NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report 32-1550

Volume I

Mariner Mars 1971 Project Final Report

Project Development Through Launch and Trajectory Correction Maneuver

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
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Preface

The work described in this report was performed by the technical divisions of the Jet Propulsion Laboratory, under the cognizance of the Mariner Mars 1971 Project.

This four-volume document constitutes the Mariner Mars 1971 Project Final Report. Volume I of this Technical Report consists of Project development through launch and trajectory correction maneuver. Volume II presents the preliminary science results derived from data evaluation to December 14, 1971. (The information contained in Volume II has appeared in *Science*, Vol. 175, January 1972.) Volume III describes the Mission Operations System and covers flight operations after trajectory correction maneuver through the standard orbital mission up to the onset of solar occultations in April 1972. Volume IV consists of the science results derived from the standard orbital mission and preliminary experimenters interpretations of the data obtained from the extended mission.

Detailed information on project organization, project policies and requirements, subsystem development, and other technical subjects has been excluded from the Project Final Report volumes. Where appropriate, reference is made to the JPL informal documentation containing this information. The development of most Mariner Mars 1971 subsystems is documented in JPL Technical Memorandums.

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Abstract

The Mariner Mars 1971 mission was another·step in the continuing program of planetary exploration in search of evidence of exo-biological activity, information on the origin and evolution of the solar system, and basic science data related to the study of planetary physics, geology, planetology, and cosmology. The mission plan was designed for two spacecraft, each performing a separate but complementary mission. However, a single mission plan was actually used for the Mariner 9 due to the failure of the first launch spacecraft.

This Technical Report, the first of four volumes of the Mariner Mars 1971 Project Final Report, describes the major pre-operational project activities, including planning, design, development, and system testing, and the operational activities from spacecraft launch through the Mariner 9 trajectory correction maneuver.

Mariner Mars 1971 Project Final Report

Project Development Through Launch and Trajectory Correction Maneuver

I. Introduction

In the exploration of space man has been extending the frontiers of his knowledge further from Earth. Early space efforts probed and examined Earth's upper atmosphere and the near-Earth space environment. Later, Moon orbiters (Lunar Orbiters) and landers (Rangers and Surveyors) studied our closest celestial neighbor, while flyby missions were sent past the planets Venus (Mariners 2 and 5) and Mars (Mariners 4, 6, and 7). With the Mariner Mars 1971 (MM'71) Project, a spacecraft, Mariner 9, has been sent to study Mars for a period of months from an orbit around the planet. A comparison of Mariner spacecraft sent to Mars is contained in Table 1.

To reduce risk and cost while progressively building a sound technological base, space exploration of the Moon and planets has evolved a series of space vehicles of increasing complexity. Following the flyby past Mars in 1965 (Refs. 1–6) by the small Mariner 4 spacecraft, two spacecraft with greater capabilities for data gathering, Mariners 6 and 7, were successfully sent past Mars in 1969 for a closer look (Ref. 7). Extensive data coverage of Mars necessitates that the spacecraft be placed in orbit around the planet (Ref. 8). NASA issued an *Announcement of Flight Opportunity* in the spring of 1968 to the science community of the impending MM'71 Project, which was approved in the summer of 1968.

As a follow-on to MM'71 in the continuing investigation of Mars, the Viking Project has been authorized to launch two spacecraft in 1975. Each spacecraft will carry a capsule which will be landed on the surface of Mars to make measurements that cannot be obtained from an orbiting spacecraft. Information gathered by Mariner 9 will greatly assist in the selection of desirable landing sites for the Viking capsules.

The Jet Propulsion Laboratory (JPL) of the California Institute of Technology was assigned project management responsibility for the MM'71 Project along with Spacecraft System, Mission Operations System (MOS), and Tracking and Data System (TDS) responsibility. The NASA Lewis Research Center (LeRC) was assigned Launch Vehicle System responsibility. Details of the management assignments and interfaces are contained in Ref. 9. Internal JPL MM'71 Project policy, requirements, and organization can be found in Ref. 10. Significant

Table 1. Mariner Mars spacecraft

Parameter	Mariner Mars 1964 Project	Mariner Mars 1969 Project	Mariner Mars 1971 Project
Spacecraft designation	Mariner 4	Mariners 6 and 7	Mariner 9
Total spacecraft weight (at launch), kg	261	384	1031
Science instrument weight, kg	23	59	68
Propellant weight (at launch), kg	10	10	476
Attitude control gas (N_2), kg	2.36	2.45	2.45
Propellant pressurant gas (N_2), kg	N/A	N/A	14.5
Computer memory capacity, words	No computer	128	512
Power supplied at Mars, W	194	380	450
Electrical part count (actual)	39,220	24,250	27,863 (approx)
Electrical part count (equivalent)	39,220	98,764	112,900 (approx)
Duration of near-Mars examination	30 min	30 min	90 days (2160 hours) minimum
Closest approach distance to Mars, km	9844	3379	1250

milestones and their completion dates are listed in Appendix A. A list of serious problems (P-List), which if uncorrected in a timely manner would have either jeopardized or significantly increased the risk of launch or the mission, was identified by the Project Manager and flagged for attention by the appropriate Division Manager, and is contained in Appendix B. Information on the planned use of resources, manpower, and dollars is contained in Ref. 9, while annual resource plans and actual expenditures were reported monthly in the Mariner Mars 1971 "Project Management Report" to the Office of Space Science.

II. Mission Objectives and Requirements

The MM'71 mission was another step in the continuing program of planetary exploration, which included the following general objectives:

(1) To search for evidence of exo-biological activity, or the presence of an environment that could support exo-biological activity.

(2) To gather information that might provide answers to questions concerning the origin and evolution of the solar system.

(3) To gather basic science data related to the general study of planetary physics, geology, planetology, and cosmology.

(4) To gather data that would assist in the planning and design of the Viking Program's Lander Mission to Mars in 1975.

Specifically, the principal scientific objectives of the MM'71 mission were:

(1) To map the topography of the planet surface at a resolution significantly higher than that achievable from Earth-based measurements and over a large portion of the surface.

(2) To study the time-variable features on the surface of Mars associated with the Wave of Darkening.

(3) To measure and understand the composition, structure, thermal properties, and gross dynamics of the planet's atmosphere.

(4) To make measurements leading to an understanding of the temperatures, composition, and thermal properties of the Martian surface including the polar caps.

(5) To perform measurements directed toward an understanding of the internal activity, mass distribution, and shape of Mars.

To meet these specific science objectives, a mission plan was designed consisting of two spacecraft, each performing a separate but complementary mission. The combination of the two missions would result in the gathering of in-depth information for all of the above scientific objectives.

These two different, complementary missions planned for MM'71 were Missions A and B.

Mission A was designed primarily as a 90-day reconnaissance mission; the spacecraft would attempt to view a large portion of the Martian surface with the highest possible resolution. It would utilize an Earth synchronous or subsynchronous direct orbit inclined about 50 to 80 deg to the Martian equator, with periapsis near the evening terminator at the time of insertion. Periapsis altitude would be in the range of 1600 to 2000 km; apoapsis altitude would be about 17,000 km. The typical orbital period would be 12 h.

Mission B was designed primarily to study the time-variable features of the Martian atmosphere and surface for 90 days. It would utilize a Mars synchronous or a harmonic of a Mars synchronous direct orbit inclined about 50 to 60 deg to the Martian equator, with periapsis near the evening terminator and apoapsis over the morning terminator at the time of insertion. Periapsis altitude would be in the range between 1800 km and the minimum altitude consistent with quarantine requirements; apoapsis altitude would be about 41,500 km. The typical orbital period would be 32.8 h.

Additional information on Missions A and B, requiring two spacecraft, is contained in Ref. 8. Because of the launch failure of the MM'71-1 spacecraft (Mariner 8), neither of these missions was flown, nor described in detail. The actual single mission plan used for Mariner 9 is discussed in *Section X*.

III. Inheritance

A. Spacecraft

The Mariner series of spacecraft has evolved in support of planetary missions since 1962. In each subsequent program, changes were incorporated (1) to adapt the previous design to unique requirements for the new mission, (2) to overcome difficulties demonstrated in the previous mission, and (3) to incorporate new technology when a major improvement in technology would provide a significant benefit in cost, weight, or reliability.

The consideration of inheritance was primarily directed toward the carryover of the overall design of the spacecraft and its ground-based support equipment, as well as their associated hardware. It should be pointed out that additional benefit was gained in areas of test, launch, and flight operation by the repeated use of experienced personnel, procedures, and documentation, as well as facilities which were carried forward from past programs. This discussion, however, will be directed primarily toward inheritance of the spacecraft design and hardware.

The spacecraft system was made up of 20 subsystems, including 4 science instruments. The discussion of spacecraft inheritance will focus on the various subsystems, some with major changes where the inheritance factor was low and the other extreme when design changes were small or nonexistent and a high inheritance factor was realized.

One of the major changes in this mission, as compared to earlier Mariners, was that the 1971 Mariner was to be an orbiter. Thus the *propulsion subsystem* was required to be completely redesigned to provide the necessary propulsion capability to inject the spacecraft into orbit upon its arrival at the planet. The new propulsion system required the capability of a 1600-m/s (about one mile per second) velocity change, and the design incorporated a 1334-newton (300-lb) thrust engine. Practically all components, i.e., valves, regulators, etc., were used on space programs previously, but never had this design as a whole been flown. The engine and the tanks were modified from existing designs. Some inheritance was, therefore, realized at the parts level by using flight-proven components, but the subsystem was a new design.

The *data storage subsystem* (DSS) was a completely new design (all digital, reel to reel) derived from earlier laboratory development efforts. This design incorporated selectable playback speeds of 16, 8, 4, 2, and 1 kilobits per second (kbps) with an 8-track capability using 2 tracks at a time. High packing density provided a total storage capability of 1.8×10^8 bits on the 168-m (550-ft) tape. Data was recorded at 132 kbps. Each playback rate was controlled to a pre-recorded speed (frequency). In this case, little or no design or hardware inheritance was realized from previous flight programs.

The *central computer and sequencer* (CC&S) design was changed primarily by the increase in memory to 512 words over the 128 words used previously. This provided the flexibility required for orbital operations to set up automatic sequences for repetitive orbital work. Lesser changes were incorporated to provide improved operations between computer and sequencer, better checking of stored information, additional systems requirements of accelerometer control and autopilot conditioning, etc.

Of the four on-board science instruments, one scientific instrument was new to the Mariner series of spacecraft. This was the IRIS, or *infrared interferometer spectrometer*. The instrument had flown before on the Nimbus Earth orbiting spacecraft. However, a new beamsplitter, additional data processing circuitry, and new power supplies were required as well as a new thermal design. New interface adaptation to the Mariner spacecraft was also required, which included power and thermal design. The inheritance was that of previous instrument design; however, from the system standpoint, it was a new instrument to be incorporated into the spacecraft design.

Another subsystem which underwent extensive change was the *television (TV) subsystem*. This subsystem employed two cameras, and much of the circuitry, optics, vidicons, etc., could be carried forward. However, the Mariner Mars 1969 subsystem had noise problems, required a great deal of processing of both analog and digital signals into usable video, had less dynamic range, and was not as adaptable as considered necessary to cope with possible variations of planet surface conditions for the orbiter. Therefore, an all digital system was developed with eight selectable filters in the wide-angle camera, automatic and commandable shutter speeds and picture sequencing, and reduced effects from aging and temperature variations. The functions of centralized timing and control were removed from the TV subsystem and transferred to the data automation subsystem. Optics were retained. Again, the experience factor with components, circuits, and functions provided a significant inheritance factor, minimized developmental costs and risk, and provided a high-performance TV subsystem.

The *attitude control (A/C) subsystem* underwent major changes to adapt to the orbital requirements. The attitude control electronics (ACE) was new to accommodate the logic changes and the new autopilot. The inertial reference unit (IRU) was redesigned to include an accelerometer to control the firing duration of the propulsion subsystem rocket engine and electronic integrators to provide both position and rate information separately from the gyros. The gyros were of a modified MM'69 design. The rocket engine autopilot gimbal actuators were new. There were considerable changes in the Canopus tracker (C/T) electronics. The Sun sensors were re-packaged to accommodate the configuration changes. The gas system was similar, with only minor modifications, to that of the MM'69 spacecraft.

The *data automation subsystem* (DAS) was a completely new logic design to accommodate the new instrument payload and mission requirements of MM'71. The integrated circuit logic family and the packaging techniques used were inherited from the Mariner Venus 67 and Mariner Mars 1969 DAS.

The *structure subsystem* underwent considerable change primarily to incorporate the new propulsion subsystem. However, in some areas inheritance was relatively significant since the changes had only moderate impact on the basic octagon structure. The structure carried over from MM'69 was modified successfully.

The *cabling subsystem* was changed considerably, but the carryover in technique, experiences, and materials provided a high degree of efficiency in cable fabrication.

The *power subsystem* underwent moderate changes to accommodate the new autopilot design, and the battery and switching requirements necessary for the peculiar aspects of the orbital mission. The design incorporated a new, high cycle life, nickel–cadmium type battery and its associated charger. A 30-V regulator was added to provide power for the autopilot, engine gimbal actuators, and the propulsion solenoid. The power switching capability was increased, as was the power capability of the booster regulators and the 2.4-kHz inverters. The solar panels were of the MM'69 design.

The *radio subsystem* carried over from MM'69 had a troubled operational history. Several key problems required correction and many lesser problems existed. A great deal of emphasis was placed on establishing a clear understanding of the problems and then deciding which ones required correction and how. A major change was made in the exciter, where a design used in Apollo was incorporated. Another change incorporated a new traveling-wave tube (TWT) in the power amplifier. Many other changes minor in nature but providing significant improvements in performance were carried out. The inheritance factor remained high, however, because a great deal of the complexity and RF idiosyncrasies were well understood or problem characteristics were reasonably established, permitting a rigorous analysis and test program to be established.

Subsystems incorporating minor changes included *command, telemetry, antennas, scan control, infrared radiometer (IRR)*, and *ultraviolet spectrometer (UVS)*. While each did have changes, they were considered of minor nature; thus a high degree of inheritance was provided.

It may be seen that the inheritance by the Mariner Mars 1971 Project of the Mariner spacecraft design varied from subsystem to subsystem. This varied from one extreme, a completely new propulsion subsystem design, to very minimal changes in the command subsystem design. The design changes which were incorporated underwent considerable review and debate prior to approval so that the maximum inheritance could be realized. The importance of a high inheritance cannot be over-emphasized in optimizing reliability, cost, and schedule.

Table 2 summarizes the spacecraft design and hardware inheritance by subsystem.

Table 2. Spacecraft design and hardware inheritance

Subsystem	Design	Hardware: Flight	Hardware: Test
Structure			
Octagon	Basic MM'69 with minor modifications	MM'69; 2 spares	MM'69; 3 test models
Propulsion support	None	None	None
High-gain antenna	MM'69	MM'69; 1 spare	MM'69; 4 test models
Low-gain antenna	Basic MM'69 with minor modifications	None	None
Solar panels	Basic MM'69 with minor modifications	MM'69; 2 spares	MM'69; 10 test models
Adapter	Basic MM'69 with minor modifications	MM'69; 1 spare	MM'69; 1 test model
Solar panel outriggers, cable trough, tank covers, engine thrust structure, NERF bar,[a] medium-gain antenna	None	None	None
Chassis and subchassis	MM'69	MM'69; 15% spares	MM'69; 30% spares
Cabling	Basic MM'69 with minor modifications	None	MM'69; structural mockups
Radio	Basic MM'69 with selected minor modifications	MM'69; 1 spare	MM'69; 2 test models
S-band antenna			
High	MM'69	MM'69; 1 spare	MM'69; 3 test models
Medium	None	None	None
Low	MM'69	None	None
Command	MM'69	None	MM'69; 1 test model
Power			
Solar panels	MM'69	MM'69; 1 spare	None
Battery	None	None	None
Conversion	Basic MM'69 with major modifications	MM'69; 1 spare	None
CC&S	Basic MM'69 with minor modifications	None	MM'69; 1 test model
Telemetry	Basic MM'69 with minor modifications	MM'69; 1 spare	MM'69; 1 test model
Attitude control			
Electronics	None	None	None
Inertial reference	None	MM'69; 2 spare roll gyros	MM'69; 3 test roll gyros MM'69; 1 test pitch and yaw gyro
Gimbal actuators	None	None	None
Canopus tracker	Basic MM'69 with major modifications	None	MM'69; 2 test models
Sun sensor	None	None	None
Gas system	Basic MM'69 with minor modifications	MM'69; 1 spare	MM'69; 2 test models
Pyrotechnics	None	None	None
Mechanical devices			
Pyrotechnics arming switch, separation-initiated timer	MM'69	MM'69; 2 spares	MM'69; 3 test models
V-band, scan platform latch	MM'69	MM'69; 2 spares	MM'69; 2 test models
Separation springs	MM'69	MM'69; 2 spares	MM'69; 1 test model
Boost damper	MM'69	MM'69; 16 spares	MM'69; 16 test models

[a]NERF bar supports top of propulsion module thermal blanket, holding it away from sharp propulsion components.

Table 2 (contd)

Subsystem	Design	Hardware	
		Flight	Test
Scan platform	MM'69	MM'69; 2 spares	MM'69; 2 test models
Deploy damp and linkage, high-gain antenna deploy	None	None	None
Propulsion			
Engines, tankage, and plumbing	None	None	None
Temperature control			
Louvers	MM'69	MM'69; 14 spares	MM'69; 12 test models
Shields	Basic MM'69 with minor modifications	MM'69; 2 sets lower channel	MM'69; 1 set lower channel
Blankets	None	None	None
Data storage	None	None	None
Data automation	Basic MM'69 with major modifications	None	None
Scan			
Electronics	Basic MM'69 with minor modifications	MM'69; 1 spare	MM'69; 1 test model
Actuators	MM'69	MM'69; 2 spares	MM'69; 2 test models
Television	MM'69 camera A lens, and camera B lens and shutter	None	MM'69; 2 test models
Ultraviolet spectrometer	MM'69	MM'69; 1 spare	MM'69; 3 test models
Infrared interferometer spectrometer	Modified Nimbus	None	None
Infrared radiometer	MM'69	MM'69; 1 spare	MM'69; 3 test models
Science support equipment	None	None	None

B. Mission Operations System

The concept of active daily operations was new to the Mariner projects. For experience and operations of this type, the Lunar Orbiter and Surveyor projects were more useful. Using the experience of those two projects, a basically new organization philosophy for operations was evolved. The organization was divided into a planning and analysis level of activities, and an execution level of activities.

The launch, maneuver, and cruise portion of MM'71 was basically similar to previous Mariners. The only significant difference was the requirement for more accuracy in navigation. The double precision orbit determination program, completed by the Mariner Mars 1969 (MM'69) Project, provided the basic tool to provide the required accuracy to reach the planet. In the orbit insertion and orbital phases of the mission, no navigation tools were available from any previous projects. A new satellite orbit determination capability and instrument pointing capability had to be generated.

Within the mission design of MM'71, the return, display, and near real-time analysis of science data were implied. None of these capabilities was available from previous projects with the exception of a limited TV data display capability from MM'69. The daily updating of spacecraft activities at Mars would also require a set of mission control programs to generate the daily command load for updating the CC&S. A very limited start on these programs was inherited from MM'69 in the Command Generation Program (COMGEN) work.

All of the data processing equipment used on MM'71 was different from that used on any previous project. Consequently, any software that was usable from any previous project had to be rewritten to be capable of operating with the new computing equipment.

C. Launch Vehicle System

The Launch Vehicle System used on MM'71 was the Atlas (SLV-3C)/Centaur, managed by the NASA Lewis Research Center (LeRC), Cleveland, Ohio. Both the Atlas and Centaur stages were manufactured under NASA Contract by General Dynamics Corporation, Convair Aerospace Division (GD/CA), San Diego, California.

Atlas/Centaur vehicles AC-23 and AC-24 were essentially the same as the vehicles used on the last three Surveyor missions and the two MM'69 launches. All of these vehicles, starting with AC-13, made use of the Atlas booster SLV-3C, which incorporated an extended propellant tank, increased propellant loading, and up-rated engines. The major difference between AC-23 and these prior launch vehicles was a further up-rating of the Atlas engines for greater thrust.

The SLV-3C Atlas/Centaur vehicle (MM'71 spacecraft encapsulated in the nose fairing) was approximately 35.7 m (117 ft) long and weighed about 147,870 kg (326,000 lb) at liftoff. The basic diameter of the vehicle was a constant 3 m (10 ft) from the aft end to the base of the conical section of the nose fairing. The configuration of the Atlas/Centaur launch vehicle is illustrated in Fig. 1. In the following paragraphs, no attempt will be made to give complete descriptions of the various items involved. More detailed information is available in Volume I of the Mariner Mars 1969 Final Project Report (Ref. 7). For purposes of this report, only the pertinent differences between the MM'69 and the MM'71 launch vehicles are described. A detailed flight report on launch vehicle performance was published by LeRC.

1. Atlas stage. The first stage of the Atlas/Centaur vehicle was a modified version of the Atlas used on many previous Air Force and NASA missions. The SLV-3C Atlas used an up-rated Rocketdyne MA-5 propulsion system which burned RP-1 and liquid oxygen in each of its engines to produce a total liftoff thrust of approximately 1,794,145 newtons (403,340 lb). This value compared with 1,757,047 newtons (395,000 lb) for MM'69. The individual seal level thrust ratings of MM'71 engines were two booster thrust chambers at 760,646 newtons (171,000 lb) each, one sustainer at 266,892 newtons (60,000 lb), and two vernier engines at 2,980 newtons (670 lb) each. An additional significant difference of MM'71 over the MM'69 launch vehicle was the conversion of the launch vehicle transmitters from P-band to S-band, the significance of this being that the launch vehicle transmitters would now operate in the same general frequency range at which the spacecraft radio operated.

2. Centaur stage. The Centaur, or second stage of the launch vehicle, was essentially the same as the ones used to launch the MM'69 spacecraft. The differences were as follows:

(1) The addition of spacecraft destructor, made necessary by the amount of propellant carried by the spacecraft.

(2) The conversion of Centaur from P-band to S-band telemetry.

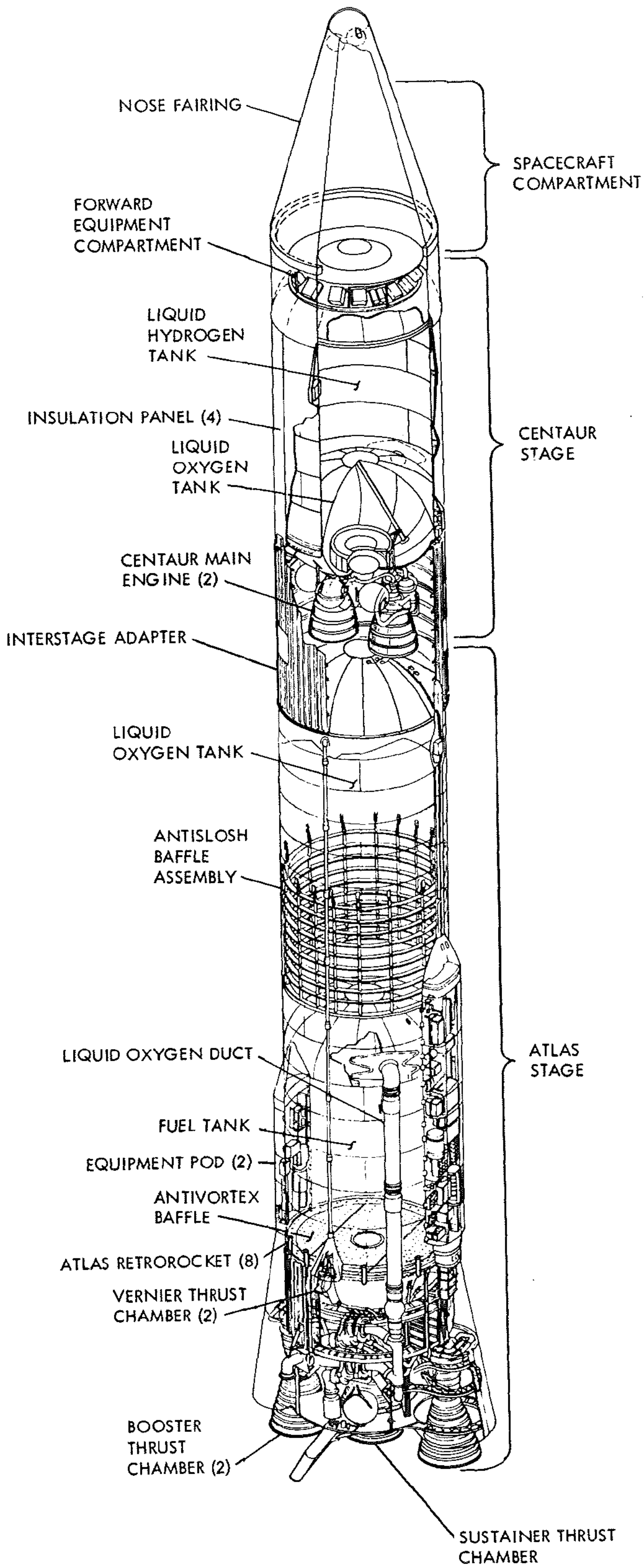

Fig. 1. Atlas/Centaur launch vehicle

(3) The routing of air conditioning ducting in the nose fairing.

(4) AC-23 and AC-24 Centaurs had only one telepak apiece as compared to two on each of the MM'69 launch vehicles. This was brought about by the fact that weight had to be saved somewhere, and the removal of one telepak was a convenient means. Having only one telepak was also reflected in telemetry channel assignments for spacecraft use. On MM'69, there were 10 channels dedicated to spacecraft purposes. MM'71 had only three channels available for spacecraft use.

(5) One further change to the Centaur was the addition of the IRIS purge line.

3. *Spacecraft adapter.* For launch, the MM'71 spacecraft was mounted on the spacecraft adapter (33 kg) (forward payload adapter) by a V-band clamp. The spacecraft adapter was attached through a bolted joint to the Centaur adapter (aft payload adapter).

The MM'71 spacecraft adapter design used MM'69 residual structures. The MM'69 dynamic test model (DTM) and flight spare structures, modified to MM'71 specifications, became the proof test model (PTM)/spare and Flight 1 spacecraft adapters, respectively. One new spacecraft adapter was fabricated for Flight 2. Modification of the MM'69 structures consisted of:

(1) Installation of adjustable shear keys on the eight upper ring support feet.

(2) Installation of nitrogen purge plumbing. This plumbing was routed from the umbilical island to a slip fit disconnect located on the IRIS instrument.

(3) Installation of two electrical strip heaters used as on-pad science stimuli for the IRR and IRIS instruments.

(4) Scarfing a portion of the thermal diaphragm to provide clearance for the scan platform-mounted IRIS instrument.

(5) Changing the 66 field joint fasteners from locking to non-locking basket nuts.

(6) Relocation of the adapter accelerometer.

(7) Installation of a sheet metal "shoe horn" guide for the deployable IRIS shade.

IV. Spacecraft

A. Spacecraft Mission Requirements

The technical design of the Mariner Mars 1971 spacecraft was derived from the interaction of a number of factors, objectives, and constraints. In addition to the constraints placed on the spacecraft implicitly by those mission design and science objectives, NASA headquarters approved these specific guidelines and constraints in the program plan:

(1) Spacecraft would be an orbiter.

(2) Mission should survey major portion of Mars surface.

(3) Mission orbital lifetime requirement was 90 days.

(4) Use of MM'69 spacecraft design and hardware was to be maximized.

(5) The missions would utilize two identical spacecraft.

(6) Low total cost was a firm requirement.

(7) Spacecraft would not be sterilized.

(8) Launch vehicle would be the Atlas/Centaur.

(9) The MM'69 nose fairings would be used.

(10) The DSN would support the two missions with one 26-m (85-ft) net and one 64-m (210-ft) antenna.

(11) Only developed and flight-qualified science instruments would be used. The science payload to be employed should consist of:

   (a) Two television (TV) cameras.

   (b) An infrared radiometer (IRR).

   (c) An infrared interferometer spectrometer (IRIS).

   (d) An ultraviolet spectrometer (UVS).

(12) In addition to experiments utilizing the above instruments, the following experiments would be performed:

   (a) Celestial mechanics.

   (b) S-band occultation.

Implicit in the science payload were requirements placed on the spacecraft design to implement the desired science capability. Four of the six experiments (television, infrared radiometry, infrared spectroscopy, and ultraviolet spectroscopy) required specific on-board instrumentation to achieve scientific objectives. The S-band occultation and celestial mechanics experiments utilized the spacecraft radio subsystem and specialized ground equipment for processing of the doppler and ranging data. The four on-board experiments also utilized specialized ground processing of the received telemetry data to transform the raw data to an interpretable form and to remove instrumentation effects and bias by application of calibration information. On-board formatting of the instrument data output and control of the instrument functions were shared by the instruments and the data automation subsystem.

It was recognized that one of the most difficult aspects of the mission was the reliability of operation during planetary insertion and orbit. The design, therefore, took advantage of the equipment and experience gained in the previous Mariner projects and in state-of-the-art, thoroughly qualified, new equipment to the greatest extent practical. (See *Inheritance, Section III.*)

The MM'69 design and equipment (Ref. 7) were adapted for use in the MM'71 spacecraft to the maximum possible extent. Any design modifications to this equipment were limited to changes: (1) required by the different payload complement, (2) required to achieve the necessary reliability for a nominal cruise and a 90-day operational lifetime in orbit, (3) required for the spacecraft to perform the orbiting mission, and (4) that increased performance within the resource allocations and schedule constraints of the Project.

B. Spacecraft Design

The spacecraft design evolved into the Mariner Mars 1971 spacecraft, Mariner 9, which was inserted into orbit around Mars. The MM'71 spacecraft was composed of 20 subsystems (Refs. 11–26): 4 science instruments, 2 directly supporting the science subsystems, and 14 contributing to the operation of the spacecraft as a semi-automated device. The features in the MM'71 design included:

(1) A 3-axis attitude control subsystem with a high accuracy autopilot for maneuvers, orbit insertion, and orbit trims.

(2) A programmable central computer and sequencer with a 512-word memory.

(3) A 1334-newton (300-lbf) propulsion subsystem capable of performing a minimum of five maneuvers.

(4) An all-digital data storage and handling subsystem.

(5) A multiple-channel telemetry subsystem with variable high-rate telemetry.

(6) A two-way communication and command capability based on the use of a low-gain, a medium-gain, and a two-position high-gain antenna mounted to the spacecraft.

(7) Four solar power panels, power storage and conversion equipment.

(8) Temperature control equipment.

(9) A computer-controlled two-degree-of-freedom scan platform for holding and pointing the science instruments.

(10) Planetary science instruments.

1. Structure and configuration. MM'71's basic structure was an 18.4-kg eight-sided (octagon) magnesium framework with eight compartments (Fig. 2). The electronic assemblies fastened within the compartments provided additional support to complete the spacecraft primary structure. Total weight of all spacecraft structures was 131 kg.

Four solar panels, each 2.15 m long and 0.90 m wide, were attached on outriggers to the top or sunward side of the octagon structure.

The gas bottles and regulators for the dual attitude control subsystem were also attached to the top of the basic structure. The propellant tanks for the liquid bipropellant propulsion were supported on top of the octagonal structure by a support truss structure with the rocket nozzle located on the Z-axis (roll axis) above the propellant tanks.

Four sets of attitude control jets, consisting of two pitch jet assemblies and two roll/yaw assemblies, were mounted on the ends of each solar panel. These jets were the actuating elements of the three-axis attitude control subsystem.

The high-gain antenna was attached to the side of the propulsion support truss structure. The aluminum honeycomb parabolic reflector was 1.02 m in diameter and weighed about 2.04 kg. A pyro-actuated device provided two-position capability, which enabled optimum pointing toward the Earth during the latter half of the interplanetary flight and for 90 days after orbit insertion.

The low-gain omnidirectional antenna aperture was located at the end of a circular aluminum waveguide; it was 10 cm in diameter and extended 1.45 m from the top of the octagonal structure.

