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June, July, and August 1965

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report No. 32-782

Mariner IV Mission to Mars
(Part I)

Jack N. James et al

These papers present the results of one phase of research carried out at the Jet Propulsion Laboratory, California Institute of Technology, under Contract No. NAS 7-100, sponsored by the National Aeronautics and Space Administration.

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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INTRODUCTION

By Dan Schneiderman
Project Manager, *Mariner-Mars* Mission

This technical report is a series of individual papers documenting the *Mariner-Mars* project from its beginning in 1962 following the successful *Mariner-Venus* mission. Part I is *pre-encounter data.* It includes papers on the design, development, and testing of *Mariner IV*, as well as papers detailing methods of maintaining communication with and obtaining data from the spacecraft during flight, and expected results during encounter with Mars. Part II, *post-encounter data,* to be published later, will consist of documentation of the events taking place during *Mariner IV's* encounter with Mars and thereafter. The *Mariner-Mars* mission, the culmination of an era of spacecraft development, has contributed much new technology to be used in future projects.

Managing the Mariner Mars Project

By JACK N. JAMES
Acting Assistant Laboratory Director, Lunar and Planetary Projects, JPL

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The rules of order imposed in this development brought the best out of everyone who supported the project, while keeping hard and fast commitments for a complex spacecraft mission

Management can easily become an impersonal tool. By extending the practices of Ranger and the Mariner Venus 1962 projects to the Mariner Mars 1964 Project (MM64), we think it cut through the tough problems that confront any major engineering development.

To apply their experience, many individuals stepped directly into MM64 upon completion of the 11-month, crash Mariner Venus effort.

This article recounts important management features of MM64. It is not presumed that they constitute a panacea for other projects; and if they seem obvious, then perhaps that is what made them significant.

Projects are not started from scratch. They generally are preceded by a variety of important study and proposal efforts that probe related and frequently more-ambitious efforts. Such was the case for MM64. In the summer of 1962, about the time of launching Mariner 2 to Venus, it became evident to NASA and JPL that a more ambitious Centaur-based launch to Mars in the fall of 1964 would not be possible, as the Centaur launch vehicle would not be ready. Studies were initiated by JPL and other NASA centers to determine what kind of mission might be achieved based on the operational Atlas-Agena launch vehicle. In November 1962, NASA approved a mission-concept proposal by the JPL and authorized the MM64 Project. NASA authorized two launches owing to the risk associated with the mission and the safety provided by vehicle redundancy.

With the approval of the Project, five major actions were taken concurrently:

1. Ordering Long-Lead-Time Parts. Since the brief launch opportunity to Mars was to occur two years to the month after Project approval, an immediate survey of certain subsystems was necessary to initiate procurement for long-lead-time parts. This was possible, even though the system design had not been completed, because the MM64 effort was, in most cases, an extrapolation from the technological base established with Mariner 2 and its predecessor design, Ranger.

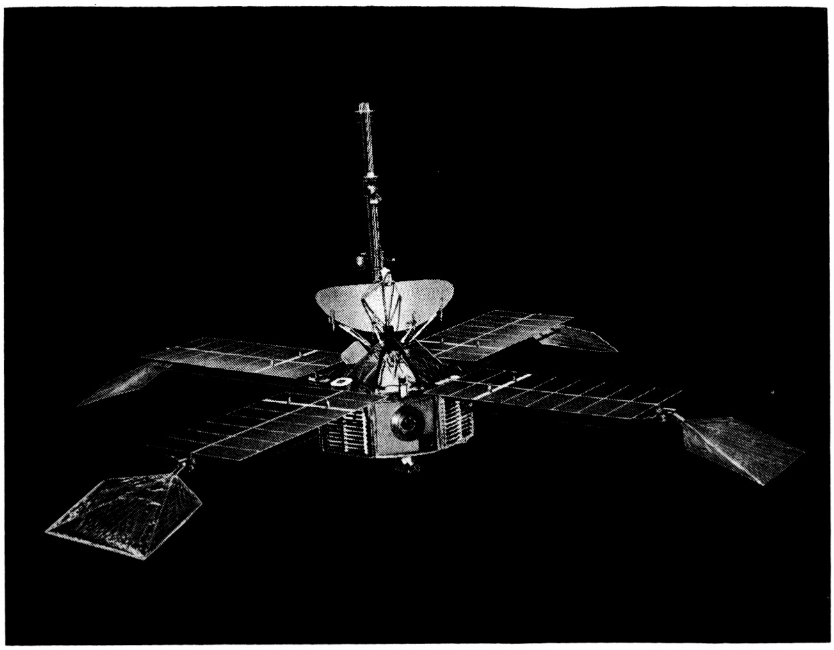
2. System Design. The conceptual and preliminary spacecraft design work that had been underway sufficiently demonstrated feasibility, but it was hardly an adequate reference from which to initiate hard subsystem design; so the study was reformed into a Spacecraft System design effort. A system-design team was formed, composed of some 30 of the best engineers available, representing all of the JPL discipline activities contributing to the design.

The output of this effort was to be the Spacecraft Design Specifications (SDS) Book, the governing document for the spacecraft throughout the life of the Project. The team was given three months, until February 1963, to produce it.

The book, in three parts, represents control by three different levels in the Project's organizational structure.

Part I, "Mission Objectives and Design Criteria," very brief, is prepared and controlled by the Project Manager.

Part II, "Spacecraft Design Characteristics and Restraints," is prepared by the Spacecraft System Manager and his Project Engineer and controlled



MARINER 4
IN CRUISE CONFIGURATION

by the Spacecraft System Manager.

Part III, "Spacecraft Function Specifications," contains some 40 subsystem and spacecraft-related specifications prepared by the JPL Technical Divisions and controlled by the Project Engineer.

Every effort on a project can influence its success or failure, including the specific risk at launch; but the importance of the system-design team's judgment cannot be overemphasized. At JPL there is an attempt to employ on this team many individuals who will have a continuing responsibility throughout the life of the Project. They must follow through on, and will be held accountable for, their early judgments.

3. Establishing the Launch Period. For a planetary flight, the launch period is one of the first efforts in mission design that the Project Manager must clearly specify.

As is common knowledge, Mars comes within shooting range of Earth every 25 months. The launch period depends on the geocentric energy (C_3) required to reach Mars each day of that particular calendar year, the spacecraft injected weight, and the ability of the launch vehicle to inject the spacecraft to the required velocity.

Each of these three areas was examined quickly by mission analysts; the project leaders had to take a firm stand on spacecraft weight and launch-vehicle requirements very early, based on judgment as to what would be achievable in the time available with a balancing of risks.

Analysis of launch-attempt statistics indicated the launch period should be 18 days at the minimum and 44 days the desirable. Less than 44 days required preparing two launch pads at Cape Kennedy. The 44 days represented a hedge against a prior, non-project abort eliminating one of the two pads. The two project launches could then still be conducted sequentially off the remaining pad. A long launch period also gave time to correct any failure cropping up in the first launch.

In view of this, two pads were established at the outset in the Program Requirements Document, the official means for placing requirements on the Air Force Eastern Test Range (AFETR) at Cape Kennedy.

Energy for a launch to Mars in 1964 did not differ much from the 1962 Venus flight. But the spacecraft weight had to increase by 131 lb. The Atlas/Agena launch vehicle therefore had to be further modified over its 1962 performance for the mission. (It had already been uprated to meet the 1962 Venus requirements.) A "shopping list" of desirable performance changes—some were already being considered—was developed for the launch vehicle, and an estimate was made of the probability of each change being flight-qualified in time.

Spacecraft weight specified by the Project Manager in Part I of the SDS Book was 570 lb. The actual Mariner 4 weight injected was 575.37 lb. The launch vehicle also met its minimum C_3 of 10.2 kms^2/sec^2.

Weight and performance control engineers exchanged published estimates on a monthly basis. Progress was tracked, and weight-saving drives instituted from time to time.

To achieve this spacecraft weight, designers had to take extreme measures. For example, the panels to which the solar cells were bonded were 0.003 in. thick. Indeed, the spacecraft structure proved so fragile that a special training film was prepared for all personnel involved in its handling. Mariner 2 and MM64 compare in basic piece-parts, such as fasteners, transistors, and so forth, as follows:

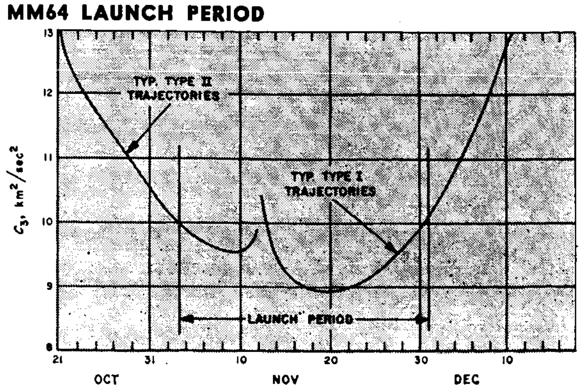
Spacecraft	Piece-part count	Injected wt, lb	Parts/lb
Mariner 2 (Venus)	54,000	444	121
Mariner 4 (Mars)	138,000	575	240

The graph at bottom shows the actual launch period achieved employing both trajectories that extend less than halfway around the Sun (Type I) and trajectories extending greater than one-half way (Type II). This launch period amounted to 28 days. A project of any kind may well slip one month after being underway for two years. A slip of one month in the MM64 Project would have meant failure. One of the functions of project management was to keep schedule discipline.

A specially assigned engineer maintained the planning document on launch constraints as an advisory function to the Project. He collected, studied, and summarized all design or policy factors that constrained in any way the launch period and the launch window on each day of the period. His advisory function extended right through the critical launch countdowns, advising the Project Manager of the impact of such factors as weather and tracking-station outage on the launch window.

4. The Project Development Plan. Shortly after its formation by the Space Act of 1958, NASA recognized that it needed a mechanism for marshalling the diversified talents of the NASA centers to meet a project objective. For this reason it drafted and issued NASA Management Instruction (NMI) 4-1-1. A project must draw upon, and force into a complementary shape, many ongoing programs and R&D efforts—many of which tend towards standardization. The Project Manager needs the edge of authority to force the pieces of a project to complement each other; for when challenging mission objectives have been set forth, you can be sure the pieces won't fit. The "pieces" of the Project are identified by NMI 4-1-1 to be "systems." Each system was to have a

MM64 LAUNCH PERIOD



System Manager, with similar responsibility.

The authority the Project Manager needed was established in NMI 4-1-1 by requiring that System Managers comply with his decision; if they objected, to appeal—but meanwhile to comply. Thus NASA recognized that such authority was the "grease" necessary to cause projects to move ahead. NMI 4-1-1 calls for a Project Development Plan (PDP) to be prepared, negotiated, and signed first by the Project Manager and then by the System Managers and authorities at NASA Headquarters. The PDP outlines the mission objectives, conceptual design, cost and manpower estimates, and special requirements, and it divides the job into systems and states which system performs which functions. A clear assigning of the tasks among systems is necessary at the outset of a project to save time. The chart on page 37 summarizes the system assignments for MM64.

As shown, Project Management and three of the four systems were assigned to JPL. The launch-vehicle system was assigned to Lewis Research Center. A major interface exists between the Project as a whole and the Air Force Eastern Test Range. Additionally, the Project Manager reported into NASA Headquarters through the NASA Program Manager assigned to MM64. The NASA Program Management function was also established by NMI 4-1-1. For Mariner, it provided an excellent management environment.

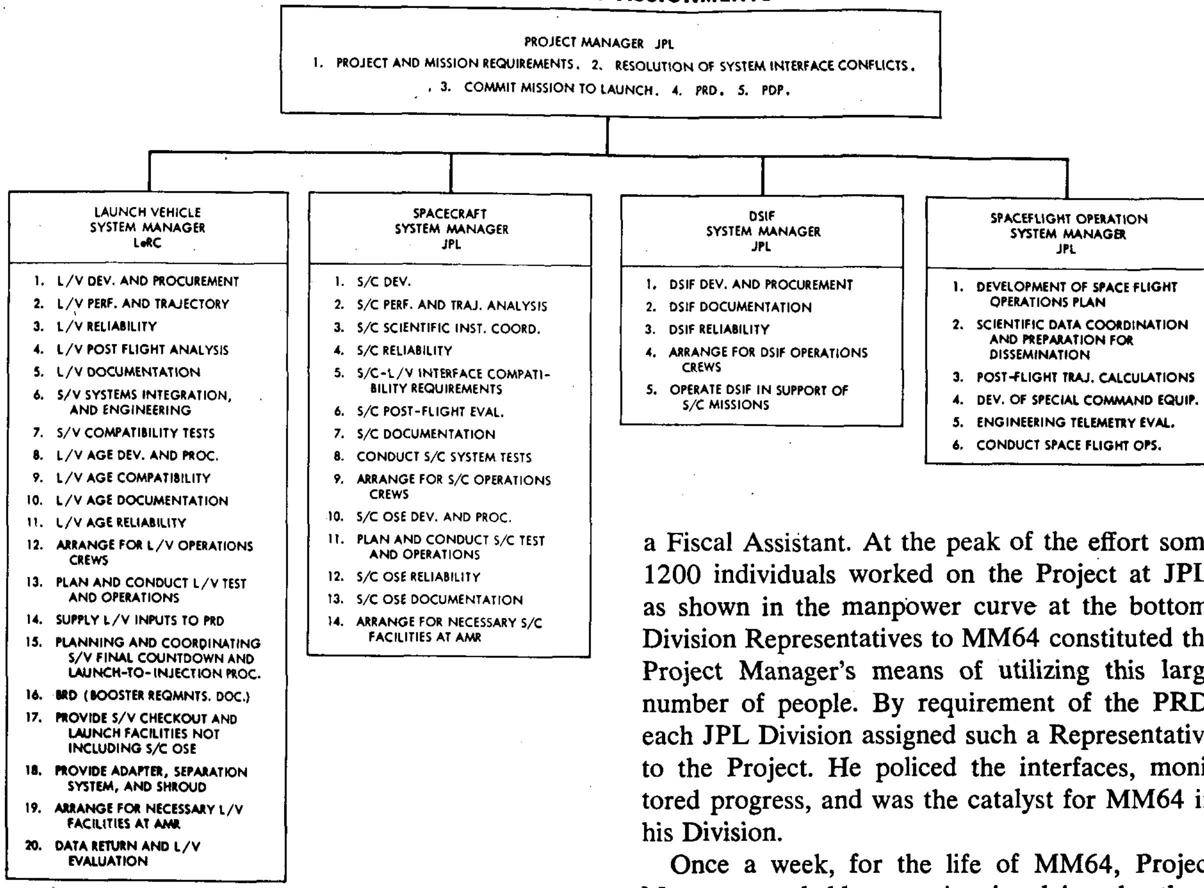
5. Policy and Requirements Document (PRD). The MM64 Project exploited the talents of the JPL discipline divisions to carry out the Project Management, Spacecraft, Space Flight Operations (SFO), and Deep Space Instrumentation Facility (DSIF) system assignments.

The Project engaged the JPL Divisions in a matrix organization, and the effort was conducted in an "in-house" fashion. That is, the Project levied its policies and requirements on the JPL Technical Divisions, what it wanted and by when. The Divisions engaged subcontractors who best complemented their talents. This approach involved a variety of subcontractors having special capabilities; and indeed, 75% of the funds authorized to JPL went to these subcontractors.

The means by which the policies and requirements of the Project were levied upon the JPL Divisions was the PRD—a management document authored by the Project Manager and two assistants. It established the obligations of each Division and the means of organizational interfacing with the Project.

This PRD, the keystone for the Project throughout its life, stressed clarity and simplicity to minimize misunderstanding as to what was wanted. Sixty pages long, it described the number of equipments to be built, assembled and tested, configuration control, quality control, parts control, test

MM64 SYSTEM ASSIGNMENTS



requirements, key planning and control documents required, procurement requirements, organization, and schedule control. The document covered virtually every aspect of the Project and its every sentence packed meaning. Yet it was intentionally held to a relatively few pages so that everyone working on the Project could easily become thoroughly familiar with it, and this they were enjoined to do.

The more complicated the Project, the more complicated the communication problems among people–and thus the more strenuous an effort management should make to simplify the guideline requirements, to ensure that an informed team moves towards the same goal and to permit an easier measuring of achievements. In MM64 it was necessary time and time again to *reconfirm* the basic guidelines and requirements to bring about their understanding. And it was made very hard to modify the basic requirements.

Once these five actions were carried out—ordering long-lead-time parts, initiating system design, establishing the launch period, making the PDP, and enacting the PRD—the Project was moving ahead on all fronts. Let's look now at some of the subsequent actions.

