required orbit-determination accuracies are tracking station location uncertainties, "effective" doppler noise, and computer program availability. During the encounter phase of the *Mariner* Mars 1969 missions, currently the most demanding navigational accuracy requirement is that specified by the need for precise pointing of the ultraviolet airglow spectrometer. The accuracy obtainable is limited by uncertainties in the locations of the tracking stations and by the "effective" quality of the tracking data (which includes the degradation introduced by the incompleteness of the model used by the computer program to simulate the "real universe"). The extent to which each of these error sources will affect the navigational accuracy is dependent upon the computer program to be used.

The current single-precision orbit-determination program will be capable of using an "effective" noise, on counted two-way doppler averaged over 1 min, of 3 to 6 mm/sec. Because of its inadequate model for Earth spin-axis wandering and the imperfect ephemerides, this program will not be able to make use of station location uncertainties below the 5- to 10-m level, even though it is expected that the station coordinates will be known to within 1.5 to 3 m by 1969. The double-precision orbit-determination program now being developed will be able to utilize data noise levels of 1 to 3 mm/sec and actual station location uncertainties of 1.5 to 3 m. These levels of uncertainty determine bounds for the accuracies obtainable. Under the operational conditions expected in 1969, other error sources, such as small nongravitational forces, mass-of-Mars uncertainty, and ephemeris errors, should not provide major hindrances in achieving the required accuracy near encounter.

2. Planetary Quarantine

Certain precautions must be taken to ensure a high probability that Mars will not be contaminated with terrestrial organisms prior to the successful completion of an adequate spectrum of biological experiments during the *Mariner* Mars 1969 missions. It is possible to impact Mars with either large populations of organisms (the unsterilized *Centaur* stage or the *Mariner* Mars 1969 spacecraft) or very small populations (various effluxes such as attitude-control gas, spalling, outgassing, propulsion subsystem exhaust, and micrometeoroid ejecta). For the first category, impact is assumed tantamount to contamination, even though many feel the Martian environment is sufficiently hostile to terrestrial organisms (even in large populations) that a low probability of survival and replication will result. For the second category, impact is not interpreted as tantamount to contamination, and the probability of contamination is dependent upon the product of a number of other related probabilities. The total

allowed probability of Mars contamination must be allocated between the two *Mariner* Mars 1969 launches, and, for each launch, among the *Centaur* stage, the spacecraft, and the efflux sources.

Since the probable necessity of having to perform a second trajectory correction is much more sensitive to the quarantine level early in the launch period, it was recommended that the planetary quarantine requirement be stated to apply to the launch opportunity rather than to each launch. This would allow more flexibility in mission design.

As a result of analyses and other studies, the following criteria have been established for the *Mariner* Mars 1969 missions: (1) The probability of contaminating Mars shall not exceed 6.0×10^{-5}, with the maximum probability for an individual flight being 5.9×10^{-5}; (2) the probability that a population of viable organisms will be released, given that impact has occurred, shall be assumed to be unity; and (3) the probability that a population of viable organisms will grow, given that impact has occurred, shall also be assumed to be unity.

C. Guidance and Control

1. Central Computer and Sequencer (CC&S)

About 400 hr of testing was performed on the *Mariner* Mars 1969 CC&S breadboard unit, with nearly half of this time spent at temperature extremes of -10 to $85°C$ with voltage extremes of $\pm 20\%$. A new design correcting certain deficiencies in the performance margins of the magnetic memory has been generated, and a new breadboard memory is 80% complete. To correct the only other area where design performance was weak, the sensitive monostable multivibrators have been replaced by gated counter techniques in the computer timing logic and have been masked by less-sensitive circuitry in the peripheral input/output logic. A contract to fabricate a flight prototype CC&S was initiated, with delivery of the unit expected in May 1967.