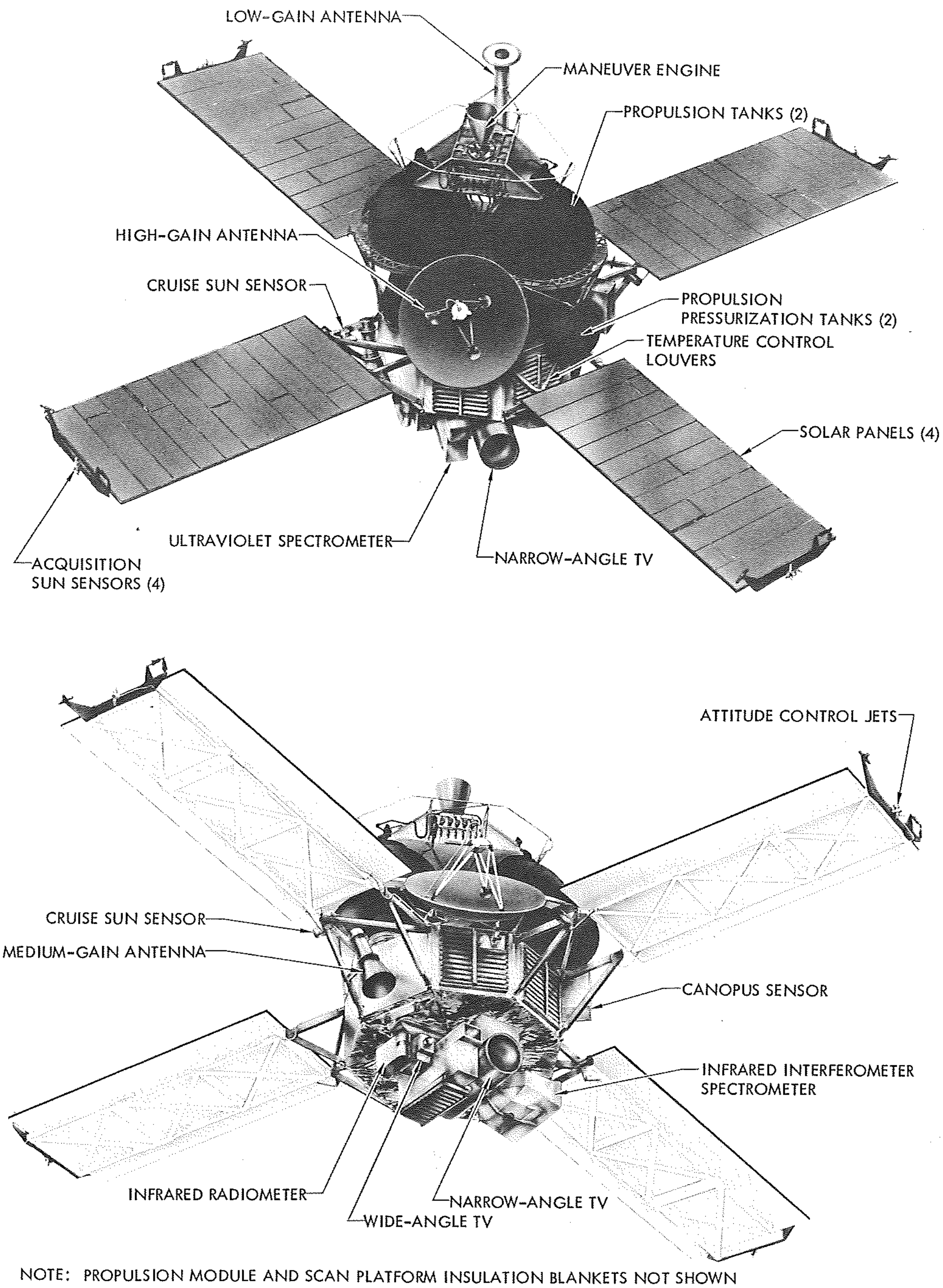

Fig. 2. Mariner Mars 1971 spacecraft

The medium-gain antenna, similar in construction to the low-gain antenna but shorter in length and of narrower aperture, was mounted to a solar panel outrigger adjacent to Bay III of the octagon. It was oriented to provide communication coverage during orbit insertion.

The Canopus star tracker assembly was located on the upper ring structure of the octagon, between solar panels, for a clear field-of-view. Acquisition Sun sensors were located at the out-board end of each solar panel, providing a composite view that subtended the full celestial sphere. A cruise Sun sensor and Sun gate assembly, of narrower field-of-view, were located on the Bay III outrigger and were aligned parallel to the Z-axis.

The eight compartments of the octagon structure contained the following electronic assemblies:

(1) Bay I: Power regulator electronics assembly.

(2) Bay II: Power conversion/IRIS/scan electronics assembly.

(3) Bay III: Attitude control/central computer and sequencer electronics assembly.

(4) Bay IV: Command/telemetry electronics assembly.

(5) Bay V: Data storage electronics assembly.

(6) Bay VI: Radio electronics assembly.

(7) Bay VII: Data automation/television electronics assembly.

(8) Bay VIII: Battery assembly.

Six of the electronics compartments were temperature-controlled by lightweight louver assemblies on the outer surfaces. Thermal shields covered most of the remaining area. The octagon propulsion module enclosure was insulated by multi-layer Mylar thermal shields (blankets) at both the top and bottom. Electrical wiring for transmitting power and signals as needed between subsystems and assemblies formed the cabling subsystem (35 kg).

The spacecraft carried four planetary science instruments. These were a TV subsystem consisting of a wide- and narrow-angle camera, an ultraviolet spectrometer, an infrared radiometer, and an infrared interferometer spectrometer, all mounted on a movable scan platform below the octagon.

2. *Radio-frequency subsystem.* The radio-frequency subsystem (26 kg) received commands and ranging signals transmitted by the deep space stations on Earth, and transmitted science data, engineering data, and ranging signals back from the spacecraft.

Received and transmitted frequencies were in the S-band part of the spectrum. When no uplink signal (i.e., a signal transmitted from Earth "up" to the spacecraft) was being received, the transmitted frequency was governed by an on-board auxiliary crystal oscillator. When the receiver detected and achieved phase lock with an uplink signal, the transmitted frequency from the spacecraft was coherent with and 240/221 times the received frequency. When the spacecraft-transmitted frequency was under control of the uplink received frequency, two-way doppler tracking was accomplished.

The telemetry data and demodulated ranging signal phase-modulated the transmitter carrier. The spacecraft transmitter had two exciters and two traveling-wave tube amplifiers (TWTAs) to increase reliability through redundancy. The TWTA output level was either low power (10 W) or high power (20 W).

Three S-band antennas (high-, medium-, and low-gain antennas) comprising the antenna subsystem were used by the radio-frequency subsystem. Early in the MM'71 spacecraft design, two switchable low-gain antennas and a high-gain antenna were examined. Due to cost and performance considerations, the current low-, medium-, and high-gain antenna configurations evolved.

The high-gain antenna (HGA) was a two-position, pyrotechnic-activated device that would allow optimum pointing of the antenna toward the Earth late in the cruise phase and during the standard orbital mission.

The low-gain antenna (LGA) would be used to receive uplink signals throughout the entire mission and to transmit downlink signals when the spacecraft–Earth range was not so large as to require the additional gain of the HGA. It was mounted on the sunward side of the spacecraft and had a hemispherical pattern almost symmetrical about the roll axis.

The medium-gain antenna (MGA) was coupled passively to the LGA for both receiving and transmitting. It was mounted almost diametrically opposite the LGA and would be used, when the spacecraft is turned off Canopus and the Sun, for propulsion motor firings.

During periods when a ground station is transmitting the S-band uplink to the spacecraft, the spacecraft receiver, operating continuously throughout the mission, would receive the uplink signal via the low- or medium-gain antenna. The uplink signal might be the S-band carrier alone or the S-band carrier containing command and/or ranging information. The carrier containing command and/or ranging modulation would then be processed and routed to the appropriate user location. The command data would be sent to the flight command subsystem; the ranging data, when present in the uplink signal, would be sent through the radio-frequency subsystem ranging channel to modulate the spacecraft-transmitted signal in order to provide the turnaround ranging function. The ranging channel would be turned off and on either by ground command or the on-board CC&S (Command 2A).

3. ***Flight command subsystem.*** The spacecraft received ground commands from Earth in addition to commands from the on-board CC&S (see Table 3). The radio subsystem received these ground commands and relayed them to the flight command subsystem (5 kg), which detected and decoded the commands and issued them to the appropriate subsystem. Ground commands would be required to execute trajectory corrections, to update functions related to spacecraft orbital operations, or to choose redundant elements in the event of certain component failures.

Table 3. Ground command list

Symbol	Name	Subsystem destination	Symbol	Name	Subsystem destination
	Direct commands			Direct commands	
DC-1	Engineering mode	FTS	DC-25	RT science No. 1 mode	FTS
DC-2	CC&S readout select (T)	FTS	DC-26	Spare	PYRO
DC-3	Playback mode	FTS DSS	DC-27	Initiate maneuver sequence	CC&S
DC-4	DSS ready mode	DSS	DC-28	Select scan stow position	SCAN
DC-5	Engineering data rate switch (T)	FTS	DC-29	Accelerometer scale factor	CC&S
DC-6	FTS redundant elements switch (T)	FTS	DC-30	Computer inhibit	CC&S
DC-7	Power amplifier switch (T)	RFS	DC-31	Computer enable	CC&S
DC-8	Exciter switch (T)	RFS	DC-32	Computer maneuver initiate	CC&S
DC-9	Ranging on/off (T)	RFS	DC-33	Tandem maneuver	CC&S
DC-10	Transmit low	RFS	DC-34	Scan on/off (T)	PWR
DC-11	Transmit high	RFS	DC-35	Select variable scan reference	SCAN
DC-12	Adaptive gate step	A/C	DC-36	Initiate TV mapping sequence	DAS
DC-13	Maneuver inhibit	A/C CC&S	DC-37	Boost mode enable inhibit (T)	PWR
DC-14	Maneuver enable	A/C	DC-38	Battery charger on/off (T)	PWR
DC-15	Canopus gate override	A/C	DC-39	DSS slew mode	DSS
DC-16	DSS record mode	DSS	DC-40	Gyros inhibit	A/C
DC-17	Canopus cone angle step	A/C	DC-41	Select scan cone position	SCAN
DC-18	Inertial roll control step	A/C	DC-42	TWT high power	RFS
DC-19	Canopus roll control	A/C	DC-43	TWT low power	RFS
DC-20	Roll control inhibit	A/C	DC-44	RT science No. 2 mode	FTS
DC-21	Roll search step	A/C	DC-45	Platform unlatch	PYRO
DC-22	Select CW reel direction	DSS	DC-46	TV cover deploy	TV
DC-23	Select CCW reel direction	DSS	DC-47	DSS on/off (T)	PWR
DC-24	Spare	PYRO	DC-48	Spare	IRR
			DC-49	High-gain antenna update	PYRO

Table 3 (contd)

Symbol	Name	Subsystem destination
	Direct commands	
DC-50	Battery test load on/off (T)	PWR
DC-51	Disable tolerance detector	CC&S
DC-52	Computer flag 7 interrupt	CC&S
DC-53	Spare	PYRO
DC-54	Downlink on	PWR
DC-55	Downlink off	PWR
DC-56	Select 16-kbps PLBK rate	DSS
DC-57	Select 8-kbps PLBK rate	DSS
DC-58	Select 4-kbps PLBK rate	DSS
DC-59	Select 2-kbps PLBK rate	DSS
DC-60	Select 1-kbps PLBK rate	DSS
DC-61	Simulate Sun gate	A/C
DC-62	Select pre-aim backup mode	A/C
DC-63	Roll gyro on	A/C
DC-64	Switch pre-aim backup bias	A/C
DC-65	Open press, P1/Prop. O1, F1	PYRO
DC-66	Close pressurant P2	PYRO
DC-67	Close propellant O2, F2	PYRO
DC-68	Open pressurant P3	PYRO
DC-69	Open propellant O3, F3	PYRO
DC-70	Close pressurant P4	PYRO
DC-71	Close propellant O4, F4	PYRO
DC-72	Open pressurant P5	PYRO
DC-73	Open propellant O5, F5	PYRO
DC-74	Charger auto. switchover on/off (T)	PWR
DC-75	Propulsion heater on/off (T)	PWR
DC-76	DAS on	PWR
DC-77	UVS and IRR on/off (T)	PWR
DC-78	TV on/off (T)	PWR
DC-79	IRIS on/off (T)	PWR
DC-80	Science off	PWR
DC-81	Select battery charge rate (T)	PWR
DC-82	Spare	DAS
DC-83	Switch RTS No. 2 data rate (T)	DAS
DC-84	Computer flag 6 interrupt	CC&S
DC-85	Enable tolerance detector	CC&S
DC-86	Computer flag 8 interrupt	CC&S
DC-87	High-power slew	DSS
	Quantitative commands	
QC-1	Platform cone slew positive	SCAN
QC-2	Platform cone slew negative	SCAN
QC-3	Platform clock slew positive	SCAN
QC-4	Platform clock slew negative	SCAN
	Coded commands	
CC-1	Computer load	CC&S
CC-2	Computer load	CC&S
CC-3	Word interrogate	CC&S
CC-4	Sequencer load	CC&S
CC-20	DAS coded command	DAS

4. ***Power subsystem.*** The power subsystem (75 kg) provided a central supply of electrical power to operate the electrical equipment on the spacecraft. It also provided the required switching and control functions for the effective management and distribution of that power, as well as a timing function for some elements of the spacecraft. The power, which was derived from four photovoltaic solar panels and a rechargeable battery, was converted and distributed in the following forms:

(1) 2.4-kHz, single-phase, square-wave power for engineering and science subsystems, and for the propulsion module and cone actuator heaters as required.

(2) 400-Hz, three-phase, quasi-square-wave power to the attitude control subsystem for gyro motors.

(3) 400-Hz, single-phase, square-wave power for the scan platform actuators.

(4) Regulated 30-Vdc power for the engine valve and gimbal actuators.

(5) Unregulated dc power to the battery charger, other heaters, and radio-frequency subsystem (RFS) for the TWTA power supply.

5. ***Central computer and sequencer subsystem.*** The central computer and sequencer (CC&S) subsystem (10 kg) provided timing and sequencing services for the other spacecraft subsystems. The memory capacity of MM'71 CC&S was increased from the 128 words for MM'69 to 512 words. This new MM'71 design capability provided

for verification of the contents of the CC&S memory and the contents of the fixed sequencer.

The sequencing was generated by a special-purpose computer with fixed sequencer redundancy in the maneuver mode. Timing and sequencing (except the fixed sequencer) were programmed into the CC&S before launch and could be modified during flight by coded command (CC).

The trajectory correction maneuver was, in the normal operating mode, fixed in sequence with roll and yaw turns, and spacecraft velocity change (motor firing) in that order. Duration and magnitude of turns and magnitude of motor burn were variable by coded commands. The normal operating mode, defined as the tandem mode, operated the computer part of the CC&S concurrently with the fixed sequencer and required that events coincided between each part. Either the computer or fixed sequencer might execute the maneuver independently if so directed by direct command (DC).

The CC&S was capable of sending the commands to the subsystems listed in Table 4. Timed events would be initiated in six basic sequences:

(1) *Launch.* Sequence started with the loading of the CC&S program before liftoff, and ended when the spacecraft became fully stabilized in flight. The first event after launch was programmed to occur one hour or more after CC&S inhibit release. Launch events were normally programmed with minutes resolution after the first event.

(2) *Cruise.* Sequence started at the same time as the launch sequence and lasted for the duration of the mission. Launch, maneuver, and orbital sequences were essentially superimposed on the cruise sequence. Cruise events were normally programmed with hours resolution.

(3) *Maneuver.* Sequence started by DC command or by computer event 5A (see Table 4). CC&S logic was modified for MM'71 to include this automatic maneuver and orbit insertion capability. Four modes of maneuver sequencing were possible: the tandem mode (normal), the fixed sequencer mode, the computer mode, and the parallel mode. The maneuver sequence was programmed for seconds resolution between events. The fixed sequencer maneuver could be interrupted by computer event 5B (see Table 4).

(4) *Pre-orbital insertion.* Sequence was comprised of calibration sequences for the scan platform and TV cameras, conducted during interplanetary cruise. Several days before insertion into Mars orbit, science data acquisition would begin with science instrument turn-on.

(5) *Orbital insertion.* Sequence started by DC command or by computer event 5A. The orbit insertion sequence was the parallel mode where the computer and sequencer operated in parallel. The sequence required both minutes and seconds resolution between commands.

(6) *Orbit operations.* Sequence started after a correct orbit was attained and continued for the life of the mission, or until changed by coded command. The sequence required hours, minutes, and seconds resolution with timing controlled by CC&S and DAS signals.

6. Flight telemetry subsystem. The flight telemetry subsystem (10 kg), by suitable modulation of the radio subsystem RF signal, enabled the formatting and transmitting of data on any of three channels: one for engineering, one for high-rate science data, and one for low-rate science data. While the engineering channel transmitted continuously, only one of the two science channels could be on at any one time. This subsystem also provided data rate, data mode, and modulation index switching.

7. Attitude control subsystem. The attitude control (A/C) subsystem (30 kg) provided continuous spacecraft flight stabilization and orientation after separation from the launch vehicle. During normal cruise and orbital operations, it would automatically orient the spacecraft with respect to the lines of sight to the Sun and the star Canopus and maintain that orientation by means of the A/C cold gas mass expulsion system.

Upon receipt of commands from the CC&S, the A/C subsystem would orient the spacecraft to align the propulsion subsystem thrust axis in the direction commanded for the trajectory correction maneuver, orbital insertion maneuver, or orbital trim maneuvers. During the maneuver rocket engine firing, the A/C subsystem would maintain spacecraft orientation and stability in pitch and yaw by two-axis gimbal control of the rocket engine and in roll by the attitude control roll jets. An accelerometer signal was provided to the sequencer in the CC&S for control of the velocity magnitude. At the end of a maneuver sequence, a signal from the CC&S would initiate attitude control reorientation of the spacecraft to the Sun and Canopus.

Table 4. CC&S command list

Symbol	Name	Destination	Symbol	Name	Destination
2A	Test radio	RFS	8A	Deploy solar panels	PYRO
2B	Transmit low	RFS	8B	Spare	PYRO
2C	TWT low power	RFS	8C	Platform unlatch	PYRO
2D	TWT high power	RFS	8D	High-gain antenna update	PYRO
2E	Transmit high	RFS	8E	Open press, P1/propellant	PYRO
4A	Select battery charge rate (T)	PWR	8F	Close pressurant P2	PYRO
4B	Battery charger on/off (T)	PWR	8G	Close propellant O2, F2	PYRO
4C	DAS on	PWR	8H	Open pressurant P3	PYRO
4D	UVS and IRR on/off (T)	PWR	8J	Open propellant O3, F3	PYRO
4E	TV on/off (T)	PWR	8K	Close pressurant P4	PYRO
4F	IRIS on/off (T)	PWR	8L	Close propellant O4, F4	PYRO
4H	DSS on/off (T)	PWR	8N	Open pressurant P5	PYRO
4J	Scan on/off (T)	PWR	8P	Open propellant O5, F5	PYRO
4K	DAS/TV 2.4 kHz off	PWR	8M6	Open/close engine valve	PYRO
4L	Propulsion heater on/off (T)	PWR	16A	Select 16-kbps PLBK rate	DSS
4M	Downlink on	PWR	16B	Select 8-kbps PLBK rate	DSS
4N	Downlink off	PWR	16C	Select 4-kbps PLBK rate	DSS
5A	Initiate maneuver sequence	CC&S	16D	Select 2-kbps PLBK rate	DSS
5B	Sequencer maneuver interrupt	CC&S	16E	Select 1-kbps PLBK rate	DSS
5C	CC&S B frame start enable	CC&S	16F	DSS ready mode	DSS
6A	Engineering mode	FTS	16G	DSS record mode	DSS
6B	Engineering data rate switch (8⅓)	FTS	16H	Playback mode	DSS
6C	Engineering data rate switch (33⅓)	FTS	16J	Advance to track 1 LEOT	DSS
6D	RT science No. 1 mode	FTS	20B	Switch RTS No. 2 data rate (T)	DAS
6E	RT science No. 2 mode	FTS	20C	Initiate TV mapping sequence	DAS
7A	A/C on	A/C	20D	Take TV picture pair	DAS
7B	Canopus sensor on	A/C	20G	Reset DAS orbit logic	DAS
7C	Adaptive gate step	A/C	20H	IRIS IMCC mirror enable	DAS
7D	Canopus cone angle step	A/C	20J	TV beam current on/off	DAS
7E	Autopilot on	A/C	31A	Platform clock slew positive	SCAN
7F	CC&S stray light signal	A/C	31B	Platform clock slew negative	SCAN
7G	Gyros inhibit	A/C	31C	Platform cone slew positive	SCAN
7M1	Gyros on	A/C	31D	Platform cone slew negative	SCAN
7M2	All axes inertial	A/C	31E	Select scan stow position	SCAN
7M3	Turn polarity set	A/C	31F	Select scan cone position	SCAN
7M4	Roll turn	A/C	31G	Select variable scan reference	SCAN
7M5	Yaw turn	A/C	36A	TV cover deploy	TV
7M6	A/C maneuver mode	A/C	38A	IRR mirror stow	IRR

Pre-aim logic was added for MM'71 to point the engine thrust vector through the spacecraft center of mass. A digital word that represented the required initial gimbal actuator displacement was sent from the CC&S and appropriately biased the null extension of the actuators before the commanded turns for a specific maneuver.

The orbital mode of operation was similar to the transit cruise mode except that a CC&S signal would be provided during expected stray-light conditions. The stray-light signal would be generated to coincide with conditions where the lighted crescent of Mars or reflected light from the Martian satellites entered the Canopus sensor stray-light field of view. If uncorrected, these conditions could cause the Canopus sensor error signal to be unreliable. The mode of operation during these stray-light conditions would be one in which the A/C subsystem logic, upon receiving the stray-light signal, turned on the roll gyro and started the fixed-interval timer. The inertial reference unit would be commanded into the *roll inertial hold* mode after the fixed interval, and the Canopus sensor signal would be replaced as the position referenced by signals from the inertial reference unit. When the stray-light signal is reset, the Canopus sensor would be commanded to reacquire Canopus and would be switched back into the control loop. Flyback and sweep logic was added for the MM'71 attitude control subsystem to prevent unnecessary roll searches during this sequence. All intentional spacecraft torques would be produced by the A/C gas system by expelling small amounts of cold gas (nitrogen) from the jets at the ends of the solar panels.

*8. **Pyrotechnics subsystem.*** Electrically initiated explosive devices were used for spacecraft separation from the Centaur, solar panel release, high-gain antenna position change, scan platform release, and propulsion system valve actuation. Functions were initiated by either direct command to the spacecraft or by commands stored in the spacecraft CC&S. Pyrotechnic firing was accomplished by capacitor discharge into the intended device. The pyrotechnic subsystem, including its electronics, weighed 4 kg.

*9. **Mechanical devices subsystem.*** The devices used in this subsystem (25 kg) were associated with latching, structural damping, nonservo-controlled actuation, planetary experiment support, and separation-activated switching and release. Mechanical devices included:

(1) Solar panel boost dampers.

(2) Solar panel deployment and cruise damper, including the latch and switch assemblies for indications of deployment of panels.

(3) High-gain antenna deployment mechanism.

(4) Planetary scan platform and scan platform latch.

(5) Pyrotechnic arming switch.

(6) Separation-initiated timer.

(7) Spacecraft-separation mechanisms.

(8) Spacecraft V-band clamp (separation from Centaur adapter) and ejection springs.

(9) Medium-gain antenna energy attenuation plug and deployment device.

*10. **Propulsion subsystem.*** The function of the propulsion subsystem was to provide directed impulse, upon command, to accomplish in-transit trajectory corrections, an orbital insertion maneuver at encounter to transfer from a flyby to an orbiting trajectory about the planet Mars, and subsequent orbit trim maneuvers. Empty weight of the propulsion subsystem was 87 kg; weight at launch was 577 kg including 476 kg of propellant and 14 kg of pressurant gas.

This storable hypergolic bi-propellant propulsion subsystem was an integrated, pressure-fed, multi-start, fixed-thrust subsystem that used nitrogen tetroxide (N_2O_4) oxidizer and monomethylhydrazine (MMH) fuel as propellants. Early in the propulsion subsystem design, four propellant tanks were considered and discarded in favor of two tanks (Ref. 27). The primary subassemblies of this design were a dual-tank nitrogen reservoir, a pressurant control assembly that provided pressurant isolation and regulation, two check and relief valve assemblies, two propellant isolation assemblies, a gimballed 1334-newton (300-lbf) thrust rocket engine assembly with an electrically operated bi-propellant valve, and the propulsion module structure. The rocket engine contained a thick beryllium combustion chamber which conducted heat rapidly and was cooled by fuel sprayed on the inside walls. The nozzle was made of high-temperature steel and was radiantly cooled during firings.

The subsystem was pressurized by gaseous nitrogen from high-pressure storage tanks. Welded or brazed tubing and component connections were used. Metal seals were used to minimize the effects of irradiation, hard vacuum, temperature, and long-term storage on critical subsystem joints. Multiple pyrotechnic valves, arranged in three groups with normally open and normally closed valve branches, provided the capability to isolate propellant and pressurant for the long periods of space storage. The subsystem was capable of being fueled, pressurized, and monitored before installation on the spacecraft.

At launch, the propellants and high-pressure gas supply were isolated by the pyrotechnic valve assemblies. Before the trajectory correction maneuver, the engine valve was opened momentarily to bleed the air trapped between the normally closed propellant pyro valves and the engine valve. Actuation of the first set of propulsion valves allowed the propellant tanks to build up to operating pressure and allowed propellant flows to the engine valve. Commanding the main engine valve open caused the propellants to flow into the thrust chamber and mix, undergo hypergolic ignition, and continue to burn until the engine valve was closed when the desired velocity increment was obtained. The propellant and pressurant lines were then closed to guard against leakage when tracking data confirmed that no more propulsion maneuvers would be required before the nominal time of another trajectory correction or for Mars orbit insertion (MOI).

After tracking data confirmed correct orbital characteristics, the propulsion fluids would be isolated by the operation of pyro valves for the rest of the cruise mission.

Commands for event sequencing originated from the CC&S and/or the flight command subsystem. Actuation of pyro valves and management of solenoid power was accomplished by power switching in the pyrotechnics subsystem. Thrust vector control was provided by the A/C subsystem through the use of gimbal actuators for pitch and yaw control and cold-gas jets for roll control.

11. Temperature control. The temperature of the spacecraft was controlled by the temperature control subsystem (13 kg) so that all equipment would function correctly in the potentially damaging flight thermal environment. The four major variables that affected the temperature of spacecraft elements were incident solar radiation, electrical power dissipation, thermal transfer between components, and thermal radiation of the spacecraft into space. Various passive (shields, thermal blankets, paint, polished surfaces) and active devices (variable-emittance louver assemblies) were used to achieve temperature control.

Multi-layer thermal blankets were employed on the sunlit and anti-solar (top and bottom) sides of the spacecraft. Both blankets were lightweight thermal boundaries. The purpose of the top blanket was to isolate the propulsion module and bus from the Sun; the bottom blanket minimized thermal gradients within the bus and forced the internally dissipated power to be rejected to space through the louvered bay faces. A third blanket controlled heat losses from science instruments on the planetary scan platform.

Thermostatically actuated louvers were installed on all spacecraft bays except Bays IV and VI. Bay IV was covered with a polished, low-emittance aluminum shield, and Bay VI was covered with high-emittance white paint.

12. Data storage subsystem. Many times during the mission, the spacecraft would acquire data faster than the data could be transmitted to Earth. The data storage subsystem (DSS) (11 kg) stored the data on a digital tape recorder until it could be transmitted to Earth at a slower rate. This "all digital" tape recorder was added to the MM'71 design because of its greater compatibility with the science instruments and on-board data handling equipment.

The DSS recorded data supplied by the data automation subsystem in the form of a serial stream of pulses. The data, recorded at a rate of 132.2 kbps, consisted primarily of digitized video from the television subsystem formatted with data from the other science instruments. About 32 TV pictures could be stored on the 1.8×10^8-bit capacity tape. Recording was automatically stopped (1) when the tape recorder was filled, (2) by command from the CC&S, or (3) by ground command. When the ground antennas of the DSN were ready to accept the data, the data were played back through the flight telemetry subsystem to the radio-frequency subsystem at a slower rate than recorded. Five playback data rates (16.2, 8.1, 4.05, 2.025, and 1.0125 kbps) were available and selectable by commands from the flight command subsystem or the CC&S.

13. Data automation subsystem. The data automation subsystem (DAS) (6 kg) acted as the signal interface between the science instruments and all other subsystems of the spacecraft. This subsystem:

(1) Controlled and synchronized the science instruments within a fixed timing and format structure and sent commands to the instruments as required so that the instrument internal sequencing was known.

(2) Provided the necessary sampling rates, both simultaneous and sequential, to ensure meaningful science data.

(3) Performed the necessary conversions and encoding of the several forms of science data, and placed them in a suitable format.

(4) Buffered the science data and sent it either to the flight telemetry subsystem at 50 bps, 8.1 kbps, or 16.2 kbps, or to the data storage subsystem at 132.2 kbps for later playback as appropriate.

(5) Issued and received commands that pertain to the operation of the science instruments to and from other spacecraft subsystems.

(6) Issued timing "cues" to CC&S for orbital operations sequencing.

14. Scan control subsystem. The scan control subsystem (8 kg) provided precise angular pointing control of the two-degree-of-freedom (clock and cone axes) gimballed support structure or platform, upon which the science instruments were mounted.

At launch, the scan platform was secured in the stowed position. One day after launch, a direct ground command or CC&S event signalled the pyrotechnics subsystem to unlatch the scan platform.

The scan platform would be used in the following modes: pre-orbital television, orbital science, and orbital cruise. In the pre-orbital television mode, the platform would be moved so that a series of television pictures could be taken of the planet. In the orbital science mode, the platform would be stepped sequentially through a series of pointing directions. The scan pointing positions would be directed in flight, and during the orbital sequence, by CC&S commands or ground quantitative commands.

Reference potentiometers would control the clock and cone angles for the start of the pre-orbital and orbital science sequences. The reference potentiometers were coupled through a gear train to step motors. Identical clock and cone sequencing circuits supplied pulses to turn the step and reference potentiometer motors. In a typical scan operation, the sequencing circuits received either clockwise or counterclockwise pulses, spaced one second apart, from either the flight command subsystem or CC&S. Each pulse resulted in a ¼-deg platform motion.

The scan platform could be pointed to within ½ deg of a desired direction (all error sources considered including the A/C limit cycle uncertainty); after moving the platform to the desired position, the actual direction could be ascertained to within ¼ deg on both cone and clock angles. Achievement of these accuracies was the result of a combination of prelaunch ground calibrations and inflight calibrations (Ref. 28).

15. Science instruments and experiments. The Mariner Mars 1971 experiments included television (TV), ultraviolet spectroscopy (UVS), infrared radiometry (IRR), infrared interferometer spectroscopy (IRIS), S-band occultation, and celestial mechanics. Data for the latter two experiments would be obtained by using the radio subsystem. Instruments for each of the other experiments were mounted on the scan platform and, along with the wide-angle camera (A), boresighted with the television narrow-angle camera (B).

a. Television. The television subsystem (26 kg) used in this experiment consisted of two television cameras (wide- and narrow-angle) mounted on the spacecraft's planetary scan platform. The camera optics and some parts of the supporting electronics were identical to the equipment used on Mariners 6 and 7. The wide-angle camera (Fig. 3)

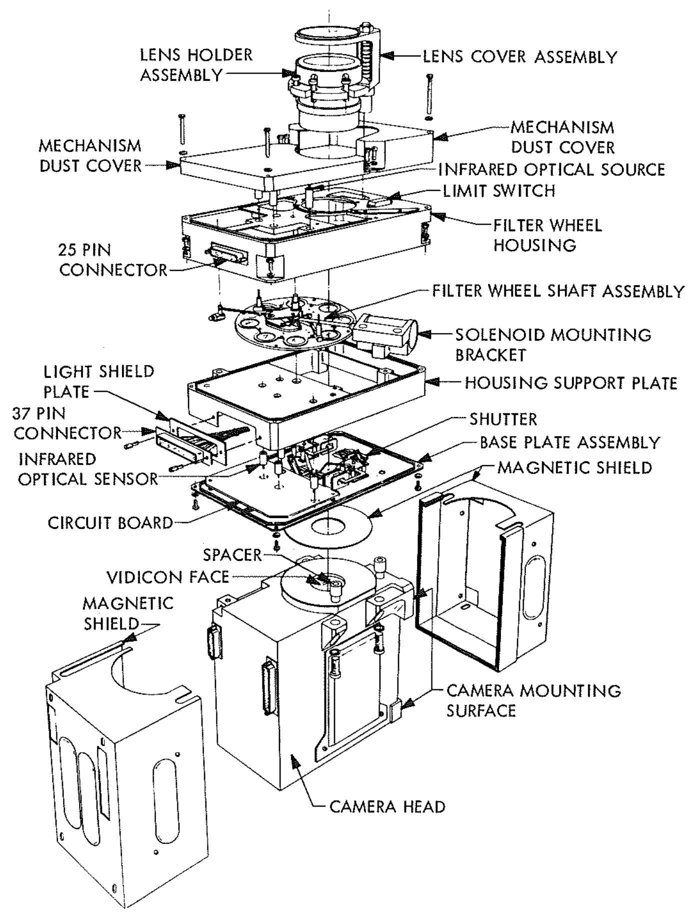

Fig. 3. Wide-angle television camera

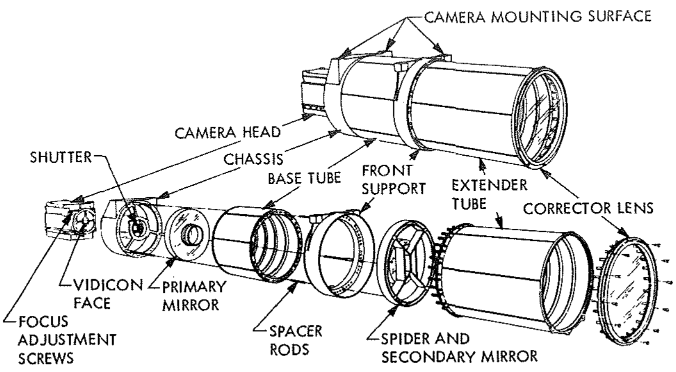

Fig. 4. Narrow-angle television camera

has a rectangular nominal field of view of 11 × 14 deg and a nominal focal length of 50 mm. The narrow-angle camera (Fig. 4) has a rectangular nominal field of view of 1.1 × 1.4 deg and a nominal focal length of about 500 mm. The resolution of objects in the field of view of each camera is dependent on the line-of-sight range to the object. With the cameras looking vertically downward at the surface and the spacecraft at an altitude of 1250 km, the wide- and narrow-angle cameras can detect objects under about 1 and 0.1 km, respectively. The television camera parameters are summarized in Table 5.

b. Ultraviolet spectrometer. The UVS subsystem (16 kg) used an Ebert–Fastie type of spectrometer. The optical view is a front surface (mirror) telescope through which ultraviolet light enters and is split into its component wavelengths by a reflection diffraction grating. Two exit slits allow two measurement channels. The detectors are photomultiplier tubes with specific photocathode and window materials that provide additional wavelength discrimination.