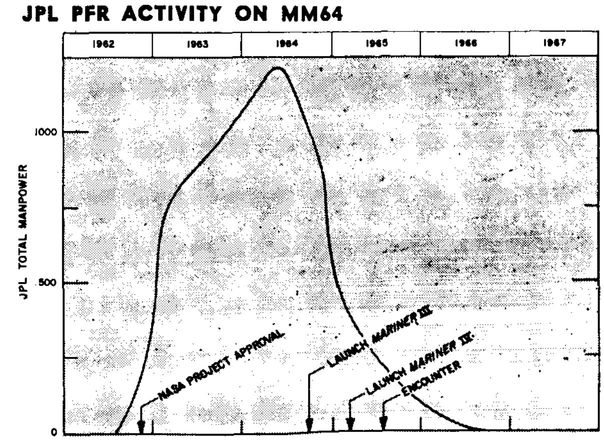
Organization and JPL Meetings. Only three individuals reported administratively to the Project Manager—two Assistant Project Managers and a Fiscal Assistant. At the peak of the effort some 1200 individuals worked on the Project at JPL, as shown in the manpower curve at the bottom. Division Representatives to MM64 constituted the Project Manager's means of utilizing this large number of people. By requirement of the PRD, each JPL Division assigned such a Representative to the Project. He policed the interfaces, monitored progress, and was the catalyst for MM64 in his Division.

Once a week, for the life of MM64, Project Management held a meeting involving the three JPL System Managers or their delegates and the Division Representatives. Each Division Representative was asked to report briefly what he considered to be the most serious problems impeding progress. From these reports, action assignments were made and carried in meeting minutes until the action was completed.

Although the subject matter was not of interest to all Representatives through the entirety of the meeting, they were expected to stay to develop a broad understanding of the total project. It is believed that this was the most important means by which a MM64 team was forged.

Anticipating Crises. For certain, a project will encounter crises. One should list the "firsts" being

JPL PFR ACTIVITY ON MM64



attempted in a project to try to anticipate where the crises are apt to occur. There follows a rough listing of MM64 "firsts" prepared in March 1963:

First NASA dual launch preparation on two pads.

18-44-day launch period must be met or mission fails.

Changes to Atlas D to increase performance.

First NASA use of Agena D.

Changes to Agena D to increase performance.

First Agena D two-burn employing C/S system.

New shroud and ejection system.

New spacecraft-Agena separation system.

New spacecraft involving—

- *First use of the star Canopus for attitude stabilization.*
- *First use of S-band communication system.*
- *First use of dual midcourse maneuver capability.*
- *First mission requiring nine months of successful spacecraft operation to achieve mission success.*

To hold schedule, a project must have modest reserve resources to commit; the Project Manager must have some funds in reserve to buy his way out of emergencies. About 6% was held by Project at beginning of each fiscal year on MM64, in spite of many forces, from within and from without the Project, acting against such a reserve. Of course, the proper time for committing reserves is a matter of judgment. The Project Manager will likely need to commit reserves as a consequence of each major phase of testing, (e.g., subsystem proof testing and spacecraft system tests) and as field operations approach.

Committing reserve talent can be as important as committing funds. An organization, with a base of supporting research and advanced development and other projects underway, can swing talent from one project activity to another when crises occur.

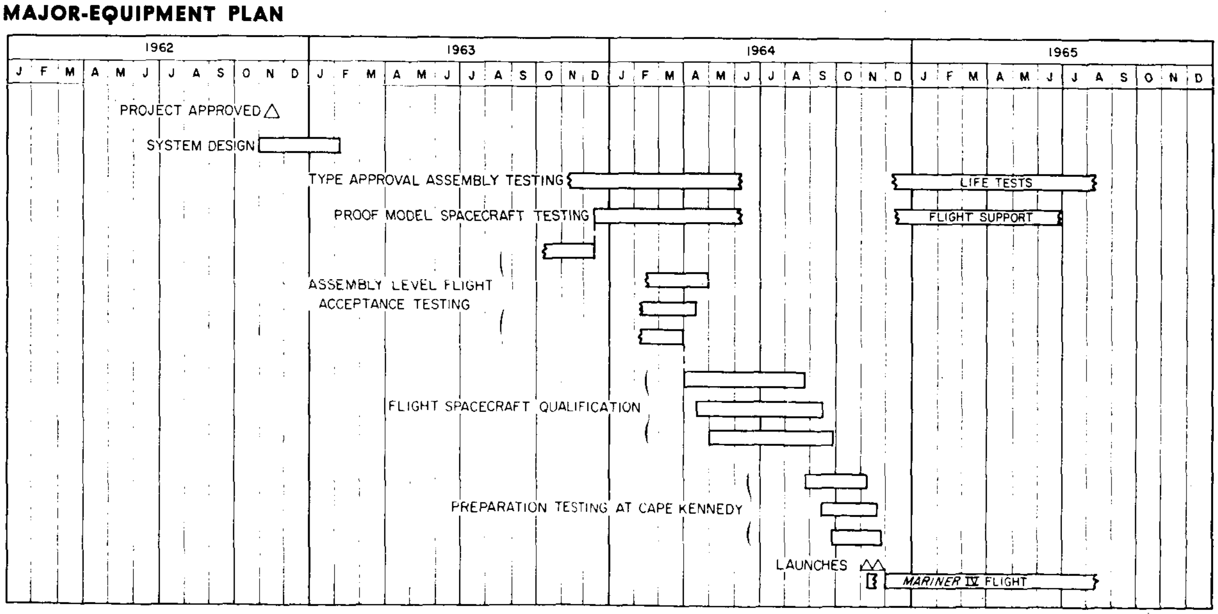
In the JPL over-all organization, divisions had many activities other than MM64, and the MM64 Division Representative held influence over only resources committed to MM64 by the Division Manager. Recognizing the limited authority of the Division Representatives, the Project instituted a series of monthly meetings with the Division Managers—the individuals who controlled all the outside talent—to inform them on the Project's progress, to have them report on their Division's contribution to it, and to solicit their advice. The long-term aim was an intimately informed Division Manager, better prepared to bring manpower to bear in a pinch.

The Division Managers came through for MM64 when the crises hit, among them these notables: (1) The early Canopus-tracker photomultiplier tubes could not pass the launch-vibration environment; (2) pellet resistors of the spacecraft's Data Automation System deteriorated under high-temperature testing; (3) the S-band power amplifier triode degraded slowly over hundreds of hours of operation; and the fiber-glass shroud ruptured in the Mariner 3 launch. Each crisis involved a "first" in the sense that the device was being flown for the first time on any mission. Each was solved by grouping together special task forces of considerable size and ability.

Schedules. Prior experience had shown that projects lose much time in the early phases, where significant milestones are more difficult to establish than in the hardware phase.

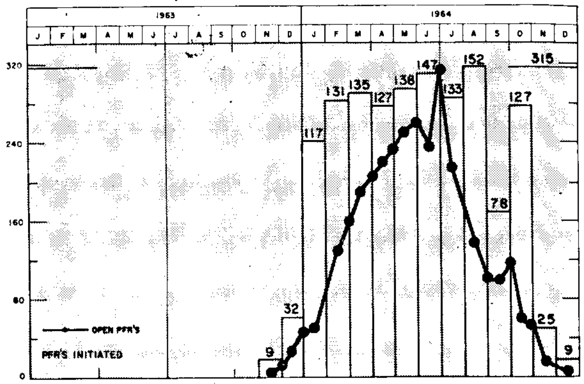
Since meeting schedules were mandatory for MM64, schedule preparation and reporting were given careful consideration. Project Management

MAJOR-EQUIPMENT PLAN

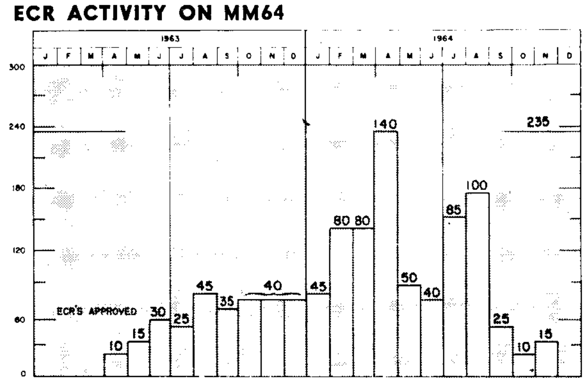


PFR ACTIVITY ON MM64

Additional activity occurred at subcontractors.



ECR ACTIVITY ON MM64



was concerned with three tiers of schedules.

With launch date fixed, the Project Manager prepared a single master control schedule with no more than about 15 line entries and milestones. This schedule entered the PRD, with a clear definition of each milestone.

Conforming to the master schedule, System Managers and Division Representatives had to prepare the following second tier of schedules in milestone format and make written and verbal reports on them periodically:

Spacecraft Development
GSE Design and Preparation
Spacecraft Assembly and Operations
Eastern Test Range (ETR) Operations
Spacecraft ETR Facilities
Launch Vehicle
Launch-Pad Preparation and Utilization
Launch Vehicle/Spacecraft/ETR Interface Documentation
Trajectory, Instrumentation, Range Support
DSIF Modification and Preparation
Spaceflight Operations Plan
Major Contracts Schedule

Once again, these were milestone schedules, put together at the system-management and staff level by the Managers estimating what could be achieved in a certain time.

The period when a spacecraft development will most likely get out of control comes between system design and delivery of flight-qualified assemblies for spacecraft assembly and test. To control this period, a third tier of schedules was prepared, more detailed, but intentionally not so detailed as to require handling by a computer. They were constrained to conform to the first- and second-tier schedules.

Taking the case of the spacecraft, these schedules resembled PERT in format and were called flow charts. A separate flow chart was assigned for preparation and reporting to a particular Division Representative for each item on the "Equipment List." This list had to be prepared as early as possible after system design by the Spacecraft System Manager. It catalogued the spacecraft assemblies, in the same configuration as they were to be carried as spares, tested environmentally, and delivered to the Spacecraft Assembly Facility. Since an assembly would typically involve modules from more than one Division, it was necessary to make an assignment to one Division Representative for each flow chart, with the other Representatives supplying inputs.

Standard milestones for all equipment were defined in the PRD for use on the flow charts; for it appears to be quite easy to misunderstand the meaning and intent of schedule milestones if they are allowed to become too numerous and to be defined by everyone involved as he sees fit.

The Division Representatives were required to submit filled-out forms on their flow-chart progress biweekly. The delinquent milestones pertinent to a particular Division were sent to the Division Managers as an "overdue bill."

Equipment Requirements. MM64 spacecraft had to operate properly for at least eight months to meet the primary mission objectives. One means of achieving such long life was believed to be a thorough parts-selection and -screening program. The PRD required that, if at all possible, "Hi-rel"-rated parts were to be used for spacecraft and that every piece-part was to be screened to a specification and tagged. No requirement was imposed for serializing and maintaining a log on parts, however, in view of the expense and time limitations. The parts-screening goal was virtually achieved for all equipment. But it was not unusual in screening to obtain a yield of only 20% of certain parts.

The chart on page 38 outlines the major equipment plan, which had three chief segments:

1. Type Approval: A complete set of spacecraft assemblies were fabricated and subjected to a series of environmental tests over ranges exceeding those expected in flight to qualify design. The time scale caused considerable overlap of this type of qualification testing with the flight-equipment acceptance testing, making it particularly difficult to work in needed changes.

An environmental-requirements group performed staff support to the Project in preparing a general environmental specification. All Divisions were then required to prepare test specifications in conformance with the governing specification for the Type Approval and flight-acceptance testing. Deviations were approved by the Spacecraft System Manager.

The Type Approval set of assemblies was put on a life-test regimen in a thermal-vacuum environment to give a comparison with Mariner 4 flight experience.

2. Proof Test Model: A complete spacecraft was constructed and subjected to an extensive test program as a forerunner of the actual flight version. Its assemblies were flight-qualified. Once assembled, this spacecraft accumulated approximately 1000 hr of testing before the launchings, as compared with 6000 hr required in the real mission.

3. Flight Spacecraft. Three sets of equipment were flight-qualified at the "assembly flight acceptance level" before being introduced into spacecraft. These tests were the same as Type Approval except the levels were reduced to be more consistent with nominal flight conditions.

Three spacecraft were prepared to cover the loss of at least one by accidental damage, as well as a means of completely qualifying a set of spares. Each of the three spacecraft had an average of 800 operating hours at the time of launch and had been put through reduced-level tests, which had been checked out on the proof-test model.

Even this heavy testing schedule produced only a fraction of the mission in terms of hours of operation.

Problem Failure Reports (PFRs). A strong tool in the MM64 Project, PFRs had to be written on any anomaly whatsoever associated with spacecraft, beginning with the testing of the first modules for ultimate use in type approval, proof-test model, or flight spacecraft. Preparation of PFRs continued into the space-flight phase. A PFR had to be written as soon as possible after observing an anomaly and before attempting to assess the cause.

In the case of spacecraft-associated PFRs, the Spacecraft Project Engineer would then assign the analysis of a PFR to the appropriate cognizant Division. It in turn was to take some corrective action to prevent reoccurrence of the problem. The analysis and corrective action were given an independent review by Project designees. Not a new technique—but the closeout and independent review of all PFRs on MM64 was vigorously enforced. The graph on page 39 shows the action.

The Phased Freeze. Determining when to permit changes and when to deny them, and by what means, is difficult on any project. If an item was "frozen" on the MM64 Project, it simply meant that changes were discouraged, and might be approved after an Engineering Change Request (ECR) and review by the Spacecraft System Management. Appeals could be made to the Project Management should the change not be approved by Spacecraft System Management. No instance of reversal of the Spacecraft System Manager's decision is recalled, however.

To effect control, a "freeze" list was issued at the conclusion of "system design" on all the functional specifications in the SDS. This meant there would be freedom on the part of the Division cognizant engineers to change their designs as long as functional specifications were not violated.

Progressively, other controlling documents were placed on the freeze list, such as the spacecraft-envelope-control drawings and circuit-interface drawings. In general, interfaces were resolved and controlled as rapidly as possible until December 1963, at the time of the delivery of the Proof Test Model equipment for assembly, when a total freeze was imposed. After this date, it was considered that any change would have a significant impact on the rest of the system, and only changes regarded as mandatory for mission success were permitted. Still they were considerable, as the second graph on page 39 shows.

Motivation. The MM64 Project had a natural appeal; it would be the first U.S. attempt to reach Mars. Yet knowledge of the mission and its challenges, such as meeting schedules, had to be brought to the attention of everyone possible who could, by his actions, influence the success or failure of the mission. It was a policy on the Project up until launch that, if any speeches were made regarding the mission, they were to be made to those who could help it succeed. Launch-vehicle contractors, spacecraft contractors, DSIF tracking stations, and the Eastern Test Range were visited to the extent possible and briefings given on the mission.

The P-List. The P-List (P: Problem, or Panic, as the case might be) was employed on the Mariner Venus Project in 1962 and was instituted anew on MM64 in the spring of 1964.

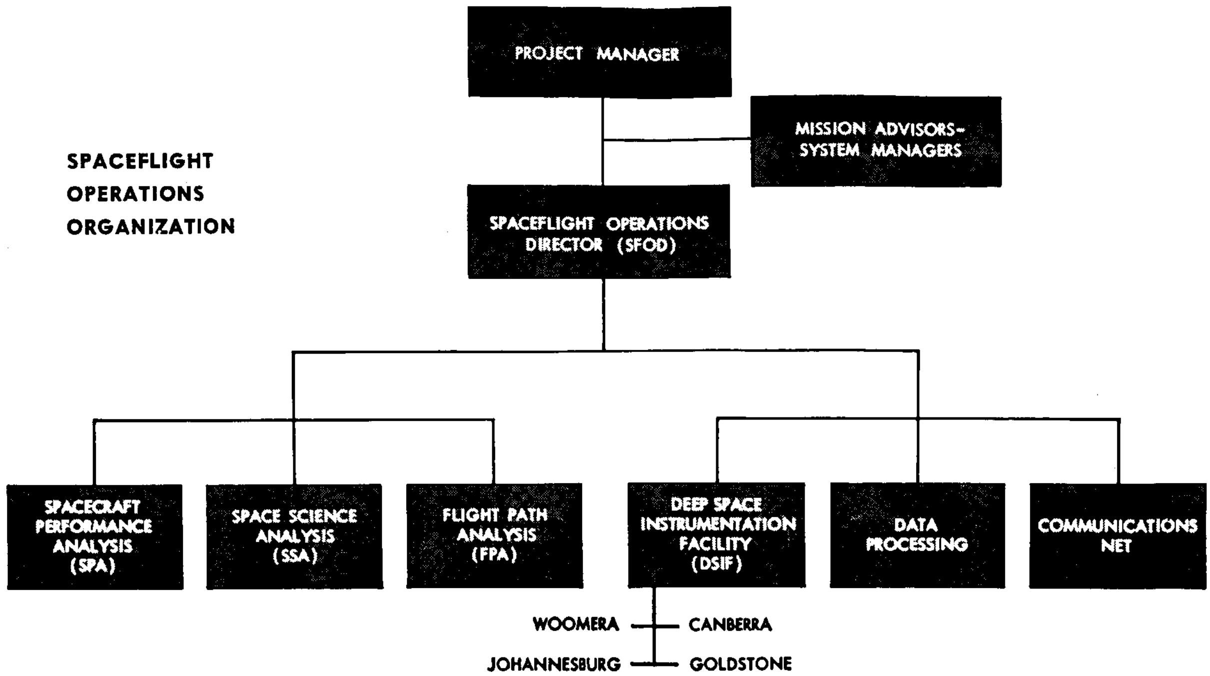
This list was the Project Manager's personal bulletin identifying major problems thought to jeopardize the mission. By agreement with the Technical Divisions, once an item was identified on this list, the most effective people available were to go on it.