DSN Capabilities and Facilities

THE DEEP SPACE NETWORK

N67-25096

Established by the NASA Office of Tracking and Data Acquisition and under the system management and technical direction of JPL, the Deep Space Network (DSN) is responsible for two-way communications with unmanned spacecraft travelling from approximately 10,000 miles from the Earth to interplanetary distances. [Earth-orbiting scientific and communications satellites and the manned spacecraft of the *Gemini* and *Apollo* Projects are tracked by the Space Tracking and Data Acquisition Network (STADAN) and the Manned Space Flight Network (MSFN), respectively.] NASA space exploration projects supported, or to be supported, by the DSN include the following: (1) *Ranger, Surveyor, Mariner* Mars 1964, *Mariner IV, Mariner* Venus 67, *Mariner* Mars 1969, and *Voyager* Projects of JPL; (2) *Lunar Orbiter* Project of the Langley Research Center; (3) *Pioneer* Project of the Ames Research Center, and (4) *Apollo* Project of the Manned Spacecraft Center (as backup to MSFN).

Present DSN facilities permit simultaneous control of a newly launched spacecraft and a second spacecraft already in flight. In preparation for increased U.S. activities in space, a capability is being developed for simultaneous control of either two newly launched spacecraft plus two in flight, or four spacecraft in flight. Advanced communications techniques are being implemented to enable obtaining data from, and tracking spacecraft to, planets as far out in space as Jupiter. The main elements of the DSN are described below.

A. Deep Space Instrumentation Facility

The Deep Space Instrumentation Facility (DSIF) is composed of tracking and data acquisition stations around the world. The deep space stations (DSS's) of the DSIF and the deep space communication complexes (DSCC's) they comprise are as follows:

DSS	DSCC
Pioneer Echo Venus Mars	Goldstone
Woomera Tidbinbilla Booroomba[b]	Canberra
Johannesburg	
Robledo Cebreros[c] Rio Cofio[b]	Madrid[a]
Cape Kennedy (spacecraft monitoring)	
Ascension Island (spacecraft guidance and command)	

[a]Planned.
[b]Station not yet authorized.
[c]Station not yet operational.

These stations are situated such that three may be selected approximately 120 deg apart in longitude in order that a spacecraft in or near the ecliptic plane is always within the field of view of at least one of the selected ground antennas. JPL operates the U.S. stations and the Ascension Island DSS. The overseas stations are normally staffed and operated by government agencies of the respective countries, with the assistance of U.S. support personnel.

The Cape Kennedy DSS supports spacecraft final checkout prior to launch, verifies compatibility between the DSN and the flight spacecraft, measures spacecraft frequencies during countdown, and provides telemetry reception from liftoff to local horizon. The other stations obtain angular position, velocity (doppler), and distance (range) data for the spacecraft and provide command control to (uplink) and data reception from (downlink) the spacecraft. Large antennas, low-noise phase-lock receiving systems, and high-power transmitters are utilized. The 85-ft-diameter antennas have gains of 53 db at 2300 MHz, with a system temperature of 55°K, making possible significant data rates at distances as far as the planet Mars. To improve the data rate and distance capability, a 210-ft-diameter antenna was built at the Mars DSS, and two additional antennas of this size are planned for installation at overseas stations. In their present configuration, all stations except the Johannesburg DSS are full S-band stations. The Johannesburg DSS receiver has the capability for L- to S-band conversion.

It is the policy of the DSN to continuously conduct research and development of new components and systems and to engineer them into the network to maintain a state-of-the-art capability. Therefore, the Goldstone DSCC is also used for extensive investigation of space tracking and telecommunications techniques, establishment of DSIF/spacecraft compatibility, and development of new DSIF hardware and software. New DSIF equipment is installed and tested at the Goldstone DSCC before being accepted for system-wide integration into the DSIF. After acceptance for general use, the equipment is classed as Goldstone Duplicate Standard in order to standardize the design and operation of identical items throughout the system.

B. Ground Communication System

To enable communications between all elements of the DSN, the Ground Communication System (GCS) uses voice, teletype, and high-speed data circuits provided by the worldwide NASA Communications Network between each overseas deep space station, the Cape Kennedy DSS, and the Space Flight Operations Facility (SFOF, described below). The NASA Communications Network is a global network consisting of more than 100,000 route mi and 450,000 circuit mi interconnecting 89 stations, of which 34 are overseas in 18 foreign countries. Entirely operationally oriented, it is comprised of those circuits, terminals, and pieces of switching equipment interconnecting tracking and data acquisition stations with, for example, mission control, project control, and computing centers. Circuits used exclusively for administration purposes are not included.

Voice, teletype, high-speed data, and video circuits between the SFOF and the Goldstone DSCC are provided by a DSN microwave link.