Table 5. Camera performance

Characteristic	Wide-angle camera	Narrow-angle camera
Focal length, mm[a]	50	500
Focal ratio	f/4.0	f/2.35
Shutter speed range, ms[a]	3 to 6144	3 to 6144
Automatic shutter speeds, ms[a]	48, 96, 192	6, 12, 24
Angular field of view, deg[a]	11 × 14	1.1 × 1.4
Active vidicon target raster, mm[a]	9.6 × 12.5	9.6 × 12.5
Scan lines per frame	700	700
Picture elements per line	832	832
Bits/picture element	9	9
Frame time, s	42	42

[a]Nominal values.

The ultraviolet spectrometer on Mariner 9 (Fig. 5) was basically the same as that on Mariners 6 and 7, with some modifications. The channel 1 photomultiplier is an F tube instead of an N tube and has a spectral range of 145 to 350 nanometers (1450 to 3500 angstroms). A step gain amplifier incorporated with this channel provides control over the expected range of surface brightness. The spacial resolution was maximized by reducing the field of view to 0.17 × 0.48 deg from 0.25 by 2.5 deg used by Mariners 6 and 7. The channel 2 photomultiplier tube (G) has a spectral range from 110 to 190 nanometers (1100 to 1900 angstroms) and a field of view of 0.17 × 1.20 deg. One spectral sweep of each channel would be recorded each 3 s.

c. Infrared radiometer. The infrared radiometer subsystem (3 kg) used an instrument (Fig. 6) to provide brightness temperatures of the Martian surface by measuring the energy radiated in the 8- to 12-μm and 18- to 25-μm wavelength bands. By using refractive optics, infrared radiation is focused on detectors, which use 13-junction bismuth–antimony thermopiles, in two independent channels. The channels have fields of view of 0.5 and 0.7 deg, respectively, and provide resolutions of about 11 and 15 km at a range of 1250 km. Although the Mariner 9 radiometer was basically the same as that flown on Mariners 6 and 7, it had been modified to provide clearer definition of the fields of view.

Infrared radiometric measurements of the Martian surface temperature would be made in each wavelength band at 1.2-s intervals.

Inflight calibration would be performed once each 42 s by pointing the 3-position scan mirror alternately at an internal thermal reference surface (whose temperature is independently monitored) and at deep space (which serves as a zero-energy reference).

d. Infrared interferometer spectrometer. The instrument used by the IRIS subsystem (23 kg) was a Michelson interferometer spectrometer similar to that used for the Nimbus III and IV meteorological Earth satellites, with modifications made in the mechanical, optical and electrical components (Fig. 7). An essential part of the instrument is the cesium iodide beamsplitter, which divides the incoming radiation into two approximately equal components. After reflection from the fixed and moving mirrors, respectively, the two beams are recombined and

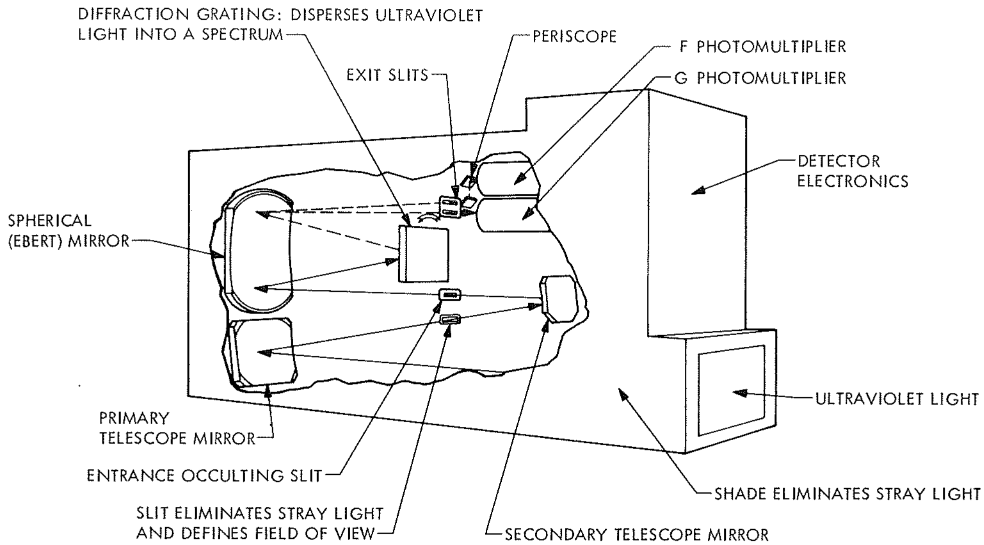

Fig. 5. Ultraviolet spectrometer

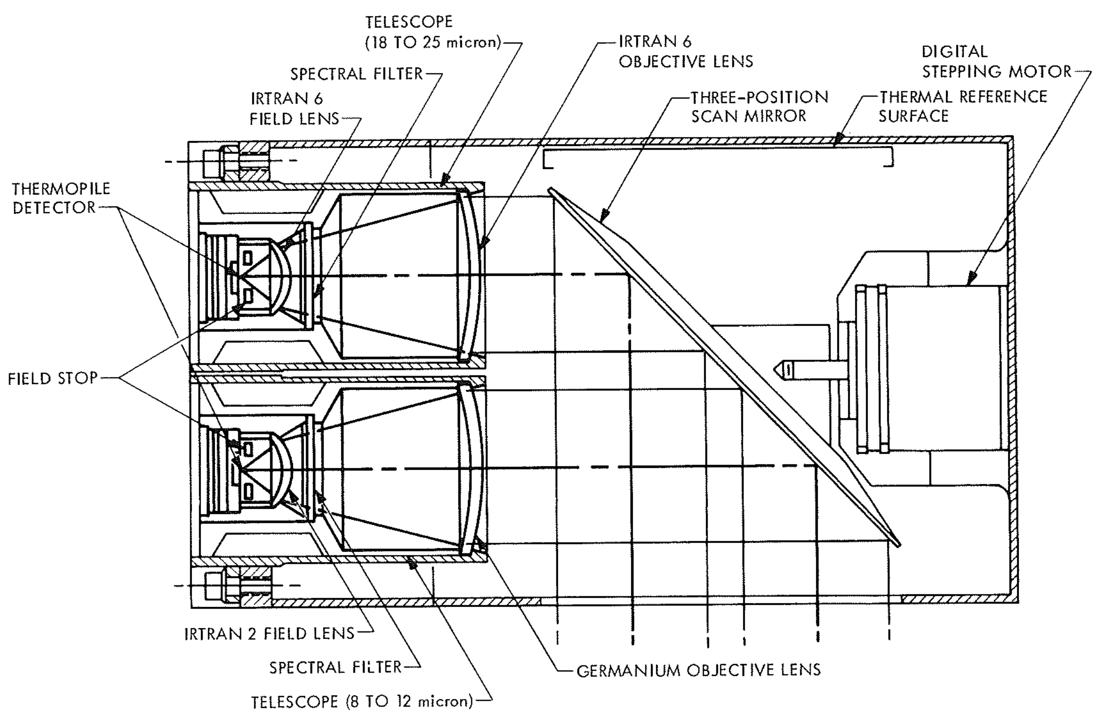

Fig. 6. Infrared radiometer

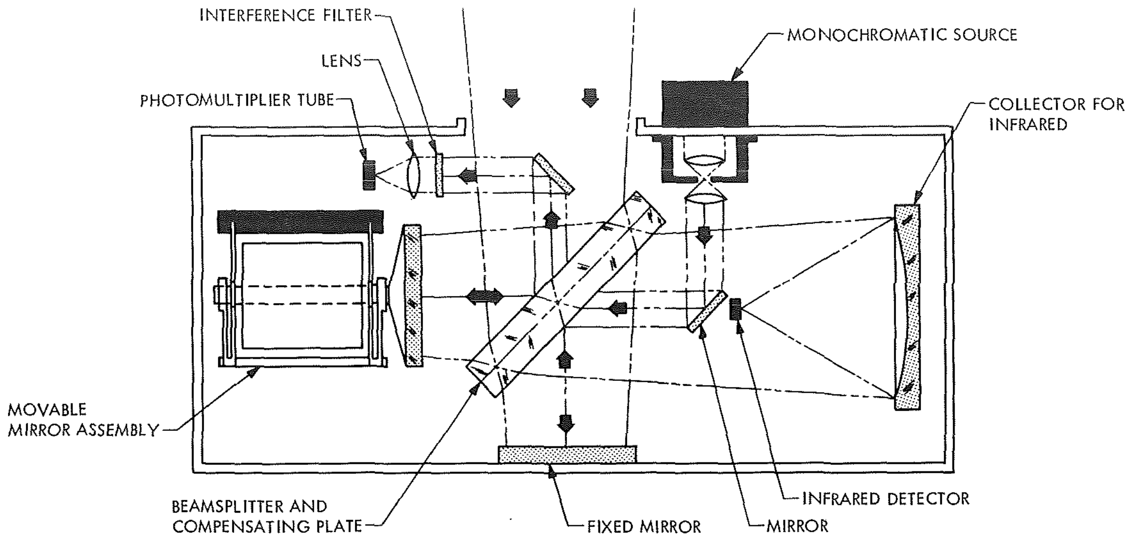

Fig. 7. Infrared interferometer spectrometer

form an interference pattern with a phase difference proportional to the optical path difference between both beams. The interference pattern is focused on the detector, where the intensity is recorded as a function of path difference. Radiometrically accurate spectra are recovered by means of extensive computer processing on the ground.

The spectral range covered by the Mariner 9 instrument is 6 to 50 μm (1600 cm^{-1} to 200 cm^{-1}), with 2.4-cm^{-1}-wide (apodized) spectral resolution elements. The spatial (geometric) resolution is about 100 km for an altitude of 1250 km with the 4.5-deg conical field of view. One spectrum (interferogram) would be recorded each 21 s.

16. Spacecraft/launch vehicle interface. Peculiar requirements placed on the Launch Vehicle System were classified in two categories:

(1) Interface (hardware) design.

(2) Mission requirements.

Hardware requirements were documented and both document and drawings were subject to negotiation and sign-off by JPL and LeRC. A panel known as the Mariner Mars 1971 Spacecraft/Launch Vehicle Hardware Interface Panel was established to have responsibility over the technical contents of the Interface Control Documents, and to ensure the timely execution of these requirements. A working group, the Atlas/Centaur/Mariner Mars 1971 Performance, Trajectory, and Guidance Working Group, was established to have responsibility over the mission requirements interface.

a. Hardware Interface Panel. Members of the Hardware Interface Panel, appointed from JPL and LeRC, were under two co-chairmen, one from each organization. Efforts of the Panel were directed primarily at the interfaces between the spacecraft and the launch vehicle, and between the spacecraft and its operational support equipment at the launch complex. Panel areas of responsibility included environmental requirements and design criteria, electrical and mechanical design requirements, electromagnetic interference, and ground support equipment requirements.

The Hardware Interface Panel held 11 meetings during the period from December 11, 1968 to February 12, 1971 at approximately 6-wk intervals at JPL in Pasadena, at LeRC in Cleveland, or GD/CA in San Diego. The Panel was superseded during launch preparations at Cape Kennedy by the activities of the Launch Operations Working Group, which had similar functions and structure, and more frequent meetings.

b. Performance, Trajectory, and Guidance (PT&G) Working Group. The members of the PT&G Working Group were comprised of personnel appointed from JPL, LeRC, GD/CA, and TRW; the Chairman was from LeRC. The purpose of this working group was to continually review, compile, evaluate, interchange data, and assess the performance of the trajectory launch period and the guidance and control requirements of the Atlas/Centaur for the Mariner Mars 1971 Project. In addition, the working group was to pinpoint problems in these areas, determine possible courses of action, and recommend solutions.

C. Spacecraft Sequence Implementation

The design of spacecraft flight sequences, in response to the Mission Design Team's identification of orbital science sequences for the two complementary missions, suggested early in the project the desirability for a set of spacecraft functional sequences that could be used as basic building "blocks" with which each science sequence could be implemented. These sequences were identified and broadened to cover the pre-orbital phase of the mission, and to include all spacecraft operations which required more than a single event to implement a given function. The spacecraft blocks were documented in the Spacecraft Block Dictionary (Ref. 29). In addition, an 1108 computer program, called SEQGEN, was designed on the basis of blocks to facilitate the production of orbital sequences using required science data (TV pictures, times, clock and cone angles of the scan platform, etc.) as inputs. These tools were transferred to the MOS organization during operations, and used extensively throughout the mission.

1. Spacecraft blocks. The set of spacecraft blocks was identified through combined efforts of the Mission Design and the Spacecraft Design Teams, and established as the basis for all spacecraft operations in flight. Since all operational sequences were required to be compatible with blocks, the latter became the standard for establishing the feasibility of all required spacecraft operations. As such, they were used as the criterion for satisfying the requirement that the system test program verified the spacecraft's capability to perform the mission. When flight operations began, the blocks were transferred to the control of the MOS, with the stipulation that any new block identified during this period had to be tested on the proof test model (PTM) before being used in the flight spacecraft.

2. SEQGEN. The 1108 program SEQGEN included all of the blocks necessary for the acquisition and return to Earth of science data. The prelaunch program was used to produce the baseline sequences generated by the Mission Sequence Working Group, and the spacecraft baseline sequences for the Spacecraft Flight Sequence (Ref. 30). After orbit insertion, SEQGEN became the primary planning and documentation tool for the spacecraft team in generating daily operational sequences, and, as such, was updated whenever new blocks were incorporated or changes were required to existing blocks.

D. Spacecraft Design Verification

1. *Environmental tests*

a. Introduction. The Mariner Mars 1971 Environmental Test Program was established within the project requirement that MM'69 basic technology and procedures be adopted to the extent possible without compromising the environmental test objectives. A successful program was conducted in compliance with this requirement.

b. Program objective and philosophy. Environmental testing of MM'71 spacecraft demonstrated that the spacecraft design was capable of performing satisfactorily throughout exposure to mission-type environments while retaining its functional integrity.

In pursuit of this objective, the environmental requirements philosophy embodied type approval (TA) of a design model and flight acceptance (FA) testing of flight equipment on both the spacecraft subsystem/assembly and system levels. Type approval tests provided equipment design verification under environmental conditions exceeding those expected during the actual mission. Flight acceptance testing demonstrated the flight equipment to be representative of the equipment design and quality verified in the TA tests.

c. Origin of test requirements. Dynamic launch environments consist of the internal nose fairing acoustic field, random vibration, and booster-induced transients. Estimates of these environments were derived from Surveyor and MM'69 flight data. The vibration test series consisted of a *swept sinusoidal* and *random vibration* combination and an *acoustic* noise test at assembly levels. In addition, swept *sinusoidal* vibration, *acoustic* noise, and pyrotechnic *shocks* were imposed at the spacecraft system level.

Thermal vacuum estimates were derived for the most critical environments anticipated for the spacecraft. These estimates were based on data obtained during past programs, on the thermal characteristics of the spacecraft, mission trajectory, and various ground operations. *Thermal vacuum*, *thermal shock*, and *temperature* tests were developed from these estimates for assembly-level qualification. System-level environmental testing consisted of exposure to thermal vacuum.

Electromagnetic compatibility (EMC) test requirements originated from environmental estimates of Cape Kennedy RFI sources, umbilical and separation connector electrical transient measurements, launch support equipment long line electrical transients, and internal spacecraft noise. The tests consisted of RF simulation, spacecraft sequencing, and umbilical and separation connector transient tests. These tests were conducted at the spacecraft system level.

d. Comparison with MM'69 Environmental Test Program. A significant portion of the MM'71 spacecraft system design and hardware was carried over from the MM'69 program. Where possible, the design verification status from MM'69 TA tests was maintained for MM'71. As a result of this commonality and program resource constraints, fewer TA tests were conducted for the MM'71 Project than for MM'69. Limited spacecraft hardware availability also dictated that MM'71 PTM equipment serve the dual role of TA test article and flight spares. Table 6 presents a comparison of the number of TA and FA assembly-level tests performed on the Mariner projects.

Another departure from past programs was the substitution of acoustic testing for random vibration testing for both PTM and flight system-level tests. The MM'71 system-level acoustic testing provided a well-controlled, more realistic set of simulations of the liftoff environment. PTM and flight system-level sinusoidal vibration was performed in the spacecraft Z-axis only. In the past, sinusoidal vibration testing was required in three axes. DTM data revealed that cross coupling was present, producing a multi-axial response of the spacecraft. The conservation of schedule time and avoidance of test control problems were also influencing factors in the decision to test in a single axis.

During the derivation of the MM'71 system-level temperature test requirements, additional consideration was devoted to those subsystems that required the system-level environment for an adequate temperature vacuum test. This approach resulted in a clearly established set of test requirements that fulfilled the objectives of spacecraft thermal vacuum testing in a minimal time duration.

A final basic difference between the MM'71 program and past programs was the conduct of formal Project reviews of the MM'71 subsystem required tests. All segments of the Project were invited to participate. This resulted in a wider understanding throughout the Project of the background relating to the establishment of the subsystem environmental test matrix. The investigation and closure of the action items from the Review Board provided additional confidence that the proposed test program was effective and complete.

Table 6. Mariner Mars 1971 assembly tests/spacecraft

Project	Number of spacecraft	Total number of tests	Tests per spacecraft
MM'71	3	474	158
MM'69	4	981	245
MV'67	2	248	128
MM'64	4	1167	292

e. Program results (assembly tests). The subsystem/assembly-level environmental test program adequately met the guidelines set forth in the test philosophy and objectives. Maximum utilization of available subsystem schedule time and Project resources was strived for throughout the program. The assembly-level tests identified the need for some design improvements that were incorporated prior to system-level testing. Additionally, a data base on subsystem response under environmental exposure was generated. This data enhanced the understanding of subsystem in-flight behavior.

f. Program results (system tests). The environmental dynamic test program demonstrated the environmental integrity of the MM'71 design and the flight worthiness of each flight spacecraft. Each system-level dynamic test adequately achieved its objectives. Deviations from the specified requirements did occur in some instances, but were thoroughly evaluated and determined to be acceptable.

The system-level space simulator testing had adequately met the MM'71 Project objectives within the guidelines set forth by the test philosophy. The space simulator system test demonstrated the capability of the total system in the thermal vacuum environment. The assembly equipment which was not tested in a thermal vacuum environment at the assembly level received this exposure on the spacecraft. The system test further demonstrated the performance of those assemblies whose assembly-level test requirements were reduced.

The Air Force Eastern Test Range (AFETR) RFI Simulation Test generally met the desired objectives. Test-level deviations occurred on two radiation sources during PTM testing, but were sufficient to evaluate the spacecraft functional performance within this environment.

The Umbilical and Separation Connector Transient Test was accomplished with some adjustments to specified requirements. The test avoided any danger of damaging circuits that could have been susceptible to the test implementation approach, but were not considered to be of concern in the expected real environment. The test met its intended objectives.

The Umbilical and Separation Connector Removal Test successfully accomplished its objective of verifying that the connector removal would not induce transients on the interrupted circuits.

The spacecraft tests, repeated frequently, provided the test for electromagnetic sensitivity and sources within the spacecraft itself, and results were very satisfactory in providing acceptance of the design.

2. *Functional tests*

a. General. Functional testing of the Mariner Mars 1971 spacecraft was accomplished by the spacecraft system test program, the purpose of which was to:

(1) Plan the process of electrical and mechanical assembly.

(2) Demonstrate subsystem specified performance and subsystem interface.

(3) Verify the design of the spacecraft in simulated mission exercises.

(4) Confirm interfaces with launch vehicle, MOS, TDS.

(5) Define normal performance for mission reference.

(6) Perform problem diagnosis and reverification of impaired hardware.

(7) Provide two spacecraft systems, tested, prepared, qualified, and ready to launch at the prescribed dates for the 1971 launch opportunity.

(8) Provide a third spacecraft system (PTM) tested to qualify spares to support the flight systems, to demonstrate design acceptance, and to serve as a pathfinder for the entire test program.

The ideal system test program would verify all parameters on the spacecraft in all operating modes and with all expected environments. However, the ability to accomplish this ideal was constrained by Earth environments, facility and support equipment limitations, simulation limitations, and inability to perform hazardous or destructive-type operations. Therefore, compromises induced by constraints were compensated by subsystem-level tests, analytical verification, or demonstration by similar units.

System test operations began with the inspection, certification, and assembly of subsystems into a complete spacecraft system; included the performance of subsystem integration tests, system tests, environmental tests, interface tests, and special tests; and concluded with launch operations at the Air Force Eastern Test Range (AFETR).

The spacecraft systems test, the major test element, was designed as a comprehensive exercise and verification of the system performance of the complete spacecraft through all mission phases and modes including backups. This test, together with the spacecraft readiness test and space environment simulation test, exercised all ground commands (direct, quantitative, and coded) and all commands generated on board the spacecraft. All spacecraft blocks were tested during the operational sequences included in these tests.

The controlling document for all spacecraft systems tests and operations was the Mariner Mars 1971 Spacecraft System Test and Operations Plan (Ref. 31). A series of Test Phase Directives (Ref. 32) based on this plan was written which formed the basis for detailed test procedures and other test material.

b. Proof test model (PTM) objectives and requirements. The PTM spacecraft (MM71-3) was used for spacecraft design verification. It was subjected to a variety of tests, some of which were conducted under more severe conditions than would normally be expected in flight (e.g., type approval tests) in order to verify design and operating margins. In general, when PTM testing revealed the need for a design change, the change was incorporated and tested on the PTM before incorporation on the flight spacecraft.

The primary objective of PTM testing was to verify spacecraft design and thereby determine the degree to which the assembled spacecraft met its design criteria. Supplementary objectives that needed to be fulfilled in order to attain the primary objective are shown in Table 7.

The test plan for the Mariner Mars 1971 PTM spacecraft was developed to assure efficient use of the scheduled test time providing the maximum knowledge of spacecraft performance. As a result, approximately 8 mo were allocated for PTM testing at Pasadena. The test schedule gave precedence to the following types of tests:

(1) Those tests that were likely to reveal design deficiencies requiring lead-time for investigation, rework, and retest.

Table 7. Proof test model spacecraft supplementary test objectives

Type of objective	Detailed objective
Interface compatibility and system performance	Verify that each subsystem, while operating on the spacecraft, meets the requirements of its functional requirement and design requirement.
	Verify that all analog, digital, and power signals from subsystem to subsystem are within tolerance and remain so throughout the test program.
	Demonstrate that the spacecraft will perform all "logic functions" in all appropriate operational modes.
	Verify that the interface circuits meet the requirements for grounding and isolation.
	Demonstrate the operational capability of all spacecraft-to-Earth data links and Earth-to-spacecraft command links.
Parameter variation and failure mode	Demonstrate the ability of backup circuits to correct for failures in primary functions.
	Verify that backup circuitry or redundant paths, when in their quiescent modes, do not interfere with the proper operation of the primary circuitry.
	Determine the effect on the spacecraft system of certain selected failure modes and develop possible corrective procedures.
	Evaluate system performance characteristics during a planned series of parameter variations of extremes of tolerance.
Environmental performance	Demonstrate that the spacecraft system will perform to specification during and after having been subjected to environments in excess of those expected during test, launch, and mission.
	Demonstrate spacecraft system operability when subjected to a simulated space environment where temperature levels exceed those expected during a nominal mission.
Miscellaneous	Verify spacecraft failure protection logic.
	Determine failure rates within the time available.
	Develop and verify system test procedures.
	Develop and train personnel.

(2) Those tests that provided data that allowed for an evaluation of the overall spacecraft design.

(3) Those tests upon which other tests were dependent.

(4) Repeat testing for development of data for statistical use.

Secondary considerations used to determine test priorities were:

(1) A greater portion of the allotted test time would be given to those spacecraft elements which were new in concept and had no previous flight experience.

(2) Emphasis would be given those tests which required a spacecraft environment, as opposed to those which could be performed off the spacecraft.

The Pasadena operations schedule was based on the considerations outlined above. When conflicts regarding the use of the Mariner Mars 1971 PTM arose, priorities were used to resolve the conflict. This schedule was developed in November 1969, but revisions were necessary during the course of test operations due to a variety of causes, such as late equipment deliveries, equipment return for repair and rework, equipment failures, problem investigations, etc.

c. Flight spacecraft objectives and requirements. The primary objective of the flight spacecraft test plan was to provide two fully qualified, flight-accepted spacecraft for the first day of the available launch period of the 1971 Mars launch opportunity.

To meet the primary objective, the total Pasadena assembly and test time (about 5 mo) was scheduled based on the following considerations:

(1) The testing was not for design verification as it was for the PTM spacecraft, but rather to verify that the equipment operated normally to design specifications. The testing was primarily concerned with acceptance and used the test data obtained from the PTM spacecraft as a standard against which to check flight spacecraft performance.

(2) The spacecraft had to operate through a complete mission sequence in a simulated space environment without major failure. A major failure was defined as one that would prohibit the successful completion of the mission; e.g., a power failure would be considered major, whereas a temperature transducer failure would be considered minor.

(3) The spacecraft would be shipped to AFETR only upon successful completion of the preshipment system test at JPL. A successful system test was defined as one in which no major failure occurred.

(4) The same procedures and careful initial power application and subsystem interface tests were to be performed on all three spacecraft (PTM and two flight spacecraft).

(5) Verification tests were required on all subsystems to ensure that their performance was within specifications and compared closely to those of the PTM spacecraft.

(6) The environmental tests, both vibration and space simulation, were scheduled to be performed as close to the shipping time as possible. The object was to give maximum assurance that the equipment which would be committed to the mission was used to meet the qualifications stated in (1) above.

(7) The testing had to ensure compatibility of the spacecraft and all other elements utilized in the Mariner Mars 1971 mission. Spacecraft data were made available to the Mission Operations System and Deep Space Network, and, in addition, formal compatibility tests were run. Compatibility tests between the spacecraft, Mission Operations System, Deep Space Network, launch complex equipment, Centaur shroud and adapter, Centaur RF systems, and supporting personnel were performed at various times during the overall test program.

The test plan recognized the importance of the PTM spacecraft in establishing the standards and procedures that were used to qualify the flight spacecraft. Therefore, the amount of time that was spent on various phases of the flight spacecraft test plan was, in general, less than that used for the PTM spacecraft. As in the case of the PTM, the flight spacecraft test schedule also underwent revision to accommodate various problems.

In general, the flight spacecraft test sequence followed the same pattern as the PTM spacecraft program. However, unlike the PTM spacecraft, where one objective was to become acquainted with the overall system behavior as early as possible, the flight spacecraft were required to successfully pass each test in the sequence before advancing to the next test in the sequence. The flight spacecraft ordinarily were not used for detailed troubleshooting as was the PTM spacecraft, but, rather, the problems were transferred to the PTM spacecraft when possible. Exploratory testing that required test setups ordinarily was not permitted on the flight spacecraft.

If the problem was peculiar to the flight spacecraft and could not be examined on the PTM or flight spare spacecraft or at the subsystem level, the problem was necessarily investigated in its own spacecraft environment. When this was the case, the cognizant engineer and quality assurance personnel gave special attention to the test to preclude equipment damage.

d. Test team organization. The spacecraft test and operations program was carried out under the direction of the Test and Operations Manager who utilized two Test Direction Teams for performance of test operations. One team was responsible for the PTM (MM71-3) and Flight 1 (MM71-1) activities and the other team handled Flight 2 (MM71-2) activities.

The test teams were supported by a documentation group, a data system group, and an operations support group. The functional organization for test activities is depicted in Fig. 8.

e. Test and operations. The Mariner Mars 1971 test and operations program began on March 10, 1970. The initiating event was the receipt of the octagon structure, scan platform, and the upper ring, lower ring, and power wiring harness for the PTM. Receipt of subsystem equipment and buildup for electrical testing continued through March 17, concurrent with buildup of the system test complex in preparation for test of the power subsystem followed by initial power application to the spacecraft which occurred on March 25, 1970. As equipment became available, it was integrated into the spacecraft system, concurrent with test operations which continued through the remaining test phases at Pasadena. The space simulation test for the PTM spacecraft was conducted in two phases. Phase I was conducted to demonstrate spacecraft functional performance while operating in a simulated space environment. Phase II was conducted to verify the temperature control design of the spacecraft. The space simulation test for the flight spacecraft was conducted in one continuous test.

Vibration testing of the PTM spacecraft was performed in the Z-axis at flight acceptance (FA) and type approval (TA) test levels. The flight spacecraft vibration tests were conducted at flight acceptance test levels only.

Flight spacecraft testing proceeded generally in the same order as the PTM operations. Simultaneous testing of two spacecraft required full participation of each test team and a two-shift operation during much of the test

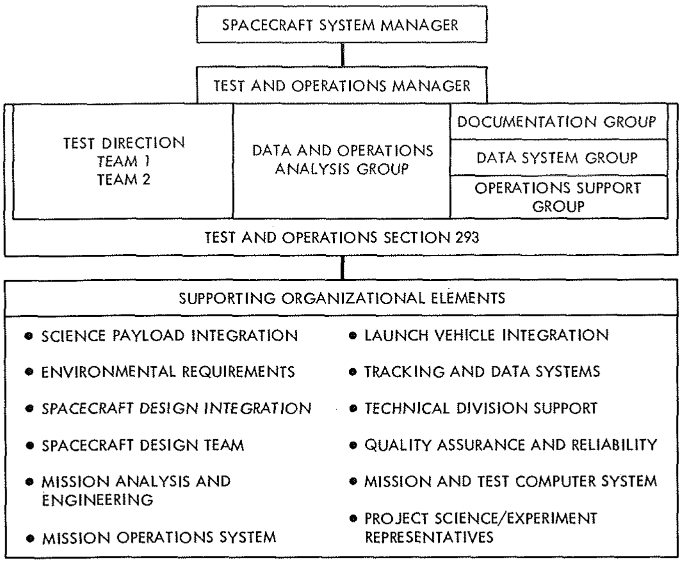

Fig. 8. Functional organization for test activities

period. The number of problems or failures was considerably less than those on the PTM and thus proved the value of a good PTM test program. Details on the problems and failures may be found in Ref. 33.

f. Mission and test computer system operations. The Mariner Mars 1971 Project selected the mission and test computer (MTC) to process the spacecraft data during system testing launch operations and mission operations. The MTC is an outgrowth of the spacecraft computer checkout facility (SCCF) used in the Mariner Mars 1964 Project, the Surveyor Project, and the latter part of the Mariner Mars 1969 Project. The SCCF was expanded by the addition of one dual processor computer and renamed the MTC prior to the start of system test operations. The SCCF included two UNIVAC 1219 computers and one UNIVAC 1218. A UNIVAC 9300 input/output terminal was added during the MM'69 Project. The MTC retained all of those computers and added a UNIVAC 1230 mobile tactical computer for the MM'71 Project. The 1230 is a dual processor used primarily as a preprocessor for high-rate data.

The MTC included the mission and test video subsystem (MTVS) for both system test and mission operations. The MTVS consisted of two media conversion film recorders (MCFR), scan converters, a data disk, a high-resolution (1000-line) monitor system, and film processing capability. The system test operations used only the two film recorders and the associated film processing. The film recorders were basically the two recorders from the Surveyor video data processing subsystem (TV-1 and TV-11). The two recorders were extensively modified and rebuilt to improve the quality of the product. One of the film recorders (formerly TV-11) was mounted in an air-ride trailer van, which also included a film processing capability. This stand-alone van was used at the Air Force Eastern Test Range (AFETR) during the MM'71 prelaunch test operations.

The MTC processed a wide variety of raw data which included five telemetry streams, two ground command streams, one on-board command stream, many discrete status and event signals, one multiplexer and analog-to-digital converter output, two frequencies requiring counting, and several parallel shifted signals. With the raw data, much subsequent data processing was required. This special processing included the data from the science spectral instruments and the television subsystem.

The MM'71 Project instituted the concept of a centralized source of system test data processing requirements. The concept was implemented by the origin of a Data Processing Review Board (DPRB). The DPRB chairman worked directly for the Mission Operations System (MOS) Manager. The MOS Manager was responsible for the development of all project data processing including the MTC system test activities. The DPRB included permanent members from the Data Systems Division, Science Ground Data Handling Sections, the spacecraft design area, and the mission operations area. In addition, *ad hoc* membership was used for special purposes.