To preserve its emphasis, the list was intentionally not issued until the time for spacecraft delivery to ETR was only about five months away. The list could duplicate in some instances a PFR, but it was broader in scope.

Judiciously used and issued rapidly, the P-List can be quite an effective tool.

Reliability Studies. Reliability analyses were made both within JPL and on contract.[1,2] Although coming after MM64 was somewhat defined,

SPACEFLIGHT OPERATIONS ORGANIZATION



and having modest influence on the actual design, these studies exerted a valuable motivating influence on the Project personnel, for they confirmed the importance of parts screening, a heavy test program, and the design redundancy.

NASA Quarterly Reviews. NASA instituted quarterly reviews of the MM64 Project, and as they progressed the Project recognized them to be a beneficial experience. On these quarterly occasions the Project was obligated to organize its thinking and expose itself to review, and indeed many of the Project personnel took pride in presenting their progress. Communication between the Project and NASA, particularly the NASA Program Manager, was quite active during the quarter between reviews. Biweekly written reports were made to NASA along with telephone conferences. On-the-spot reviews were performed by the Program Manager for tests, shipping, operations.

Space Flight Operations (SFO). The organization for SFO, shown at top here, was patterned after that employed on Ranger and Mariner 2.

The SPA, SSA, and FPA all recommended action to the SFOD. The SFOD was the "Pilot" and took action on all matters such as data modes and priority, except for transmitting commands to the spacecraft, which the Project Manager had to approve.

The eight-month flight posed particular problems such as tedium and a danger that personnel might become lax. To offset this tendency, the SPA, SSA, FPA, SFOD, and Project Management functions were grouped together in a single mission area, where progress displays were set up to inform interested visitors, who were encouraged to make visits there. Regular meetings were held to permit total knowledge of progress among each of the flight groups on the Project.

Concluding Remarks. As a result of the MM64 experience, we think that some of the management practices that are applicable to any project may be summarized as follows:

A clear documenting by management as to what it wants done, by whom, and by what time.

Personal attention and participation by management to motivate and keep continually before the project participants what the job is that's to be done.

Management attention to a thread of key events during every phase of the project.

The structuring and use of scheduling, fiscal, configuration, and reliability performance controls with feedback to the various responsible managers as to the project management's evaluation of their progress.

Anticipating crisis areas and planning reserve resources to cope with crises.

And finally, a conscientious effort on the part of management to emphasize simplicity and accuracy in communication exchanges and in the control methods.

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Mariner 4—
A Point of Departure

By J. R. CASANI, A. G. CONRAD, and R. A. NEILSON
Jet Propulsion Laboratory

Mariner 4—
A Point of Departure

By J. R. CASANI, A. G. CONRAD, and R. A. NEILSON
Jet Propulsion Laboratory

Signaling the culmination of an era in spacecraft development, it represents a powerful and refined body of new technology certain to affect the designer's work in the future

Mariner 4 crowns a spacecraft design philosophy codified in 1959 to meet the initial objectives for unmanned exploration of the Moon and near planets Venus and Mars, its precursors in the development line being the Rangers and the Mariner 2 spacecraft to Venus. As we write, Mariner 4 continues its near-perfect performance on the last lap to Mars. It should not take much exercise of imagination to guess how we feel as we review the design of Mariner 4 not knowing the outcome of the flyby itself. But as far as concerns the spacecraft engineering, we will count it a success, discuss what went into it, and consider factors in the design and development that may affect future unmanned interplanetary vehicles. The reader should see our remarks against the broad picture of project management drawn by Jack James on page 34, the explanation of structural and mechanical engineering by J. D. Schmuecker and J. N. Wilson on page 26, and other Mariner 4 articles in this issue of *A/A* and the October issue, which will evaluate flight results in terms of design in more detail. Between our writing and your reading, history has been made, an era in spacecraft development culminated, and, as history has a somewhat arbitrary air, a new era might be said to have begun in interplanetary exploration, with Mariner 4 the point of departure. It represents a powerful and refined body of technology certain to affect the designer's work in the future.

Requirements. The Rangers and Mariners had similar basic requirements. Besides surviving the launch and space environments, such unmanned spacecraft must be able to operate completely unattended for at least about 66 hr in a lunar mission and over 6000 hr in a mission to Mars. They must, moreover, transmit data gathered by the instrument payload back to Earth. These are actually the only two basic mission-independent requirements for unmanned spacecraft—survive while operating unattended and return data to receivers on Earth.

Many approaches might satisfy these requirements. What approach the designer takes depends largely on the specific mission—Earth satellite, lunar probe, or interplanetary flyby or orbiter. The mission objectives will determine the lifetime, communication distance, and type and quantity of data to be transmitted, while various mission restraints, such as launch-vehicle capability and launch opportunity, will determine such factors as total spacecraft weight and the development schedule. Jack James discusses these in some detail in his article on page 34.

The Rangers and Mariners embody a set of basic design choices fairly well known by now: Full and continuous attitude stabilization, using the Sun as one reference and Earth or a prominent star as the second; photovoltaic power; and a fixed, or pointable, high-gain antenna for most of the communications functions.

At the inception of JPL's work on these spacecraft back in 1959, other possibilities were considered, such as spin stabilization, omnidirectional antennas, radioisotope thermal-electric power, and hibernating and intermittently operating spacecraft; many others exist. But the core of design just cited seemed to offer the most advantages for the existing and projected missions in which JPL expected to be involved. The table on page 17 summarizes the advantages of this core design in terms of full attitude stabilization.

Spacecraft Description. Mariner 4 represents an extension of Mariner 2 design and has the same flight sequence: Launch by an Atlas/Agena, establish a cruise mode, perform early in flight a trajectory correction as necessary, gather interplanetary data, and turn on planetary experiments and gather data about the planet Mars. Its major differences from Mariner 2 reflect the longer flight to Mars—eight rather than three months—and thermal loads steadily decreasing.

The spacecraft, fully stabilized in attitude, uses the Sun and the star Canopus as references. Cold-gas jets point the spacecraft in all three axes, and external torques are counteracted in two axes by changing the aspect of movable vanes to the Sun.

Its power system uses photovoltaic cells arranged on panels with a body-fixed orientation for cruise operations and a rechargeable battery for launch, trajectory-correction maneuvers, and backup. Power-conversion equipment delivers regulated 2.4-kc square wave, 400 cycle, and unregulated-DC electricity for distribution to the spacecraft subsystems. A Central Computer and Sequencer (CC&S) provides sync signals for frequency regulation, and performs the sequencing of onboard switching.

The spacecraft has a guidance system permitting trajectory-correction maneuvers and a propulsion system capable of executing two such corrections; these will be covered in an October *A/A* article.

A two-way S-band communications system carries telemetry to Earth, commands to the spacecraft, and angle-tracking, doppler and ranging information for orbit determination. It has two antennas—a low-gain and a fixed high-gain—either of which may transmit or receive. Switching between antennas may be done by onboard logic or by ground command. The command subsystem detects and decodes incoming command messages and passes them to the various onboard equipment.

FULL ATTITUDE STABILIZATION
Advantages of design philosophy adopted in 1959.

Permits high-gain antennas. For communications over great distances, this brings a significant reduction in required radiated power and consequently a lesser demand for raw electrical power—a critical factor in most spacecraft even yet.

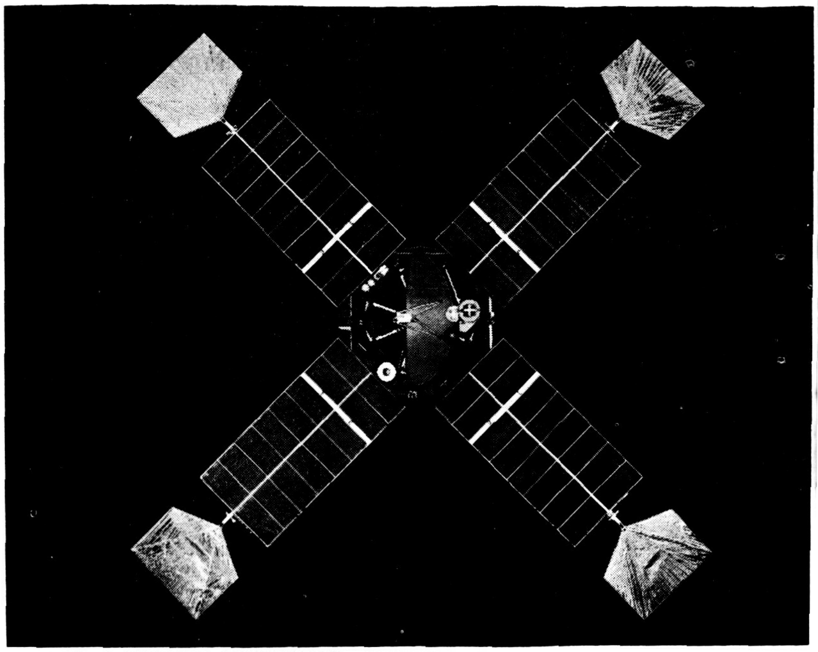
Gives inertial-reference directions automatically, from which in-course corrections to the flight path can be made. This is significant for missions requiring greater targeting accuracy than the launch vehicle can provide.

Permits 100% utilization of solar-cell area for photovoltaic power conversion systems. Compare this to a randomly oriented spacecraft, for instance, which for a sphere-shaped configuration would be only 25% efficient.

For interplanetary missions, the changing heat input from the Sun creates difficulties in temperature controlling randomly oriented or spin-stabilized spacecraft having spin axis off the Sun line. Sun-line orientation, on the other hand, permits parasol or shading techniques, which make it relatively simple to isolate the effects of a changing solar input.

Provides a stable platform, with orientation as functions of time precisely known, for carrying scientific instruments. This is particularly advantageous when a requirement exists for directing the axis of an optical or scanning instrument at a particular place on the target.

Note: Disadvantages—Position sensors and an associated torquing system are required to maintain orientation. The additional complexity of the attitude stabilization and the dependence of the power and communication system on it are considered to be the most serious drawbacks of full attitude stabilization, as compared with spin stabilization or random orientation.



MARINER 4 SUN-LINE ORIENTATION

Two types of commands may be used: Direct commands (DCs), which result in direct action by the receiving subsystem, and quantitative commands (QCs), which are transferred to the CC&S to be stored for later use. A Data Encoder subsystem formats, sequences, and, as necessary, provides analog-to-digital conversion of the telemetry data.

The spacecraft carries a number of scientific instruments to measure fields and particles between Earth and Mars and in the vicinity of Mars—cosmic-ray telescope, cosmic-dust detector, trapped-radiation detector, an ionization chamber, a plasma probe, and magnetometer. It also carries a television subsystem. A Data Automation Subsystem furnishes control and synchronization, performs necessary data conversions and encoding functions, and buffers the science data, transmitting it to the Data Encoder at the various appropriate rates and times.

The performance of this instrumentation will be discussed in the October issue of *A/A* following the encounter.

Mission Profile. As described by Jack James on page 34, the two Mariner launches were scheduled to take place at the Air Force Eastern Test Range (ETR) during November 1964, using the Atlas/Agena D. The projected flight times were 265 and 220 days to planetary encounter, plus additional time for stored-data transmission.

During the launch-to-injection phase, data reception from the spacecraft depends on the availability of the required range instrumentation. From liftoff until shroud ejection, the spacecraft radiates through a parasitic antenna in the shroud. The low-gain antenna maintains communications after shroud ejection. During this phase, the radio operates at decreased power, and the science instruments are not operating, to prevent a high-voltage breakdown as the spacecraft passes through the critical-pressure region. Moreover, to avoid vibration-induced problems during boost, injection, and separation, a holding current is passed through selected latching relays, the tape recorder is running, and power is not applied to the pyrotechnic firing and control circuitry. At separation from the Agena D, the cruise-science instruments are turned on, the communication-system radiated power is increased from 1 to 10 watts, power is applied to pyrotechnic firing and control circuitry, the relay holding current is removed, and electronic logic circuits are set that will turn off the tape recorder at the proper time.

Shortly after separation from the Agena D, spacecraft onboard logic commands pyrotechnic deployment of the four solar panels and the release of a support which holds the scan platform motionless during the ascent phase. (See the discussion by Schmuecker and Wilson on page 26.) Attitude-stabilization electronics then begin to operate and the spacecraft begins to seek its Sun reference.

A battery provides spacecraft power during the launch and solar-stabilization phases of the mission. Upon its acquisition, the solar panels are presented to the Sun, and they begin to supply power. It takes about 20 min, after injection, for the spacecraft to acquire the Sun.

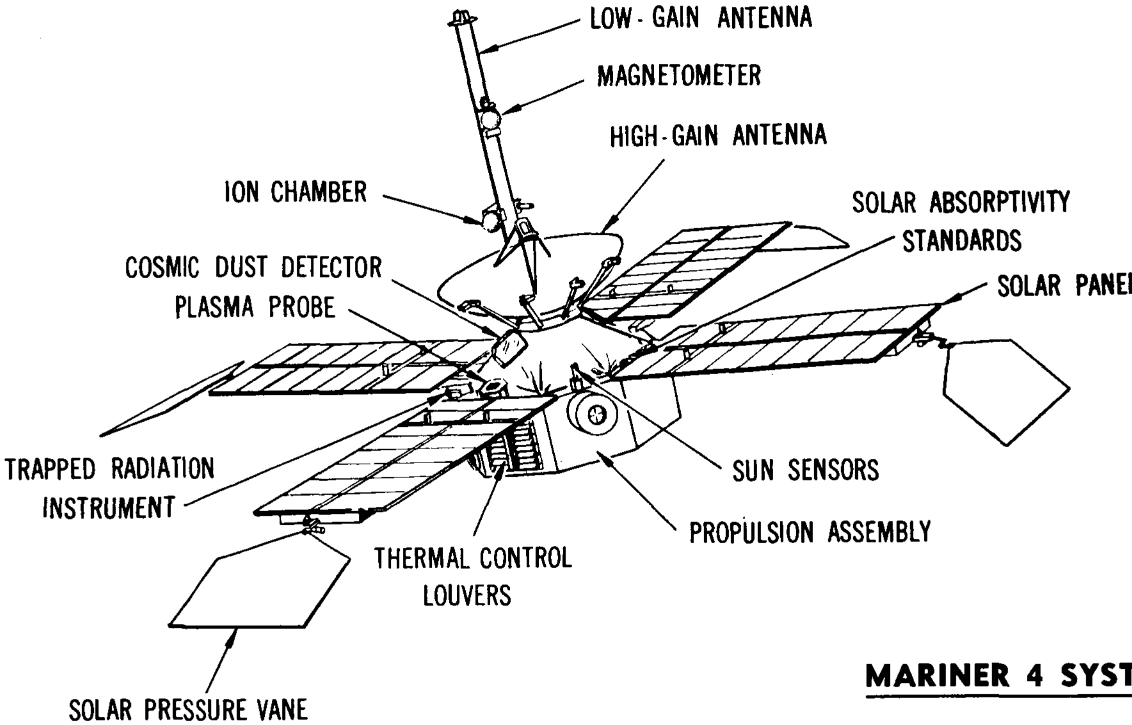
After acquisition, the spacecraft's attitude-control logic applies a constant roll-error signal that produces a controlled roll rate. Designed to furnish calibration information for the magnetometer while the spacecraft is still in the Earth's magnetosphere, this roll continues until 997 min after liftoff, when roll stabilization using the star Canopus begins. Canopus search involves rolling the spacecraft at approximately half the roll rate for magnetometer calibration until an object which meets the Canopus brightness requirements passes into the field of view of the Canopus tracker. At this acquisition point, the spacecraft gyros are turned off. Should the object be something other than Canopus, acquisition can be overridden by a ground command.