C. Space Flight Operations Facility

During the support of a spacecraft, the entire DSN operation is controlled by the Space Flight Operations Facility (SFOF) at JPL. All spacecraft command, data processing, and data analysis can be accomplished within this facility. Operations control consoles, status and operations displays, computers, and data processing equipment are used for the analysis of spacecraft performance and space science experiments. Communications facilities are used to control space flight operations by generating trajectories and orbits and command and control data from tracking and telemetry data received from the DSIF in near-real time. The telemetry, tracking, command, and station performance data recorded by the DSIF are also reduced at the SFOF into engineering and scientific information for analysis and use by scientific experimenters and spacecraft engineers.

THE DEEP SPACE NETWORK

DSIF Development and Operations

N67-25097

A. Flight Project Support

1. Surveyor Project

Preparations at the Pioneer DSS for tracking the third *Surveyor* spacecraft are being carried out concurrent with the DSN system operational testing of the equipment in the MSFN wing. An on-site data processing test and an SFOF/Pioneer DSS simulation test of midcourse and spacecraft maneuver anomalies were performed.

Contact with *Surveyor I* was established on November 8, 1966, during the sixth lunar day after the spacecraft's landing on June 1. Pioneer DSS transmitted 566 commands to the spacecraft, but no additional pictures were received. Telemetry recorded during the contact was forwarded to JPL for processing.

2. *Mariner* Mars 1964 Project

Tracking of the *Mariner IV* spacecraft, which completed its second operational year in space on November 27, continues to be performed by the Mars DSS. The experimental maser at the station is mounted in a cone modified from an 85-ft-diameter antenna unit and is mated to the 210-ft-diameter antenna on a special adapter section. This maser and an experimental receiver installed in the alidade control room are used exclusively in the *Mariner IV* tracking operations in conjunction with the normal DSIF S-band antenna drive. Resulting telemetry is microwaved to the Venus DSS for certain experiments involving a bistatic arrangement of the two stations.

3. *Lunar Orbiter* Project

The first acquisition by the Echo DSS of the *Lunar Orbiter II* spacecraft, launched November 6, was accomplished during the first pass. The midcourse maneuver was performed on November 8 during the second pass. Deboosting of the spacecraft for the initial lunar orbit required 39 min, 45 sec. The spacecraft then remained in the initial high orbit until the tenth pass on November 16, when stored commands previously issued by the Robledo DSS were executed by a command from the Echo DSS and a photographic low orbit was achieved. Picture-taking began immediately after a stable orbit was confirmed, and picture readouts were recorded from November 18 to December 7. The spacecraft was then boosted into a higher orbit, and tracking continued.

4. *Pioneer* Project

The *Pioneer VI* spacecraft, which completed its first year in space on December 16, continues to be tracked by the Mars DSS 210-ft-diameter antenna. Multimission telemetry and command processing is provided by the Echo DSS.

Tracking responsibility for the *Pioneer VII* spacecraft was returned to the Pioneer DSS on November 3, with the Echo DSS providing telemetry and command processing. Although committed to *Lunar Orbiter II* tracking, the Echo DSS continued *Pioneer VII* multimission support through December 13. The Echo DSS reassumed the prime tracking of *Pioneer VII* during the next pass, while continuing to track *Lunar Orbiter II.*

B. DSS Equipment Installation and Testing

1. Pioneer DSS

Operational testing of the equipment in the MSFN wing is in progress. Final installation testing was performed using a NASA target airplane flying a prescribed course to determine antenna pointing and control-room equipment operation. Boresight testing of the acquisition-aid antenna was performed, and intersite interface testing between the Goldstone MSFN Station and the Pioneer DSS is essentially complete.

Installation of the multiple-mission support area is continuing. Currently installed are the telemetry and command data transfer racks and one of a pair of computers for the telemetry and command processing II (TCP II) subsystem. Interfaces between the multiple-mission support area/TCP II and the *Surveyor* ground operational equipment are undergoing operational testing. The TCP II equipment for the Tidbinbilla DSS was operationally tested before shipment in December.

Preparations for the *Mariner* Venus 67 mission are also in progress. The *read, write, verify* command equipment and ground telemetry equipment racks have been installed (Fig. 1), and power-on and internal subsystem tests have been performed.