The functions of the DPRB were to generate a cohesive, integrated set of data processing requirements and to accept the subsequent processing capabilities for use in the system test operations. The data processing requirements consisted of both the spacecraft system-level requirements and the special processing requirements of the spacecraft subsystems and science instruments. The DPRB contributed to the design of the MTC software system where the handling of the spacecraft data was concerned.

The MTC was capable of supporting two spacecraft simultaneously throughout the system test operations. Each UNIVAC 1219 could support one spacecraft. The 1219 concurrently processed the two low-rate telemetry channels as telemetry or as direct access, the ground and on-board command streams, the myriad of diverse direct-access data, and the data from the analog-to-digital converter/multiplexer (ADC/MUX). In addition, at any one time, the MTC could process one of (1) the science 132.3-kbps umbilical data stream, (2) the autopilot data, (3) the data storage subsystem playback, or (4) the central computer and sequencer (CC&S) load and verify data. These four processors were programmed to be overlays because of 1219 magnetic core storage limitations. The CC&S load and verify processor was deleted during the system test operations because of known hardware interface and possible software problems. When the UNIVAC 1230 mobile tactical computer became operational near the end of calendar year 1970, it could process the high-rate telemetry channel from one spacecraft. The dual spacecraft support capability in the 1230, though available, was never made operational because the spacecraft system test schedule never required it. The UNIVAC 1218 was used only for nonreal-time processing throughout the system test operations.

The MTC and MTVS were shipped in stages to AFETR for launch operations beginning in early February 1971 with the MTVS van. One 1219 and associated peripherals were sent with the PTM spacecraft and the STC. The second 1219, the 1230, and the 9300 were shipped with the second flight spacecraft.

A 50-kbps wideband full-duplex telephone line for data transmission was provided by the DSN for intra-MTC use. The wideband data line was used for two purposes: to transmit data from computer to computer or to drive the character printers and low-speed line printers at JPL. One of the two 1219s was used at AFETR and the 1218 at JPL for the inter-computer link. The printers at JPL used to display AFETR data were those in the mission support area in the Space Flight Operations Facility (SFOF).

The MTC experienced considerable difficulty at AFETR in establishing acceptable hardware operations for spacecraft support. This was believed to be caused by the poor electrical grounding system which was available for MTC use and a high-temperature problem which affected the interface electronic equipment. Much effort was put into establishing temperature stabilization at acceptable levels and in electrical grounding. The effects of the problems were reduced significantly but never totally resolved.

One 1219 and the associated peripherals and interface electronics were shipped back to JPL immediately after the second launch. This 1219 was used to support the midcourse maneuver. The remainder of the MTC and the MTVS van were returned to JPL with the PTM spacecraft.

3. Reliability and quality assurance

a. Introduction. The Mariner Mars 1971 Project Plan (Ref. 34) was the governing document for the Project and contained, among others, the requirements for the reliability and quality assurance programs. Reference was made in this plan to the NASA Project Approval Document (PAD) which required Project surveillance over quality assurance and reliability assurance programs.

b. Quality Assurance Program. The Mariner Mars 1971 Quality Assurance Program was based on a series of requirements and implemented activities which took place during all program phases, and were designed toward ensuring maximum mission success. The provisions of NPC 200-2 were implemented by JPL and JPL's subcontractors, with Ref. 35 as a basis for the MM'71 quality assurance system. These provisions were reflected in individual subsystem Quality Assurance Plans which were contractor-generated and JPL-approved.

One of the major tasks accomplished by quality assurance was the surveillance of parts screening vendors and a rigorous control of parts from the screening vendor to the subsystem builder. Since the major part of the flight subsystem hardware was built by subcontractors, a rigorous quality assurance inspection task was undertaken. This task included the inspection of all hardware at mandatory control points by JPL quality assurance personnel, and a final review of all as-built documentation prior to shipment. These disciplines were also imposed on science instruments.

Once at JPL, the hardware was independently inspected by quality assurance engineers to assure flight worthiness. The hardware was subsequently maintained in flight status by constant quality assurance surveillance. This surveillance was imposed until launch.

In summary, the MM'71 Quality Assurance Program was a hardware-oriented, closed-loop inspection system. This type effort continues to be the most appropriate for low hardware volume planetary programs.

c. Reliability Assurance Program. The MM'71 Reliability Assurance Program (Ref. 36) consisted of the following tasks:

(1) Developing system-level reliability plans.

(2) Participating in design reviews.

(3) Monitoring reliability plans.

(4) Problem failure report (PFR) activity.

(5) Monitoring system PFR activities.

(6) FMECA (failure mode, effects, and criticality analysis).

(7) Parts and materials activity.

(8) Engineering support to FMECA, problem/failure analyses, etc.

(9) Appraising the MM'71 Project Office of risk concerns.

(10) Reporting significant deviations.

Of these many tasks, three were felt to have had significant impact on the Project. These were design review, electronic parts reliability, and problem/failure reporting. In addition, the FMECA activity had special emphasis.

d. Reliability activities

(1) *Design reviews.* The three basic formal design reviews utilized on the MM'71 Project (Ref. 37) were functional, detail, and hardware reviews. Reliability considerations were an important facet of each of these reviews, including:

(1) Review of MM'69 PFRs as applicable.

(2) Failure mode analysis.

(3) Parts reliability.

(4) Review of MM'71 PFRs (hardware reviews only).

The functional design reviews for the spacecraft subsystems were relatively unchanged from MM'69 requirements. The Reliability Assurance Office took an early lead in discussing and describing the functional design reviews to be held, and performed an evaluation of the effectiveness of the functional design reviews by use of an evaluation criteria work sheet. This work sheet provided a shopping list of potential deficiencies, and early corrective action was achieved to strengthen the agenda and topics to be covered at the remaining design reviews.

Participation in the three series of reviews gives an indication of importance and intensity of effort brought about by the reviews. Participation was as follows:

Reviews	Number of reviews	Average board members	Average attendance
Subsystem:			
Functional	12	9	35
Detail	20	8	35
Hardware	63	5	12
System:			
Functional	1	9	70
Detail	1	9	70

(2) *FMECA.* To effect some standardization in failure mode, effect, and criticality analyses (FMECA) performance, a guideline document (Ref. 38) was developed early and utilized as a source of "how" to conduct an FMECA. Several design changes and test planning changes resulted from MM'71 FMECAs.

The effectiveness of the FMECA task was not so much in its documentation but in doing the analysis, wherein the documentation showed the results. To summarize the known studies, a report (Ref. 39) of subsystem FMECAs was issued in October 1969. This report, along with logic and state diagrams, was used as a tool by systems personnel to understand and identify operating modes and failure modes of the spacecraft.

(3) *PFR Center management.* The JPL PFR Center maintained responsibility for PFR distribution, filing, approval coordination, and status reporting. The quality and efficiency of all of these functions were significantly improved over previous projects. A total of 2423 PFRs was processed within about two years (through launch) and distributed to 25 different combinations of cognizant personnel. Readability was greatly improved over previous projects due to the use of offset printing rather than "ditto." The most significant advance, however, was the implementation of the Mark IV System, a new file management software program which was used for recording PFR data and for generating periodic standard reports and special reports.

The Mark IV System has been an extremely valuable tool for the MM'71 Project. All data from the PFR form were recorded, except the text. A three-line summary was recorded for the "Description," "Verification and Analysis," and "Corrective Action." These data were then sorted in many logical combinations and printed out as status, management, or analytical reports. About 75 special types of report format were generated.

(4) *Electronic parts reliability.* Electronic parts reliability activities were a major effort throughout the MM'71 Project. These activities included the Component Parts Investigation Committee (CPIC), special parts, and electronic part reliability analyses and special tests. The CPIC meetings and special parts review were conducted primarily during 1969, whereas electronic part evaluation was conducted throughout the MM'71 Project.

(a) *Component Parts Investigation Committee.* The purpose of the CPIC was to review inadequate component parts in subsystems which were planning to use existing hardware (with or without modifications) and/or build copies of existing MM'69 designs for MM'71. The subsystems reviewed contained 17,213 component parts, 70% of the total number of MM'69 baseline component parts. Discussions and action items were concerned primarily with component parts which had an MM'69 Parts Control Program rating of Class 4, or a high ground test failure rate on the MM'69 Project. Of these parts, 185 were upgraded and 21 parts were left unchanged.

(b) *Special parts.* Special (proprietary) parts were the subject of many of the most significant MM'69 PFRs throughout this Project. These parts are not subjected to the same extensive part specialist review and testing (qualification, life, and screening tests) as experienced by electronic parts. The MM'71 Reliability Office was therefore assigned responsibility for developing a complete list of MM'71 special parts. Design, development, and testing were monitored for 72 part types, including approximately 32,000 individual parts. Each part was classified periodically, with the same Class 1 through 4 system used for electronic part classification (Ref. 40). Project and Division Management attention was focused on special parts which were a risk to the MM'71 mission. Nearly all serious problems on MM'71 special parts (such as IRIS pyroelectric detector, TV vidicon, RFS TWTs, and propulsion bladder) were identified and solved at an early phase of the Project.

(c) *Electronic part reliability analyses and special tests.* Throughout the MM'71 Project, JPL reliability engineers maintained a careful observance of part failures, NASA alerts, and reports from other NASA Centers. The purpose of this effort was to eliminate unreliable parts from the MM'71 spacecraft design at the earliest possible phase of the Project. Particular emphasis was placed on identifying generic parts deficiencies associated with a particular manufacturer or a specific lot. The use of Mark IV EDP printouts of all MM'69 part failure data and periodically updated MM'71 data was extremely valuable in this effort. The following analyses and special tests are examples of the activities related to eliminating unreliable parts:

(1) Motorola transistors with one-mil aluminum wedge-bonded leads were identified by a NASA alert as a potentially unreliable part in some circuit applications. An extensive study of testing at several NASA Centers and NASA contractors, MM'71 circuit applications, MM'69 and MM'71 part failure data, etc., resulted in replacement of several of these transistors.

(2) Concern over poor bonding and metallization defects in National LM 709, 710, and 711 integrated circuits (ICs) resulted in analysis of samples from several subsystems, replacement of LM 710s in the infrared interferometer spectrometer, and procurement of spare ICs from another manufacturer.

(3) Other extensive analyses and testing were conducted for UVS Vitramon ceramic capacitor capacitance drift, TV and IRU CRC polycarbonate capacitor shorting, TV TI 2N2905A transistor "channeling," IRIS 2N5093 solid-state device transistors with bad bonds, CC&S Teledyne relay failures, Signetics IC failures due to chlorine contamination, IRIS reverse bias on Sprague "wet slug" tantalum capacitors, TV Custom capacitors, FTS TI FET failures due to electrostatic discharge, and transformer redesigns for IRR, UVS, power, and TV.

Each of these parts problems was resolved through cooperative efforts of part specialists, cognizant engineers, reliability engineers, quality assurance engineers, and the MM'71 Project Office Product Engineer.

(5) *Problem/failure reporting.* This task was the prime task and involved the greatest level of effort. The PFR effort received considerable interest and continued support from the Project Manager. PFRs became the singular means of logging and identifying problems and inciting action to correct the problems. Details of how the mechanics of the system were organized and conducted are described in Ref. 33.

The Problem/Failure Reporting Program implemented by the Mariner Mars 1971 Project provided a closed-loop procedure for reporting, analyzing, defining corrective action, and verifying the accomplishment of correction. Project requirements provided for the initiation of a report to document all incidents of failure, problem, malfunction, anomalies, and nonstandard or unexpected results. Reports were initiated by the person who observed the problem/failure of all deliverable hardware items, starting from the first functional checkout of devices or subassemblies subsequent to part screening. Electrical check of two or more parts was defined as the start point for problem/failure reporting for electrical/electronic equipment. Many subsystems began reporting for developmental or prototype hardware, with a total of 161 PFRs for 17 different subsystems. PFRs for support equipment (SE) were required from at least the start of equipment element functional checkout, prior to use with spacecraft system equipment, and through all operations in conjunction with or associated with its use on deliverable spacecraft equipment.

PFRs were automatically flagged in bi-weekly status reports as "delinquent" or "red flag" if they were not closed out within the 30-, 60-, and 90-day time limits specified by Ref. 41. The "red flag" classification was also assigned for problem/failures considered critical in respect to achievement of spacecraft performance requirements. This system provided an excellent source of Project and division management visibility, to ensure that problems were being resolved in a timely manner.

(a) *PFR form changes.* Three significant changes were incorporated in MM'71 PFR forms, which have greatly enhanced PFR solution, trend analysis, and risk evaluation. These three changes from the MM'69 PFR form were as follows:

(1) System test and flight PFRs have a blank for Greenwich Mean Time (GMT), to record the time the anomaly occurred.

(2) Cause of failure categories, including subcategories, was specifically defined.

(3) Four classes (Ref. 41) were established for rating combined subsystem and spacecraft system "cause status and residual risk."

(b) *Subsystem PFR summary.* Table 8 provides a subsystem summary of all MM'71 prelaunch PFRs.

(c) *MM'69 versus MM'71 PFR comparison.* Nearly all MM'71 subsystems exhibited a significant decrease in the total number of flight hardware PFRs compared to MM'69. This is probably due to the fact that many MM'71 subsystems utilized MM'69 spare flight hardware and produced new subsystems from MM'69 drawings, with few design changes. The completely new propulsion subsystem was the only subsystem that experienced a large increase in quantity of PFRs (from 51 on MM'69 to 213 on MM'71).

The total number of prelaunch PFRs was about the same for the MM'69 and MM'71 SE, but about 900 less for MM'71 flight hardware PFRs. Only three flight configuration units were built for MM'71 subsystems, compared to four for MM'69. A review of Fig. 9 clearly indicates a more rapid increase in total PFRs during the early subsystem test phase of the MM'71 Project. This is attributed to the fact that MM'71 subsystem testing began earlier due to "carryover" of MM'69 hardware. As a result, a much greater percentage of design and manufacturing deficiencies was discovered and corrected prior to start of spacecraft assembly.

(d) *Cause of prelaunch flight hardware PFRs.* Of the 1844 prelaunch flight PFRs, analysis determined that the failure causes were: *design* = 550 or 29.8%; *workmanship* = 268 or 14.5%; *parts* = 224 or 12.1%; *manufacturing* = 143 or 7.8%; *operator error* = 128 or 6.9%; *support equipment* = 136 or 7.4%; *damage* = 49 or 2.7%; *adjustment* = 32 or 1.7%; and *other* (inclusive of unknown) = 314 or 17%. Within the *design* category, 210 or 38.2% were functional or application-induced and 164 or 29.8% were associated with specification or tolerance callout inadequacies.

The number of EMI (electromagnetic interference) PFRs was significantly reduced from a total of 351 on MM'69 to 168 on MM'71. The most significant reason for this reduction was the formation of an EMC Panel which accomplished the following:

(1) Upgraded EMC requirements (design and test) based on MM'69 problems.

(2) Gave special attention to MM'71 hardware similar to MM'69 hardware which had EMC problems.

(3) Carefully examined new hardware for potential EMC problems and assessed the need for design changes or special tests to evaluate for EMC.

The specific cause was not determined for 59 part problems and 170 subsystem/spacecraft problems. In all cases, the failure was analyzed and determined to be no risk or constituted an acceptable risk to the mission.

(e) *Science instrument PFRs.* Several interesting observations can be made in reviewing the curves on Fig. 10 for cumulative PFRs for the MM'71 data automation subsystem (DAS) versus the four science subsystems. The

Table 8. Prelaunch PFR summary report

Reference designator	Action responsibility	Total PFRs[a]	Development total[b]	System flight[c]	Flight total[d]	System SE[e]	SE total[f]	Spacecraft risks[g]				Safety[h]
								1	2	3	4	
2000	Systems	68	1	48	48	19	19	50	18			2
2001/ 2101	Structure	48	13	16	31	1	4	46	2			
2002/ 2102	RFS	440	20	44	289	40	131	321	108	4	5	1
2003/ 2103	FCS	124	1	7	71	16	52	98	13		13	1
2004/ 2104	Power	71	6	14	41	14	24	65	5	1		2
2005/ 2105	CC&S	165	2	24	144	14	19	126	21	2	16	
2006/ 2106	FTS	116	3	4	62	40	51	96	17	2	1	
2007/ 2107	A/C	357	15	16	286	19	56	324	30	2	1	4
2008/ 2108	Pyro	46	4	8	23	7	19	41	5			2
2009/ 2109	Cabling	39		17	26	13	13	32	6		1	2
2010/ 2110	Prop	213	31	3	163	1	19	183	11	4	15	
2011/ 2111	Temp control	17	5	8	12			15	2			
2012/ 2112	Devices	27	2	8	23	2	2	24	3			
2016/ 2116	DSS	120	14	11	88	7	18	91	20	3	6	
2017	S-band antenna	15	5	3	10			13	2			
2020	DAS	66	13	1	51		2	60	3	1	2	1
2031	Scan	23		1	23			23				
2034	UVS	44	2	9	42			37	4	3		
2036	TV	111	22	30	81	1	8	87	11	8	5	
2038	IRR	38	2	6	29		7	33	5			
2039	IRIS	146		46	143	1	3	113	19	5	9	
21XX	STCE - SE	33				31	33	32	1			
2120	Science-SE	38				38	38	23	15			
2500	ETE	58					58	55	3			1
		2,423	161	324	1,686	264	576	1,988	324	35	74	16

[a]Total PFRs = development + flight + SE PFRs.

[b]Development total = total PFRs for development or prototype hardware.

[c]System flight = total PFRs which occurred during spacecraft system testing on the PTM and two flight subsystems. A low number in this column indicates early design maturity.

[d]Flight total = total PFRs on above 3 units which occurred in subsystem and system testing.

[e]System SE = all SE PFRs which occurred during or in support of spacecraft testing.

[f]SE total = total SE PFRs in subsystem and system testing.

[g]Spacecraft risk 1 through 4 = total PFRs classified in spacecraft risk categories 1 through 4.

[h]Safety = PFRs with a real or potential safety hazard to personnel.

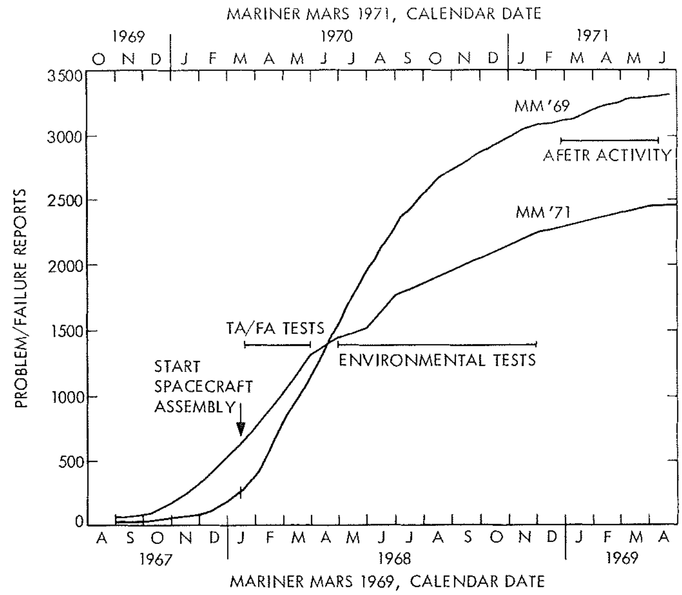

Fig. 9. Mariner Mars 1969 and Mariner Mars 1971 total problem/failure reports vs date, spacecraft system flight hardware, and OSE

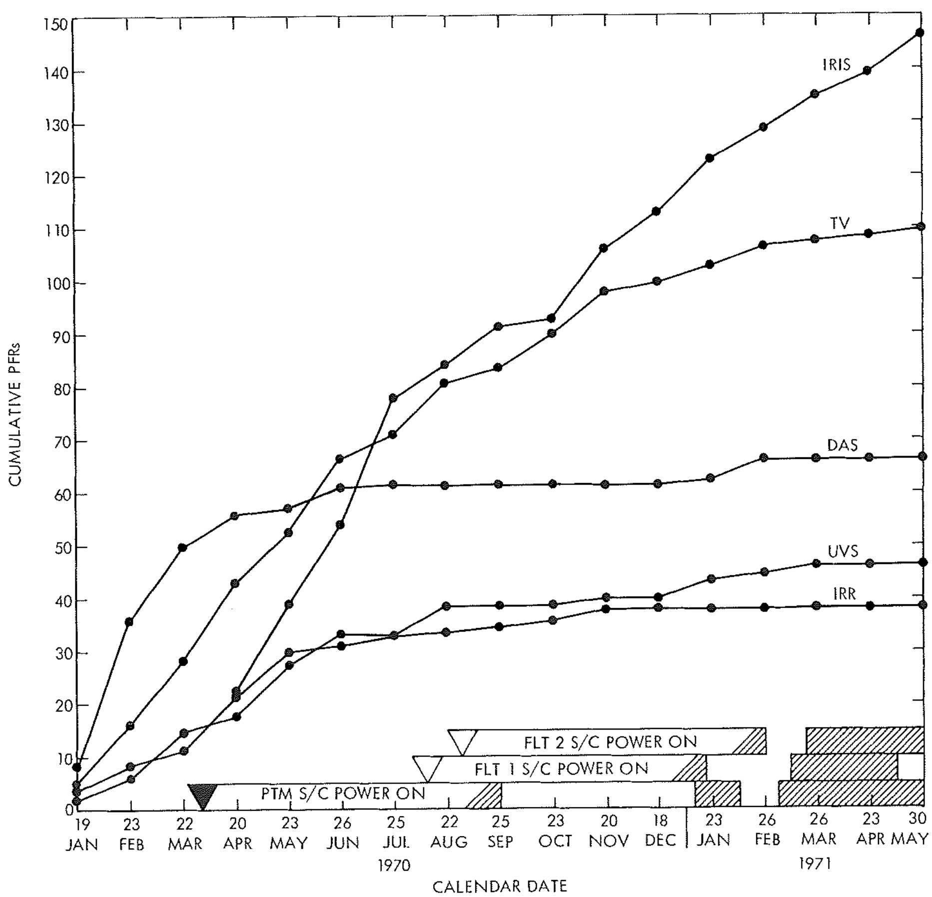

Fig. 10. Cumulative PFRs: DAS vs science subsystems

DAS and the scan control subsystems were extremely reliable during all prelaunch spacecraft system testing, with only one flight system hardware PFR written against each subsystem. The DAS cumulative PFR curve, therefore, represents the "ideal" situation, wherein nearly all deficiencies are discovered and corrected during subsystem testing. Note that the infrared radiometer (IRR) and ultraviolet spectrometer (UVS), with carryover MM'69 hardware and only slight design changes, experienced a PFR history similar to the DAS. Both subsystems experienced only a few significant PFRs during system testing.

The MM'71 television subsystem (TVS) incorporated several significant design changes from the MM'69 configuration and had almost twice the number of electronic parts (2633) as the UVS (872) and IRR (532) combined. The rate of TV PFR initiation decreased gradually during system testing. Most of the seven TV PFRs that required individual design changes occurred early in subsystem and system testing.

The infrared interferometer spectrometer (IRIS) was flying for the first time on a Mariner spacecraft. The IRIS has several complex, critically aligned optical and electromechanical subassemblies and almost as many electronic parts (2347) as the TVS. The critical alignment, extreme sensitivity to EMI, and parts problems contributed to many of the 30 IRIS PFRs, which resulted in separate design changes. These were also major causes for the IRIS total of 46 PFRs during spacecraft system testing, the largest number for any MM'71 subsystem.

The IRIS was also the only MM'71 subsystem for which problem/failure reporting did not start from the first functional checkout of devices or subassemblies subsequent to parts screening. IRIS problem/failure reporting covered only malfunctions (except those attributed to workmanship or operator error), starting with subsystem integration testing, the first time that all units of a complete instrument were operated together, both electrically and mechanically. This late start (about 3 mo) in problem/failure reporting was a significant factor in limiting Project visibility of early IRIS design and development problems.

E. Planetary Quarantine

The objective of the NASA planetary quarantine policies as applied to Mars is to prevent the transfer of terrestrial life to Mars, a planet of biological interest, so that life detection experiments will not be invalidated and the planet's environment will not be irreversibly altered.

The Mariner Mars 1971 Project in accordance with its Planetary Quarantine Plan (Ref. 42) analyzed the probability of contaminating Mars with viable terrestrial microorganisms carried on or ejected from the spacecraft. A mathematical model was constructed to allocate and to estimate probability of contamination associated with identified contaminating sources or events. Mission strategy, including aiming point biasing and orbit periapsis altitude selection, was developed to satisfy the probability allocations for accidental spacecraft impact. The results of the prelaunch analysis were published in Ref. 43.

Based on the planetary quarantine analysis, large surface areas of the spacecraft were determined to be principal sources of microbiological ejecta which could result in Mars biological contamination. Four zones of the spacecraft (high-gain antenna, solar panels, structural elements, and thermal blankets) were designated as critical areas. The microbiological monitoring and cleaning activities were concentrated on these surfaces.

To assure that the upper permissible microbial level at the time of encapsulation would not be exceeded, the spacecraft were assembled, tested, and encapsulated in Class 100 laminar downflow tents (Fig. 11). Also, clothing and access restrictions for personnel were established, and an extensive cleaning program using isopropyl alcohol on critical spacecraft surfaces was implemented. These measures were part of the spacecraft contamination control effort as delineated in the Contamination Control Plan (Ref. 44). Microbiological assays (Fig. 12) were taken using the swab–rinse method in accordance with Mariner Mars 1971 Microbiological Assay and Monitoring Plan. The United States Public Health Service verified the assays. The final encapsulation estimate for Mariner H was 1.3×10^5 estimated spacecraft microbial burden on exposed surfaces, which did not exceed the permissible Mission A upper limit of 3.1×10^6. The estimated microbial burden on the exposed surfaces at encapsulation of the Mariner I spacecraft was 3.1×10^4, which was well within the permissible limit of Mission B, or of any proposed mission plan.

The analysis and microbiological assay results indicated that the planetary quarantine constraints for the orbiter mission were satisfied.

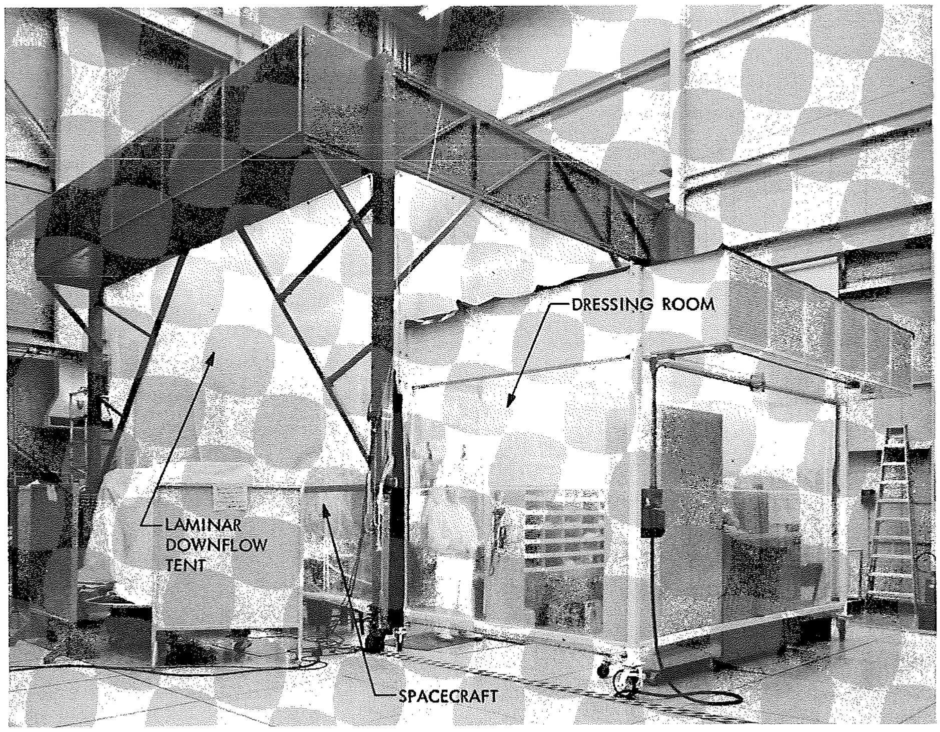

Fig. 11. Spacecraft laminar downflow tent

Fig. 12. Microbiological sampling of the Mariner Mars 1971 spacecraft

V. Mission Operations System

A. MOS Requirements

The MOS design was most significantly affected by a basic philosophy of the Mariner mission design. Rather than develop a mission design directed toward guaranteeing a specified level of performance or science return, the Mariner Mars 1971 mission design activities were directed toward providing a broad framework within which the mission could respond to actual performance and to the character of the science data as received and analyzed. Thus, the mission operations design had to be sufficiently flexible to allow modification of the mission plans to take full advantage of flight and ground equipment capabilities and to accommodate to the maximum extent possible any shifts in science emphasis. It was also a primary mission objective to:

(1) Develop necessary hardware, software, strategies, and procedures for, and to demonstrate the capability of, conducting orbital operations at planetary distance with two spacecraft simultaneously. Orbital operations were defined to include insertion into Mars orbit, orbital trim maneuver(s), science data acquisition, engineering and science data transmission to Earth, orbital metric data acquisition, ground data handling, processing and analysis, and spacecraft command and control in orbit.

(2) Develop and demonstrate the capability of conducting orbital operations in an adaptive mode whereby the data from one spacecraft revolution is used to influence the operation of the spacecraft on subsequent revolutions. The adaptive mode was intended to provide for the enhancement of science data value and to permit full exploitation of targets of opportunity.

(3) Develop a mission design which provided the maximum degree of achievement of mission objectives, given a degraded spacecraft performance, and also allowed enhancement of objectives, given better than nominal performance.

These requirements implied the development of navigation capabilities of sufficient accuracy to deliver the spacecraft into the proper orbit and to determine the location and viewing characteristics of the science instrument data. The single accuracy requirement most important to the navigation effort was the requirement to control instrument pointing to 0.5 deg relative to inertial space.

The mission requirements also implied the generation of new types of mission control programs. These programs were required for the daily conversion of scientists' desires of targeting into the required CC&S program to execute those desires on the spacecraft. These programs started with the scientific desire to target the instruments on a certain location on the planet and with certain viewing conditions at that location. The navigation programs, knowing the location of the spacecraft and planet, generated the spacecraft instrument pointing requirements and passed these requirements onto engineering programs. The engineering programs converted the pointing requirements to the scan platform stepping requirements, shuttering requirements, filter setting requirements, etc. These spacecraft activities were then passed to a program that generated the CC&S program, which caused those spacecraft activities to be executed. This same program then generated the set of commands which must be sent to the spacecraft to properly load the on-board computer.

The requirement to use data during the orbital mission to affect future activities implied: a requirement to display the data as it was obtained and to provide analysis capabilities with the data in a short time scale relative to the 90-day orbital missions. The quantity of data obtained also implied a requirement for complete records, not only of the telemetry data returned, but of all the supporting data which accompanied each telemetry measurement.

B. Organizational Plan

The length of MM'71 orbital operations made it necessary to organize for mission operations in such a way as to minimize the demand for specialists in technical disciplines (such as navigation, and spacecraft and instrument design) to support ongoing operations.

The organizational plan, therefore, was to employ a two-tier concept in which one tier was occupied by those organizational elements principally involved in mission operations analysis and planning and the other tier by those elements involved in continuous, real-time monitoring and operation of the spacecraft and the ground system. The responsibilities of the former normally were satisfied by scheduled daily participation; the latter required continuous support of varying levels through the missions. Support requirements were geared to mission phase and status, with minimum staffing during the cruise phase, normal staffing for orbital operations, and extra staffing for maneuvers.

The mission operations organization, as shown in Fig. 13, was headed by the Chief of Mission Operations

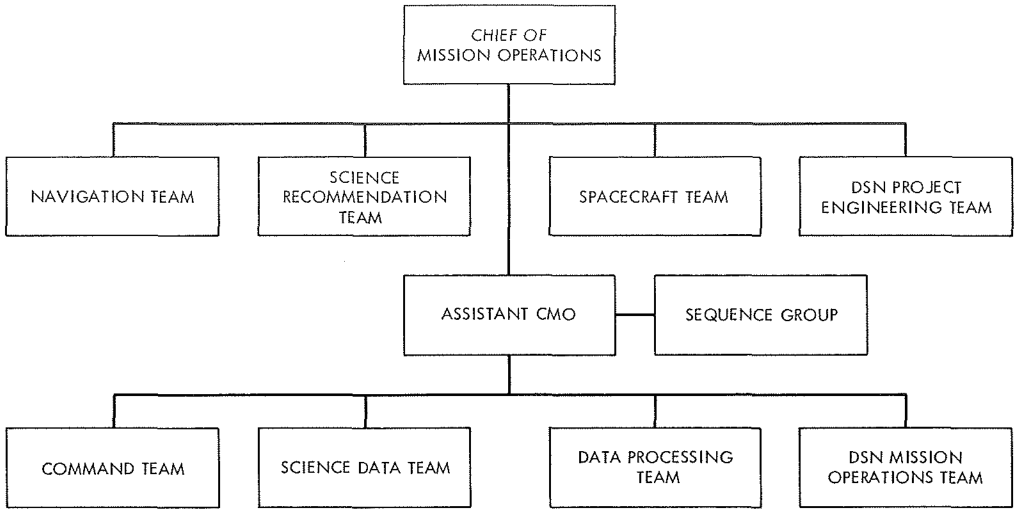

Fig. 13. Mariner Mars 1971 mission operations organization

(CMO), who was responsible to the Mission Manager and MOS Manager for the conduct of the missions. Mission conduct was construed to mean mission operations plan development, approval, and execution.