Now the spacecraft enters the cruise phase, its major condition during the transit to Mars, attitude-stabilized and transmitting continuously. Provisions are made for onboard automatic reacquisition should either of the two references be lost. The transmitted information consists of commutated engineering-data frames alternated with

SYSTEM REDUNDANCY SUMMARY

All "block" redundancy except 9, 10, 11, and 12, which have "functional" redundancy.

Item	Description
1. RF power amplifiers	Two complete power amplifiers plus associated power supplies switchable either by internal logic or by ground command. Only one operates at any time.
2. RF exciters	Two identical exciters switchable either by internal logic or by ground command. Only one operates at any time.
3. Pyrotechnic assemblies	Two identical half-systems, both on line continuously after separation.
4. Pyrotechnic arming	Two separation-activated switches — pyro-arm switch and separation - initiated timer. Either will power both pyro half-systems.
5. Attitude-control gas system	Two half-systems and pressure bottles. Both operate continuously after A/C turn on.
6. Power booster/regulator	Failure-sensing circuit will detect over/under voltage condition at main-booster output, and will switch maneuver booster on, main booster off.
7. Power frequency control	Primary control—38.4 kc sync from CC&S; in the absence of this, sync is derived from an oscillator internal to power. As a tertiary path, the inverters will free run at approximately 2.4 kc.
8. Analog-to-digital converter and pseudo-noise code generator	Two identical units switchable by ground command, only one operating at any time.
9. Cruise-science control relay	Separation or DC-2 control the primary cruise-science power supply relay. A secondary relay is driven at MT-7 or DC-25 to ensure cruise science on during encounter.
10. Science cover drop	Solenoid-actuated via pyro control assembly for primary; backup via lanyard.
11. End of TV record sequence	Primary, 2nd "end-of-tape" signal; secondary, inhibition of start-record commands after 11 non-real-time science frames.
12. Spacecraft cruise and encounter events	CC&S events for primary; backup via ground command for all.
13. Midcourse trajectory correction	Duplicate pyro - actuated valves provide the capability for two mid-courses.



MARINER 4 SYSTEMS

frames of experimentally collected scientific data.

Within the first few days after injection, the cruise phase is interrupted for a trajectory-correction maneuver. Usually at least one correction will be made to ensure that the spacecraft passes the planet at the most advantageous position and time. The spacecraft has the capacity for a second trajectory correction, should it be necessary to improve aiming-point dispersion. (See "Mariner 4 flight path to Mars" in the June *A/A*, page 28.) Normally, however, this capability would not be used before encounter.

During the interplanetary cruise, several events will occur, as a result of trajectory.

1. The data rate will be reduced from 33⅓ to 8⅓ bits per second to maintain the telemetry margin above threshold.

2. The Canopus tracker cone angle will be electronically updated four times during the mission to adjust it to the rotation of the spacecraft about the Sun.

3. The communications-system transmitter will be switched from the low-gain to the fixed, high-gain antenna as the communication margin diminishes with increasing spacecraft-Earth distance. The use of a fixed, high-gain antenna for this design was possible because of the favorable characteristics of the Mars '64 transfer orbit.

A 13⅓-hr period bracketing closest approach to the planet Mars defines encounter. At the beginning of this period the TV subsystem and the scan-actuator electronics are switched on by onboard logic or by ground command. The TV subsystem takes pictures during the entire encounter, but only pictures taken during the approximately 20 min when the planet is in the field of view of the camera are of interest. During that time, the TV pictures and selected real-time science data are stored on a tape recorder having a capacity of 5.24×10^6 bits. The recorder can store approximately 21 TV pictures. From the time that the planet comes into the field of view of the wide-angle sensor of the scan actuator until the end of the TV sequence, the spacecraft telemetry consists entirely of scientific data, the engineering-data frame being supplanted by a special planetary-encounter science data frame.

Mariner 4's trajectory in the vicinity of Mars will cause the planet to occult the spacecraft for as much as an hour some time between 1 and 4 hr after closest approach. During the last few minutes before occultation, the RF signals transmitted between the spacecraft and the Deep Space Instrumentation Facility (DSIF) will refract through the atmosphere of Mars. Kliore et al. describe the significance of the occultation experiment in the July *A/A*, page 72.

The stored data will be read out after the encounter phase, alternating at intervals with real-time engineering data. Each interval of stored data will include one TV picture plus some other stored science data, while each interval of engineering data will contain sufficient information for a brief engineering analysis of spacecraft performance and for DSIF operational purposes. These transmissions will be repeated cyclically until Earth stations have the full record of the experiments.

With the spacecraft back in the cruise mode, the option exists for an experiment to determine the feasibility of performing further trajectory correc-

ALTERNATE AND DEGRADED OPERATIONAL CAPABILITIES

APPROACH	OPTIONS
Cruise Science On	
Functional redundancy with an alternate mode of operation at encounter.	(a) Power switched on at separation when holding current removed from power-subsystem relay. (b) Ground command (DC-2); also switches encoder to data mode 2. (c) Alternate power-route activated at turn on of encounter science, ground command (DC-25), or CC&S (MT-7). (d) Before separation, primary mode may be exercised by turning off maneuver boosters.
RF Power Up	
Alternate mode.	(a) Removal of separation connector releases holding circuit. (b) Turnoff of maneuver booster releases holding circuit.
Tape Recorder Off	
Alternate mode.	(a) Removal of separation connector activates stop logic. (b) Maneuver booster turnoff removes tape drive power.
CC&S Relay Hold Off	
Alternate mode.	(a) Removal of separation connector interrupts holding current. (b) Maneuver booster turnoff removes off holding-current. Relays will reset if turned on again.
Solar-panel Deployment	
Functional redundancy.	(a) Separation-initiated timer. (b) CC&S.
Attitude Control On	
Functional redundancy; turns on Sun sensors and switching amplifiers.	(a) Pyro-arming switch. (b) CC&S. (c) Ground (DC-13); also turns on Canopus tracker and sets maneuver-inhibit logic.
Canopus Tracker On	
Functional redundancy.	(a) CC&S; also turns on solar-vane electronics. (b) Ground command (DC-13); also turns on attitude control and sets maneuver-inhibit logic.
Switch Data Rates	
Functional redundancy.	(a) CC&S (MT-6) switches to $8\frac{1}{3}$ bps. (b) Ground command (DC-5) switches bit rates (toggle).
Canopus Cone-Angle Update	
Functional redundancy.	(a) CC&S (MT-1, MT-2, MT-3, and MT-4). (b) Ground (DC-17).
Antenna Switching	
Functional redundancy plus alternate mode. The initial state of system determines reaction to any switching input.	(a) CC&S (MT-5) switches to transmit via high-gain antenna and enables CY-1 logic. (b) Ground command (DC-10) switches to transmit via high gain and receive via low gain. (c) Ground (DC-11) switches to transmit and receive via high gain. (d) Ground (DC-12) switches to transmit and receive via low gain. (e) Gyros on

tion maneuvers. The end of the Mariner 4 mission will occur 20-40 days after encounter, when the combined effects of increased range and antenna-pointing error should exceed the communications threshold.

Design Goals and Philosophy. The MM64 approach involved accepting a modest or conservative set of mission objectives and applying to these the maximum in terms of equipment redundancy and alternate operational modes. Several "design philosophies" followed from it and enormously influenced spacecraft system mechanization.

Probably the most significant of these philosophies is the notion of the "automatic spacecraft"—a spacecraft able to complete its entire mission from launch to end of life without ground-based intervention or support, except for orbit-correction maneuver. Since the required velocity increment can only be determined by ground-based observation following the launch, information regarding the magnitude, direction, and time for applying this velocity increment must be transmitted to the spacecraft from the ground. And to satisfy this requirement, the spacecraft must have the necessary sequencing, logic, and control capability to program itself through the various phases of the mission profile.

There are two corollaries to this philosophy of the "automatic spacecraft": First, the constraint that the mission profile and flight sequence be fixed before launch; and second, that ground commands not be required for in-line functional support, except to provide quantitative information to the spacecraft which cannot be determined before launch.

Another of the philosophies requires every discrete function or event critical to the success of the mission to be supported by at least two independent means of initiation. These functions include opening the solar panels, initiating attitude stabilization, updating the Canopus tracker cone angle, and initiating the encounter sequence. Usually radio commands plus CC&S commands provided the independent initiation; but many other techniques were used, as will be discussed later.

A third and very important philosophy requires continuous functions critical to mission success to be supported by redundant or alternate modes. A corollary to this philosophy requires the minimizing of subsystem and functional interdependency; subsystem failures that would not otherwise be critical to mission success must also not cause a loss of a second subsystem or critical function. What follows amplifies this philosophy.

Application of Redundancy. The most straightforward method of increasing reliability of a system is to exercise great care in the design, in the selection of component parts, and in the manufacturing process. If the system is composed of series elements, an increase in reliability of 1% in a single element increases the reliability of the total system by a like amount. This type of reliability increase is attained by providing adequate design margins, instituting parts qualification and selection programs, carrying out extensive inspection procedures at both the parts-manufacturing and subsystem-assembly levels, and submitting the assembled flight items to well-designed prototype and flight-qualification testing. The advantage to this type of program is that it ensures the highest reliability for a given set of weight and performance requirements. The limiting factor, however, is that the law of diminishing returns sets in, precluding any major increase in system reliability over that attainable through best engineering and manufacturing practice.

Although this type of reliability, which attempts to guarantee that failures will not occur, is certainly necessary, it is not sufficient. For example, the Mariner 4 spacecraft has 35,000 electrical components. Assuming that each has a mean time between failures of 100-million hours, the probability that at least one component failure will have occurred by the end of 6000 hr of operation is 0.84. The spacecraft design, then, must provide for proper operation in spite of failures. This capability exists to some extent in any design; components in noncritical applications may fail with only slightly deleterious effect, or with no effect at all. The failure of others may serve only to reduce protection against nonstandard operating conditions. The failure of a critical component, however, may not only result in the loss of a single function, but may induce such nonstandard operating conditions that other components are overstressed and the failures propagate throughout the system.

Under these constraints, redundancy will be the only practical way to provide the required system reliability. Redundancy may involve the paralleling of two or more physically identical systems or the paralleling of two or more functionally identical but physically different systems. Each type of redundancy has its advantages. The decision to employ one or the other must carefully consider the particular application and possible consequences of the choice.

Block Redundancy. Paralleling two identical units protects against loss of function as the result of component or piece-part failure. If the failure is casual, however, as would be a design failure or an overstress on components due to some external perturbation of the system, block redundancy becomes less desirable, since both units would be likely to experience the same failure, defeating the original purpose of redundancy. This problem can be avoided by operating only one

APPROACH	OPTIONS
	switches to receive via low gain. (f) Two consecutive CY-1 cyclics with receiver out of lock switches receiver to other antenna. Each subsequent consecutive CY-1 with receiver out of lock will switch receiver.
Roll Control	
Alternate modes of operation.	(a) Normal; initiated by CC&S and ground command. Canopus tracker on with brightness gates in effect. This mode may be restored after use of an alternate mode by ground command. (b) Ground command (DC-15); Canopus tracker on, brightness gates removed from circuit. Will track any detectable light source. Gyros disabled. (c) Ground (DC-18); inertial roll control. Tracker off, gyros on. Pitch and yaw gyros in rate mode. Sun sensors unaffected. (d) Ground (DC-20); no roll control. Tracker off, gyros off. (e) Rate mode; prior to Sun gate only. Gyros on in rate mode. (f) Magnetometer calibrate roll mode; subsequent to Sun acquisition, before star acquisition. (g) Normal search; initiated by CC&S or ground command (L-3 or DC-13). Until initial acquisition only. (h) Acquisition override, by ground (DC-21), causes normal search.
Data-Mode Switching, Data Mode 1	
Functionally redundant for midcourse phase.	(a) Ground (DC-27) switches to data mode 1, starts midcourse sequence. (b) Ground (DC-1).
Data-Mode Switching, Data Mode 2	
Functionally redundant.	(a) Reset of CC&S (M-1) relay during midcourse; also initiates reacquisition. (b) Ground (DC-2); also turns on cruise science. (c) Ground (DC-25); also turns on encounter science. (d) Tape recorder second end of tape or end of DAS NRT frame No. 11, whichever first. (e) CC&S (MT-7).
Data-Mode Switching, Data Mode 3	
	(a) Wide-angle acquisition. (b) Narrow-angle acquisition or TV planet-in-view if no WAA.
Data Mode Switching, Data Mode 4	
Functionally redundant. If no data present on VSS tape, data is automatically formatted to mode 1.	(a) CC&S (MT-9); tape playback and data mode 5. (b) Ground command (DC-4); tape playback and data mode 4.
Encounter Science On	
Functionally redundant.	(a) CC&S (MT-7). (b) Ground (DC-25).
Encounter Science On	
Functionally redundant.	(a) CC&S (MT-8). (b) (DC-26).
Tape-Recorder Tape Track Change	
Functionally redundant.	(a) Internal logic triggered by end-of-tape signal. (b) DC-22.

half-system at any given time, and inserting a failure-detection and switching unit serially with the redundant pair of units. The insertion of this detection and switching unit, however, decreases system reliability to the extent that it enters as a serial element. One other aspect of block redundancy is always unfavorable to some extent; two units operating in parallel cost approximately twice as much in weight and volume as one unit. If they are time-sharing units, controlled by a switching element, they cost more power than one unit by the amount of power consumed by the switching element; and if they operate simultaneously, they draw twice as much power as a single unit.

Functional Redundancy. Paralleling physically different but functionally identical subsystems not only has the advantage of offering greater protection against casual failures than does block redundancy, but also it can often be done at relatively low cost in terms of weight, volume, power, and complexity. The prime objective in employing functional redundancy is to provide at least two separate and independent paths by which a critical operation can be performed.

A simple example of this design philosophy in the Mariner 4 is found in the deployment scheme for the protective cover of the scan and TV optics. The primary path for deploying this cover involves operating a solenoid to unlatch the cover, and then allowing the spring-loaded cover to drop. A secondary path has a lanyard connected to the cover, so that motion of the scan platform will unlatch it. The primary path then depends upon the correct operation of the pyrotechnic firing and control assemblies and the solenoid, while the secondary path depends on structural integrity and proper operation of the scan-platform actuator. The paths are separate and independent, with the result that no noncatastrophic failure can be postulated which will fail both paths, short of a failure in the cover itself.

A second example of functional redundancy is found in the philosophy that all critical onboard switching functions should be backed up by ground commands. It would have been possible to include a redundant central computer and sequencer in the Mariner 4 design, but any advantages would have been outweighed by the bulk of the redundant unit, the complexity of an onboard failure-detection network, and the operational inflexibility of such a system. These disadvantages could be avoided to some extent by performing the failure detection via telemetry and the switching function via ground command. This would then require that the telecommunications system be inserted in series with the CC&S in any consideration of system reliability. A much simpler and more direct method of increasing the reliability of onboard switching functions is to increase the capability of the already existing command subsystem to provide backup command capability. Again a set of separate and independent paths are provided for critical functions. The value of this approach was well proven during the encounter of the planet Venus by Mariner 2.

The block-redundancy approach is not without merit, however. Certain subsystems lend themselves better to block redundancy than any other technique. Subsystems whose design is well understood and tested, and whose probable-failure modes involve component failures unprovoked by the design environment itself, are prime candidates for block redundancy. For that reason, Mariner 4 was designed to utilize two completely independent nitrogen-gas systems for attitude stabilization. Assuming flight equipment is well-enough controlled during manufacture, assembly, and testing to preclude any severe weaknesses or manufacturing flaws, the most probable failure mode would be loss of gas pressure due to random sticking of the gas valves. In this case, the redundancy of gas valves between the two half-systems permits normal operation if a valve fails closed; and if it fails open there are two opposing valves to counteract it, one in its own half-system and one in the redundant half-system. If the gas load is sized properly, after all of the gas in the failed half-system has been expended, enough remains in the redundant half-system to allow the completion of the mission.

Block redundancy is used in a number of other areas in the spacecraft. Two pyrotechnic firing and control assemblies are employed in parallel, each operating its own pyrotechnic devices or at least separate burn wires in the same device. The pyrotechnic half-systems are designed to operate simultaneously and independently, and each is sufficient for all pyro functions through the end of midcourse maneuver.