2. Echo DSS

The multiple-mission support recording and analog instrumentation subsystems were operationally tested for power-on, and internal tests were also performed before the subsystems were shipped to the Johannesburg DSS.

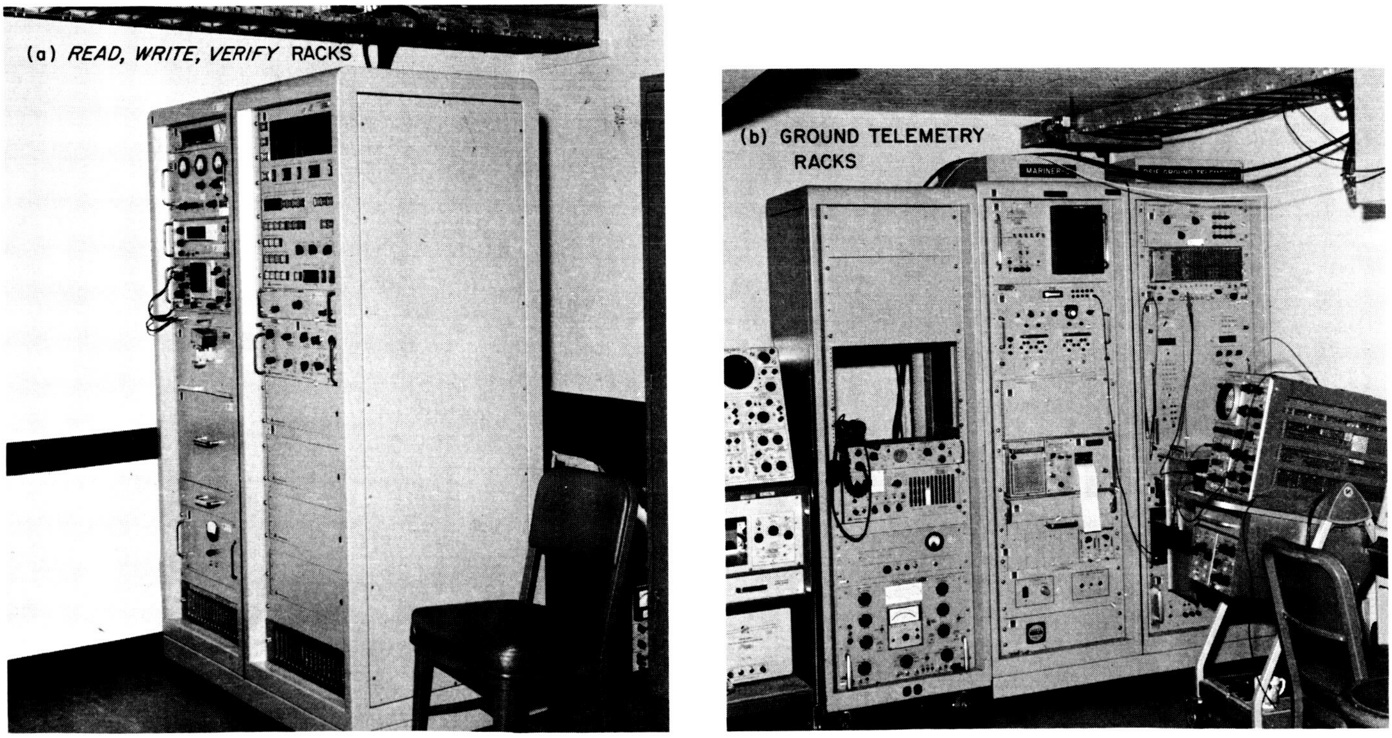

Fig. 1. ***Mariner*** **Venus 67** ***read, write, verify*** **command equipment and ground telemetry equipment at the Pioneer DSS**

3. Venus DSS

From October 15 to December 12, the Venus DSS was primarily engaged in bistatic planetary radar experiments (with the Mars DSS) and in reception from and transmission to the *Mariner IV* spacecraft. Additionally, the 85-ft-diameter azimuth–elevation antenna was used for radio source evaluation at both S- and X-band (2388 and 8448 MHz, respectively). Utilization of the 85-ft-diameter antenna at X-band was made possible by the construction and installation of a standard cassegrain feed cone equipped with a traveling wave maser and associated closed-cycle refrigerator, two follow-up receivers and various components such as waveguide switches, terminations, and signal sources as required for operation and evaluation. The 30-ft-diameter azimuth–elevation antenna was used for 22- to 23-GHz radiometer observations of Venus.