During orbital operations, the Chief of Mission Operations prepared daily an updated and revised mission operations plan. This was based on planning teams' analyses and recommendations, and on Science Recommendation Team inputs concerning desired science operations and priorities for the ensuing period. The science plan was reviewed by the CMO and the other planning and analysis team chiefs for compatibility with spacecraft and ground operations limitations and Project guidelines. That review resulted in an updated operations plan which the CMO delivered to the Assistant Chief of Mission Operations (ACMO) to be translated into specific spacecraft command sequences and directions for ground activities to be carried out by the mission operations team.

The mission operations organization consisted of two major groupings of elements: the planning and analysis teams and the mission operations teams.

1. Planning and analysis teams. The planning and analysis teams consisted of:

(1) A *Navigation Team*, whose principal functions were spacecraft navigation and scan geometry analysis.

(2) A *Science Recommendations Team*, whose principal functions were to analyze science data and recommend science operations' plans and priorities.

(3) A *Spacecraft Team*, whose principal functions were spacecraft performance evaluation and prediction, and spacecraft sequence design and validation. The Spacecraft Team was responsible for maintaining, and updating as required authoritative compilations of spacecraft commands, spacecraft blocks, spacecraft sequences, telemetry calibrations and conversion coefficients, and operating margins.

(4) A *DSN Project Engineering Team*, whose principal functions were DSN resources allocation, operations planning, and configuration control.

2. Mission operations teams. The mission operations execution functions were embodied in four teams and a group under the direction of an Assistant Chief of Mission Operations (ACMO):

(1) A *Command Team*, whose primary function was to operate the spacecraft in accordance with the mission operations plans specified by the ACMO. This team issued commands to the spacecraft and provided real-time evaluation and control of spacecraft performance.

(2) A *Science Data Team*, whose primary function was to determine and specify science data processing requirements, and to collect, catalogue, and disseminate science data products and maintain a science data library.

(3) A *Data Processing Team*, whose primary functions were planning, scheduling, coordinating, and trouble-shooting all data processing for the Project.

In addition, this team furnished and scheduled personnel to operate Project-supplied hardware and computer programs.

(4) A *DSN Mission Operations Team*, whose primary function was the operation of the DSN and coordination of near-Earth phase tracking (MSFN, AFETR) and NASCOM support in accordance with Project direction and DSN operating policies and procedures.

(5) A *Sequence Group*, whose primary function was to prepare mission and spacecraft sequences, CC&S program updates, and files of commands for subsequent transmission to the spacecraft in accordance with instructions from the ACMO.

C. Operational Readiness/Simulation

MOS readiness, simulation, and training discussed pertain solely to readiness for launch and the first trajectory correction maneuver (TCM). Training and demonstration of readiness for orbital operations had always been planned for and were conducted during the cruise phase following the first TCM and will be documented in Volume III of the Mariner Mars 1971 Project Final Report (to be published).

The MOS readiness preparations were divided into several phases. The first phase of these preparations consisted of nine lectures covering organizational, software, communications, tracking, telemetry, and command systems. Following these lectures a series of 17 interteam training exercises were conducted. Ninety-five hours of training went into this individual team training. The next series of training tests combined the entire MOS organization with the SFOF portion of the DSN. Two of these tests were conducted. The next series of tests combined the MOS with the entire TDS, especially to work out the handling of the telemetry and command systems. One of these tests was run with each prime deep space station and one with MSFN Ascension. Four more training tests were conducted between the MOS and TDS to work out all the procedures for launch, cruise, and first maneuver. All of the tests and training to this point were dedicated to working out a final set of procedures which could accomplish the mission design. The next set of five tests called *Operational Demonstration Tests* was especially designed to incorporate anomalies into the training and thereby stress the operations organization. These tests included long-duration training. A total of 148 h was devoted to these tests.

Several compatibility tests were also conducted between the MOS/TDS and the spacecraft. Fifty hours of testing with the spacecraft were especially dedicated to compatibility testing. In addition, the MOS/TDS supported the terminal countdown demonstration at AFETR with the PTM spacecraft and supported the J-FACT testing with both Mariners H and I. The compatibility testing, in addition to its value for compatibility, provided excellent training in the recognition of the real signature of the spacecraft.

All of the training exercises were severely hampered by two major problems. The first of these problems was the late availability of the 360/75 capabilities. The final launch version of the 360/75 was not available until April 8, 1971. The continual change of the 360/75 software configuration during the tests preceding this date required training, retraining, and test work for the whole period. The second difficulty was with the simulation system. A simulation math model of the MM'71 spacecraft was generated which was adequate for proper training, but unfortunately the 6050 computer and the 6050/1108 interface, within which the math model had to work, proved to be highly unreliable. In fact, it seemed to always fail in the most crucial part of the test.

As a final part of the launch readiness verification, two operational readiness tests were conducted before the launch of Mariner H. These tests demonstrated readiness of the MOS/TDS to support launch. A third operational readiness test was conducted prior to the Mariner I launch to demonstrate the launch readiness.

VI. Intersystem Compatibility

In addition to testing the individual MM'71 systems and demonstrating their design and capabilities separately, it was necessary to test and verify the interface between systems. This section describes the program to validate intersystem compatibility needed between (1) the spacecraft and the Tracking and Data System, (2) the Mission Operations System and the Tracking and Data System, (3) the spacecraft and launch vehicle, and (4) the spacecraft and Mission Operations System (Ref. 45).

A. Spacecraft/Tracking and Data System[1]

1. Approach. The approach to spacecraft/TDS compatibility testing on the Mariner Mars 1971 Project was to

[1]See Ref. 46 for details of TDS support for the Mariner Mars 1971 mission.

demonstrate first a compatible RF interface between the spacecraft and a deep space station (DSS) telecommunications system. Next, the compatibility of the spacecraft and the DSN Telemetry and Command Data Systems was demonstrated by the proper processing of data. The operational interface was then verified by conducting typical flight sequences with representative operational procedures. These tests constituted the design compatibility test (Fig. 14) and were conducted at JPL between the spacecraft, located in the Spacecraft Assembly Facility (SAF) or Environmental Test Laboratory (ETL), and Compatibility Test Area (CTA 21). Further compatibility testing verified the design compatibility established at JPL by RF verification tests conducted at Cape Kennedy between the spacecraft (in Building AO at the AFETR) and DSS 71.

2. ***Test program.*** Spacecraft/TDS compatibility testing was divided into three phases:

(1) *Phase I, Design Compatibility.* Phase I tests were conducted with the fully assembled PTM spacecraft. The purpose of these tests was to verify that the spacecraft design and the TDS were mutually compatible. Tests were conducted with the spacecraft located in the SAF or ETL with a communications link to CTA 21.

(2) *Phase II, Design Compatibility Verification.* Phase II tests involved each flight spacecraft in conjunction with CTA 21 and DSS 71. The purpose of the tests was to verify data from the Phase I design compatibility tests of the PTM, confirming that each flight spacecraft performance was acceptable as referenced to the PTM and specified requirements.

(3) *Phase III, Mutual Interference Compatibility.* For the first time, two spacecraft were to have been tracked simultaneously by one DSS. Phase III tests were conducted to determine if there was any interference when commanding either spacecraft or processing two telemetry data streams.

These compatibility tests included RF System, Telemetry System, Command System, and Ranging System tests.

The PTM/CTA 21 compatibility tests were conducted from July 1 to September 14, 1970. The MM71-1 flight spacecraft/CTA 21 compatibility tests were performed in two phases: (1) from December 14 through 17, 1970, and (2) from February 8 through 10, 1971. The MM71-2 flight spacecraft/CTA 21 compatibility tests were run from February 24 to 26, 1971. The MM71-1 flight spacecraft/DSS 71 compatibility tests were performed at AFETR on March 24 and 25, 1971; MM71-2 flight spacecraft/DSS 71 compatibility tests were performed from March 26 to 30, 1971.

Significant problems experienced during flight spacecraft/TDS compatibility tests and the solutions to the problems are listed in Table 9.

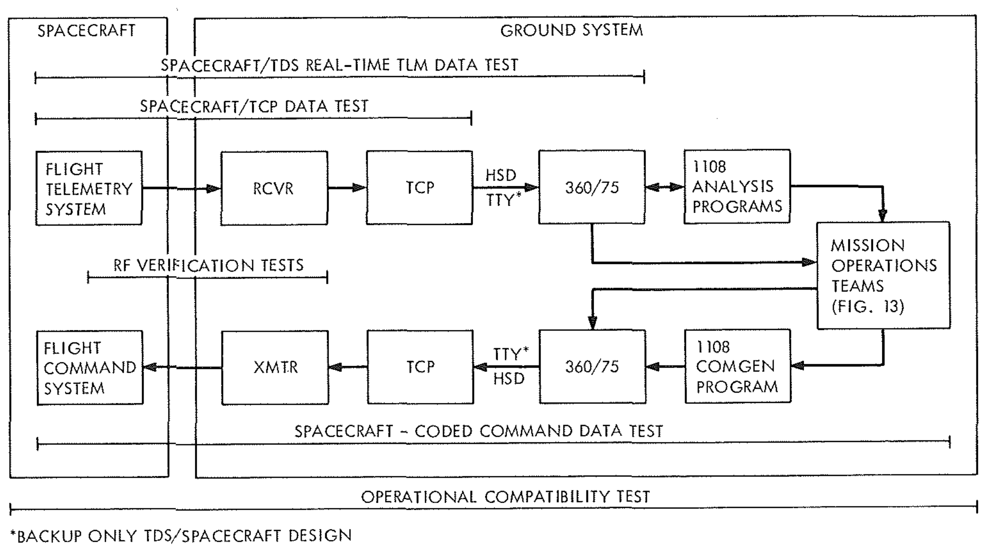

Fig. 14. Spacecraft/TDS design compatibility tests

Table 9. Flight spacecraft/TDS telecommunications compatibility — significant problems and solutions

Test	Problem	Solution
RF System	Spacecraft "best lock" frequency tests at CTA 21 and DSS 71 revealed that procedures were inadequate for determining uplink acquisition frequency.	Uplink "sweep" procedures were developed, documented, and utilized successfully in tests.
Telemetry System	During dual carrier–multiple subcarrier tests at CTA 21, using the MM71-2 spacecraft and a breadboard model at TDL, the signal-to-noise ratio of the 8⅓ bps was 3.36 dB lower than predicted.	This problem was due to an incorrect setting of modulation index on the engineering subcarrier. The dual carrier–multiple subcarrier operation was not required until planet operation.
Command System	During preparation for MM71-1 compatibility tests at CTA 21, it was noted that commands would always abort if command modulation and ranging modulation were on simultaneously.	Investigation revealed that confirmation detection was not compatible with having the command and ranging modulation on simultaneously. The operational program was modified to disable the confirmation detector.
	During MM71-2 spacecraft tests at CTA 21, two command aborts occurred. In each case, the abort reason was a "bit-by-bit" verification failure.	Investigation concluded that this problem was caused by a noisy channel in the $F_s/2F_s$ comparison circuitry. The command modulator assembly (CMA) tolerance on this measurement was modified from 1 to 5 μs.
	Several command "bit-verify" aborts occurred during MM71-1 and -2 spacecraft tests.	Intensive troubleshooting revealed that the problem was an inherent CMA design fault, which was isolated to noisy CMA input lines. Incorporation of noise suppression diodes and capacitors in each of the 48 lines rectified the problem. 10,000 commands were transmitted from the modified CMAs at CTA 21 during a "proof soak" test without any alarms or aborts, and 7000 commands were sent successfully from DSS 14.
	The spacecraft command system apparently dropped phase lock for 51 s during the MM71-2/DSS 71 compatibility test on March 29, 1971.	This was not a DSN problem. The spacecraft had experienced the same phenomenon using ground support equipment. It was established that the performance was normal for the conditions of test. No further action was necessary.
Ranging System	Compatibility tests at CTA 21 and DSS 71 with the MM71-2 spacecraft revealed that the ranging acquisition threshold was degraded by 1 to 1.5 dB from the predicted threshold. The MM71-1 spacecraft was at the predicted ranging threshold.	No solution was required since the tolerance for this measurement was ±2 dB.
Operational Program	During compatibility tests with the MM71-2 spacecraft at CTA 21, the telemetry and command processor was reloaded in the investigation of a telemetry problem. This operation caused the spacecraft command detector to drop lock. This was an operational constraint.	For all applicable operational procedures, "Command Modulation" was removed from the exciter prior to reloading the telemetry and command processor operational program.

B. Mission Operations System/Tracking and Data Acquisition System

The compatibility testing between the MOS organization and the TDS facilities was a necessary part of the MOS/SFOF and MOS/TDS testing and training. The basic purpose of these tests was for the operations personnel in the MOS to learn to use the TDS equipment and facilities. Consequently, all of these tests were in fact compatibility tests and no special, additional compatibility tests were required by the MOS and TDS.

C. Spacecraft/Launch Vehicle

Intersystem compatibility between the MM'71 spacecraft and the launch vehicle was demonstrated through development, design verification, and prelaunch operations tests. These tests were necessary to show that the established functional, electrical, mechanical and environmental interface design requirements and constraints had been met with the fabricated hardware.

1. Test program

a. Development tests. Chief in this category of development tests were tests performed at JPL to determine the dynamic compatibility between the Centaur adapter and the payload assembly (spacecraft adapter and spacecraft) for purposes of determining modal shapes and response characteristics of the spacecraft and adapter assembly. Modal surveys and vibration testing were conducted utilizing a full-scale spacecraft model, a prototype spacecraft adapter, and a prototype Centaur adapter which had been furnished by Lewis Research Center.

b. Design verification tests

(1) *Match-mate.* First under this category was the match-mate conducted at GD/CA, San Diego, with a full-scale spacecraft model. The purpose of this test was to verify that the clearances and fit between spacecraft, adapter, and nose fairing were satisfactory; to validate the encapsulation procedures; to validate ground handling equipment; and to gather data on the RF environment generated by the spacecraft and Centaur radios.

(2) *Pyrotechnic design verification.* Testing was conducted at GD/CA to verify that the launch vehicle firing unit was compatible with the requirements of the spacecraft release device squibs.

(3) *Flight adapter compatibility test.* This test was conducted at GD/CA to demonstrate the electrical and mechanical compatibility and interchangeability of the flight forward payload adapters with the launch vehicles.

(4) *Telemetry system compatibility tests.* Testing was conducted at GD/CA to verify that the spacecraft flight telemetry system (FTS) was compatible with the Centaur telemetry system.

(5) *RF compatibility tests.* Testing was conducted at GD/CA to determine the effects produced by the spacecraft radio on Centaur systems.

c. Prelaunch operations tests

(1) *Launch Complex Checkout*

(a) *Electrical checkout.* Following installation of launch complex equipment (LCE) in the blockhouse, spacecraft functions were verified through all interface cabling to the field joint.

(b) *RF system checkout and calibration.* End-to-end checkouts consisting of calibration of the RF systems were performed to verify both air and landline links.

(c) *Air conditioning and cleanliness.* Prior to each spacecraft/launch vehicle mate, the conditioned air was sampled for particulate cleanliness.

(2) *Terminal countdown demonstration (TCD).* A spacecraft participated in a TCD with each launch vehicle to verify systems operational compatibility under cryogenic tanking and actual launch environmental conditions. The TCD test objectives were to:

(a) Demonstrate the capability of the ground and airborne propellant systems and associated support systems to support a launch.

(b) Demonstrate operational autopilot and guidance systems during simulated flight under cryogenic conditions.

(c) Demonstrate the performance of the instrumentation systems with the Service Tower removed.

(d) Demonstrate the ability to de-tank propellants.

(e) Verify that no RFI is caused by the interaction of the launch vehicle, the spacecraft, and the range radars.

(3) *Joint flight acceptance composite test (J-FACT).* Each flight spacecraft participated in the J-FACT with its launch vehicle to verify systems operational compatibility under simulated flight conditions from initiation of countdown through completion of Centaur retromaneuver. Further, the test demonstrated on an integrated basis the operation of all airborne electrical systems during a simulated flight, with launch vehicle guidance in flight mode using telemetry and the gantry test rack for event monitoring.

(4) *Composite readiness test (CRT).* Each flight spacecraft participated with its launch vehicle in system-level simulated flight to demonstrate launch readiness of all electrical and RF systems with a minimum of system violation, utilizing launch control GSE, landline instrumentation and telemetry for event monitoring. This test was the final composite systems test prior to launch.

(5) *Electromagnetic interference/radio frequency interference (EMI/RFI) test.* Each flight spacecraft participated with its launch vehicle in system-level tests to ensure interference-free operations and to develop the characteristic RF signature of that particular spacecraft/launch vehicle combination. Each spacecraft was counted

down to a predetermined state and monitored for EMI/RFI while launch vehicle, launch complex, and range systems were activated.

2. *Schedules.* Accomplishment of schedules for testing intersystem compatibility between spacecraft and launch vehicle as described above is shown in Table 10.

D. Spacecraft/Mission Operations System

The MM'71 spacecraft/MOS compatibility plan was documented in Ref. 47. The spacecraft and DSN had performed design compatibility tests, and the MOS approach was to validate the MOS software and computer interfaces that were new for this mission.

It was recognized early in the program that it would be difficult to arrange spacecraft time for MOS compatibility tests, because of the tight spacecraft schedule; however, it was also recognized that it was necessary that such compatibility tests be performed. Other difficulties associated with developing the test schedule were: (1) the 360/75 implementation plan was behind schedule, and (2) it was the MOS's desire to use the closest 360/75 flight version that was possible. These two facts presented a profound conflict.

The compatibility test program was designed so that there would be a minimal duplication of tests performed on Flight 1 (MM71-1) and Flight 2 (MM71-2) spacecraft (see Table 11). It should be noted that the only test duplication was the composite test, which exercised the spacecraft in as many RF states as possible that could be anticipated during flight operations.

The spacecraft was in SAF, and CTA 21 was configured as nearly as possible to an operational tracking station. Only the telecommunication signal levels on the uplink and the downlink were not realistic; this was because the CTA 21 antenna gain and noise temperature were different from the operational tracking stations.

The composite test 1 sequence stepped through the various telecommunication data modes (RTS-1, RTS-2, playback, and engineering) and the data rates twice, requiring nearly six hours. It started out in the RTS-1 mode, with 50-bps science and 33⅓-bps engineering telemetry. The RTS-2 mode, both 16.2 and ending at 1.0125 kbps, was exercised. Interspersed in the sequence were engineering channel data rate switches from 33⅓ bps to 8⅓ bps, and back again, and also CC&S memory readouts. This portion of the test concluded with a second sequence of RTS-1, RTS-2, playback, and engineering data modes.

Table 10. Spacecraft/launch vehicle intersystem compatibility test accomplishment

Test	Date accomplished
Match-mate	5/14/70
Pyro design verification	10/6/70
Telemetry system compatibility	10/6/70
RF compatibility	10/6/70
Flight adapter compatibility	10/7/70
Launch complex 36B checkout	1/25/71
Launch complex 36A checkout	2/25/71
PTM/AC-23 TCD	3/11/71
PTM/AC-23 J-FACT	3/18/71
PTM/AC-24 TCD	4/8/71
PTM/AC-24 J-FACT	4/13/71
MM71-1/AC-24 J-FACT	5/4/71
MM71-1/AC-24 CRT	5/5/71
MM71-2/AC-23 J-FACT	5/17 and 23/71
MM71-2/AC-23 CRT	5/24/71
MM71-2/AC-23 TCD	5/25/71

Table 11. Spacecraft/MOS compatibility tests

Priority	MM71-1	MM71-2
1	Composite test 1 (6–10 hours)	Composite test 2 (6–10 hours)
2	Science sequence by CC&S (4–5 hours)	Science sequence by ground commands (4–5 hours)
3	Tandem maneuver sequence (4–5 hours)	Insertion maneuver sequence (4–5 hours)

Due to continuing 360/75 development problems, the CC&S readout could not be performed at JPL, and this was recognized as a lien on the spacecraft during the JPL preshipping review. However, a CC&S readout was accomplished by using the PTM spacecraft at hanger AO and by utilizing DSS 71 at Cape Kennedy, with support by the SFOF. The readout was performed successfully and was judged a demonstration of compatibility of all elements required for flight support.

As a result of the compatibility test program, the systems were declared to be compatible prior to launch of the spacecraft.

VII. Science Planning and Implementation

The science instruments and experiments were selected as a result of an *Announcement of Flight Opportunity* for MM'71. Three of the instruments were eliminated when the scope of the mission had to be narrowed due to budgetary considerations (April 1969). The eliminated instruments were the following:

(1) Charged-particle telescope (principal investigator: J. Simpson, University of Chicago).

(2) Multiple-frequency receiver (principal investigator: Von R. Eshelman, Stanford University).

(3) X-ray particle detector (principal investigator: J. Van Allen, University of Iowa).

The final set of science experiments and the investigators for these experiments are shown in Table 12. A brief description of the instruments is given in *Section IV-B-15*.

References 48 and 49 established MM'71 Project science requirements. Reference 48 lists basic science requirements by defining experiment objectives and listing requirements that impact spacecraft design and significantly guide the choice of mission and Mission Operations System designs. Reference 49 lists the specific requirements necessary for payload integration, spacecraft system testing, calibration, and detailed design of mission operations. The experimenters and project science prepared a definitive set of papers (Ref. 50) that described the experiments, scientific objectives, and capabilities of the MM'71 mission.

The principal science philosophy was one that not only considered the instruments an integral part of the spacecraft design from the beginning but also considered the investigators an integral part of the project team. Each experiment had a principal investigator; if there were more than one principal investigator, a team leader was named. All of the instrument teams depended on JPL experiment representatives who served as the day-to-day interface between the experiment and the project. Some of the representatives were co-experimenters while others later became co-experimenters.

Due to the size of the television team, it was possible and necessary to organize the team in terms of discipline groups and task groups (Table 13) so that they might effectively contribute to their project responsibilities. All of the teams participated in the instrument design, testing, and calibration. They also helped in the preparation for flight data processing.

Table 12. Experimenter teams

Experiment	Investigator	Affiliation
Television	H. Masursky[a]	U.S. Geological Survey
	D. Arthur	U.S. Geological Survey
	R. Batson	U.S. Geological Survey
	W. Borgeson	U.S. Geological Survey
	M. Carr	U.S. Geological Survey
	J. McCauley	U.S. Geological Survey
	D. Milton	U.S. Geological Survey
	L. Soderblom	U.S. Geological Survey
	R. Wildey	U.S. Geological Survey
	D. Wilhelms	U.S. Geological Survey
	J. Lederberg[b]	Stanford University
	E. Levinthal	Stanford University
	J. Pollack	Ames Research Center
	C. Sagan	Cornell University
	J. Veverka	Cornell University
	G. de Vaucouleurs[b]	Texas University
	A. Young	Jet Propulsion Laboratory
	G. Briggs[b]	Jet Propulsion Laboratory
	E. Shipley	Bell Telephone Laboratories
	B. Smith[b]	New Mexico State University
	J. Cutts	California Institute of Technology
	M. Davies	Rand Corporation
	W. Hartmann	Planetary Science Institute
	R. Leighton	California Institute of Technology
	C. Leovy	University of Washington
	B. Murray	California Institute of Technology
	R. Sharp	California Institute of Technology
Infrared interferometer spectrometer	R. Hanel[b]	Goddard Space Flight Center
	B. Conrath	Goddard Space Flight Center
	W. Hovis	Goddard Space Flight Center
	V. Kunde	Goddard Space Flight Center

[a]Team leader, principal investigator.

[b]Principal investigator.

Table 12 (contd)

Experiment	Investigator	Affiliation
Infrared interferometer spectrometer (contd)	G. Levin	Biospherics, Inc.
	P. Lowman	Goddard Space Flight Center
	C. Prabhakara	Goddard Space Flight Center
	B. Schlachman	Goddard Space Flight Center
	J. Pearl	Goddard Space Flight Center
	T. Burke	Jet Propulsion Laboratory
Infrared radiometer	G. Neugebauer[b]	California Institute of Technology
	S. Chase	Santa Barbara Research Center
	H. Kieffer	University of California at Los Angeles
	E. Miner	Jet Propulsion Laboratory
	G. Munch	California Institute of Technology
Ultraviolet spectrometer	C. Barth[b]	University of Colorado
	C. Hord	University of Colorado
	A. Lane	Jet Propulsion Laboratory
	I. Stewart	University of Colorado
Celestial mechanics	J. Lorell[a]	Jet Propulsion Laboratory
	J. Anderson	Jet Propulsion Laboratory
	W. Martin	Jet Propulsion Laboratory
	W. Sjogren	Jet Propulsion Laboratory
	I. Shapiro[b]	Massachusetts Institute of Technology
	R. Reasenberg	Massachusetts Institute of Technology
S-band occultation	A. Kliore[b]	Jet Propulsion Laboratory
	D. Cain	Jet Propulsion Laboratory
	G. Fjeldbo	Jet Propulsion Laboratory
	S. Rasool	NASA Headquarters
	B. Seidel	Jet Propulsion Laboratory

The project science support was divided into two groups. A Science Evaluation Team (SET) was established to perform the long-range data analysis which would be incorporated into science recommendations for mission operations. The SET was chaired by an assistant project scientist and consisted of all the experimenters. For the real-time mission operations interface, the Science Recommendation Team (SRT) was created under another assistant project scientist.

The organization of the teams for the adaptive mode mission operations was paramount in the prelaunch period. The SRT, which served as the interface between the scientists and mission operations, was organized and had a representative from each experiment team as a member of the SRT. Initially, the experiment representatives served this SRT function. The scientists participated on the mission design team whose function was to definitize and optimize orbital parameters and integrate science inputs into sequences.

The two-launch mission design resulted in one spacecraft which was to be primarily utilized for mapping, geodesy, and polar coverage and the other for variable surface features, atmospheric studies, and global coverage. The missions were a 12-h orbit period at an 80-deg inclination and a 20-h orbit period at a 50-deg inclination, respectively. After the launch failure of Mariner 8, a new plan was developed by the project and experimenters. The compromise orbit, described in *Section X*, was a 12-h orbit period with a 65-deg inclination.

This was a building, organizing, and planning period for experiments and experimenters. Primary functions were preparation of instruments, hardware, and software, and preparing the experimenters for adaptive mode orbital operations.

VIII. AFETR Prelaunch and Launch Operations

A. Plan

The MM'71 basic Launch Operations Plan provided for the launch of two Mariner Mars 1971 spacecraft (MM71-1 and MM71-2) and Atlas/Centaur vehicles (AC-23 and -24) from Launch Pads 36A and B at the Cape Kennedy Air Force Station, Air Force Eastern Test Range (AFETR). The first launch was scheduled for May 7, 1971, to be followed by the second launch no later than 10 days. Arrival dates at Mars were November 14 and 24, 1971. The nominal launch period available was May 7, 1971 through June 4, 1971. Launch windows varied from 30 to 90 min. The dominant constraint to the launch window and period was the launch vehicle performance.

The proof test model (PTM) spacecraft was shipped to the AFETR for utilization as a pathfinder and a source of spares.

Table 13. MM'71 television team organization chart[a]

Discipline Groups					
Atmospheric phenomena	Geodesy/ cartography	Geology	Satellite astronomy	Variable surface features	Physics of polar phenomena
C. Leovy[b]	G. de Vaucouleurs[b]	J. McCauley[b]	J. Pollack[b]	C. Sagan[b]	B. Murray[b]
G. Briggs	D. Arthur	M. Carr	D. Milton	J. Cutts	R. Leighton
E. Shipley	R. Batson	W. Hartmann	M. Davies	J. Lederberg	J. Lederberg
B. Smith	W. Borgeson	R. Sharp	W. Hartmann	E. Levinthal	C. Leovy
R. Wildey	M. Davies	L. Soderblom	C. Sagan	J. Veverka	R. Sharp
J. Pollack	R. Leighton	D. Wilhelms	J. Veverka	R. Wildey	L. Soderblom
A. Young	A. Young	J. Cutts	B. Smith	A. Young	D. Milton
	R. Wildey	D. Milton	A. Young	G. Briggs	
		B. Murray		M. Carr	
		C. Sagan		G. de Vaucouleurs	
				J. Pollack	
				B. Smith	

Task Groups			
Hardware	Mission Analysis	Data Processing and Process Control	Mission Operations
B. Murray[b]	G. Briggs[b]	E. Levinthal[b]	B. Smith[b]
W. Borgeson	W. Borgeson	D. Arthur	R. Batson
J. Cutts	M. Davies	R. Batson	G. Briggs
R. Leighton	D. Milton	G. Briggs	M. Carr
B. Smith	J. Pollack	J. Cutts	W. Hartmann
R. Wildey	C. Sagan	M. Davies	C. Leovy
A. Young	B. Smith	E. Shipley	J. McCauley
		B. Smith	B. Murray
		L. Soderblom	C. Sagan
		J. Veverka	
		R. Wildey	
		A. Young	

[a]H. Masursky (Team Leader) and B. Smith (Deputy) are ex-officio members of all groups.
[b]Group leaders.

B. Organization

The MM'71 organization at AFETR is described in detail in the Launch Operations Plan (Ref. 51). In general, a satellite Project Office was set up at AFETR, with the Assistant Project Manager (Near Earth) acting as Launch Manager and Mission Director for the launch. The Spacecraft System Manager and his organization moved to AFETR with the spacecraft for the entire launch operations period. Elements of the Tracking and Data System (TDS), i.e., Near-Earth TDS, were in residence at the AFETR; these consisted of AFETR, MSFN, and DSN stations. The Mission Test Computer and Video System elements of the Mission Operations System (MOS) were in residence to support spacecraft testing and launch preparations.

The Launch Vehicle Manager was in residence at AFETR during launch preparations and launch. However, the actual launch vehicle preparations were delegated to the Kennedy Space Center Unmanned Launch Operations (KSC/ULO), under whose direction General Dynamics, Convair Aerospace (GD/CA) performed the actual operations on the Atlas and Centaur vehicles.

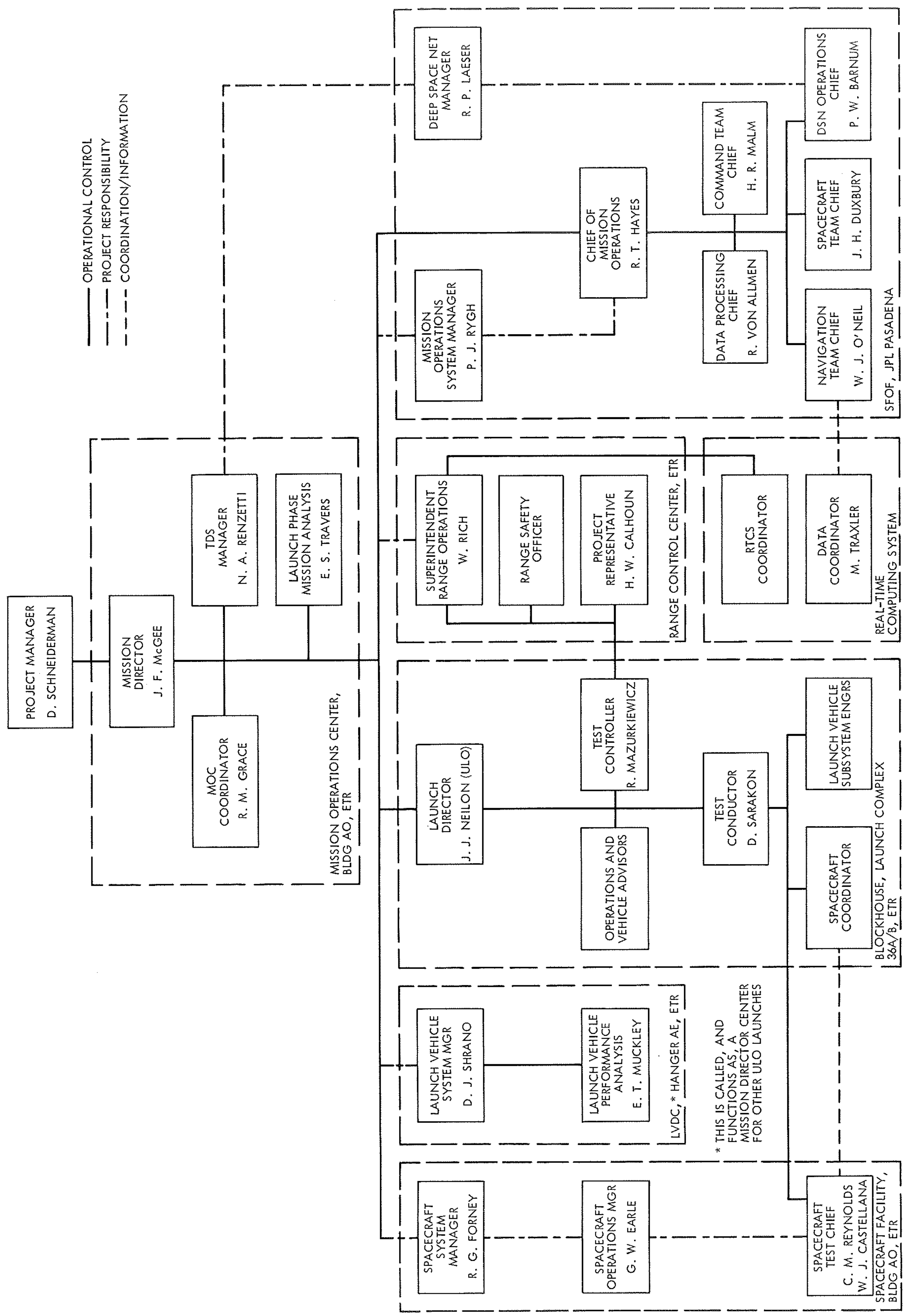

Fig. 15. Mariner Mars 1971 countdown operations organization

Project meetings at AFETR were held weekly on Wednesday, and results were transmitted immediately after the meetings by telecon to the JPL Pasadena MM'71 Project Meetings. Minutes were also distributed.