Pyrotechnic-initiated functions include solar-panel release, scan-cover release, and motor-ignition and termination events. These events are typical of a class which is time critical as well as functionally critical, and the application of block redundancy does not necessarily improve the reliability of initiating the required event on time.

In a number of areas where simultaneous operation of redundant units is not permissible, time-sharing redundant elements are employed, switched on and off line either by onboard failure-sensing devices or by ground command. The RF power amplifier is continuously monitored in a control unit on the spacecraft. Should the power output become degraded, the next 66⅔ hr cyclic pulse issued by the CC&S will cause the control unit to switch the degraded power amplifier and its associated power supply off line and switch trom an ascent to a cruise configuration. During the ascent, two environmental factors—the high vibrational loadings and the passage through the critical-pressure region—impose severe constraints on the spacecraft. Consequently, its structures are clamped to prevent damage, no high-voltage circuits are operating, no power is applied to the pyrotechnics subsystem, magnetic latching relays are either electrically held or designed to reset at power application, and the video-storage-tape drive is running to prevent possible tape spillage.

All of these must attain a normal cruise state between separation and initial star acquisition. Most of the events that must take place after injection are initiated by removal of electrical shorts across the Agena separation connector or by activating spring-loaded pressure switches at the separation interface. The CC&S relay-hold function, the tape drive-power, and the "RF power up" and "cruise-science on" commands are all mechanized with shorting plugs across the separation connector. In each case, power is supplied through the separation connector to operate circuitry holding gear in the launch state. Upon removal of this power, the gear reverts to its normal cruise state. Failure of the separation connector or associated cabling is still a possibility. And short circuits can occur internal to the spacecraft. In such a case it is necessary to find some alternate way to return the subsystems to a cruise state.

The key to finding an alternate mode of operation for this particular case lies in the fact that an interruption of power through the separation connector is the requirement for normal operation of the subsystems involved. The spacecraft has a power supply—namely, the maneuver booster—normally turned off for all but a few periods during the mission, one being the launch phase. Supplying power from this maneuver booster to hold subsystems in the launch condition establishes a secondary means for initiating cruise mode. For drive to the tape recorder and "cruise science on," this secondary means gives a true functional redundancy, since neither is affected by maneuver-booster operation after it has been turned off once. This is true for the "cruise science on" command because it is undirectional; that is, although cruise science would be turned on every time the maneuver booster cycled in the event of a separation-connector failure, there is no means of turning cruise science off via the same path. For the drive to the tape recorder, a judicious design has enabled the circuit to remove itself from operation once it has functioned.

In the other two cases—RF power up and the relay-hold circuit—this is an alternate operating mode in which the spacecraft can perform, but somewhat degraded. RF power will stay high when the maneuver booster is on if the short circuit is to spacecraft ground, the only degradation being an additional one watt of power drawn from the maneuver booster. If the short is across the separation connector, however, RF power will cycle up and down with the maneuver booster on and off. As a result, telemetry will be lost each time the gyros are turned on in this mode; however, the RF radiated power returns to normal each time the gyros are turned off again. As to the CC&S relay-hold circuit, all of the relays are reset each time the gyros come on, and none can be set until the gyros go off again. The primary degradation to spacecraft performance from this comes as the loss of ability to perform a trajectory correction.

A number of other examples of the alternate-operating-modes approach exist in Mariner 4 design, principally at the subsystem level. The table on pages 20 and 21 summarizes the alternate modes with appropriate remarks.

Concluding Remarks. At this writing, the flight of the Mariner 4 has so far substantiated the merit of this design philosophy. All functions have been performed normally by the spacecraft, and a measure of operational flexibility has been added to the flight operations not previously available in JPL missions. In addition, one of the alternate operational modes has been brought into action in response to an environmental effect that could have proved catastrophic; the Canopus tracker brightness gates were removed by ground command (DC-15) after a number of high brightness gate violations—possibly the result of the "fireflies" first noted by the Mercury astronauts—caused temporary losses of roll stabilization.

There is every reason to believe that if it proves completely valid, the Mariner Mars 1964 design approach will be increasingly useful for future missions.

As the mission objectives become more ambitious, so will the spacecraft which perform them become larger and more complex, the susceptibility to problems increase, and the penalty for a problem which compromises mission success become more severe. Under these constraints, design philosophy must ensure that the spacecraft operate normally under the widest spectrum of conditions possible and beyond that still be able to operate with only slight degradation, even if the implementation of such a philosophy reduces payload below the maximum possible. After all, even the most sophisticated scientific instruments are of no value if the basic rules for unmanned spacecraft design are violated—that the spacecraft must survive while operating and must return the data gathered to the Earth.

on the redundant amplifier and power supply. Switching may also be accomplished by ground command—an example of the integrated use of block and functional redundancy to produce the highest possible system reliability. This method is also employed with the radio subsystem's dual RF exciters.

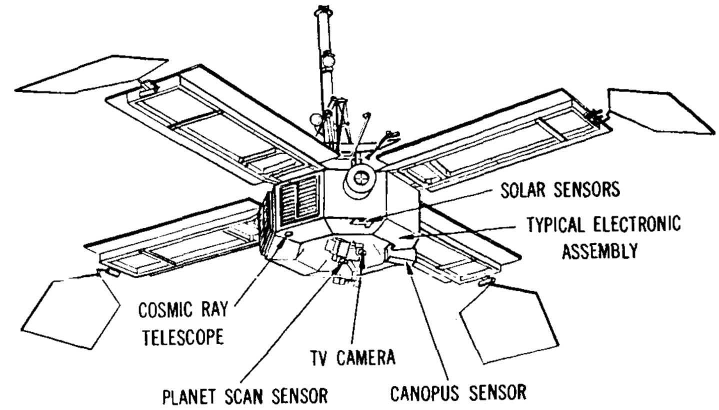
The spacecraft power has redundancy techniques in two areas at the system level—power boosting and regulation, and timing. The Mariner 4 design includes two separate power booster/regulators, one primarily for fixed loads and the other for varying and intermittent loads. This permits the fixed-load, or main, booster to be sized for peak efficiency during the power-critical period at encounter. Efficiency is not a prime consideration for the intermittent, or maneuver, booster, since its duty cycle is extremely low and normally it is not operating for extended periods near encounter. Therefore it was sized so that it could assume all of the spacecraft loads in the event of a main-booster failure. A failure-detection circuit which senses an overvoltage and undervoltage operates the switching network that removes the main booster and inserts the maneuver booster. A redundant timing system is also provided for frequency synchronization of 2400- and 400-cycle power. Primary timing is derived from the central computer and sequencer; should this path fail, there is a backup oscillator in the power subsystem capable of assuming the timing function.

A second backup is provided through the design of the inverters, which have a natural frequency nominally equal to the synchronization frequency, so that near-normal operation can be maintained in the absence of any external timing signals.

Finally, two units are provided in the telemetry encoder for the conversion of analog data to a digital format and for the generation of the pseudo-noise code. Switching between units is done by ground command.

In Mariner 4, subsystems critical for all or most of the mission objectives are the ones to which system-redundancy techniques have been applied, as summarized in the table on page 18. The items listed are those for which two or more redundant units of hardware have been included in the design, using either the block or functional approach. In each case, either of the redundant items can be used. Not included in this table are the functions or techniques employing alternate or degraded modes of operation, which will be covered next.

Alternate and Degraded Operational Modes. Somewhat apart from the general spacecraft reliability considerations are a number of problems involving degraded operation for reasons of failures or abnormal environments, operational flexibility, and design and operational procedures for time-critical functions. With Mariner 4, the approach to these has been to create or to exploit some alternate mode of spacecraft operation. Generally, for each function considered one or more alternates have been available, with a minimum of design perturbation.



PLANETARY-SCAN INSTRUMENTS AND SENSORS

It is obvious that alternate modes of operation underwrite continuous operation against failures or abnormal environments. Furthermore, if the modes can be selected by ground command, the system gains operational flexibility.

The alternate-mode approach can also prove especially valuable in treating time-critical functions. Suppose, for instance, an event is constrained to occur within an interval of time, and it occurring outside this interval will be potentially catastrophic. Using redundancy techniques alone, any action taken to increase the probability that the event has occurred by the end of the interval will also increase the probability that it occurs before its beginning. Conversely, an attempt to preclude any occurrence before the interval will lower the probability that the event has occurred by the end of the interval.

The dilemma which this presents is nicely circumvented by the alternate-mode approach. The system is mechanized to lower the probability of premature occurrence to an acceptable level, and the attendant risk of late occurrence is accepted. To offset the effects of late occurrence, an alternate mode is provided that permits all of the mandatory functions associated with the event to be performed at the expense of some possible system degradation. Since this approach generally involves multiple interfaces among subsystems, the large number of possible combinations of subsystem states provides a relatively wide spectrum of operational modes, some number of which can be selected by ground command.

The phases of the mission which lend themselves best to an alternate-mode design approach are the spacecraft/Agena separation and initial acquisition phases, which cover the period in which the spacecraft must make the transition

Structural and Mechanical Design of Mariner Mars

By JAY D. SCHMUECKER and JAMES N. WILSON
Jet Propulsion Laboratory

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Many important design lessons have been learned, and relearned, with this spacecraft, which may well prove a model for future interplanetary vehicles

Since Mars launch opportunities occur only at 25-month intervals, the Mariner 4 spacecraft had to be designed, fabricated, and tested on a tight schedule, with no slippage possible, to meet a launch in the designated time period, as discussed by Jack James on page 34. This firm but short schedule, severe weight restrictions, and very high reliability levels dictated seemingly incompatible requirements of making the spacecraft light in weight, yet using proven techniques.

This article describes how the problem was resolved. It includes description of the various subsystems, along with reasons why designs or approaches were selected. Philosophy will be discussed, problems uncovered and solved during the development cycle reviewed, and recommendations for future projects made. Primary emphasis will be placed on new developments and less on those more routine.

Configuration. Developing the mechanical configuration of a spacecraft—a fascinating process—involves unifying all its discrete elements, most of which confront the designer with conflicting requirements. A good configuration minimizes problems as development proceeds. For example, solar panels in the wrong place may necessitate extra structural support and increase panel dynamic environment. And allowing many spacecraft elements to protrude beyond a relatively smooth exterior may unduly complicate temperature-control analysis and testing, from several points of view.

Some of the requirements to be satisfied by a configuration include the following: The envelope dictated by the launch-vehicle shroud cannot be violated; adequate and usable volume for all equipment must be provided; and view-angle constraints imposed by solar cells, antennas, attitude-control sensors, and scientific instruments must be met. Temperature-control design constraints must also be satisfied, and operational requirements considered.

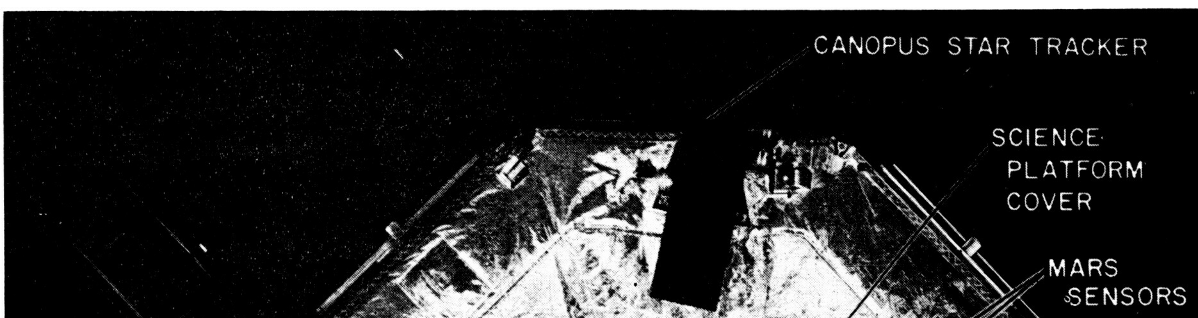
Nearly 20 different approaches were investigated during the Mariner 4 configuration development, leading to the flight configuration shown on page 19. Built around an octagonal structure, 18 in. high and 4½ ft in diam, the spacecraft has electronic equipment housed around its periphery in seven of eight bays. The eighth bay houses the post-injection propulsion system.

Location of the propulsion system, approximately normal to the roll axis, represents a departure from previous practice. During the early phases of the configuration design, the capability for doing a "restricted" trajectory-correction maneuver was desired. The maneuver could thus be mechanized using only a roll turn, never losing the Sun attitude reference and solar power. As the design evolved, this requirement disappeared; but during the interim configuration iterations, many other desirable features of so locating the midcourse motor more than offset the disadvantages. Placing the motor in this position allowed it to be easily installed at the last moment as a fully fueled unit, and space was available in the traditional motor location for the planet-oriented scientific instruments. The maneuver attitude-control system was compromised slightly by this position.

On top of the basic octagon structure ride the four deployable solar panels, restrained during boost by eight concentrics tube viscous dampers. Power requirements called for 70 sq ft of panel area, made rectangular and of modular dimensions to allow dense solar-cell layout. The low-gain antenna sits on top of the octagon, and the fixed, high-gain parabolic antenna on top of the superstructure. During the initial portion of the flight, the low-gain antenna illuminated the forward hemisphere in which the Earth was always located, even during the trajectory-correction maneuver. This indicates another advantage of having the motor normal to the roll axis; had it been parallel to the spacecraft–Sun line, another antenna would have been needed for communication during the then different maneuver.

On the tips of the solar panels are mounted the cold-gas attitude-control jets. Despite the design and operational complexity of having the attitude-control plumbing cross the solar-panel hinge line, placing the jets at the tips of the panels increased their moment arm enough to allow significant reductions in the total gas weight. Dual gas supplies are situated within the octagon. Also at the tips of the solar panels are movable solar-pressure vanes for attitude stabilization. These vanes cause the spacecraft to be stable under solar pressure and by their movement allow balancing of upsetting moments. The vanes stow in a furled position behind the solar panels during boost.

Attitude-control Sun, star, and Earth sensors ride the basic octagon structure, so placed to give minimum interference in viewing fields. To minimize light reflections into it, the sensitive Canopus star tracker rides in the shade of the aft side of the spacecraft, with its field of view extending between two solar panels. The cruise scientific instruments are mounted around the periphery of the octagon, on the superstructure, and those requiring physical isolation from the spacecraft are mounted on the low-gain antenna. Originally mounted on the tips of the solar panels these latter

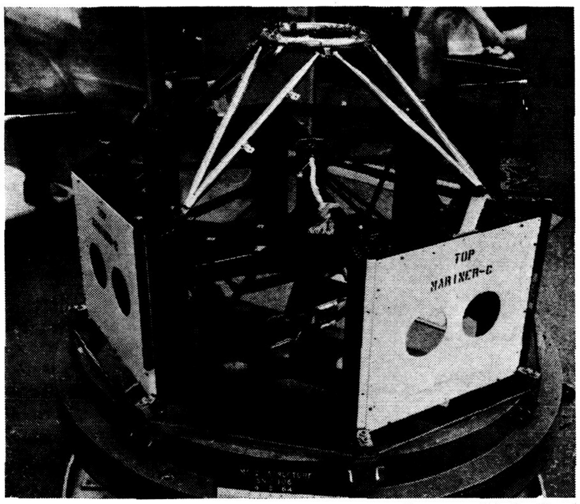


Multi-layer aluminized-mylar shields on the sunward side of the spacecraft isolate it from the changing solar intensities and on the aft side, shown here, minimize heat losses to space.

instruments were later relocated with some small compromise in data quality.

TV camera and planetary sensors ride on a one degree-of-freedom rotating structure, mounted on the center line of the spacecraft with the instruments facing 120 deg away from the Sun line. An actuator in the basic structure can rotate the structure 180 deg about an axis parallel to the Sun line.

Most of the temperature-critical spacecraft components ride in the primary octagon. Various surface finishes and six sets of thermostatically controlled louvers mounted on the sides of the octagon control spacecraft temperature. Multilayer aluminized-mylar shields on the sunward side of the spacecraft isolate it from the changing solar intensities and on the aft side, shown on page 27, minimize heat losses to space. Polished aluminum shields control the heat losses from the unlouvered sides of the spacecraft. Combinations of surface finishes control the temperature of elements outside of the basic octagon.



Basic spacecraft structure consists of two 4½-ft-diam machined magnesium rings coupled by eight longerons. The outboard surface of the electronic-packaging chassis provides shear webs around the periphery.