4. Mars DSS

Installation of the DSIF S-band equipment is continuing concurrent with tracking commitments for *Mariner IV* and *Pioneer VI*. Calibration of the master equatorial and the independent reference structure is being performed using a series of star tracks and the *Pioneer VI* spacecraft. Equipment for the antenna pointing subsystem is undergoing operational testing in preparation for use with the master equatorial and independent reference structure subsystems.

C. Communications Development and Testing

1. Temperature Compensator Shunt Assembly for Traveling-Wave-Maser Magnets

Traveling wave masers (TWM's) in JPL installations use permanent magnets to provide the magnetic field necessary for maser operation. Because these magnets are temperature-sensitive, a maser frequency instability problem has been encountered.

Laboratory tests of a TWM magnet were made to determine the change in magnetic field caused by changes in temperature in the range from 80 to 125°F. This temperature range was of interest because temperature controllers in existing TWM installations work well in this range. The measured magnetic-field change of a magnet without compensation caused the TWM center frequency to change at the rate of 0.57 MHz/°F.

A temperature compensator shunt assembly has been added to eliminate the temperature sensitivity of the magnet (Fig. 2). This assembly may be installed without removing the magnet from the TWM package. Installation of the shunt reduces the magnetic field by approximately 25 gauss (at 100°F). This field change can be corrected by a change in trim coil current of 0.1 amp.

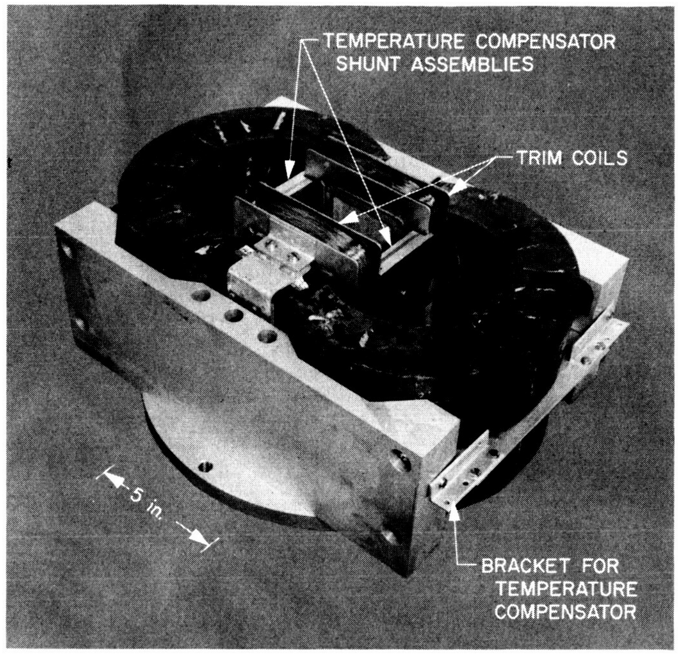

Fig. 2. TWM magnet assembly showing location of temperature compensator shunts

Laboratory tests of the temperature compensator shunt assembly have shown that the magnetic-field change caused by temperature changes in the range from 80 to 125°F can be reduced to the point where maser instability caused by temperature changes will not be noticeable. Thus, the combination of temperature compensation and temperature control can eliminate maser instability caused by the temperature sensitivity of the permanent magnet.

2. Digital Frequency and Phase Meter

The measurement of the frequency of a pair of sinusoidal voltages and the phase difference between them is difficult, since the frequency of each sine wave is considerably less than 1 Hz and its amplitude is less than 1 v. Most frequency meters and phase meters are not able to measure signals of such low frequency, and special equipment for these measurements is complicated and costly.

A new method has been developed which uses a digital computer to make the frequency and phase

measurements. It is designed to give the best possible accuracy for any amount of data which is available. Error due to noise in the data is minimized as much as possible. Fundamentally, the method is very simple. First, the two signals are sampled and read into the computer. Then, the computer fits a sine wave to each set of data by the method of least squares. Frequency and phase are found directly from each of the fitted sine waves.

An iterative method is used for fitting the sine wave to the data to avoid the necessity for solving nonlinear equations. The equations used are easy to program on a digital computer, and they converge to the answer rapidly when the starting values are good. Iterations may be terminated when correction terms become insignificant, provided the error is not unreasonably large.