Other organizational elements that contributed greatly to the launch operations were the Launch Operations Working Group (LOWG) and the Near-Earth Tracking and Data System (NETDS) Working Group.

During the prelaunch phase of AFETR operations, the LOWG acted as the prime mechanism for coordinating space vehicle launch preparations, including launch complex preparations. The LOWG was chaired by KSC/ULO acting in behalf of LeRC and met weekly at AFETR. Membership in LOWG consisted of representatives of LeRC, ULO, JPL, and associated contractors, major subcontractors, and AFETR program management personnel. Actions of the LOWG that affected the Mariner Mars 1971 Project were reported to the Launch Manager and were subject to his review and approval. Group responsibilities included:

(1) Review of the day-to-day schedule status.

(2) Review of day-to-day problems.

(3) Launch operation test records and schedules.

The NETDS Working Group, chaired by the JPL/ETR Assistant Field Station Manager (NETDS Manager), was responsible for the planning of operations required to provide near-Earth tracking and data support of the Mariner Mars 1971 Project. The support requirements were defined in Refs. 52 and 53. The NETDS Working Group coordinated the efforts of the AFETR, Manned Space Flight Network, Goddard Space Flight Center, and those elements of the DSN needed for near-Earth support. The group carried out all necessary interfaces with the DSN Project Engineer with regard to the flow of voice and data between the near-Earth and the deep space facilities.

The MM'71 Launch Readiness Review was originally scheduled to cover all systems for both launches. However, following the failure of the first launch (MM71-1), a second Launch Readiness Review was conducted prior to the second launch.

Two days prior to each launch, a Launch Readiness Verification Meeting was held at the AFETR, and a conference call was made to JPL Pasadena.

The countdown operations organization is shown in Fig. 15. Project operations were controlled from Building AO at the AFETR.

C. Prelaunch Preparations and PTM/LV Tests

The MM'71 Launch Operations at Cape Kennedy, AFETR, actually commenced with the delivery of the Atlas for the Atlas/Centaur 23 (AC-23) to AFETR on December 5, 1970; and the Atlas for AC-24 was delivered on February 10, 1971 (see Fig. 16). The launch vehicles were erected on Launch Pads 36B and A, respectively, and were prepared for launch. Spacecraft activities commenced with the delivery of the propulsion subsystem "pathfinder" on January 15, 1971. The first MM'71 spacecraft, the PTM (MM71-3), was delivered to AFETR on February 17, 1971, followed by the two flight spacecraft, the MM71-1 on February 28, 1971 and the MM71-2 on March 14, 1971.

The basic philosophy followed during launch operations was to check out each system (Launch Vehicle, Spacecraft, Near-Earth Tracking and Data System, Mission Operations System) independently and then conduct intersystem compatibility tests. The PTM spacecraft (MM71-3) was utilized as a "pathfinder" for the flight spacecraft to proof equipment, facilities, and test procedures. The PTM was also mated to each of the launch vehicles on Launch Pads 36A and B as early as possible (beginning on March 9, 1971) to participate in interface tests, such as Terminal Countdown Demonstration (TCD), Joint Flight Acceptance Test (J-FACT), Radio Frequency Interference Test (RFI), Electromagnetic Interference Test (EMI), and Composite Readiness Test (CRT).

In addition, these interface tests provided the environment to test the spacecraft compatibility with the actual launch complex equipment and provided "practice" countdowns for personnel. The original intent was to mate the PTM and AC-23 on Pad 36B and then move to AC-24 on Pad 36A and complete all tests without returning to the Explosive Safe Area (ESA). However, due to the UVS problem, the PTM was demated from AC-23, Pad 36B, on March 24, 1971 and then remated to AC-24, Pad 36A, on April 5, 1971. The complete set of interface tests was successfully completed on April 14 without any major problems and provided high confidence that the two flight spacecraft would encounter no unforeseen difficulties when mated to their respective launch vehicles.

The two flight spacecraft (MM71-1 and MM71-2) were system-tested in Building AO and prepared for launch in the ESA. During the period March 24 through 30, compatibility testing between both spacecraft and DSS 71 was conducted. These compatibility tests uncovered a problem in the ground command system which was subsequently rectified; the ground command system was retested prior to launch. In addition, DSS 71 participated in testing at the ESA and on the launch pads, and during prelaunch and launch countdowns.

The NETDS, in preparation for launch, conducted subsystem testing and then participated in the Operational Readiness Test (ORT) with the MOS and TDS on April 29, 1971 in the final demonstration of readiness for launch.

The MOS participated in interface tests through DSS 71 to the spacecraft during this preparation period.

By April 23, 1971, all system and intersystem tests had been completed at AFETR except the ORT, and only the operations necessary to prepare for launch were required.

On April 23, the MM'71 Cape Launch Readiness Review was conducted at AFETR by a review board chaired by R. J. Parks, Assistant Laboratory Director for Flight Projects, with board members from both JPL and NASA.

This same board had participated in the "Spacecraft Preshipment Review" at JPL. A major item of concern was the IRIS failure that had occurred the previous day. Major action items were to (1) establish a contingency plan to use the spacecraft propulsion subsystem to extend the launch period, (2) transmit listings of spacecraft signatures to the MOS, and (3) establish a contingency plan to consider the actions to be taken in the event the planetary quarantine restraints were not met. The recommendation of the Board was to proceed with launch.

At the Cape Launch Readiness Review, emphasis was placed on the launch vehicle and spacecraft systems, near Earth TDS readiness, and intersystem interface compatibility. Subsequently, an MOS and DSN Launch Readiness Review was conducted on April 27 at JPL, Pasadena, by R. J. Park's review board.

D. Testing (Pathfinder, System Tests, Launch Preparations)

The PTM spacecraft was utilized as a pathfinder system to precede the flight spacecraft in order to evaluate procedures, train personnel, and demonstrate spacecraft compatibility with the launch complex environment and interfacing system hardware. Following initial preparation at Building AO, it was transported to the Explosive Safe Facility (ESF), where it was pressurized and encapsulated. The spacecraft then proceeded through a series of combined spacecraft and launch vehicle integrated systems tests at each launch pad. This use of the PTM for compatibility testing enabled compression of flight spacecraft test time and proved to be a valuable scheme as evidenced by the absence of any spacecraft, or spacecraft-induced, problems during combined operations with the flight spacecraft.

Flight spacecraft test operations at AFETR were designed to reduce activities to those required to complete launch preparations and demonstrate launch environment compatibility including the DSIF. Functional system tests and calibration activities were normally restricted to those which could be performed without removal of flight hardware from the spacecraft or the demating of flight connectors. Equipment shipped to AFETR on the spacecraft was removed only to complete hazardous flight preparations (e.g., propulsion subsystem fueling) and, when necessary, to replace hardware with flight-qualified provisioned spares.

Although the flight spacecraft was not intended to be demated from the launch vehicle after the first mate, spacecraft MM71-2 was demated on May 19, 1971 to enable investigation of a Centaur propellant utilization problem.

While the majority of the problems or failures that occurred during AFETR test operations was quickly understood and corrected, three problems proved to be very troublesome, and a great deal of time was consumed in troubleshooting, repair and rework, reinstallation, and retest:

(1) The engineering pedestal data on Channel F of UVS SN 003 (MM71-2) was 20 data number (DN) higher than normal. The unit performed normally during bench tests after removal but would not operate properly after reinstallation. After many hours of troubleshooting and analysis and a decision to use the PTM unit on MM71-2, the trouble was found to be a cracked epoxy lead-in resistor termination. The unit was repaired, replaced on MM71-2, and operated satisfactorily.

(2) The IRIS failed to activate when spacecraft power was applied on April 22, 1971 during a precountdown test of MM71-1. The cause was determined to be a design fault, wherein reverse bias circuitry resulted in capacitor failure. The instruments were removed from all three spacecraft and returned to

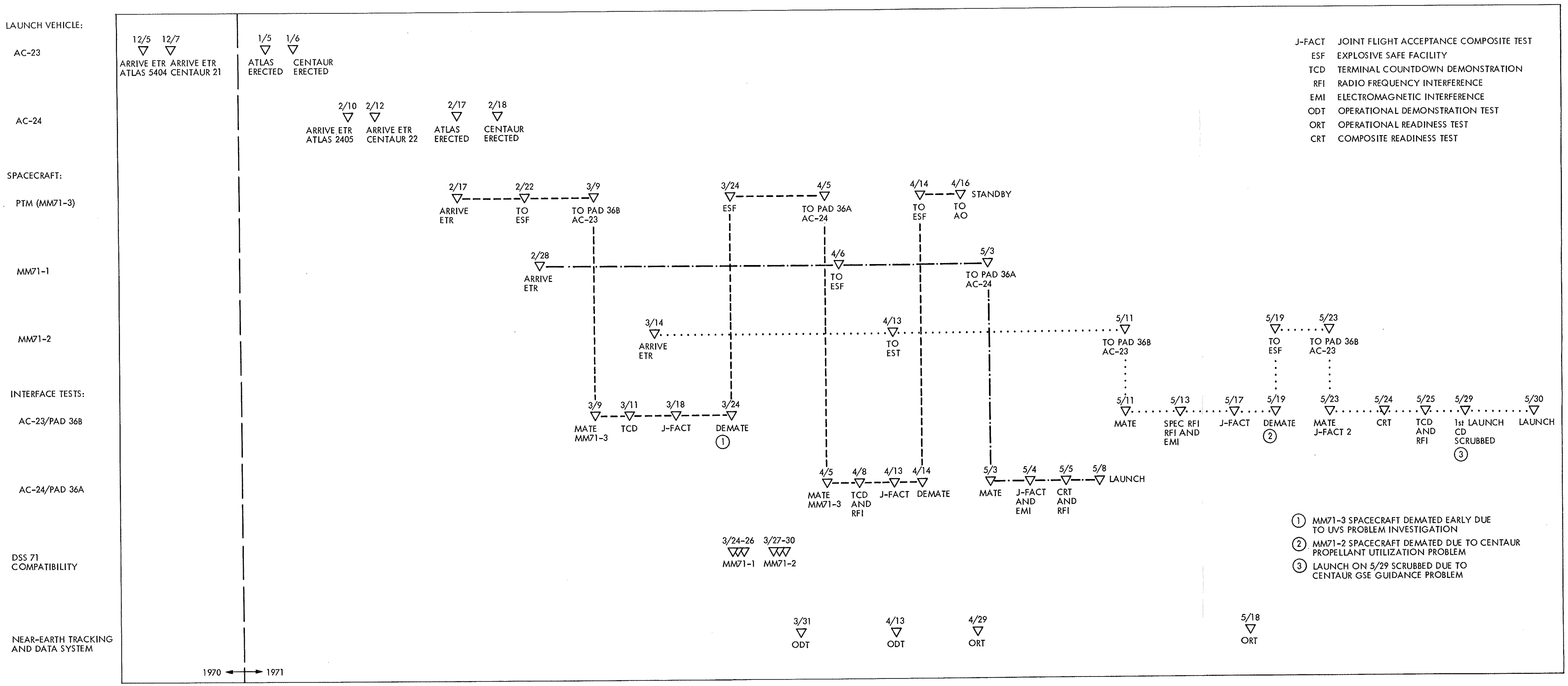

Fig. 16. ETR sequence of events

the manufacturer for rework and test. Several days of test time were lost, resulting in schedule compromises and round-the-clock operations in order to meet the desired launch date.

(3) The third problem was the in-flight failure of AC-24 and the loss of MM71-1, which resulted in a delay of the second launch to enable investigation of the cause.

The total test time, with spacecraft power on, from start of system testing at Pasadena through AFETR operations was 1111.9 h for the PTM, 812.9 h for MM71-1, and 896.2 h for MM71-2.

Complete details of test activities at AFETR may be found in the Mariner Mars 1971 Spacecraft Test and Operations Report (Ref. 54).

E. Mariner H Launch Preparations and Launch

On April 22, the IRIS on MM71-1 failed. The IRIS capacitor problem affected all units, and launch schedules were now dictated by the IRIS repair and retest schedules.

On May 3, a repaired and tested IRIS was received at AFETR and installed on MM71-1. The launch preparation schedule for MM71-1 and AC-24 was conducted as follows:

May 3 – Mate MM71-1 to AC-24 on Pad 36A

May 4 – J-FACT and EMI test

May 5 – Composite readiness test (CRT) and RFI test

May 8 – Launch (May 8, EDT; May 9 GMT)

The prelaunch tests were completed satisfactorily and the countdown for launch of Mariner H (MM71-1) was normal without any significant anomalies.

The AC-24 (Atlas SLV-3C S/N 5405 and Centaur S/N 22D) and Mariner H (MM71-1) were launched from AFETR Launch Complex 36A, May 9, 1971, at 01:11:02.294 GMT on a flight azimuth of 101:95 deg at the opening of the launch window. The weather was as follows:

Temperature: 21.1°C (70°F)

Relative humidity: 89%

Visibility: 14.5 km (9 mi)

Dew point: 18.3°C (65°F)

Surface wind: 1.03 m/s (2 knots) from 140 deg

Clouds: scattered at 4267 m (14,000 ft); altocumulus at 3048 m (10,000 ft)

Sea level atmospheric pressure: 762.254 mm (30.010 in.) of mercury

The Mariner H flight was nominal until shortly after Centaur main engine start. At this time, Centaur vehicle pitch stabilization was lost, followed by complete loss of pitch control, which subsequently resulted in vehicle tumbling and engine shutdown. The vehicle and the MM71-1 spacecraft impacted in the Atlantic Ocean approximately 600 s after launch, 1665 km (900 nmi) downrange, approximately 560 km (350 mi) northwest of Puerto Rico, resulting in the loss of Mariner H.[2]

F. Recovery From Mariner H Failure

1. Failure

a. Flight events. At about 270 s into the Mariner H flight, it became evident from radar tracking data that the space vehicle was no longer following the predicted altitude track. Post-flight data analysis revealed that at Centaur main engine start (MES), when the flight control system starts to actively control the vehicle, the pitch rate channel gain was 20 to 40% of nominal. Shortly thereafter, at MES + 4 s, when guidance steering was initiated and increased the control requirements of the flight control system, the vehicle pitch movements started to become divergent. At MES + 28 s, the engines went to fixed-pitch positions and the vehicle started to tumble. The roll and yaw channels were normal during this period. The payload data channel, Channel 13, was lost at MES + 86 s, reflecting separation of the payload due to the tumbling motion of the vehicle, and 2 s later an initial Centaur engine shutdown occurred due to propellant starvation. With the exception of the flight control failure, all other Centaur systems operated as expected under the existing conditions. A failure investigation conducted after the flight revealed that an integrated circuit failure from the pitch rate gyro preamplifier of the rate gyro package was the most consistent explanation for the flight failure and the data indications. As a result of this failure, corrective action was implemented to partially redesign the rate gyro package and to modify the test methods and procedures for flight control system. As an aid to understanding the system within which the failure occurred, a brief description follows.

[2]Mariner H has also been referred to as Mariner 8 even though the Centaur failed to inject it into the trajectory to Mars.

b. Centaur Flight Control System. The flight control system maintained vehicle stability by conditioning the appropriate rate/position data to provide command signals for positioning the main thrust chambers. The system also provided a pre-set switching sequence for maintaining the desired order of flight events. A simplified block diagram is shown in Fig. 17, separating the system into three major functional components, the rate gyro package, the servo amplifier package, and the programmer. The steering reference is supplied by the Centaur Guidance System.

The rate gyro package contained three rate gyros (pitch, yaw, and roll), which provided the signals for damping necessary to maintain control system stability. The outputs of the pitch and yaw rate gyros were amplified and summed with the appropriate guidance-generated pitch and yaw steering signals, producing proportional rate/position error signals for use by the servo amplifier package. The steering signals were electrically limited to prevent excessive turning rates, and might also be inhibited during select flight periods.

The servo amplifier canister conditioned the pitch/yaw/roll error signals via appropriate filter/integrator circuitry to provide command signals to four actuators. The command signals were summed with the individual actuator feedback transducer outputs in the servo amplifiers. The command to the actuator was the difference between the actual and desired engine position. The integrators provided engine trimming signals to ensure that the steady-state error signals were zero, negating the effect of system offset, engine thrust unbalances, rigging errors, and center-of-gravity offsets.

The programmer consisted of an electromechanical timer for sequential control and an auxiliary electronics unit which contained the required relays and logic for distributing the programmer commands. External discretes were received from the guidance and Atlas flight control systems. The switching outputs controlled various vehicle and system internal commands. The timer was driven by a 3-phase 400-Hz synchronous motor and was started at Atlas sustainer engine cutoff (SECO) by a discrete from the Atlas programmer.

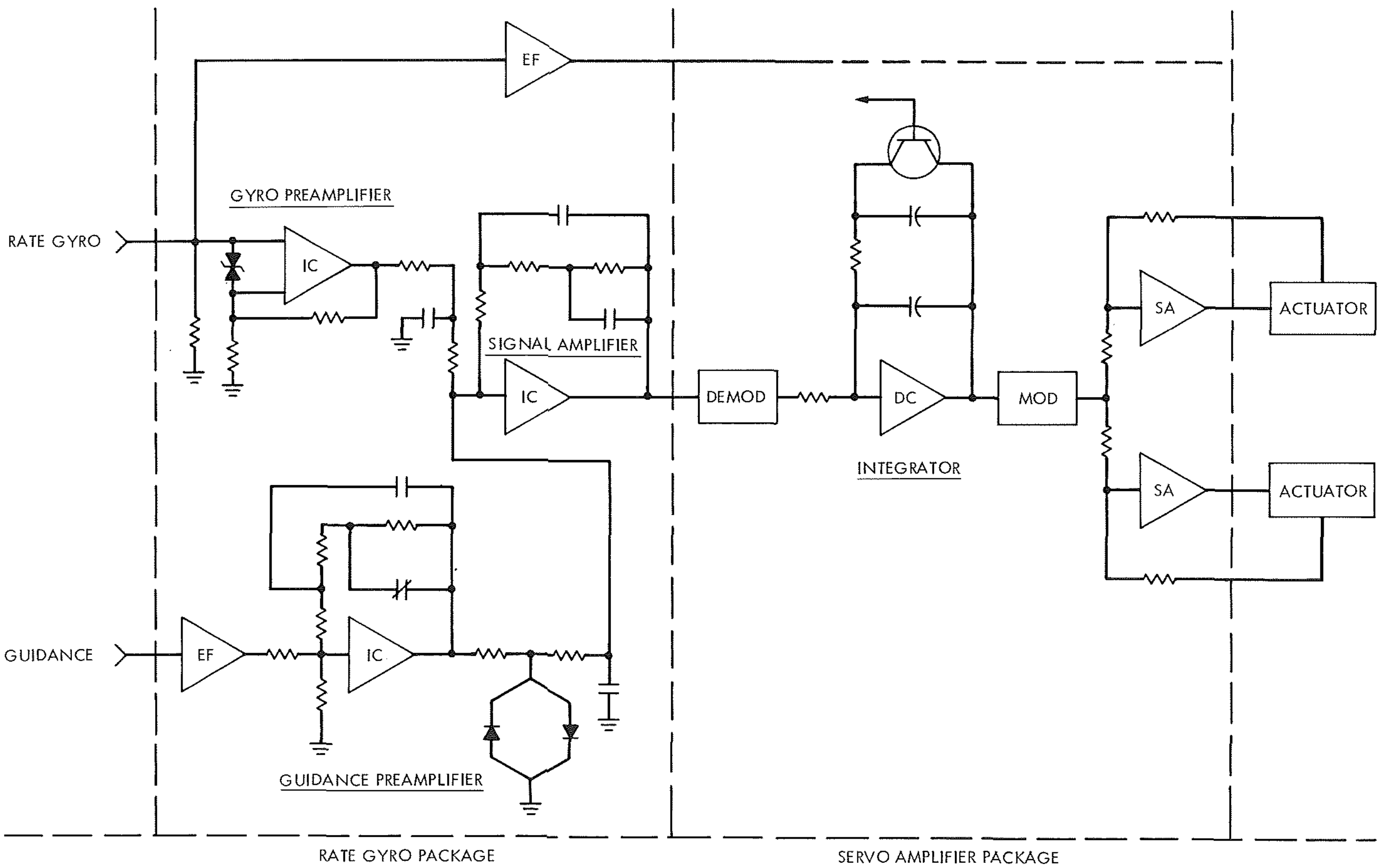

Fig. 17. Signal flow diagram

2. *Investigation*

a. Delay of second launch. Prior to the AC-24 flight failure, a nominal 10-day separation between the two launches had been planned. After the failure, a Project decision was made to delay the second launch as long as feasible to allow time for problem solution and corrective action, and design of a single spacecraft mission (see *Section X*). As a result, the second launch occurred 22 days later on May 30, 1971.

b. Investigation plans. A first-look failure review was conducted during the night of the launch. Participants were LeRC, JPL, KSC, GD/CA, Minneapolis-Honeywell, and Pratt & Whitney. The conclusion reached at this review was that failure had taken place in the pitch channel of the Centaur flight control system. The next day, a $T+12$-h status review of all systems was held by JPL. At the conclusion of this status review, LeRC announced that the launch vehicle project office and technical personnel were returning to the Lewis Research Center, to bring into play more of their technical personnel there and the facilities available at LeRC. The GD/CA personnel returned to San Diego to continue the investigation, specifically to set up a computer simulation of the Centaur flight control system. At the same time, JPL continued data analysis at AFETR and in Pasadena. The JPL investigation was twofold: (1) an analysis of those areas wherein the spacecraft potentially was the generator of the causes of the ultimate failure (in particular, mass shifts, propellant slosh, and radio signal radiation), and (2) furnishing technical assistance through our personnel who had gone to LeRC and later to GD/CA in the detailed investigation of the flight control system.

Detailed investigation of flight data and the result of the analysis revolved around the following significant flight data:

(1) Pitch channel instability.

(2) Rate only mode was stable. Rate gain was initially only 20 to 40% of the nominal.

(3) Instability began when position data was admitted.

(4) Frequency of oscillation was approximately 20% low.

(5) Engine did not go to stops, but finally remained at 1.2 deg.

(6) Peculiar signature on engine feedback traces at peak levels.

The LeRC–GD/CA general analytical approach was as follows:

(1) Review flight data for significant parameters, such as frequencies, amplitudes, and nonlinearities.

(2) Compare with previously predicted values.

(3) Perform root-loci analysis to determine possible causes.

(4) Use analog simulation to verify root-loci and further pinpoint possible causes.

(5) Install the hardware in computer simulation loop to confirm cause of failure.

Analysis of simulation reduced the potential failure cause candidates to a single one: the pitch rate gyro preamplifier. The schematic is shown in Fig. 18. Further analytical effort pinpointed the single cause, or a potential single cause, of failure to the μA709 integrated circuit used in the preamplifier. The details of this integrated circuit are shown in Fig. 19. The GD/CA simulation, using a prototype gyro canister with a failed integrated circuit, was able to reproduce the flight conditions. JPL investigation confirmed the analysis.

3. *Corrective action.* In order to ensure that the gyro canisters for future Centaur flights were good, GD/CA suggested a twofold corrective action: (1) enlarged acceptance testing of the gyro canisters, and (2) countdown procedural changes. The expanded acceptance testing proposed by GD/CA consisted of the following:

(1) Temperature cycling tests for dc offset.

(2) Elevated temperature and vibration tests.

(3) High-rate tests for dc latch-up.

(4) High-guidance input for dc latch-up.

(5) Check amplifier input diodes.

The countdown procedural changes were:

(1) Increase accuracy and rate gain tests.

(2) Perform rate gain tests later in countdown (at $T-10$ min).

(3) High-guidance input tests at $T-110$ min.

These proposed changes were implemented and the rate package which subsequently flew on AC-23 went through the series of testing. The countdown procedural changes were incorporated and used during the countdown.

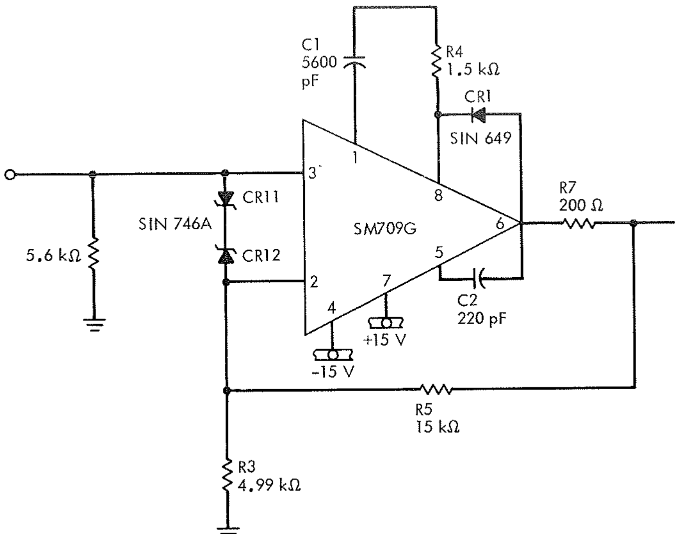

Fig. 18. Pitch rate gyro preamplifier schematic

4. NASA Headquarters review. A Failure Review Board was convened by Dr. Low of NASA Headquarters at General Dynamics, Convair Aerospace. LeRC, KSC, JPL, and GD/CA made presentations to the Review Board. The consensus of the Review Board was that the analytical approach was sound, that the results and conclusions were meaningful, and that the proposed remedial action was acceptable. The same Review Board then planned to sit in on the second Launch Readiness Review to be held at AFETR prior to the launch of the second vehicle.

5. Third spacecraft/launch vehicle. At the same time the failure investigation was going on, an additional investigation was being performed by spacecraft and launch vehicle personnel. This was an attempt to see if a third launch were possible to replace the failed one. In the spacecraft area, this would require the procurement of additional solar panels in order to bring the PTM spacecraft up to flight configuration. The launch vehicle area had several things to contend with. One was the nose fairing, of which there was not one of the right configuration available anywhere. This problem was not insoluble since there was a nose fairing available which could be modified. There was a Centaur available at GD/CA which could be modified, and an Atlas could be made available, all of these in time to support a launch by the end of the contingency launch period. Spacecraft analysis determined that the PTM spacecraft could be readied within the same time frame. In the final analysis, everyone agreed that it was possible; all it would take was money. However, the NASA decision was negative due to lack of funds. Consequently, the third launch preparations were cancelled.

6. Second launch readiness review. On May 26, 1971, a second Launch Readiness Review was conducted jointly by Mr. Parks' Launch Readiness Review Board and Dr. Low's Failure Review Board. The major items of concern were the failure of the Centaur after the first launch, the analysis of the failure, and the action, repair, and retest undertaken to qualify the Centaur for the second launch. However, the new single spacecraft mission and all of the other systems (spacecraft, TDS, and MOS) were also reviewed. The consensus of the two boards was that reasonable action had been taken on the Centaur problem and that all systems were *go* for launch.

G. Mariner I Launch

1. AFETR operations. Pending investigation of the AC-24 pitch rate gyro and to save time, MM71-2 was mated to AC-23 on Pad 36B on May 11 to conduct interface tests. On May 13, a special RFI test was conducted to verify that the spacecraft did not specifically contribute to the AC-24 failure. Results indicated no effect on AC-23. A J-FACT was conducted on May 17 with a retested rate gyro package (later replaced for launch with a more extensively tested unit) installed on the Centaur. During J-FACT, the Propellant Utilization (PU) System failed to return to the null parameters. At this time, the anomaly was attributed to lack of warm-up time. The next day during testing, the PU System was determined to have an intermittency in the electrical system. It was determined that the spacecraft would have to be demated to allow removal of a hatch and access to the inside of the hydrogen tank of the Centaur. Work continued on the PU System anomaly through May 23 when final button-up was performed. The anomaly was attributed to contamination in the PU plug at the Centaur hydrogen tank bulkhead. Concurrently, another rate gyro was being tested at GD/CA under more stringent specifications and procedures. On May 22, the retested rate gyro package was received from GD/CA and was installed on the Centaur. At this time, preparations for the second launch were scheduled as follows:

May 23 – Mate MM71-2 to AC-23 on Pad 36B

May 23 – J-FACT

May 24 – CRT

May 25 – TCD and RFI test

May 29 – Launch

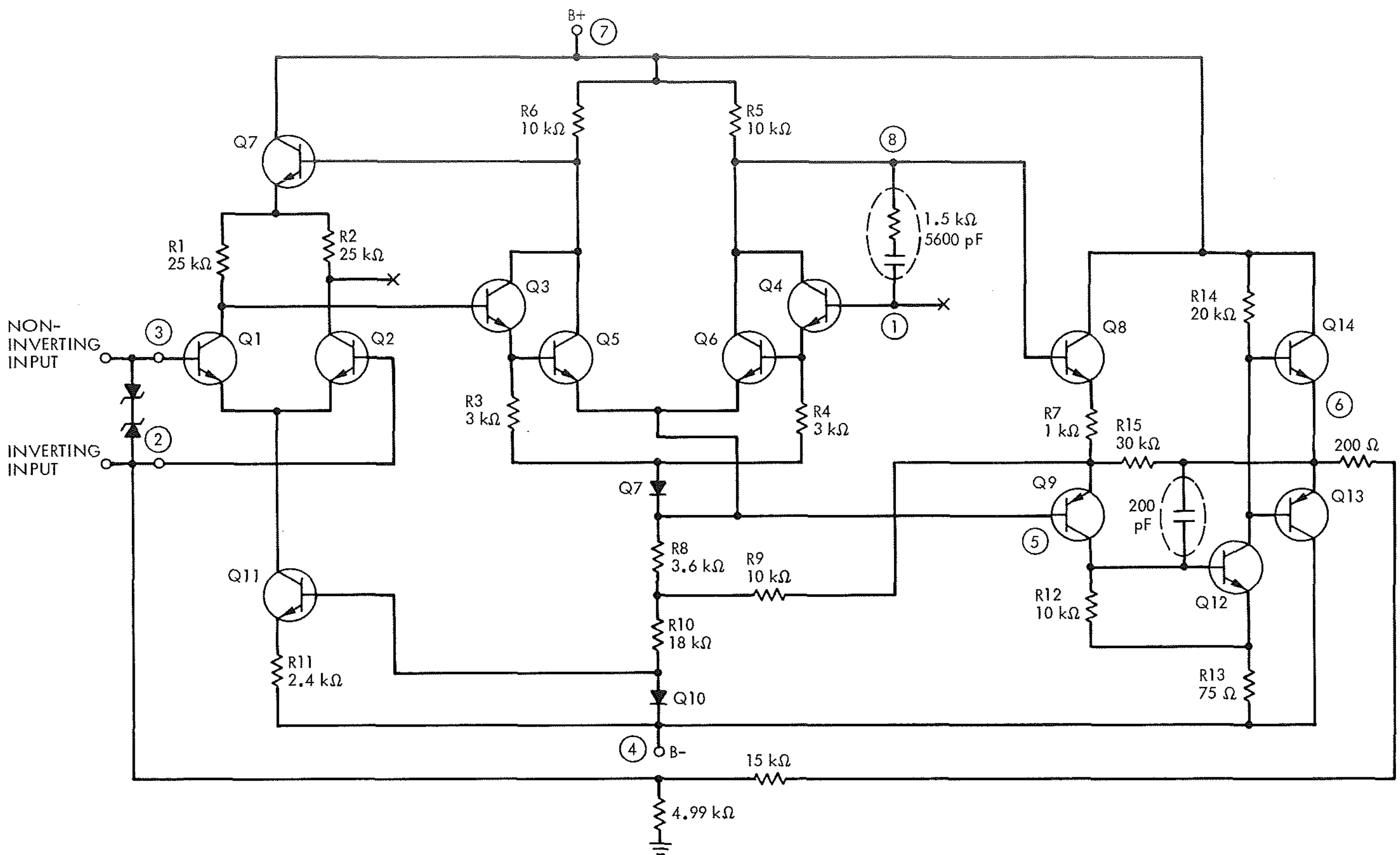

Fig. 19. Pitch rate gyro preamplifier integrated circuit

The J-FACT was required due to replacement of the rate gyro package and modifications (replacement of tantalum capacitors with four ceramic capacitors which are not degraded by back-bias voltage) in the Centaur guidance package. Another TCD was required to check the rework done on the Centaur hydrogen tank and also to demonstrate the special rate gyro torquing test, special vent valve lockup test, and careful examination of the PU System. All tests yielded satisfactory results.