Many of the requirements imposed on the configuration design by subsystems are reasonable, but a few turn out to conflict severely. The configuration designer has to review these requirements constantly to verify that they are mandatory. Many initially firm requirements can be compromised without undue degradation of the subsystem, thus allowing a more nearly optimum system design. An example of this was the requirement that the solar-plasma experiment have a 15-deg unobstructed view of the Sun. There was no place on the spacecraft which satisfied this requirement, thus creating the need for some kind of deployable structure. Consultation with the scientist and engineer responsible for the instrument indicated that it could be allowed to point a small angle away from the direct Sun line without significant compromise to the data. This allowed location on the octagon and simplified the total design. The propulsion-system location previously mentioned violated an originally firm requirement to point along the roll axis. Many other such compromises during the configuration design simplified the end result.

The table on page 29 lists Mariner 4 weights, with a detailed breakdown of the mechanical subsystem portion.

Primary Structure. The basic octagonal structure consists of two 4½-ft-diam machined ZK-60 T5 magnesium rings coupled together by eight longerons. The outboard surface of the electronic packaging chassis provides shear webs around the periphery. This basic approach had been tried successfully on a smaller scale on the early Mariner/Centaur spacecraft. Miscellaneous supporting structure within this octagon mounts the low-gain antenna, the planet-oriented equipment support structure, and the gas-pressure vessels for the attitude-control system. All attitude-control sensors ride either the upper or lower rings on integrally machined pads. Thus, no machining or adjustments after assembly was required. Extensive use of tooling to manufacture the rings and longerons allowed the basic structure to be assembled rapidly with a minimum of jigs. The spacecraft was attached to the Agena-D adapter by a V-band, which held eight shoes clamping eight feet on the lower ring of the spacecraft. The superstructure that supports the high-gain antenna and dampers for the solar panel and low-gain antenna consists of a welded aluminum truss bolted to the top of the octagon. The accompanying photo shows the primary structure and superstructure installed on a handling ring.

Solar-Panel Structures. Each of the four solar-panel structures consist of a 0.005-in.-thick aluminum substrate made rigid by corrugated stiffeners of 0.0035-in.-thick aluminum. This structure is supported by a pair of longitudinal box beam spars, their bases being attached to the primary structure by rod end-bearings. These elements were bonded together, as initial tests demonstrated that the more conventional spot-welded construction was inadequate to survive the acoustic environment of boost. Each panel was restrained during boost by two concentrics tube viscous dampers attached from the superstructure to the panel spars at their quarter points. The dampers contained a 5-mil annulus filled with silicone grease. A spring was included to statically center the damper tubes. The top photo on page 30 shows a prototype panel supported by the damped struts; and the photo below it shows the panel backside. As a direct result of the vibration reduction brought by these dampers, the solar-panel struc-

MARINER 4 WEIGHT PROFILE

Structure and Mechanisms	
Primary octagonal structure	30.86
Superstructure	3.06
Propulsion-support structure and cover	3.89
Six electronic-assembly chassis	25.45
High-gain antenna	4.44
Low-gain antenna	3.87
Solar-panel and low-gain-antenna support	3.09
Dampers	2.09
Four solar-panel structures	44.72
Four sets of panel deployment and retardation hardware	2.16
Four sets of solar-pressure-vane structure and deployment hardware	2.52
Science-platform structure and associated gear	4.12
Science-platform actuator	2.07
Science cover and deployment hardware	1.14
	131.39
Electronic Cabling	
Cable harnesses and connectors	44.72
Harness-support structures	3.80
	48.52
Temperature-Control Hardware	
Six sets of temperature-control louvers	8.16
Temperature-control shields and associated gear	7.02
Absorptivity standard	1.37
Canopus sensor light shields	0.45
	17.00
Science Instruments and Electronics	
Platform instruments and sensors	10.97
Low-gain-antenna and primary structure mounted experiments	12.37
Science electronics and data conditioning gear	34.42
	57.76
Power Conversion and Storage Equipment	
Conversion and regulation electronics	39.32
Battery	33.46
Panel solar cells and wiring	34.40
	107.18
Attitude Control and Guidance Equipment	
Electronics	18.80
Motor thrust-vector-control assembly	3.28
Attitude-control sensors	6.38
Two attitude-control gas systems	22.92
Total nitrogen gas	5.00
Four solar-pressure-vane control assemblies	2.80
	59.18
Central Timer and Sequencer	11.89
Data Handling and Storage Electronics	
Data encoder	22.45
Tape machine	17.04
	39.49
RF Communications Electronics	
Command	10.11
RF transmitter and receiver	35.50
	45.61
Propulsion System	
Propulsion hardware	23.39
Propellant	21.50
Nitrogen gas	0.92
	45.81
Pyrotechnic Subsystem	
Electronics and switches	8.51
Pinpullers and squibs	2.17
	10.68
Miscellaneous Hardware	0.86
Total pounds:	575.37

tural weight-to-area ratio was only 0.58 lb/sq ft.

After two pyrotechnic pin pullers holding each panel to the dampers fired, each panel deploys 90 deg by means of two clock springs attached at the hinge. Then a hydraulic damper absorbs the energy imparted to the panels and positions them. The dampers also minimize solar-panel excursions during propulsion maneuvers.

Antenna Structures. The high-gain antenna reflector was fabricated from aluminum honeycomb —cell material 0.0007 and skins 0.004 in. thick. The core was formed into a parabolic shape by first bending it along several lines parallel to the long direction of the antenna, thus forming it into a roughly cylindrical shape; then, by draping the core over a parabolic mold, it deflected about the short axis, allowing the original cylindrical radius to increase to the proper curvature. The skins were bonded to this formed core by applying a 1-mil sheet of adhesive to the facings. The edges of the core were filled with a foam plastic to minimize handling damage. The antenna feed truss was fabricated of unidirectional RF-transparent fiberglass tubes, and used RF-transparent Lexan plastic truss fittings.

The low-gain antenna is a circular waveguide structure, 4 in. in diameter, with the feed located at the aft end and an aperture at the forward end to shape the beam. This approach proved considerably lighter than a conventional antenna located on a support structure with a long coax cable to the antenna. Like the solar panels, this long mast was damped by concentrics tube viscous dampers. Wall thickness–0.025 in.–of the antenna was determined by the ability of the waveguide to withstand handling abuses and not from structural requirements.

Planet-Science Platform Structure and Mechanisms. The planet-oriented scientific instruments and sensors ride on a structure cantilevered aft from the primary one. An actuator atop a central torque tube rotates the platform to achieve the scan. The platform was restrained during boost by a pyrotechnic pinpuller. Cables to the instruments run through the 15-in. long central tube, loosely bundled, allowing them to twist during articulation. The actuator, a slightly modified version of the one used on Mariner 2, consisted of a 400-cycle, 26-volt, 8000-rpm synchronous motor and reduction gearing to give a 0.5 deg/sec output. Cam-operated limit switches reverse the scanning motor. The actuator has a housing pressurized with dry air to 2 atm (absolute). An aluminum honeycomb cover protects the delicate planet-oriented instruments from cosmic dust and sunlight. It can be deployed by a solenoid or independently by a lanyard triggered by platform rotation.

Electronic Packaging. Packaging design employed modifications of the proven techniques used in previous programs. The electronic components were packaged in standard-profile subassemblies. Typically, two vertical rows of subassemblies were mounted into each electronic chassis, and this bolted to the primary structure. This structurally efficient, compact design was very flexible; it allowed the spacecraft design to proceed while electronic-subsystem sizes and locations were still being determined.

Placing all the components relating to one electronic subsystem into one easily removable chassis facilitated subsystem lab and environmental testing. During spacecraft-system testing, subassembly accessibility was achieved by either removing the complete electronic assembly or removing the subassembly from within the primary structure. The photo on page 32 shows an electronic chassis, typical electronic packaging approaches, and a case harness.

Several component packaging techniques were used:

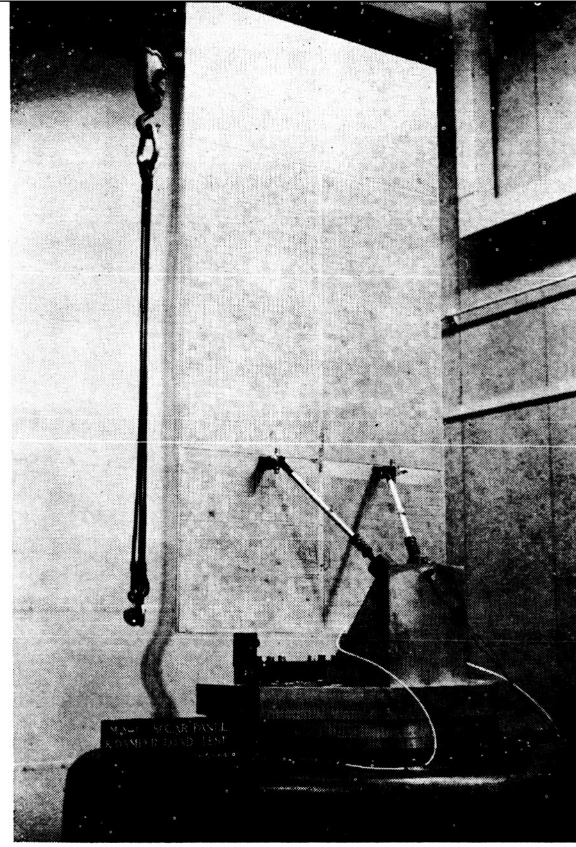
1. Planar Packaging. Components of various sizes were mounted to printed wiring boards which in turn were mounted to the magnesium subchassis. This approach gave design flexibility and eased component replacement.
2. Cordwood Packaging. Repetitive circuits were packaged in potted modules, which were then mounted to the subchassis. This approach gave high packaging density but compromised component replacement.
3. Pelletized-Component Packaging. A new technique, inserting pelletized components into a printed circuit board, saved much weight and space. This approach was used on some low-power subsystems.
4. Other Techniques. Combinations of these approaches and special techniques were used when needed.

Electronic Cabling. Cabling interconnects all the spacecraft subsystems. Case harnesses interconnect subassemblies, and through pigtails connect to a ring harness trough-mounted on top of the basic structure. This trough contains power and signal-wire bundles. Another harness, around the periphery of the spacecraft base, carries umbilical and attitude-control functions.

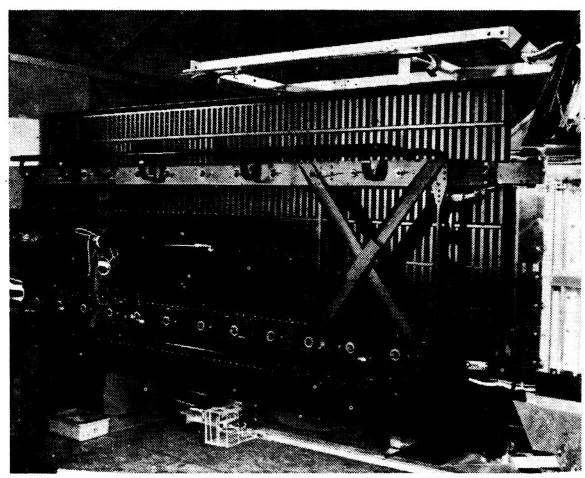
The wiring conductors consist of AWG 22 and 24 multistrand wire; insulation, of three wrapped layers of 0.003-in. polytetrafluoroethylene.

Two types of connectors were used: Bendix PT-type circular connectors for multiple mating cycles; and subminiature rack and panel Cannon D-series-type connectors in confined areas, notably on the electronic subchassis.

The cabling design had to allow for wiring changes, as many incompatibilities are not discovered until spacecraft electrical testing begins. In most cases, it was easier to change interconnecting cabling than the subassembly wiring.



Prototype solar panel supported by damped struts; below, other side of panel, showing structure.



Developmental Testing. Besides the many qualification tests on systems and subsystems, there were a number of development tests—feasibility, functional, and design-verification. These proved very rewarding, as they uncovered a significant number of possible problems in time to solve them. The qualification tests went quite routinely, very few failures being observed.

Nearly 40 different kinds of mechanical tests were made during the Mariner development cycle; these included a damped structure feasibility test, matchmate tests with the booster hardware, a test in which the softly suspended spacecraft was subjected to the propulsion maneuver controlled by the attitude-stabilization system, solar-panel-deployment tests, effects of shroud impacts on the backs of the panels, and deployment of the solar-pressure vanes. Also made were temperature-control-model and subsystem tests in a space simulator; tests of thermal-shield outgassing and of ballooning as multilayer shielding experienced decreasing pressure; a test of sunlight reflections into or near the Canopus tracker; aerodynamic-fairing purging, cooling and combined-vibration tests; and separation tests of the spacecraft from the Agena adapter.

Selected Problems and Solutions. Many problems came to light during the Mariner 4 development, but none of the *expected* ones. Since significant knowledge or learning resulted from the solutions of these problems, and since they contributed much to the project's success, they will be treated here in some detail.

One problem, partially expected, had to do with the spacecraft's very lightweight, fragile structures. Handling problems had been anticipated with the solar panels and antennas, but not of the magnitude encountered. These structures, which could be damaged by pencils falling out of technicians' pockets, a slip of a knuckle as a screw was tightened, etc.—prompted several corrective steps. Technicians were trained to avoid all common accident pitfalls. A movie depicting many of these was prepared for showing to both technicians and contractors who would handle the structures. And special procedures were evolved to repair the inevitable damage. Another problem had to do with temperature control; portions of fragile structures had to be made heavier to give adequate heat conduction.

A problem not expected at all concerned stray light being reflected into the Canopus tracker. Although the Canopus tracker was placed in a position to minimize this, analysis and design techniques did not adequately predict reflections from various surfaces, the effects of diffuse illumination, etc. The geometry involved was just too complex.

For this reason, a full-size, representative spacecraft was placed in a darkroom where portions of it could be illuminated by simulated sunlight, and the intensity of reflected light from various portions of the spacecraft was measured at the Canopus tracker. Fixes were made and evaluated during the test. The following corrective actions were taken: Tooling holes in the solar panels were plugged to minimize light shining through and illuminating their backsides. Many shades were added around the spacecraft to block light reflections. The edges of the solar panels and the solar pressure vanes were specially treated to minimize reflections, and the paint on the solar-panel backsides was changed from a dull (diffuse reflection) to a glossy finish.

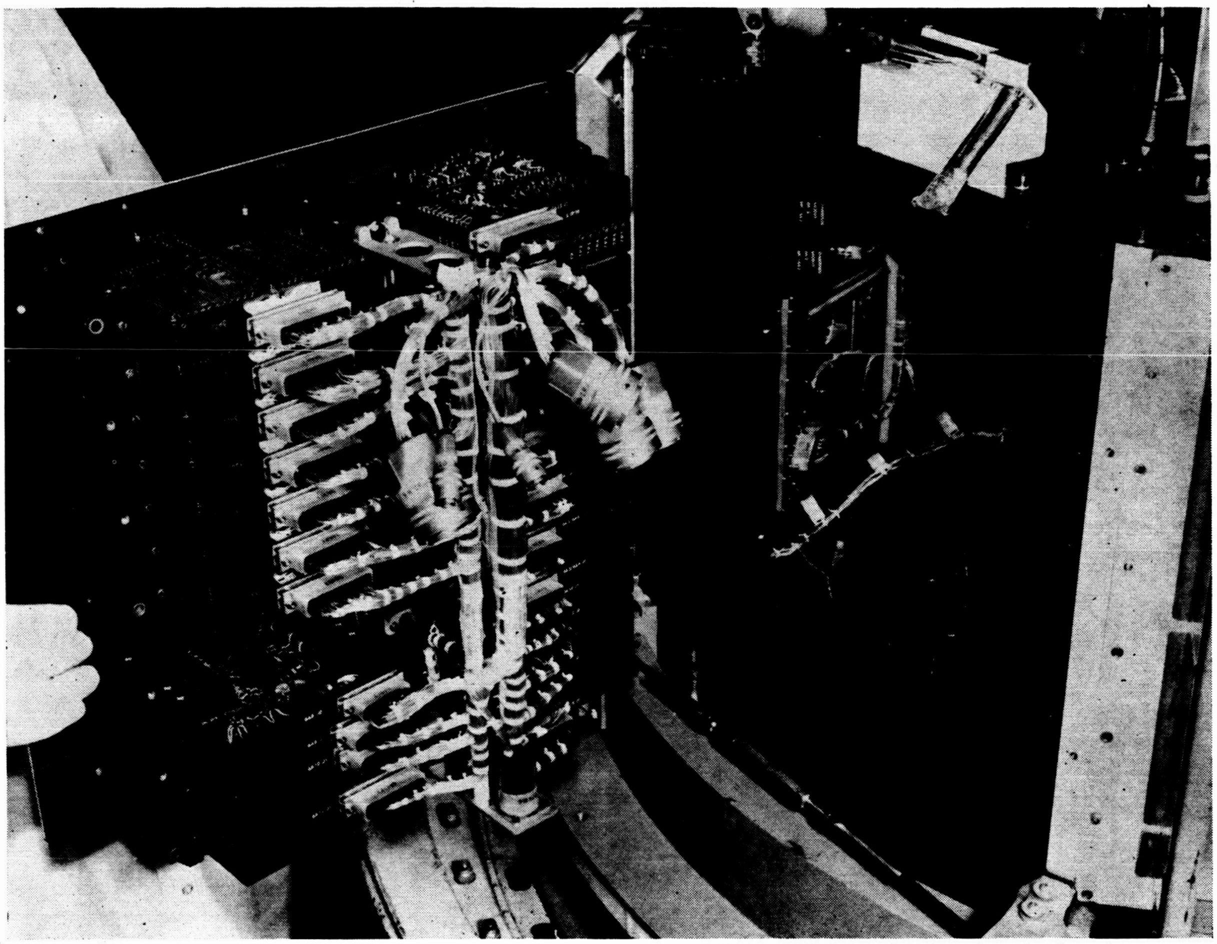
New concepts were fostered and developments undertaken to minimize solar-panel actuator weight. The Ranger and Mariner 2 philosophy of deploying the panels at a slow, controlled rate was continued, but lighter actuators were proposed and attempts were made to remove the potential hazard to the spacecraft of fluids leaking. Two actuators—one patterned after a telephone dial and another patterned after a rotary-vane motor—were devised; they worked, but presented potential reliability problems.