Two advantages result from use of the method of least squares: (1) Errors in the final results due to any type of noise in the data are minimized, and (2) an answer can be obtained with very little data. Crude answers have even been obtained with less than a full cycle of noisy waveform.

Input data are available to the computer system in the form of two analog voltages. These voltages are sampled periodically to 11-bit accuracy under control of the computer. To maintain constant spacing, the sampling is synchronized externally with a frequency standard. This removes any uncertainty in the sampling period due to the variable execution time required for certain parts of the program. Any machine having Fortran capability can run the program.

In general, the accuracy which can be obtained with this digital frequency and phase meter system depends upon the noise present in the original waveform. Perfect data can give up to 8 significant figures on each of the output quantities. Longer data runs tend to give greater accuracy when the frequency and phase are steady. A typical data run of a few hundred samples will give 3 or 4 significant figures on frequency and phase and will require between six and 10 iterations, depending upon the accuracy of the starting values.

3. *Mariner* Venus 67 Ranging Equipment

Supplemental RF equipment will be required for the range measurement to be made using the *Mariner* Venus 67 spacecraft signals and the DSIF standard receiver. Two such modules, a ×2 frequency multiplier (5 MHz × 2) and a ÷2 frequency divider (1 MHz ÷ 2), have been developed to meet the needs of the hardware portion of the experiment. The internal construction of the modules is shown in Figs. 3 and 4. The external dimensions are the same as those of the standard DSIF module. The gold-plated cavity-type construction is that used for the research-and-development configuration. The conventional feedthroughs in the signal and power paths have been replaced by slots and channels, thus eliminating access holes and plugs. Both modules have low RF-interference leakage, excellent isolation, and good thermal stability.

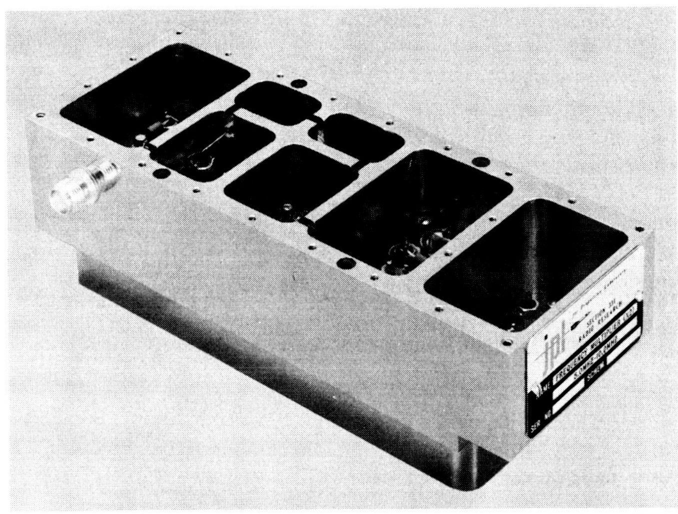

Fig. 3. ×2 Frequency multiplier (top view)

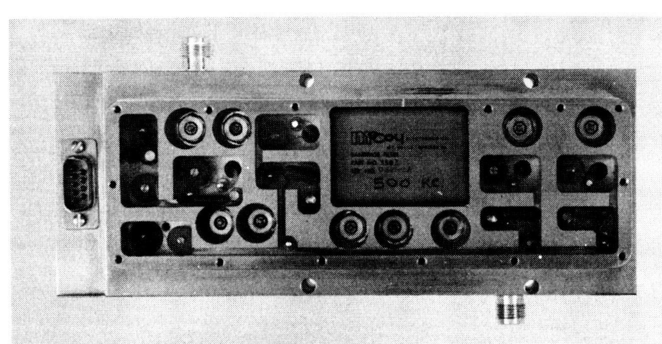

Fig. 4. ÷2 Frequency divider (bottom view)

4. Time-Synchronization System

The DSN stations located around the world are each equipped with a rubidium frequency standard and counting circuits which together provide all frequency, time interval, and time-of-day signals used in the stations. The frequency standards are such that they provide adequate accuracy and stability for the frequency

and time-interval signals; however, there is a drift in time of day due to slight frequency inaccuracies of from <1 to approximately 6 μsec/day. This, plus a certain number of failures, makes it necessary to reset the time of day from the counting circuits occasionally. The objective of this process, known as time synchronization, is to have the time of day at each station the same as that at some master station (e.g., the Goldstone DSCC or the National Bureau of Standards). In the past, time transmissions from WWV (the radio station operated by the National Bureau of Standards in Washington, D.C.) have been used, and the agreement of the times of day at the several stations has generally been maintained to within a few milliseconds.