The first countdown for launch on May 29, 1971 proceeded normally to $T-110$ min, at which time special checks were performed to verify proper operation of integrated circuits in the control loop. Data indicated a problem in the pitch channel which could not be resolved prior to the close of the launch window. The countdown proceeded to $T-72$ min and holding so as not to commit propellants to the vehicle. The launch was scrubbed at 22:01 GMT. Later in the evening, the problem was resolved to be a characteristic of the GSE (auxiliary programmer) when operated in the mode used for this special test. The launch was rescheduled for the next day.

The second countdown for launch on May 30, 1971 proceeded normally except for two launch vehicle anomalies:

(1) At $T-5$ min, landline instrumentation of the Atlas PU error demodulator voltage shifted off scale. Telemetry verified that the problem was associated with the landline and that the voltage was in tolerance.

(2) Throughout the countdown the Centaur PU System experienced erratic propellant quantity readout with the landline instrumentation system. Telemetry verified that this was a landline problem so telemetry was used as prime instrumentation.

These problems caused a hold, and launch occurred 6 min into the launch window.

The AC-23 (Atlas SLV-3 S/N 5404 and Centaur S/N 21D) and the MM71-2 were launched from AFETR Complex 36B, May 30, 1971, at 22:23:04 GMT (day 150) on a flight azimuth of 92.742 deg. The weather was as follows:

Temperature: 23.8°C (74.9°F)

Relative humidity: 60%

Visibility: 16 km (10 mi)

Dew point: 15.5°C (60°F)

Surface wind: 3.6 m/s (7 knots) from 090 deg

Clouds: sky obscured by smoke to 3048 m (10,000 ft)

Sea level atmospheric pressure: 757.301 mm (29.815 in.) of mercury

Performances of launch vehicle and spacecraft were nominal in all aspects. The spacecraft was injected into an approximate one-sigma trajectory to Mars.

2. *MOS launch operations*

a. Spacecraft. Separation in the Earth's shadow was observed on the spacecraft 13 min and 18 s after launch, by the counter 3 "pyro arm" event indication, and switching of telemetry Channel 113 from B-axis gimbal measurements to Sun sensor measurements. This was followed by subsequent confirmation that the attitude control switching amplifiers for the cold gas reaction control system were enabled, that the 30-Vdc regulator was turned off, and that power changes associated with the removal of the CC&S relay-hold function had occurred.

After a nominal time for the pyro capacitor banks to achieve full charge, the first spacecraft squib firing occurred, releasing the four solar panel tip latches at 22:40:55.962. The four panels did not unfold at the same rate, but all were deployed within 8.4 s of one another by 22:42:02.9.

The spacecraft was now in a state to acquire the Sun upon emergence from the Earth's shadow. The penumbra was first seen at 23:08:46. After 2 min and 34 s, there was a sufficient amount of sunlight so that the spacecraft started a Sun search, as evidenced by pitch rate changes. Sun acquisition was achieved at 23:15:59.

Command modulation was applied at 23:30:00, with command lock occurring at 23:36:11.804. The first command, a DC-9 to turn on the ranging channel, was sent at 23:40:00, and ranging modulation was applied shortly thereafter. Problems with the ground equipment prevented acquiring good ranging data until DSS 62 acquired good correlation at 01:23:00 on day 151, May 31, 1971.

The second "hours" scan of the CC&S at 00:09:59 produced a 16J command to put the DSS into slew mode, advancing to the left end of tape, Pass 1. Simultaneously, the data rate logic was changed from the launch configuration of 4 kbps to the orbital operations configuration of 16 kbps by CC&S command 16A. Upon completion of the slew, at 00:15:13, the DSS logic went into *ready mode*, indicating that it was now "parked" for the cruise phase.

The fourth "hours" scan produced a 7B command, turning on the Canopus sensor. Since Sun acquisition, while the spacecraft was fully stabilized in pitch and yaw, it was drifting without a reference for the roll axis. Application of power to the Canopus sensor caused it to immediately search for Canopus, the roll reference star. On the first roll, Achernar was acquired instead of Canopus, as expected. The second ground command sent to the spacecraft since the flight began, a DS-21, was transmitted to disacquire Achernar and continue the search for Canopus. The next star acquired was Canopus, and the spacecraft was fully stabilized in 3 axes at 02:25:10. Three min and 36 s later, the 3-min timer turned the gyros off.

At 03:04:57, the automatic switchover circuit was activated, and the high-rate battery charger was turned off, and the low-rate charger turned on.

The day after launch, two major activities took place: unlatching of the scan platform, followed by venting the engine supply lines to exhaust entrapped air.

The first activity consisted of putting the spaceraft into roll-inertial control by CC&S command 7F (CC&S straylight signal) to prevent potential loss of Canopus during the squib firing required to unlatch the platform.

In previous missions, such squib firings have caused bright particles to come into view of the Canopus tracker, causing loss of lock of Canopus for a period of time. By maintaining inertial control, the loss of Canopus due to bright particles is avoided.

The unlatch was accomplished by ground command DC-45 at 22:31:01, so timed that Channel 421, scan latch pressure, would be sampled 30 s into the pressure decay.

The second part of the day's activity began with a load of 10 coded command 1/2 pairs to the CC&S starting at 00:30. The purpose of CC&S update No. 1 was to remove some launch program diagnostic routines, insert a new 2A time and to enable DC-32 for the engine vent routine.

A DC-84 was sent following the load to initiate a checksum routine. The counter 2 event indicated that CC&S update 1 was properly received.

A DC-32 was then sent to initiate the engine vent sequence. This sequence was very much like a computer-only maneuver except that there were no turns programmed, and the fuel and oxidizer were not available to the engine. Engine venting allows air at one atmosphere to escape from the propellant lines, thereby providing a hard vacuum for liquid filling.

b. Navigation. Orbit determination at launch plus 28 h indicated a good trajectory with a miss distance at Mars of about 30,000 km (OD uncertainty about 300 km) at about noon on November 14. This represented slightly greater than a one-sigma injection inaccuracy.

The trajectory correction maneuver required to correct for miss distance and time of flight was predicted to be about 8 m/s at launch plus 5 or 6 days.

c. Ground systems. Ground systems conditions were generally acceptable, but less than perfect. Outstanding problems were:

(1) Hardware failure in 360/75B core memory array—machine unavailable major part of the day.

(2) DSS 51 ranging subsystem down.

(3) DSS 12 ranging subsystem down for a short period.

DSS 14 tracked for the first three days of the mission, while the spacecraft was within range of the Mark IA ranging equipment, in order to obtain a comparison of the lunar and planetary ranging systems.

IX. First Trajectory Correction Maneuver

Two trajectory correction maneuvers (TCM) were planned for Mariner 9 during its cruise to Mars. These maneuvers would eliminate an intentional target bias (required by planetary quarantine considerations) and correct for any launch vehicle-induced trajectory errors. Establishment of an accurate trajectory, time of arrival at Mars and target point, would allow insertion into the desired Mars orbit with minimum expenditure of propulsion fuel.

Final preparations for the first trajectory correction maneuver began shortly after the successful Mariner 9 launch on May 30, 1971. Successive orbit determination calculations were run, and a maneuver strategy was developed. As additional tracking data were obtained from the Earth-based antennas, knowledge of Mariner 9's trajectory became increasingly accurate. Maneuver strategy studies led to the best procedure for turning the spacecraft to the desired orientation for firing the engine. From the Sun- and Canopus-stabilized orientation, it was decided to first roll the spacecraft 141 deg (counterclockwise as viewed from the Sun) about an axis through the rocket nozzle and then turn the spacecraft about its yaw axis -45 deg (counterclockwise as viewed from above, looking down on the spacecraft toward the star Canopus). These turns would orient Mariner 9 with its rocket pointing almost toward Earth. After engine firing, the spacecraft would be returned to its previous three-axis stabilization by reversing the order and direction of the turns.

On June 3, 1971, operations planned for the Mariner 9 first trajectory correction maneuver were checked on the proof test model at the JPL Air Force Eastern Test Range facility. Beginning at 19:30:00 GMT on the same day, a series of six coded commands (CC-4) were transmitted to Mariner 9, followed by thirteen CC-1 and CC-2 pairs on one-minute centers starting at 19:36:00, to load the maneuver parameters into the fixed central computer and sequencer (CC&S). Next, the maneuver-enabled direct command (DC-14) was sent, followed by a DC-33 to put the CC&S in the tandem standby mode and a DC-29 to disable the divide-by-32 network in the accelerometer circuitry. The tandem standby mode was a necessary condition for executing a "tandem" maneuver.

The spacecraft was then placed in the roll inertial mode prior to propulsion subsystem pressurization to avoid the possibility of loss of Canopus due to bright particles released by the pressurization impulse. DC-65 was transmitted at 21:17:25 to fire pyrotechnic valves in the propulsion subsystem. With these valves open, both oxidizer and fuel storage tanks were pressurized, forcing propellants through the lines to the main engine valve. The impulse about the yaw axis was quickly damped out; no bright particles were observed. Canopus reference was restored by DC-19 at 21:48:00.

The first trajectory correction maneuver was executed on June 4, 1971. The spacecraft CC&S loads were further refined by two CC-4's and twelve CC-1 and CC-2 pairs, respectively. A time-critical DC-52 transmitted at 22:19:04 started the on-board maneuver routine; then gyros were turned on. The following table indicates the programmed and actual values of the maneuver:

Parameter	Programmed value	Actual value
Roll turn magnitude, deg	−140.806	−140.717
Yaw turn magnitude, deg	− 44.725	− 44.828
Roll turn time, s	777	777
Yaw turn time, s	247	247
ΔV imparted to spacecraft, m/s	6.731	6.723
Accelerometer pulse count	223	223

Nominal performance occurred during the spacecraft roll and yaw turns. At 00:22:00 GMT, June 5, 1971, the main engine valve was opened, and the hypergolic propellants, nitrogen tetroxide (oxidizer) and monomethyl hydrazine (fuel), burned for 5.1 s until the main engine valve was automatically closed. Spacecraft yaw and roll unwind was accomplished. At 00:48:44, after a short roll search, Mariner 9 reacquired the Sun and Canopus celestial references. The gyros were turned off 3 min and 36 s later.

Tracking data indicated that the first trajectory correction maneuver was extremely accurate and the orbit determination computations on June 14, 1972 showed:

Parameter	Targeted	Achieved	Error
ΔTCA (time of closest approach)	19 h 06 min 36 s	19 h 04 min 28 s	02 min 08 s
ΔB (B-plane target point correction), km	24,948	24,869	140 (vector error)

X. Single Mission Plan

A. Single Spacecraft Impact

Immediately following the failure to successfully launch Mariner H (MM71-1 spacecraft aka Mariner 8), intensive mission design activities were initiated. With only one spacecraft remaining to be launched, neither the plan for Mission A nor Mission B alone was adequate to meet all of the mission objectives. Consequently, a new hybrid mission had to be designed which would accomplish the mission objectives within the capabilities of the existing systems. During the time between the first and second launch, a single mission plan was developed sufficiently to provide the necessary confidence to proceed with the launch of Mariner I. All of the basic elements of the single mission plan were known and understood prior to the second launch although certain details and documentation were lacking.

This mission plan reflected a concerted effort to maximize the science value for the single mission so as to minimize the impact on the experiments by the reduction of two missions to one.

B. Orbit Design

Characteristics of the orbit (Ref. 55) are summarized in Table 14.

Table 14. Orbit characteristics

Period	11.98 h
Periapsis altitude	1200 to 1500 km
Inclination	65 deg
Apsidal orientation	136 to 142 deg
Arrival date	November 14, 1971 (GMT)

The 11.98-h orbital period would allow variable feature studies of Mars to be made. Mars revolves about the Sun at the rate of 0.538 deg of celestial longitude per Mars mean solar day (24.660 h). The 11.98-h period is 17/35 of the Martian mean solar day. Thus, after 17 Martian mean solar days (and 35 spacecraft revolutions), the orbit ground track on the surface of Mars would begin to repeat itself. The 11.98-h orbit period would be synchronized with the 0.538-deg/day motion of Mars about the Sun. After every 17 Martian mean solar days, the solar illumination conditions of any specific point on the planet as viewed from orbit would be nearly constant. This is important for studies of variable features.

The minimum periapsis altitude of 1200 km or higher was chosen to ensure that, when vertical wide-angle camera pictures are taken at periapsis, there would be front-lap (contiguity) between adjacent pictures taken every 84 s; side-lap gaps would occur between sequences of pictures taken near periapsis on adjacent orbits. However, these gaps would be filled in on subsequent 17- to 18-day longitude circuits.

The orbit inclination (65 deg) was a compromise between a higher inclination, which would provide excellent coverage of the south polar region, and a lower inclination, which would provide better viewing of variable-feature phenomena near the sub-solar point. Planetary quarantine did not affect the choice of inclination.

The apsidal orientation angle, ψ, is measured from the incoming hyperbolic direction in which the spacecraft approaches Mars (the approach asymptote) to the orbit ellipse line of apsides (Fig. 20). As ψ is increased, the latitude of periapsis would move north on the surface of Mars, and the opportunities for global TV coverage would be enhanced. Global TV coverage would consist of several low-resolution pictures, which would map the lighted disk of the planet about 2 h before periapsis. The time would be dictated by the scan platform viewing limitations. If ψ is increased, pictures could be taken at higher altitudes, which would reduce the total number of pictures needed for a mosaic of the lighted disk.

If a periapsis-to-periapsis (minimum impulsive) orbit insertion is attempted, the resulting apsidal orientation angle would be equal to about 118 deg. However, by consuming more spacecraft retro fuel at orbit insertion, the value of ψ could be increased to a maximum of 136 to 142 deg depending on the delivery accuracy of the spacecraft at the planet.

Figure 21 illustrates the characteristics of the proposed orbit. The times at which the center (or the limb) of Mars comes into or goes out of view is determined by the spacecraft scan platform mechanical limits. The cone angle is limited to the range of 96 to 165 deg; the clock angle range is 90 to 305 deg. As long as the scan platform cone and clock angles remain within these limits, the spacecraft instruments could view the planet vertically at the sub-spacecraft point on the planet. When the scan platform hits one of its limits, off-vertical viewing of some points of the planet might still be possible until the planet limb is reached. Figure 21 shows the time when the spacecraft would pass through the 60-deg solar incidence angle contour on its way to the evening terminator (90-deg incidence angle). Because of the orbit inclination

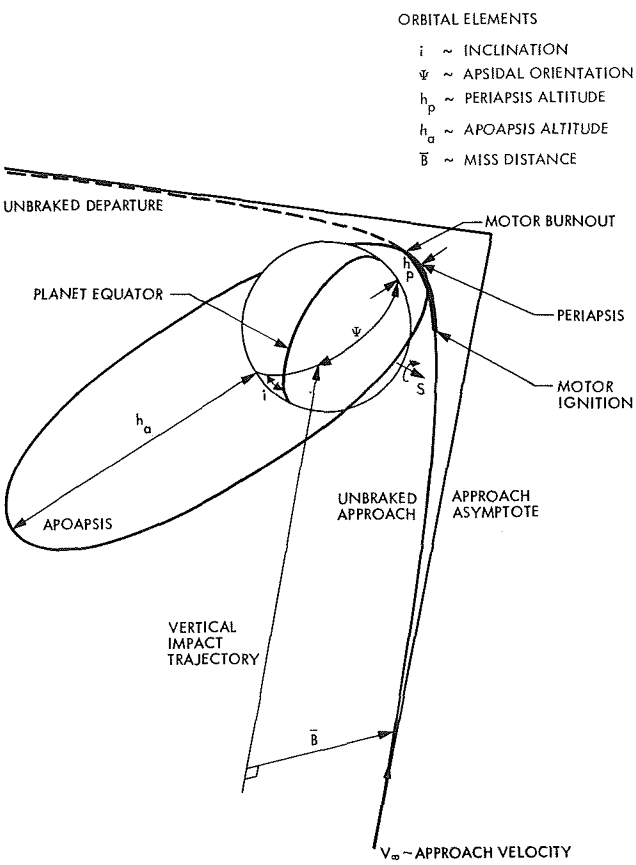

Fig. 20. Approach trajectory and elliptical orbit

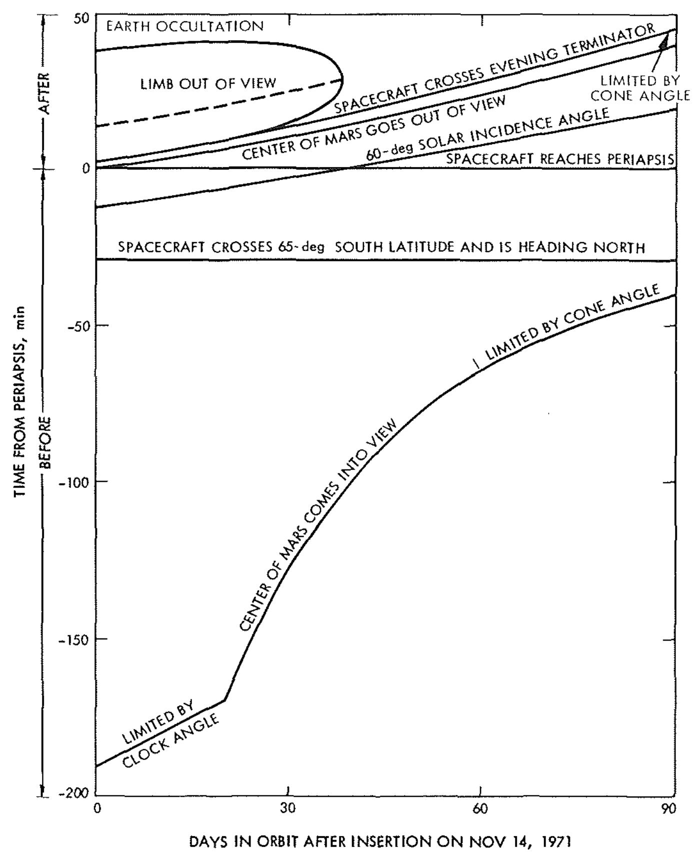

Fig. 21. Orbit characteristics

selected, the spacecraft ground track would be confined between 65 deg north and 65 deg south latitude on Mars. About 29 min before periapsis, the spacecraft would cross 65 deg south latitude and start to head north along its ground track toward periapsis.

Shortly after passing through orbit periapsis, the spacecraft would enter Earth occultation. During occultation, the planet limb would pass out of view of the scan platform, as shown by the dashed line in Fig. 21. After approximately 38 days from insertion, Earth occultations would cease for the balance of the 90-day nominal mission.

There would be no solar occultations during the 90-day nominal mission. Solar occultations would commence about 135 days after orbit insertion (during the proposed extended mission). The occultation period would not exceed 1 h and 40 min on any orbit. The solar occultations would end about 190 days after insertion. To conserve spacecraft power during the period of solar occultations when the solar panels are periodically shaded, the science instruments would be turned off.

An additional period of Earth occultations would commence about 170 days after insertion and last for about 40 days.

Reflected stray sunlight from the surface of Mars might enter the field of view of the Canopus sensor for several minutes during each orbit starting some 40 days after orbit insertion. The spacecraft would have to turn on its gyros to maintain its roll reference during these periods.

C. Maneuvers

If required, the second trajectory correction maneuver (TCM) would occur on October 26, 1971. The purpose of this maneuver would be to refine the spacecraft trajectory so that the Mars orbit insertion maneuver and the trim maneuver(s) would produce desired orbit parameters with lowest propellant expenditures. Motor burn time would be approximately three seconds, and the time to reacquire Canopus would be one hour. (Tracking data and orbit determination computations performed in September and October 1971 showed that the first TCM was sufficiently accurate to justify cancellation of the second TCM.)

The Mars orbit insertion (MOI) maneuver (Ref. 56) would occur on November 13, 1971 (November 14, 1971 GMT). The purpose of this maneuver was to decrease the spacecraft speed so that the Mars gravity field would capture the spacecraft in an orbit whose parameters were:

Period	12.5 h
Periapsis	1300 km
Inclination (to Mars equator)	65 deg

The orbit insertion maneuver also would take place over Goldstone because the high-gain antenna would be pointing off the Earth and the engineering telemetry would have to be played back over the medium-gain antenna and received over the 64-m antenna at Goldstone. The motor burn would begin about 28 min prior to closest elliptic approach. Total burn time would be approximately 16 min and the time to reacquire Canopus and initial doppler data would be approximately 2 h.

The orbit insertion maneuver would be a planar transfer from the hyperbolic orbit to the elliptic orbit. This meant that there would be no change made in the inclination at orbit insertion. Due to the required rotation of the periapsis of the elliptic orbit, the orbit insertion would not be a minimum energy transfer.

An orbit trim maneuver (Ref. 56) would occur between November 16 and 18, 1971. The purpose of this maneuver was to modify the orbit produced by the MOI maneuver from the 12.5-h period to the desired 11.98-h period and to synchronize the time of periapsis passage with the Goldstone zenith.

The orbit trim strategy was designed to place the time of periapsis of the orbit within an envelope that would provide for a low probability of overlap of playbacks with Earth occultations and portions of the planet-in-view period for the duration of the 90-day mission (Fig. 22). The size of this envelope would be constrained by the ± 0.3-min post-trim period tolerance (3σ) and by the planned asynchronism with Goldstone of the desired and nominal period of 11.98 h.

In the event of large insertion dispersions, two trim maneuvers would be executed to achieve synchronism. The second trim maneuver would be performed at periapsis 8 days after insertion. In Fig. 22, a typical single trim case is shown as a solid line. The two trim cases are indicated by dashed lines.

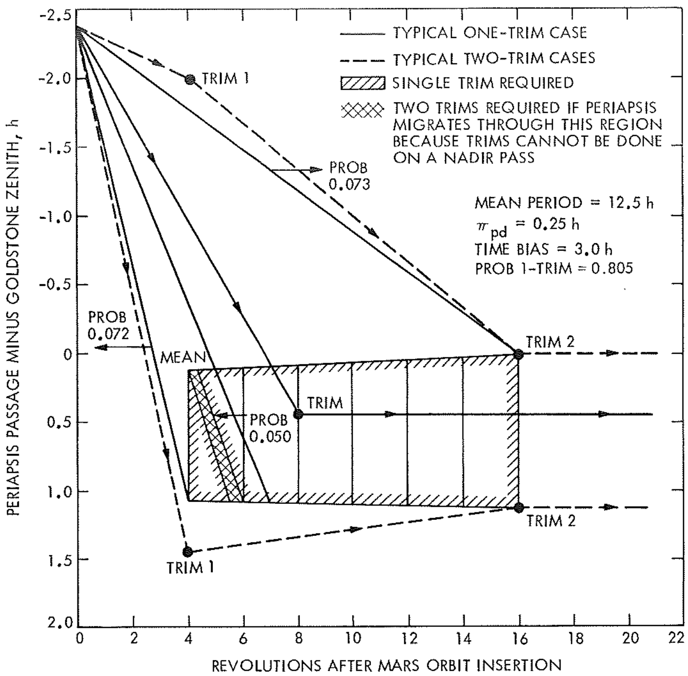

Fig. 22. Time of periapsis envelope

D. Science Sequences

1. Pre-orbit insertion. When the spacecraft is close enough to obtain scientific measurements, primarily visual imaging, the first of three pre-orbit science sequences (POS-1) would be conducted (Ref. 57). POS-1 would begin on November 10, 1971, and continue for approximately 24 h. A 3-h playback of the data over the 64-m Goldstone antenna would be conducted on November 11, 1971. Twenty-five narrow-angle TV pictures of Mars would be taken at 62-min intervals along with six narrow-angle TV pictures of the natural satellite Deimos. The Mars pictures would provide global coverage of all longitudes.

Pre-orbit science sequence 2 (POS-2) would begin on November 11, 1971 and continue for 24 h. A 3-h playback of the data over the 64-m Goldstone antenna would be conducted on November 12, 1971. Essentially the same coverage as POS-1 was planned, except that 24 narrow-angle pictures of Mars and 7 pictures of Deimos would be taken.

Pre-orbit science sequence 3 (POS-3) would begin on November 12, 1971 and continue for approximately 10 h. A 3-h playback of the data over the 64-m Goldstone antenna would be conducted, after orbit insertion, on November 13, 1971 (PST). Twenty-three wide- and narrow-angle TV pictures of Mars would be taken at 2-h intervals, producing a mosaic of most of the planet. (By this time, the spacecraft would be very close to Mars and the TV cameras could not see the entire planet.) Satellite coverage would be two pictures of Phobos and five pictures of Deimos.

2. Orbit operations. Figures 23 and 24 show typical Goldstone zenith and nadir science sequences as they might appear in the mission. As Mars rises to an elevation angle of approximately 15 deg above the local horizon at Goldstone, playback of the tape load of data taken during the preceding nadir pass would begin. Nearly 3 h would be required to play back the data at a rate of 16.2 kbps. After playback, there might be an opportunity to take global coverage pictures.

Three geodesy pictures from the wide-angle camera would be taken at 84-s intervals beginning about 1 h and 30 min before periapsis. These wide-angle camera pictures would be targeted at latitudes of about 15 deg south, 30 deg south, and 45 deg south. On one pair of orbits, the TV pictures would be taken at the sub-spacecraft longitude; on the succeeding pair of orbits, the pictures would be taken ahead (east) of the sub-spacecraft longitude to provide a stereo effect. Figures 25 and 26 show typical geodesy pictures.

Scan platform cone and clock angles would be selected so that, as the ultraviolet spectrometer slit sweeps down through the atmosphere above the brightest portion of the Mars surface, the slit would be as close to perpendicular to the local vertical as possible when it passes through an altitude of 100 km above the surface (Fig. 27). The ultraviolet spectrometer data would be transmitted in real time at 8.1 kbps. During the conduct of this bright limb experiment, three TV pictures of the atmosphere immediately above the planet limb might be recorded. On succeeding orbits, wide-angle camera pictures using different colored filters might be taken to provide spectral studies of stratified atmospheric haze.

Approximately four wide- or narrow-angle pictures might be taken when the spacecraft is orbiting near 65 deg south latitude. These pictures might be aimed toward the south polar region or, to study variable features on the surface, north (on alternate orbits) toward the latitudes where the solar illumination is highest.

Ultraviolet pressure mapping would be conducted by pointing the ultraviolet spectrometer at the lighted side

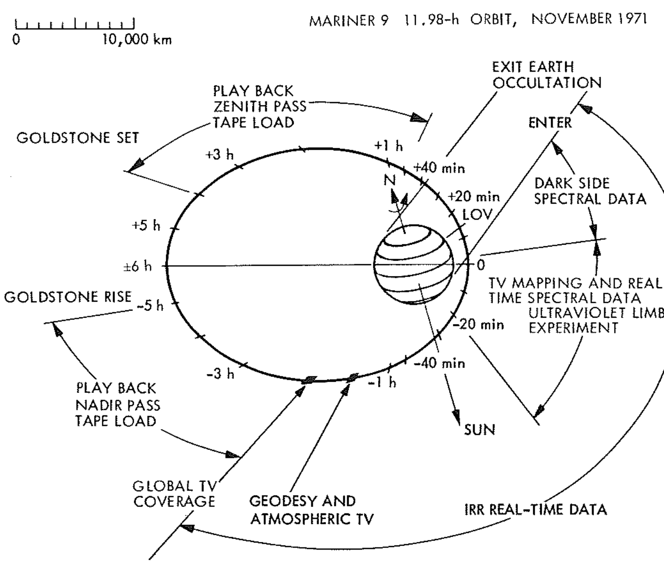

Fig. 23. Typical Goldstone zenith science sequence

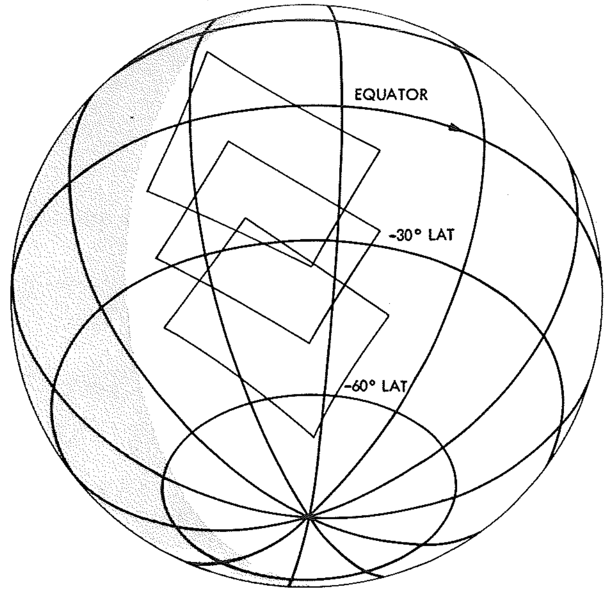

Fig. 25. Typical geodesy sequence, view near vertical

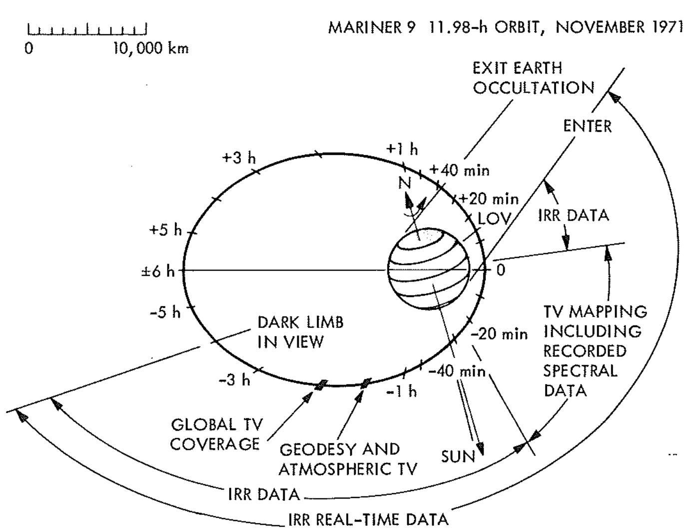

Fig. 24. Typical Goldstone nadir science sequence

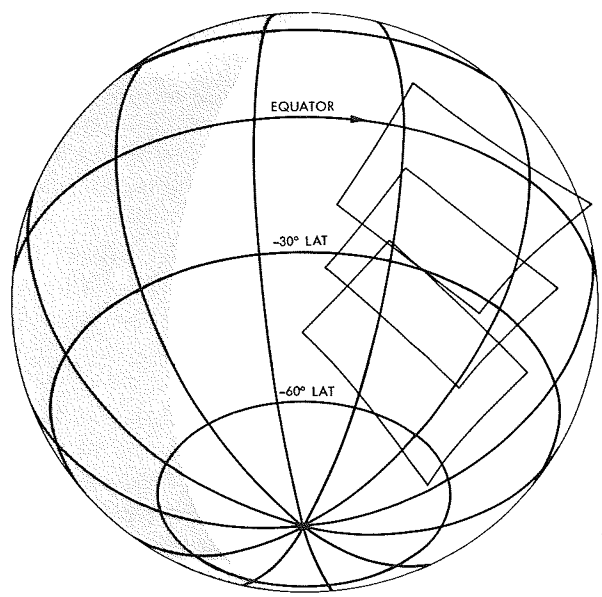

Fig. 26. Typical geodesy sequence, looking eastward

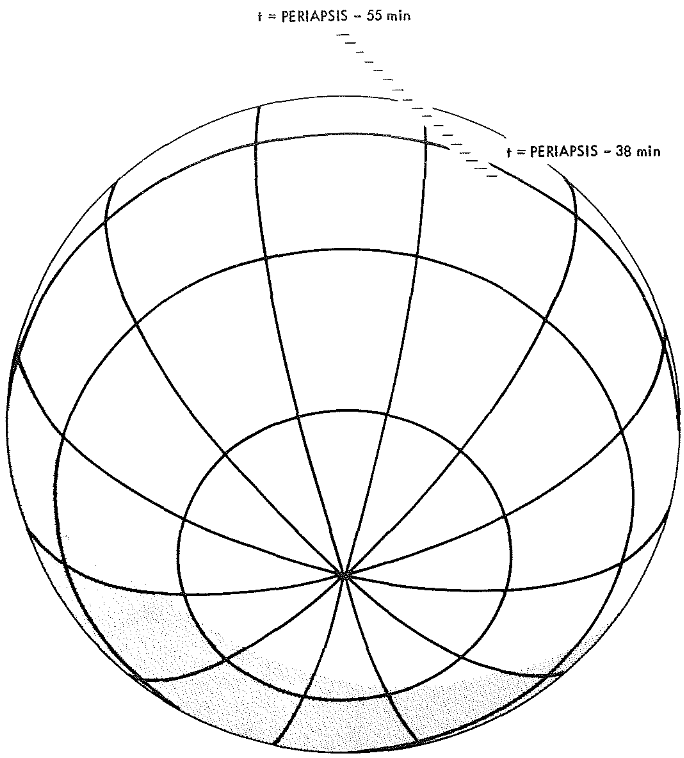

Fig. 27. View of the ultraviolet spectroscopy experiment

of the planet surface during a portion of the orbit up to within about 20 deg of the terminator. The data again would be transmitted in real time at 8.1 kbps.