A critical review of reliability and failure modes fostered a different approach late in the program. By this time, a better understanding of all spacecraft requirements permitted nearly unretarded deployment of the panels, with a gentle deceleration to the deployed position. Analysis and tests verified that the unretarded deployment had adequate margins on both the overcoming of frictional torques and on the panel-stopping loads. Negotiation with the attitude-control engineer verified that the cruise damper characteristics could be modified to allow them to perform the stopping of the panels. From this point on, the development progressed with relative freedom from difficulty.

Towards the end of preliminary design, program considerations forced a change of scientific instruments. The new instruments selected were different enough to force substantial changes to both the configuration and the detailed design of the spacecraft structure. This design iteration delayed the completion of preliminary design and had far-reaching effects felt late in the program. Schedule changes shortened development and testing time to the point that it was hard to feed results of tests back into the design. In evaluating the design changes, moreover, not all of the original considerations were reviewed, and this caused trouble, not major, but enough to complicate the development process.

The problem of spacecraft contamination had been of concern right along, but was not really felt until late in the program. Although we had selected polymeric materials thought compatible with the space environment, tests showed many to outgas products potentially deleterious to temperature-control or optical surfaces. This led to a policy of pre-outgassing everything possible, e.g., sewn aluminized-mylar thermal shields, dacron material used as the exterior surface of thermal shields, nylon cable clamps, and many other smaller components. The result of this work was an absence of contamination during qualification testing of the flight spacecraft.

Another contamination problem of concern involved the possibility of dust, dirt, paint flakes, and other particles falling into the view of the Canopus tracker. If illuminated by sunlight, it was reasoned, such particles might be sufficiently



Electronic chassis, typical electronic packaging, and case harness reveal the complexity and order of payload packaging.

bright to confuse the tracker and cause it to lose lock on the referenced star. Investigations indicated that it was not possible to insure that the spacecraft had no dust particles on it. But much was done to lessen the chance of particles from various places. Techniques were evolved to clean and reclean the spacecraft surfaces to the maximum extent possible, to use capped female fasteners wherever possible to minimize the possibility of burrs or thread chips being free to float around the spacecraft, to assemble the spacecraft in a clean facility, and to clean the aerodynamic fairing acoustically to remove as much debris as possible. Despite this regimen, flight data indicates that significant numbers of particles floated by the Canopus tracker, causing its brightness gates to be exceeded and tracking lost. This brightness gate was removed by radio command with no subsequent deleterious effects.

Dot or pellet components were used in portions of the spacecraft to minimize weight and volume. Although these had not been used on a spacecraft before, developing them for flight was felt to be understood well enough to commit them to the vehicle. The development progressed very well; but late in the program, a chemical incompatibility in these units when subjected to elevated temperatures for long periods lowered confidence in their suitability. Although all of these components could not be removed from the spacecraft, and the test conditions under which the problem arose were more severe than those expected in flight, a hurried backup design was pursued for a portion of the system. This backup used welded modules with special interconnections. It raised weight 15% and volume 35%. The modular subchassis approach used as part of the packaging scheme allowed a slight rearrangement of subassemblies to take this change this late in the program, when it was felt to be needed.

Conclusions and Recommendations. By carefully considering operational problems and applying special handling, integrated structure and packaging designs are, in fact, workable. The need for simple and adaptable designs was proven to be necessary in several cases. For instance, the program required last-minute packaging, cabling, and instrument changes, and it was possible to make these and affect only portions of the spacecraft design. Last-minute thermal shield and louver changes based on late information derived from tests were also accommodated with relative ease because provision for so doing had been incorporated into the design.

The use of dampers to minimize vibration, and so permit lightweight structures, progressed with a minimum of problems. This was a significant new development, proven by feasibility tests before it was committed to flight. A requirement to contain as many elements of the spacecraft within a thermally controlled envelope caused few problems and was easily accommodated.

Building-in mechanical alignments, rather than providing adjustments, proved adequate. The fact that flight performance and trajectory corrections have been well within tolerance verifies this approach.

In retrospect, it appears that there were significant numbers of items involving new developments which were of enough concern that they received special and prolonged attention. Most of these were relatively trouble-free. *However,* items not *thought* to be major problems did not receive the same degree of attention and they ultimately *did* turn out to be problems. Moral: Detailed and comprehensive attention needs to be paid to *all* elements of the system, no matter how minor.

Having completed the Mariner 4 design, development, and qualification tests and the majority of the flight, there are a very few things that we would like to do differently. As mentioned earlier, we relearned some old ideas and philosophies—most particularly, to not attempt to develop new items on a project schedule. Only tried and proven techniques should be used. The solar-panel actuators are a prime example wherein a new development, although reasonably successful, was fraught with problems requiring several changes ultimately consuming a great deal of effort. This particular approach seemed easy and understandable, but the fact that it had not been proven before commitment to the spacecraft indicated that there were potential problems for investigation. The use of the dot or pellet component packaging techniques fits this category, and gave similar results.

It is wise, shall we say, not to leave consideration of any element of the system until the last minute. One is continually tempted to stagger developments to best apply effort to more critical problems. However, the interactions among all elements of such a complex design is too great to ignore even the smallest element. The thermal shields on Mariner 4 exemplify this. It was felt they could be added later, almost as an afterthought, without undue complications. When their development was started, many unforeseen problems and interactions were discovered. They had to be redone many times in a hurry before a successful design was achieved. Furthermore, their design was compromised in some areas because the rest of the spacecraft was defined too completely to be changed. We would recommend that at least one cycle of the design of every element should be done at an early date. After this, if further design of certain elements needs to be delayed, at least some awareness of the interactions and problems will have been developed early.

Another recommendation would be to not make structures as light and fragile as Mariner 4's. Having successfully completed the development of these lightweight structures, it is our feeling that we have explored an extreme case where structural considerations in fact became secondary. Having strived for years to achieve lighter and lighter structures, we finally overdid it. A very few pounds of weight applied to these structures would have been well spent in terms of reducing the very careful handling procedures necessary and the many repairs required.

Materials selection and packaging techniques should be controlled much more strongly. For this, a central responsibility is needed to insure that the many hardware designs and associated techniques use acceptable procedures. These techniques and procedures, which take years to understand and assimilate, are not within the capabilities of the scattered organizations nominally selecting both materials and packaging techniques.

The importance of developmental testing cannot be overemphasized. Through such exploratory testing comes an understanding of hardware behavior which cannot be achieved by analysis alone. If this increased understanding is coupled with a tenacious refusal to leave unexplained *any* phenomena, event, or feeling, then few surprises during qualification testing and flight should be found. The recommendation is to pursue developmental testing of *all* spacecraft elements, leaving no stone unturned, and redesigning and retesting until nothing is unexplained. This approach will consume a lot of effort, and the significant monies required for its undertaking are difficult to justify to cost-conscious project managers. But considering the substantial investment of a space program and the consequences of a failure, the cost of such tests are reasonable. Example: The elimination of the Mariner stray-light reflectance test (which was undertaken only because of an engineer's suspicions) would unquestionably have caused a mission failure. Analysis only suggested that there *might* be one problem; testing allowed discovery and solution of many related ones, including the original.

Mariner 4, a good design, may well prove a model for future planetary spacecraft. The techniques used are simple and straightforward, yielding designs lightweight and efficient, with minimum operational problems. Above all, they work.

Mariner 4 Environmental Testing

By WILLIAM S. SHIPLEY and JAMES E. MACLAY
Jet Propulsion Laboratory

A stringent technical and administrative order must protect the planetary project moving toward a fixed launch date; Mariner 4's test program prompted 84 design changes

The Mariner Mars environmental test program included estimating flight environments, establishing general test specifications, reviewing detailed test specifications and selected test procedures, monitoring the status of testing, and certifying environmental requirements. The activities extend throughout evaluation of the flight environment and encounter preparation. Other articles in this issue of *A/A* describe the Mariner Mars 1964 (MM64) program and the spacecraft. Suffice it to say, by way of introduction, that the spacecraft, the mission, and the launch and flight environments transcended previous experiences. To meet program goals, environmental testing would be thorough and painstaking.

Test Program Management. The emphasis placed on environmental testing by MM64 Project Management can be seen in numerous statements in the Project Policy and Requirements Document (PPRD), a directive serving the project in much the same manner as the Constitution serves the U.S. Government. We quote this document often in the following paragraphs. (For a complete picture of MM64 management, see the article by Jack James on page 34.)

The environmental test program was established by the following PPRD statements:

"The project requirements shall be for environmental flight acceptance testing on all PTM (Proof Test Model), flight, and spare spacecraft equipment and environmental type approval (TA) testing on one complete set of TA spacecraft equipment, as a general rule at assembly level. All equipment must be made to pass these tests before it is considered acceptable for flight.

"At the spacecraft system level the PTM tests shall be performed for qualification of the system design, analogous to the TA tests at the assembly level. The flight spacecraft system tests determine the flight acceptability of each spacecraft. It is the over-all responsibility of the Spacecraft Systems Manager to establish the criteria of acceptance for these two test programs."

The Spacecraft Systems Manager was supported in the management of these test programs by an Environmental Requirements Project Representative from JPL's Environmental Requirements Section. Besides this staff function, the Environmental Requirements Representative was responsible for the technical development of test requirements, a line function.

During preliminary design of the spacecraft, a General Environmental Test Specification was prepared for the assemblies, and approved by the Project and Spacecraft Systems Managers. This General Specification established environmental conditions for tests (in some cases defining procedures for establishing the conditions), but placed no explicit requirements on the equipment to be tested.

Immediately following definition of the spacecraft's subsystem elements, an equipment list was issued identifying the equipment configurations that were to be tested as assemblies (to satisfy the General Specification) and subsequently stocked as replaceable spares. Engineers responsible for the development of the assemblies then

Mariner 4 Environmental Testing

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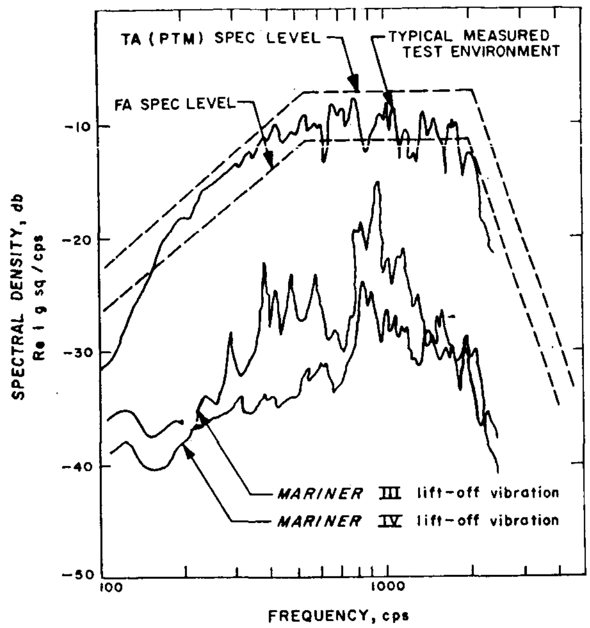
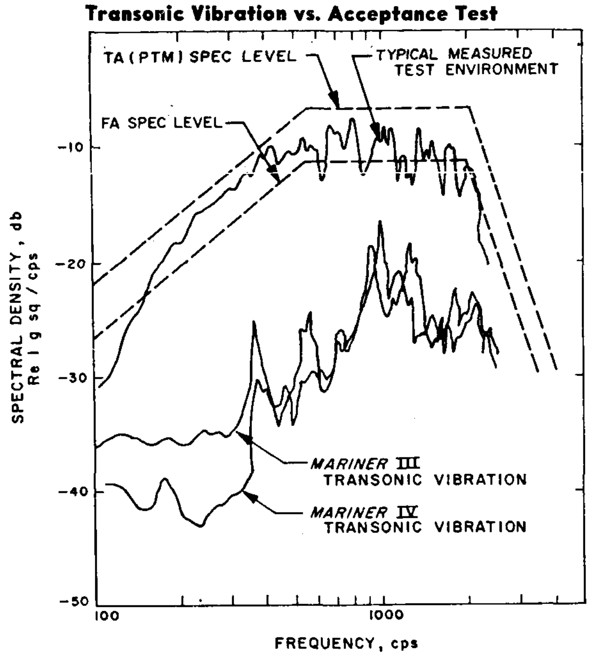
MARINER ENVIRONMENTS—TEST VS. FLIGHT

Launch vehicle: Atlas/Agena D

Liftoff Vibration vs. Acceptance Test



Transonic Vibration vs. Acceptance Test



prepared Environmental Test Specifications for them. These referred to the General Environmental Test Specification for environmental conditions, specified the performance requirements for each environment, and identified the equipment required to support the test. These Environmental Test Specifications were submitted to the Environmental Requirements Section for approval with regard to meeting the intent of the General Specification. Failure to obtain this approval required a waiver from the General Environmental Test Specification approved by the Spacecraft Systems Manager. An approved test specification or a waiver was required before any formal environmental testing. The Environmental Test Laboratory provided the control "Toll Gate" to enforce this requirement. At the conclusion of the testing of an assembly, an Environmental Test Summary report was filed with the Environmental Requirements Section for approval and reporting to the Project Manager, in accordance with the following guideline:

"For each serial number of each equipment group subjected to Environmental TA and FA Tests, a test report is to be made on a Test Results Summary Form. . . . The Summary Form is intended to keep a single and accurate record of the tests performed. The Summary Form is to be completed by the hardware supplying Division and supplied to the Environmental Requirements Project Representative immediately upon the passing of the tests. Included in the Test Results Summary Form will be the following items:

(1) Tests covered and completion dates. Reference Environmental Test Specification Numbers.

(2) Signature of the cognizant engineer or person supporting test from the Division and signature of the QA representative if monitoring the test.

(3) Environmental Lab's test report form number (a copy of the test report is to be submitted if testing is done off the Laboratory).

(4) List of all applicable failure reports.

(5) Serial numbers, reference designations and accumulated operating time.

"The Environmental Requirements Group shall be required to collect and appraise these reports, monitor the testing progress, and prepare for submission to the Spacecraft Systems Manager a Project Test Results Manual based on this information."

For the systems-level environmental tests, the General Specifications for the flight spacecraft were established, approved by the Spacecraft Systems and Project Managers (again defining environmental-test conditions), and incorporated into the appropriate portions of the Test and Operations Plan. These portions of the Test and Operations Plan required approval of the Environmental Requirements Section as did the assembly Environmental Test Specifications. Procedures were then generated by the operations personnel for the operation of the spacecraft and by the Environmental Test Laboratory for the control and monitoring of the test conditions. The latter procedures were reviewed by the Environmental Requirements Section. The systems-level environmental tests were witnessed, and the test conditions analyzed, by Environmental Requirements personnel for the Spacecraft Systems Manager.

Release of these documents counted as project milestones, and their status duly monitored. The status of all testing was maintained and reports made in accordance with scheduled milestones (see Jack James' discussion, page 34).