Studies conducted have shown the requirement for time synchronization throughout the DSN to 20 μsec and preferably 5 μsec. Tracking data requirements for present and future flight projects include a requirement for an agreement at the several DSN stations to within a few tens of microseconds or less. Since the present methods of time synchronization cannot achieve this, a time-synchronization project has been organized to determine the exact requirements and to devise a system which will meet those requirements.

Of the systems investigated, coded Moon-bounce signals were found to meet all requirements for a satisfactory time-synchronization system. Accuracies of ± 5 μsec can be obtained; the system is entirely independent of the main station and antenna; the cost is low; operation is semi-automatic; calibration can be provided daily, and the system is entirely under DSN control. An experimental system is now being fabricated using the existing 10-kw X-band transmitter at the Venus DSS and two receiving systems. The first receiver will be installed at the Echo DSS about 7 miles from the transmitter at the Venus DSS. The Venus DSS clock will be used on this receiver by means of the microwave link, and the system will be checked and calibrated for internal delays. The receiver will then be moved to JPL, and synchronization of a local clock will be demonstrated with checks against the Venus DSS clock by means of the microwave link. Upon completion of this experiment, the receiver will be reinstalled at the Echo DSS. Later, the second receiver will be installed at the Tidbinbilla DSS, and synchronization over this range will be demonstrated.

The experimental precision time-synchronization equipment now being fabricated is a bistatic X-band continuous-wave range-gated radar system. The transmitted signal will be biphase-modulated with pseudorandom code, and the received signal will be correlated with the same code delayed in time. The two-level autocorrelation function that characterizes the pseudorandom code ensures that energy is detected only when the round-trip transit time of the signal from the transmitter to the target and back to the receiver is equal to the time difference between the code transmitted and the code used at the receiver.

5. 400-kw Klystron

A 400-kw klystron is under development for future use in the DSN. The prototype tubes will be tunable over the range of 2.320 to 2.455 GHz, but the design is such that it can be scaled to produce tubes for operation at the DSN frequency (2.1 GHz) or at other lower frequencies. The design is being verified, and fabrication and assembly of the first tube is in process.

The gross weight of the klystron (including magnet and tank) is 1100 lb, and the over-all height is 80¼ in. The cathode of the klystron will be at -65 kv and will be oil-immersed to reduce the length of the cathode insulator and to improve the cooling of the electron gun. The filament power supply must be insulated for this high voltage, since one side of the filament is connected to the cathode. This supply is included in the socket tank, below the cathode, thus utilizing the same insulating and cooling oil and eliminating the need for high-voltage bushings.

Design and fabrication of a transmitter to utilize the 400-kw klystron is proceeding. Following receipt of the klystron, it will be subjected to extensive testing at the high-power facility at the Venus DSS using the new transmitter. It is expected that the new transmitting system can be placed into service on the 85-ft-diameter antenna at the Venus DSS in the third quarter of 1967.

6. 100-kHz Modules

A new family of 100-kHz modules consisting of an IF amplifier, a phase shifter, and a coherent amplitude detector has been developed. A limiting phase detector and a distribution amplifier are currently under development. An IF of 100-kHz was chosen to allow construction of narrow-band crystal filters for the IF amplifiers. The narrowest crystal filters presently available have noise bandwidths of 10 Hz.

The 100-kHz module family was originally developed to enable the construction of narrow-band carrier tracking loops and angle pointing systems. Presently, three

new systems are using these modules: the X-band time-synchronization receiver, the *Mariner* Venus 67 ranging experiment, and a mission-independent telemetry system. The reduction in predetection bandwidth will eventually allow a carrier-loop bandwidth of a fraction of 1 Hz, thus increasing the receiver threshold. To meet the more stringent requirements created by increasing the system threshold, a new set of design specifications has been generated and used in the development of the modules discussed here.