The mapping sequence was the heart of the mission. This sequence would consist largely of TV wide-angle camera pictures of selected points on the Mars surface supplemented by TV narrow-angle camera pictures at 10 times better resolution. Figures 28 and 29 can be used to visualize the mapping of Mars. In Fig. 29, the bands of latitudes to be mapped are shown in 20-day intervals beginning after the final orbit trim and continuing throughout the nominal mission. In each of the four mapping sequences shown, 360 deg of Mars longitude are covered between two parallels of latitude. As the mission progresses, these bands of latitude would shift northward to follow the evening terminator and to fill in side-lap gaps between sequences of pictures taken on adjacent orbits. For an orbit inclination of 65 deg and with an apsidal orientation angle of 136 to 142 deg, the latitude of periapsis would be initially at about 20 deg south latitude. As the mission progresses, the latitude of periapsis would decrease slightly because of a slight negative precession of the orbit line of apsides. When vertical wide-angle camera pictures are taken north or south of periapsis, the spacecraft altitude would be greater; the TV wide-angle pictures would have positive front-lap; and the resolution would be reduced. Most of the wide-angle pictures would be taken vertically. The last vertical picture could be taken at a cone angle of 96 deg. Late in the mission, the fourth mapping sequence could be extended northward to perhaps 45 degrees north latitude by fixing the scan platform at a cone angle of 96 deg and at a clock angle of 305 deg and then taking off-vertical wide-angle camera pictures up to the terminator. Thus, the planet could be mapped completely from 65 deg south latitude to 45 deg north latitude during the standard mission.

When the spacecraft is in view of Goldstone, the planet is in view of the spacecraft scan platform, and the tape recorder is not being played back, the spectral data from the infrared radiometer, the infrared interferometer spectrometer, and the ultraviolet spectrometer would be transmitted to the 64-m Goldstone station. Also, during the mapping sequence when the tape recorder is on, complementary multi-spectral data from the ultraviolet, visual, and infrared portions of the electromagnetic spectrum would be obtained.

Early in the mission, shortly after the mapping sequence ends near periapsis, the spacecraft would enter Earth occultation and communication with the Earth would be temporarily lost. However, as the spacecraft enters and exits occultation, the S-band radio doppler data obtained would ultimately be used to determine the pressure profile of the Mars atmosphere. After Earth occultations cease, communication with the spacecraft during the 40 min following periapsis passage would be maintained on Goldstone zenith passes. Between the time the mapping sequence ends and the limb of Mars passes out of view of the spacecraft scan platform, spectral data from the ultraviolet spectrometer, infrared interferometer spectrometer, and infrared radiometer could be obtained from the night side of the planet beyond the evening terminator.

After the limb of the planet passes out of view of the scan platform, the ultraviolet spectrometer could be pointed to various regions of space surrounding Mars in search of Lyman alpha particles. The data would be transmitted in real time at 50 bps. At this low data rate, continuous coverage could be provided independently from Goldstone by the world-wide network of 26-m (85-ft) diameter antennas.

The celestial mechanics experiment would continue throughout the mission. However, the most useful data

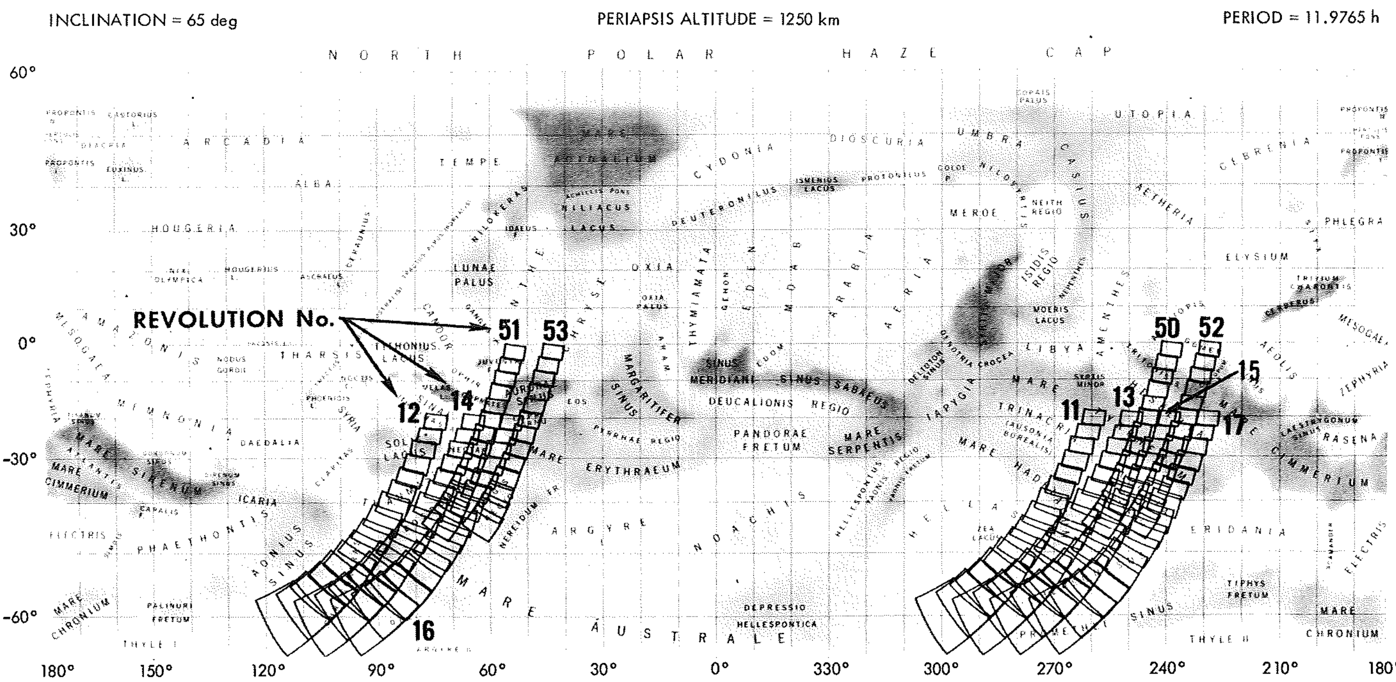

Fig. 28. Typical wide-angle camera footprints for surface mapping

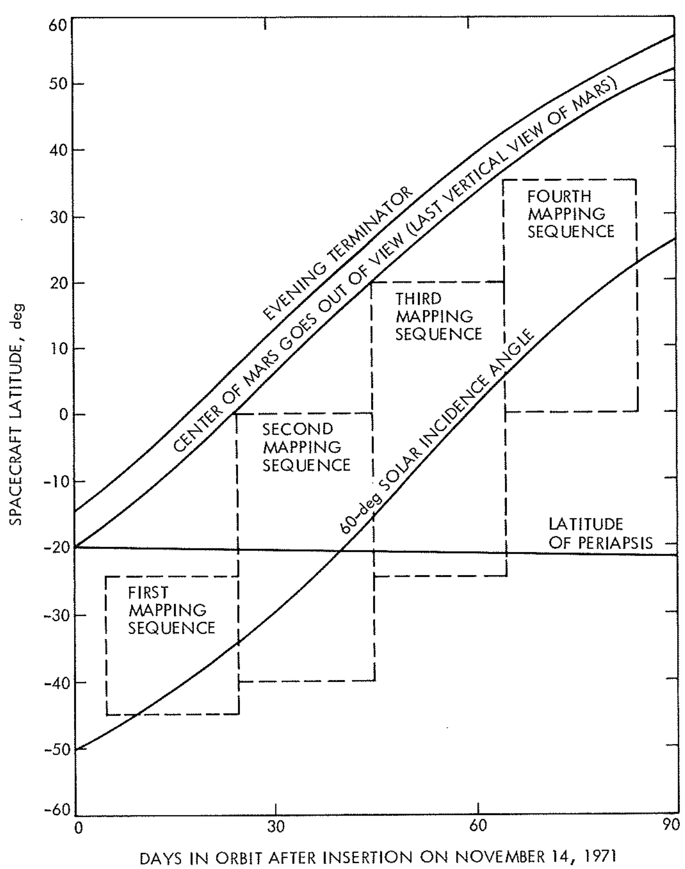

Fig. 29. Mars mapping sequence

would be obtained near each periapsis passage. One- and two-way doppler data would be obtained on Goldstone zenith and nadir passes at 33⅓ bps, except for periods of Earth occultation. However, ranging data could be obtained only on Goldstone zenith passes.

After the spacecraft exits Earth occultation, playback of the tape load of data obtained on the previous Goldstone zenith pass would be completed in about 3 h. After playback, Goldstone would set. When Goldstone is not in view, no tape recorder playbacks would occur, and no high-rate spectral data would be returned in real time.

During the period from 3 to 1.5 h before periapsis, opportunities would exist to take TV wide-angle pictures of the lighted portion of the planet disk and of the atmosphere above it. This global TV coverage would be constructed from a mosaic of the wide-angle camera pictures taken on each nadir orbit. If three pictures are taken per orbit, a single wide-angle camera color filter might be used (Fig. 30). If six pictures are taken per orbit, two TV wide-angle camera pictures of selected portions of the lighted limb and of the atmosphere above it could be taken. The best opportunities for global coverage would occur early in the mission after the limb of Mars first comes into view of the scan platform (cone angle = 96 deg and clock angle = 90 deg). As the mission progresses, the

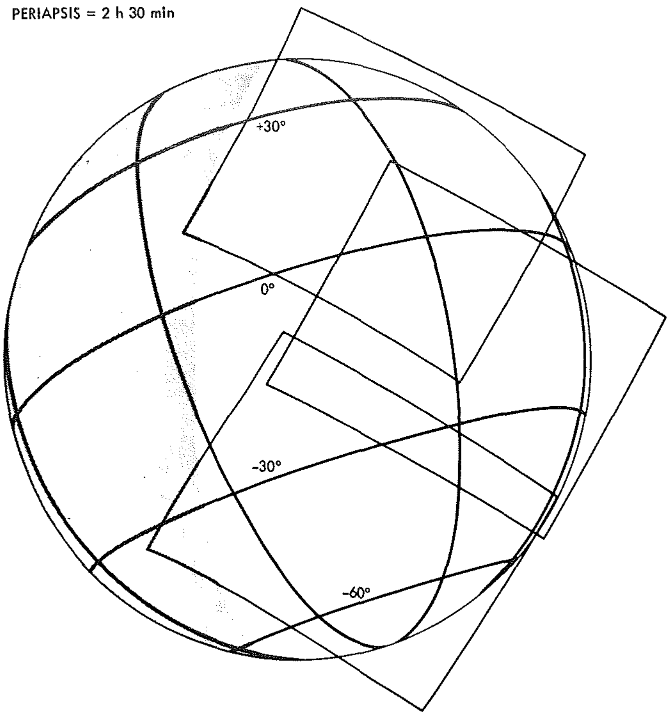

Fig. 30. Typical global coverage pictures

limb and the center of Mars would come into view when the spacecraft is closer to periapsis, the angular diameter of Mars would be larger, and the number of TV wide-angle pictures required to mosaic the lighted portion of the planet would escalate rapidly. Global coverage would probably end 25 to 30 days after insertion. The apsidal orientation angle, ψ, would be made as large as possible at orbit insertion to permit global coverage for the maximum practical period after arrival.

On the nadir pass, geodesy TV pictures would be taken about 1 h and 30 min prior to periapsis, but they could not be played back at that time because Goldstone would not be in view. South polar region TV or high-Sun TV pictures would be taken on alternate nadir orbits. TV mapping, including recorded spectral data, would be obtained on each nadir pass.

The only real-time data that could be obtained on nadir orbits would be infrared radiometer data and ultraviolet spectrometer Lyman alpha data at 50 bps or celestial mechanics one- and two-way doppler data at $33\frac{1}{3}$ bps. This data would be received by the 26-m (85-ft) diameter antenna network.

After the spacecraft is in orbit from 90 to 120 days, the north polar cap could be seen in the light by the scan platform of the spacecraft. This viewing period would occur from 0.5 to 1.5 h after periapsis.

Photography of the Martian Moons, Phobos and Deimos, would be obtained many times throughout the mission. The spacecraft would pass to within 5000 to 10,000 km of the satellites at the optimum opportunities.

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Appendix A

Project and System Milestones

Date completed	Milestone
4/30/68	Announcement of Flight Opportunity (AFO) issued
8/23	Project Approval Document (PAD) signed
9/30	Payload selection completed
10/17	Task Order issued
10/25	Project Policy and Requirements (PP&R) (PD 610-4) issued
11/1	Start Mission Operations System (MOS) Functional Design Book
12/2	Start spacecraft subsystem rework and fabrication
1/13/69	MOS and Tracking and Data System (TDS) Management Plan (PD 610-14) issued
1/31	Mission Design Review completed
2/17	Launch vehicle Centaur contract issued
2/21	Spacecraft System Functional Design and Design Review completed
4/10	Mission Design and Mission Specification (PD 610-16, Part 1) issued
4/14	Spacecraft Subsystem Functional Review completed
4/18	Cruise science experiments removed
5/1	Telecommunications Requirements Definition completed
5/1	Mariner Mars 1969 proof test model transferred to MM'71
5/21	Preliminary Support Instrumentation Requirements Document (SIRD) (PD 610-41) issued
5/23	Near-Earth Requirements Definition completed
6/1	Maneuver Operation Functional Requirements completed
6/20	Spacecraft System Detail Design Review completed
7/23	Spacecraft Operational Support Equipment Review completed
8/18	Mission Specification (PD 610-16, Part 1, Revision A) issued
8/29	Mariner Mars 1969 flight spare transferred to MM'71
9/2	MOS Functional Design and Design Book completed

Date completed	Milestone
9/5/69	Mission analysis and engineering inputs to Program Requirements Document completed
9/8	Quality Assurance Plan (PD 610-15) issued
9/12	Preflight PEGASIS program completed
9/16	Spacecraft Subsystem Detail Design and Detail Design Review completed
9/19	Contract plans and negotiations completed
9/22	Spacecraft Environmental Test Review completed
10/1	Preliminary Planetary Quarantine Estimate completed
10/1	Maneuver Programs–Interplanetary Program completed
10/1	Command, Telemetry Metric Constraints Definition completed
10/2	Spacecraft/Launch Vehicle Design Review completed
10/15	Near-Earth nominal trajectory data delivered by Lewis Research Center to Air Force Eastern Test Range (AFETR)
10/15	Flight Plan Approval Letter delivered to AFETR
10/15	Deep Space Network (DSN) Operations Plan–DSN/Spacecraft Telecommunications Interface (PD 610-74, Volume IIA) issued
10/17	Statement of Flight Approval received from AFETR
10/22	Interface Compatibility Management Plan (PD 610-93) issued
10/22	Program Requirements Document issued
10/24	MOS and TDS Functional Design Review completed
11/6	Spacecraft/Launch Vehicle Interface Schedule (PD 610-61) issued
11/13	Project Plan (PD 610-8) issued
11/14	Initial Mission Plan (PD 610-16, Part II, Draft 1) completed
12/1	Spacecraft/Centaur Launch Vehicle Hardware Interface Control Document (PD 610-17) issued
12/2	Preliminary NASA Support Plan issued
12/15	Spacecraft System Test and Operations Preliminary Plan (TOP) (PD 610-50) available

Date completed	Milestone
12/19/69	Spacecraft System Test and Operations Preliminary Plan (TOP) (PD 610-50) Review completed
1/23/70	Overall Planetary Quarantine completed
2/2	Maneuver Programs–Trim Program completed
2/2	Interplanetary Orbit Determination Accuracy completed
2/12	Mission Plans for Proof Test Model (PTM) Tests completed
2/13	Program Support Plan issued
2/20	DSN Operations Plan–DSN System Description (PD 610-76, Volume III) issued
2/25	Final Support Instrumentation Requirements Document (SIRD) (PD 610-41) issued (signed by JPL)
2/27	Maneuver Operations Program Request for Program issued
3/7	Launch Vehicle–Atlas Centaur 23, Centaur Guidance System delivered to General Dynamics, San Diego
3/11	Interplanetary Trajectory Characteristics Design and Document (PD 610-92) completed
3/15	Launch Vehicle–Atlas Centaur 23, Centaur RL 10 engine delivered to General Dynamics, San Diego
4/14	Maneuver Programs–Insertion Program completed
4/14	Maneuver Programs–Statistical Linking Program completed
4/29	Planetary Quarantine Plan (PD 610-18, Part 1) issued
4/30	Orbit Design and Characteristics Handbook (PD 610-113) issued
5/6	Launch Vehicle–Atlas Centaur 23, Centaur major weld completed
5/11	Structure test model delivered to General Dynamics, San Diego
5/14	Launch Vehicle–Atlas Centaur 23, Centaur and Structure Test Model Matchmate Test completed
5/15	DSN Operations Plan–DSN Operations Support (PD 610-70, Volume I) issued
5/15	MOS Implementation Plan issued (transit)
6/1	MOS Implementation Plan issued (orbit)
6/1	Spacecraft Assembly Facility (SAF) Test Directives issued
6/5	Spacecraft System Test and Operations Plan (TOP) (PD 610-50) issued

Date completed	Milestone
6/8/70	MM71-3 spacecraft subsystem rework and fabrication completed
6/10	Spacecraft System Test Procedures completed
6/17	Spacecraft–MOS Interface Control Document (PD 610-13) issued
6/19	Launch Vehicle–Atlas Centaur 24, Centaur major weld completed
7/1	Start MM71-3 Spacecraft–MOS Compatibility Test
7/6	MM71-3 spacecraft available for Spacecraft–MOS–TDS Compatibility Test
7/9	Space Flight Operations Plan–Policy and Requirements (PD 610-29, Volume I) issued
7/10	MM71-3 Spacecraft Subsystem Type Approval (TA) and Flight Acceptance (FA) Test completed
7/14	MM71-3 spacecraft subsystem delivered to SAF
7/15	DSN Test Plan (PD 610-77, Volume I, Part B) issued
7/15	DSN Test Plan–DSN Test Procedures (PD 610-78, Volume II) issued
7/17	MM71-3 Spacecraft System Test Analysis and Review completed
7/20	Microbiological Assay and Monitoring Plan (PD 610-18, Part III) issued
7/24	Launch Vehicle–Atlas Centaur 24, Guidance System delivered to General Dynamics, San Diego
8/1	TDS Operations Plan–Near-Earth Expected Coverage Capabilities (PD 610-133, Volume IV) issued
8/5	Launch Constraints Document (PD 610-40) issued
8/15	DSN Operations Plan–DSN/MOS Interface Control Document (PD 610-75, Volume IIB) issued
8/17	Preliminary Orbit Determination Strategy and Accuracy Document (PD 610-33) issued
8/21	Launch Vehicle–Atlas Centaur 23, Centaur final assembly completed
8/26	Final NASA Support Plan (NSP) issued (signed by JPL)
8/31	Spacecraft Contamination Control Plan (PD 610-123) issued
9/1	TDS Operations Plan–Near-Earth Phase (PD 610-137, Volume III) issued
9/9	Telecommunications Link Performance Document (PD 610-57) issued

Date completed	Milestone
9/11/70	DSN Simulation System Readiness completed (transit)
9/14	MM71-3 Spacecraft Functional and Environmental Tests completed
9/14	MM71-3 Spacecraft–MOS–TDS Compatibility Test completed
9/15	Post MM71-3 Spacecraft Test and Design Verification Review completed
9/15	DSN Test Plan—Ground Communications Facility Test Procedures (PD 610-85, Volume IV) issued
9/30	PAD Safety Report (PD 610-136) issued
9/30	MM71-1 spacecraft subsystems rework and fabrication completed
10/1	MOS Test Plan (PD 610-46) issued
10/1	Targeting Specification (PD 610-49) issued
10/1	Mission Profile Studies Through Orbit Inspection completed (transit)
10/5	Flight adapters (3) delivered to General Dynamics, San Diego
10/7	Launch Vehicle—Atlas Centaur 23, Flight Adapter Compatibility Test completed
10/8	Launch Vehicle—Atlas Centaur 24, Centaur final assembly completed
10/19	MM71-2 spacecraft subsystems fabrication completed
10/22	Launch Vehicle—Atlas Centaur 23, Atlas acceptance completed
10/27	MOS hardware and software delivered and performance demonstration completed for integration (transit)
10/29	Launch Vehicle—Atlas Centaur 24, Flight Adapter Compatibility Test completed
10/29	MM71-1 Spacecraft Subsystem Flight Acceptance (FA) Test completed
10/29	MM71-1 spacecraft delivered to SAF
10/30	Flight adapters delivered to JPL from General Dynamics, San Diego
10/30	DSN Test/Training Plan—Deep Space Instrumentation Facility (DSIF) Engineering Test Procedures (Volume VI) issued
11/1	MOS Implementation Plan (PD 610-44) issued (transit)
11/13	MM71-1 spacecraft system test and analysis completed
11/13	Launch Vehicle—Atlas Centaur 23, Centaur checkout and acceptance completed
11/13	Preliminary Maneuver Analysis Document (PD 610-34, Part 1) issued

Date completed	Milestone
11/16/70	Spacecraft—Mission Operations Complex Compatibility Test Plan (PD 610-48) issued
11/17	DSN Test Plan—Space Flight Operations Facility (SFOF) Test Procedures (PD 610-86, Volume V) issued
11/30	Launch Vehicle—Atlas Centaur 24, RL 10 engine delivered to General Dynamics, San Diego
12/1	Launch Vehicle—Atlas Centaur 23 Combined System Test completed and prepared to ship to AFETR
12/3	MM71-2 spacecraft system test and analysis completed
12/7	Launch Vehicle—Atlas Centaur 23 delivered to AFETR
12/14	MM71-1 spacecraft available for Spacecraft–MOS–TDS Compatibility Test
12/15	MM71-2 spacecraft available for Spacecraft–MOS–TDS Compatibility Test
12/15	MOS Hardware and Software Development and Performance Demonstration completed for flight certification (transit)
12/15	Start MM17-1 Spacecraft–MOS Compatibility Test
12/15	Space Flight Operations Plan (SFOP) (PD 610-29, Volume III) issued (1st issue)
12/15	Start MM71-1 Spacecraft–MOS Compatibility Test
12/17	Spacecraft Subsystem Reviews completed (prior to Preshipment Review)
12/18	Spacecraft System Review completed (prior to Preshipment Review)
12/18	TDS Near-Earth Phase Operations Plan–Interface Description (PD 610-138, Volume II) issued
12/18	Launch Vehicle—Atlas Centaur 24, Atlas acceptance completed
12/19	MM71-2 spacecraft subsystems delivered to SAF
12/19	MM71-2 Spacecraft Subsystems Flight Acceptance Test completed
12/31	MM71-1 Spacecraft System Functional and Environmental Test completed
12/31	MOS Hardware and Software Development and Performance Demonstration completed for integration (nominal orbit)
1/19/71	Spacecraft Subsystem Type Approval Test completed
1/20	MM71-3 spacecraft reassembly completed

Date completed	Milestone
1/22/71	AFETR Test Directives issued
1/25	Spacecraft support equipment sets 1 and 2 delivered to launch complex
1/25	Space Flight Operations Facility Mission Support Area available for beneficial occupancy
1/27	MM71-2 Spacecraft Functional and Environmental Test completed
1/29	TDS Hardware and Software Development and Performance Demonstration completed (model 1)
2/1	Space Flight Operations Plan–Sequence of Events (PD 610-29, Volume IIA) issued (1st issue)
2/1	Launch Vehicle–Atlas Centaur 24, Centaur checkout and acceptance completed
2/5	Launch Vehicle–Atlas Centaur 24, Combined System Test completed and prepared to ship
2/8	Launch Complex Checkout completed, AFETR PAD 36B (spacecraft peculiar)
2/8	Launch Operations Plan (PD 610-37) issued
2/8	Spacecraft Hardware Reviews completed
2/9	MM71-3 Spacecraft informal Preshipment Hardware Review completed
2/12	Launch Vehicle–Atlas Centaur 24, launch vehicle 2 delivered to AFETR
2/13	MM71-1 Spacecraft–MOS–TDS Compatibility Test completed
2/15	MOS Hardware and Software Development and Performance Demonstration completed for flight certification
2/15	MOS and TDS System Test completed for DSN (model 1)
2/17	MM71-3 spacecraft delivered to AFETR
2/18	MM-71-1, MM71-2, MM71-3 Spacecraft Preshipment Hardware Review completed
2/23	MOS and TDS Integration, System Test and Performance Demonstration completed for near-Earth (model 1)
2/28	MM71-1 spacecraft delivered to AFETR
3/1	MM71-2 Spacecraft–MOS–TDS Compatibility Test completed
3/1	TDS Hardware and Software Development and Performance Demonstration completed (model 2)
3/10	Launch Vehicle–Atlas Centaur 23, MM71-3 Spacecraft/Launch Complex Compatibility Test, PAD 36B, completed
3/11	Launch Vehicle–Atlas Centaur 23, Terminal Countdown with MM71-3 spacecraft, PAD 36B, completed
3/14/71	MM71-2 spacecraft delivered to AFETR
3/15	MOS and TDS integration completed (model 2)
3/18	Launch Vehicle–Atlas Centaur 23, Joint Flight Acceptance Test (J-FACT) with MM71-3 spacecraft, PAD 36B, completed
3/19	MOS and TDS Launch Preparations Review completed
3/26	Prelaunch Analysis of Probability of Planetary Contamination (PD 610-18, Part II) issued
4/1	MOS and TDS System Test completed (model 2)
4/1	MOS Implementation Plan, Phase B (PD 610-44, Volumes 1–5) issued
4/1	Launch Complex Checkout completed, AFETR PAD 36A (spacecraft peculiar)
4/6	Launch Vehicle–Atlas Centaur 24, Spacecraft–Launch Complex Compatibility Test completed
4/7	Mission Profile Studies Through Orbit Insertion (Preliminary Orbit) completed
4/7	Firing Tables Document issued
4/8	Launch Vehicle–Atlas Centaur 24, Terminal Countdown Demonstration with MM71-3 spacecraft, PAD 36A, completed
4/13	Launch Vehicle–Atlas Centaur 24, Joint Flight Acceptance Test (J-FACT) with MM71-3 spacecraft, PAD 36A, completed
4/14	MM71-3 spacecraft demated and decapsulated
4/21	TDS Near-Earth Readiness Review completed
4/23	Launch Readiness Review completed at AFETR
4/23	Launch Readiness completed
4/23	Midcourse Maneuver Policy (PD 610-34, Part II) issued as an interoffice memorandum
4/27	MOS Operations Training and Readiness completed (transit)
4/27	MOS and DSN Launch Readiness Review completed
4/30	Mission Plan (PD 610-16, Part II) issued (transit)
4/30	TDS Near-Earth Phase Operations Plan–Launch Operations (PD 610-182, Volume I, Part A) issued
4/30	TDS Near-Earth Phase Operations Plan–Expected Coverage Capabilities (PD 610-133, Volume IV, Revision A) issued
5/3	Space Flight Operations Plan–Operations Procedures (PD 610-29, Volume III) issued

Date completed	Milestone
5/3/71	MM71-1 spacecraft available for mate and encapsulation
5/3	Final mate with Mariner H (MM71-1 spacecraft, Atlas Centaur 24), PAD 36A, completed
5/4	Joint Flight Acceptance Composite Test (J-FACT)/Electromagnetic Interference Test (EMI), PAD 36A, completed
5/5	Space Flight Operations Plan–Sequence of Events (PD 610-29, Volume IIB, modification for Misison A) issued
5/5	Mariner H Radio Frequency Interference (RFI) Test, PAD 36A, completed
5/7	MM71-2 spacecraft available for mate and encapsulation
5/8	Mariner H launch[a]

[a]Mariner H was designated Mariner 8 even though the Centaur failed to inject it into the trajectory to Mars. Mariner I was designated Mariner 9 after launch.

Date completed	Milestone
5/11/71	Final mate with Mariner I (MM71-2 spacecraft-Atlas Centaur 23,) PAD 36B, completed
5/13	Electromagnetic Interference (EMI) Test with Mariner I, PAD 36B, completed
5/15	DSN Simulation System Readiness completed
5/18	Launch Vehicle–Atlas Centaur 24, Launch Failure Review completed at General Dynamics, San Diego
5/23	Mariner I Joint Flight Acceptance Composite Test (J-FACT)/EMI, PAD 36B, completed
5/24	Mariner I Composite Readiness Test completed
5/25	Mariner I RFI and Launch Vehicle EMI Test completed
5/26	Mariner I Launch Readiness Review completed
5/30	Mariner I Launch[a]
6/4	Mariner 9 trajectory correction maneuver completed

Appendix B

Mariner Mars 1971 Problem List (P-List)

Number	Problem description	Assigned to	Date assigned	Required closure date	Date closed	Status
P1	Delivery of the radio subsystem is dangerously late due to: (1) Corona problem–fix needs to be qualified (pacing problem). (2) TWT delivery and changeout.	R. Stevens	11/13/70	2/15/71	2/24/71	IOM 3300-71-097, Stevens/Hunter to Schneiderman, dated 2/22/71.
P2	Confidence in the propulsion system is reduced due to a series of check valve failures, some of which may be contamination in the valves. In at least one case, contamination is not the cause.	R. Rose	11/13/70	2/16/71	2/12/71	IOM MM71-71-086, Rose to Schneiderman/ Forney, dated 2/9/71.
P3	The CC&S operation is questionable due to the following unexplained problems: (1) Motor burn variation times. (2) Missing commands. (3) Five-second timing error. Schedule delays are in prospect if the causes of these problems are not understood in the near future.	J. Scull	11/13/70	2/16/71	2/12/71	IOM 3611-71-29, Scull to Schneiderman, dated 1/27/71.
P4	Confidence in the IRIS ability to operate reliably and to be delivered on schedule is lacking.	J. Small/ R. Hanel	11/13/70	2/16/71	3/17/71	IOM, Small to Schneiderman, Subject: "MM71 P-List," dated 3/9/71.
P5	Failure of MOS/TDS simulation system to meet readiness dates for support of testing and training. Specific current problems include 6050 upper memory failure and 6050/1108 interface delays.	G. Lairmore	12/8/70		12/22/70	IOM 295-70-393, Rodriguez to Rygh.
P6	360/75 launch capability schedule does not allow sufficient training for reasonable risk launch support.	G. Lairmore	12/16/70	4/1/71	4/2/71	IOM GEL-71-86, Lairmore to Schneiderman dated, 3/31/71.
P7	360/75 schedule does not allow time to incorporate into orbital operation the capabilities required to meet the mission objectives, particularly in the generation of data records.	G. Lairmore	12/16/70	7/1/71	2/1/71	IOM GEL-71-31, Lairmore to Schneiderman, dated 1/29/71.
P8	DSS tape degradation causes some loss of data (actually, an alignment problem damaged tape).	J. Scull	1/4/71	2/16/71	2/12/71	IOM 3634-71-028, Grumm to Schneiderman (approved by Scull), dated 2/11/71.

Number	Problem description	Assigned to	Date assigned	Required closure date	Date closed	Status
P9	IRIS electronics failure caused loss of data (Flight 2 IRIS failed in solar thermal vacuum).	J. Small/ R. Hanel	1/19/71	2/16/71	3/17/71	IOM, Small to Schneiderman, Subject: "MM71 P-List," dated 3/9/71.
P10	DSS drive belt breakage (PTM DSS).	J. Scull	1/19/71	2/16/71	4/16/71	IOM, 3634-71-083, Grumm/Scull to Schneiderman, dated 4/12/71.
P11	CC&S relay failure–mechanical mounting suspect (Flight 1 CC&S).	J. Scull	1/19/71	2/16/71	2/12/71	Relay mounting changed in all units–all units have successfully passed environmental requalification tests and have been delivered to SAF.
P12	MM71-2 radio circulator switch anomaly repair, retest, and delivery to Cape is critical.	R. Stevens	2/24/71	3/15/71	3/24/71	MM71-2 switch retest and delivery accomplished. IOM 33M1-71-014, 3300-71-171, from Stevens/Hunter to Schneiderman, dated 3/23/71.
P13	Insufficient reliability of DSIF Command System resulting in inability to complete command sequences.	R. Stevens	4/16/71	4/30/71	5/7/71	IOM 3300-71-270, Stevens to Schneiderman, dated 5/6/71.
P14	Radio exciter power output decrease not understood; model needed which duplicates spacecraft and ground test data.	R. Stevens	8/16/71	10/8/71	10/28/71	IOM 33M1-71-033, 330-71-572 from Stevens/Hunter to Schneiderman, dated 10/8/71.
P15	CC&S Assembler/Compiler Program has not been completed. This program is a serial program in the Adaptive Mode Planning Set (AMPS) and is delaying readiness for orbital operations.	J. Scull	9/28/71	10/10/71	10/25/71	IOM 360-71-104 from Scull to Schneiderman, dated 10/22/71.
P16	The on-site telemetry and command processor (TCP) currently has faults in the command system which could cause loss of the flight spacecraft. A version of the TCP which corrects these faults has not been delivered.	R. Stevens	9/28/71	10/11/71	11/19/71	IOM 3300-71-623 from Stevens to Schneiderman, dated 11/10/71.
P17	Radio receiver appears to have taken a static offset of best lock frequency; evaluation of cause and projection of effect needed.	R. Stevens	11/19/71	11/26/71	1/13/72	IOM 33M1-72-001 from Stevens/Hunter to Schneiderman, dated 1/3/72.

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