The Environmental Test Specifications will be recognized as material included in Part IV of "De-

ASSEMBLY-LEVEL TESTING

TA TEST LEVEL	FA TEST LEVEL
Bench Handling	
Free-fall corner drop	N.A.
Drop Test	
Height variable to weight	N.A.
Transportation Vibration	
1.3 g, 2-35 cps	N.A.
3.0 g, 35-48 cps	N.A.
5.0 g, 48-500 cps	N.A.
Explosive Atmosphere	
Fuel and air during assembly operation	N.A.
Humidity	
75% humidity and varied temperature	N.A.
Shock	
Five 200-g, 0.7 ± 0.2 millisec pulses, three axes	N.A.
Static Acceleration	
± 14 g, three axes, 5 min	N.A.
Vibration-Low-Frequency (All Assemblies)	
± 1.5 in., 1-4.4 cps, 3 min; 3-g peak from 4.4-15 cps	N.A.
Vibration-Complex Wave (Assemblies ≥ 10 lb)	
16.4g rms noise: 3 min	9.0-g rms noise: 6 sec
5.0-g rms noise plus 2.0-g rms sine, 15-40 cps 9.0-g rms sine, 40-250 cps *4.5-g rms sine, 250-2000 cps 600 sec	3.0-g rms noise plus 1.5-g rms sine, 15-40 cps 6.0-g rms sine, 40-250 cps **3.0-g rms sine, 250-2000 cps 200 sec
Vacuum/Temperature (9g rms noise: 6 sec)	
−10 C (14 F): 4 hr	0 C (32 F): 2 hr
75 C (167 F): 12 days	55 C (131 F): 40 hr
$<10^{-4}$-mm Hg	$<10^{-4}$-mm Hg
Thermal Shock (for External Assemblies)	
75 C to −46 C (167 F to −50 F)	N.A.

*(9.0 g for assemblies ≤ 10 lb). **(6.0 g for assemblies ≤ 10 lb).

TEST PROGRAM RESULTS

Test	No. Tests	No. Failures
TA Test Summary (Assembly Level)		
Bench handling	39	0
Package drop	39	0
Transportation vibration	85	0
Humidity	51	5
Explosive atmosphere	19	0
Shock	116	3
Static acceleration	89	0
Low-frequency vibration	90	6
Complex-wave vibration	154	24
Vacuum/temperature	95	19
Thermal shock	28	1
Total	805	58
FA Test Summary		
Vibration	538	26
Vacuum/temperature	310	28
Special thermal	21	1
Total	869	55

sign or Detailed Specs" in a standard military system. The need of separate specifications may be questioned by some. There are specific advantages to individually identifying this test documentation:

1. Separation permits it to flow through a special approval channel, where it can be handled expediently yet with appropriate technical consideration.
2. It relates test requirements to the test configuration, which often differs markedly from the design manufacturing configuration.
3. It is well suited to status monitoring carried throughout the program.

This enumeration of procedures may strike the reader as tedious. Such procedures, we agree, do not kindle romance, but they do constitute the control essential for confidence in the excellence of a test program.

Test Requirements. The General Environmental Specifications for MM64 Spacecraft Assemblies are summarized in the table at the left. The Type Approval (TA) tests were applied to samples of each assembly to qualify the designs. Flight Acceptance (FA) tests were applied to each flight item as part of the certification of flight readiness. Environmental tests were likewise performed on the complete Proof Test Model (PTM) spacecraft for design qualification and on the flight spacecraft for certification. The table on page 45 summarizes the system-level tests.

The assembly-level TA tests were to qualify the equipment against all environments that the assemblies would encounter, either as assemblies or part of the spacecraft. The TA tests were more severe than the extremes of the expected environments, with a margin to allow for variations in equipment strength.

The assembly-level FA tests, limited to two environments in which equipment operation could be evaluated, and which could be simulated in a well-controlled manner, were intended to be as severe as the extremes of the expected flight conditions. Passage of the FA tests demonstrated the ability of the equipment to perform satisfactorily in the selected environments at levels at least as severe as flight. This information, added to inspection results, provided implicit evidence that the equipment was similar to the TA units and thus capable of performing in all environments.

The PTM testing included all environments modified by the presence of elements of the spacecraft. The assembly TA testing provided qualification against environments independent of the spacecraft, such as humidity, static acceleration, explosive atmosphere, etc. The assembly and the system designs then required further qualification, as a system, against environments with which the spacecraft interacts, such as shock, vibration, vacuum thermal conditions, and environmental RF radiation. In selecting these tests, emphasis was placed on environmental interactions, rather than performance interactions. Protection against performance interactions was provided by the assembly-level tests by monitoring assembly performance against specification requirements and by assembly and subassembly margin testing. Besides providing design qualification, the PTM tests (and assorted system-level mockup tests) provided environmental data to increase the confidence in the adequacy of assembly-level tests or identify deficiencies in the designs, processes, or materials.

Flight-spacecraft tests gave confidence in the ability of each complete spacecraft to perform its function in the extremes of environments expected to interact with the system.

JPL policy is to set FA tests at the 95% level of the flight environment and the TA tests at a more severe level—above the 99% level—to the degree that such definition of the environment is possible. Even such a statistical statement requires elaboration for a specific environment. Consider random noise vibration. Here JPL has followed an approach used by several other aerospace organizations—flight-acceptance testing to a random spectrum constructed to envelop the 95% points of *maximum* power-spectra densities from each 50-cps bandwidth of the existing data. A test based on such a spectra envelope will have a rms level typically 10 db above the mean rms level of the flight data in question. The duration of the test is then set equal to the time that the wideband vibration level is within 10 db of the maximum. Such a test is obviously a more severe experience than will be encountered on any one flight; but experience indicates that there is usually one peak in the power-spectra density from a flight measurement within 3 db of the test level constructed in this manner.

These paragraphs should at least impart the spirit of MM64 environmental-test selection. The sources of information on which the tests were based will now be examined.

Some of the tests were based on military or aerospace requirements—for example, the bench-handling, drop, transportation-vibration, and explosive-atmosphere tests. If equipment was not compatible with these tests, special handling procedures were established and the test requirements were waived. Other environmental-test requirements were based on spacecraft design information, experience, or special studies. The vacuum temperature test requirements for the assemblies were based on experience and on design data. The temperature test-range has been used for JPL spacecraft projects with only minor changes since Explorer 1. The technology of spacecraft thermal design permits a temperature environment well within this range for all inboard parts of the space-

SYSTEM LEVEL TESTING

TA LEVEL	FA LEVEL
Space Simulator:	
Part I—(Systems validation)	Launch through encounter and playback
10 days at 10^{-6}-mm Hg (or less)	250 hr at 10^{-6}-mm Hg (or less)
Space Simulation	
Part II—Temperature-control verification	
108 hr at 30-134 watts simulated solar intensity	134 hr at 30-134 watts simulated solar intensity
Vibration—Sinusoidal	
Roll Axis	Roll Axis
5-15 cps, 1.5-g rms: 1.6 min 15-450 cps, 1.5-g rms: 4.8 min 450-800 cps, 5.0-g rms: 0.8 min 800-2000 cps, 10.0-g rms (and reverse sweep): 1.3 min	20-200-20 cps, 0.5-g rms 3¼ min Two Lateral Axes 20-200-20 cps, 0.5-g rms 3¼ min per axis
Three Lateral Axes 5-150 cps, 0.75-g rms: 4.9 min 150-450 cps, 1.25-g rms: 1.5 min 450-800 cps, 5.00-g rms: 0.8 min 800-2000 cps, 10.00-g rms (and reverse sweep): 1.3 min	
Vibration—Noise	
Roll Axis & Three Lateral Axes	Roll Axis & Two Lateral Axes
Shaped spectra, 18.1-g rms over-all 550-2000 cps, 0.2 g^2/cps: 3.0 min 3 db/octave rolloff below 550 cps	Shaped spectra, 10-7-g rms over-all 550-2000 cps, 0.07-g/cps: 1 min 3 db/octave rolloff below 550 cps
Vibration—Torsional	
Two 69-cps pulses, 205 rad/sec², 0.14 sec 20-150-20 cps, 12.86 rad/sec², 6.0 min 50-150-50 cps, 154 rad/sec², 5.5 sec	N.A.
Acoustic	
Approximately 142-db shaped spectrum, 90 sec	N.A.
Shock	
Shroud V-band release firing	N.A.
S/C separation V-band release firing	N.A.
All S/C pyrotechnics fired	All S/C pyrotechnics fired
Electromagnetic Interference—RF Susceptibility	
Launch-complex RF Agena-telemetry RF C-band beacon	Launch-complex RF Agena-telemetry RF C-band beacon

VACUUM/THERMAL AND VIBRATION TEST RESULTS

Type	Vibration				Vacuum/Thermal				Over-all fail rate
	No. of Tests	Pass	Fail	Failure rate, %	No. of Tests	Pass	Fail	Failure rate, %	
TA	244	214	30	12.3	95	76	19	20.0	14.5
FA	538	512	26	4.8	310	282	28	9.0	6.3

craft. Outboard parts, such as the solar panels and antennas, required special testing beyond the scope of the General Specifications, but completely defined in the environmental test specification for their assemblies. MM64 spacecraft as satisfactory in thermal design was verified by systems-level space simulator tests.

Certain tests, such as vibration and shock, required special studies. At the outset of MM64, vibration tests for the assemblies were based on vibration data from the Ranger project, since both used the Atlas/Agena launch vehicle. Examination of data from other vehicles lead to the conclusion that the spacecraft vibration environment was strongly influenced by the shape of the vehicle nose fairing, or shroud. Unfortunately, the Mariner Mars nose fairing was different than Ranger's. An examination of wind-tunnel data, again from other vehicles, indicated that the Mariner Mars vibration environment could be noticeably different than Ranger's. With little time available for research, the vibration test-levels, spectrum, and duration were changed. In after-the-fact support of this change, scale-model wind-tunnel tests were made by Lewis Research Center and the data used by JPL for estimating transonic acoustic fields. Acoustic-admittance measurements were made on a Mariner shroud and mockup spacecraft, and the resulting data applied to estimates of the acoustic field to predict flight-vibration environment. The results of these studies supported the change, and when combined with test and flight data provided basic information for predicting spacecraft environments.

Although tables on pages 44-45 summarize the basic tests for the assemblies and spacecraft, a number of additional tests were made on selected assemblies as a result of continued study of the environment. These included shroud RF-attenuation and coupling tests to determine power levels to which the spacecraft would be exposed, low-speed (handling) impact tests to investigate the qualification of the complete spacecraft relative to certain features of the operations environment, high-gain-antenna combined heating and vibration tests to qualify the antenna against the aerodynamic heating and vibration environment during the second-stage operation, and electron-radiation tests. These are but a few of many special tests documented in the "Mariner Mars Test Results Manual," a document which in summary form is an inch-and-a-half thick, supported by volumes of test results.

Test-Program Results. One way of evaluating an environmental test program is by examining the failures that occurred and their effect on the hardware flown. The tables on pages 44-45, 47 summarize the Mariner environmental test program.

The following comments on the results of the assembly-level testing provide some insight to the nature of the problems encountered and reaction of the spacecraft hardware to the various environments created for testing.

1. The five failures in TA humidity were due to the selection of an unrealistic, overly severe test environment. The test condition resulted in measurable quantities of water inside the test item. The spec was modified. There were no failures observed in subsequent testing.

2. Of the 24 failures in TA complex wave vibration tests, 14 resulted in redesign and seven were caused by workmanship errors in manufacturing process.

3. Of the 19 failures during TA vacuum/temperature tests, eight resulted in redesign and three were caused by workmanship errors in manufacturing process.

4. Four of the TA vibration failures involved vacuum tubes (a photomultiplier and a vidicon tube); these are the only vacuum tubes in the spacecraft, so this represented an abnormally high-failure incidence. The failures were localized to the dynode structures, which seemed to be quite susceptible in the range of 400-600 cps. Both units eventually passed TA vibration test, one by making the dynode structure more rugged and the other by isolating the vacuum tube itself.

5. Of the 55 FA failures, 14 resulted in redesign; the rest were caused by workmanship flaws or operator errors.

6. The high incidence, proportionally, of failures in complex-wave vibration and vacuum/thermal tests indorse the selection of these environments for FA tests.

The environmental test program definitely affected improvement in the design and quality of the equipment flown. A total of 84 design changes were incorporated as a result of environmental testing. The trend of improvement between TA and FA testing is evidenced by the reduction in failure rate shown in the table on page 44. The improvement achieved by proper testing is somewhat clouded by the fact that some of the FA failures reflected design deficiencies that would have been detected if TA had preceded FA, and the corrective actions incorporated before production of flight equipment rather than after experiences in testing.

The improvement in equipment quality can more readily be seen by comparing the number of failures encountered in subsystem testing with that encountered in system testing, as has been done in the table on page 47. (It must be remembered that over half of the assemblies FA-tested were later subjected to systems-level environmental tests.) The improvement is even more striking when it is realized that only three of the 16 failures in system-level tests appear to have been

SYSTEM-LEVEL TEST FAILURES

Test	Number of Failures	
	PTM	FA
Space Simulator		
Part I	4	6
Part II	1	1
Vibration	2	1
Acoustic	1	0
Shock	0	0
EMI	0	0
Total	8	8

amenable to detection at the assembly level. The remainder were associated with environmental interactions or ancillary equipment unsuitable for assembly tests.

Independent of the test failures, the technical adequacy of an environmental test program can be appraised by a comparison of the environment to which equipment is subjected during test X with that encountered during flight. Although knowledge of the actual flight environment is limited by telemetry capacity, certain features of the launch dynamic environment and space-flight thermal environment are reasonably defined by existing data. The graphs on page 43 indicate an adequate margin between the vibration from the liftoff and transonic portions of flight and the test requirements. The figures present power-spectra densities of the vibration measured in the axial direction at the attachment between spacecraft adapter and launch vehicle. The test environment for the same axis as the flight measurement is represented by the mean power-spectra density of six measurements of the input acceleration, i.e., at the spacecraft adapter to test fixture interface. In the table at bottom, flight temperatures at 25 locations on the spacecraft are treated statistically and compared to the assembly-level vacuum/temperature test requirements.

Conclusions and Implications. MM64 spacecraft equipment was subjected to an orderly and conservative environmental test program. The rigorous use of clearly defined general-testing requirements enabled attention to be focused on identified problems, and contributed to the timely

FLIGHT VS. TEST TEMPERATURES
All temperatures, C.

Mean temp	1-σ variation	FA test temp	TA test temp
Temperatures at 18 locations interior to spacecraft.			
23	5	55 and 0	75 and −10
Temperatures at six locations exterior to spacecraft.			
23	28	55 and 0	75 and −10 plus thermal transient of 75 to −46

accomplishment of the test program. The regimentation of the testing activity permitted Project Management to approach the launch with confidence in the integrity of the spacecraft.

From the data accumulated in this program a number of conclusions can be drawn, administrative and technical. Although the specter of "limited sample" hangs over these conclusions, they do not defy common sense and indeed support rules of thumb which have been used for many years. Some observations and inferences:

1. Many of the TA environmental tests caused no failures. There is a strong implication here that if a unit passes certain key environmental tests, it will pass all tests that involve the same type of stress. Complex-wave vibration and vacuum/temperature would be the key tests. It is not unreasonable to assume that a barebones TA environmental test program for Mariner could have deleted all the handling environments, explosive atmosphere, and static acceleration and still have produced quality hardware, provided operational restraints (such as shipping containers and conformal coating) were substituted. A cautionary word here; we do not advocate simply dropping certain tests. We suggest that by purposefully structuring certain key tests and acknowledging certain operational constraints, it appears reasonable to drop less-productive tests.

2. The humidity test, which resulted in five failures, was admittedly overly severe. It is further true that one plans for a controlled atmosphere throughout the equipment-use cycle, yet a unit which had waived the humidity test failed in the lab when the air conditioning was shut down for servicing. It seems advisable to retain this test and thereby avoid severe operational restraints.

3. The shock test caused three failures, all latching-relay malfunctions. (There were no relay malfunctions during vibration.) Although this represents a small failure rate, since each shock test involved 15 repetitions and the spacecraft complement contained dozens of relays, there is an indication of fragility of latching relays in a shock environment. This shock test was to simulate the effects of the firing of pyrotechnic devices—a high-*g*, high-frequency phenomenon. Nothing should be inferred from this about other shock phenomena, such as cushioned impact, which is a low-frequency phenomenon. The mechanism of failure is not fully understood; it may even be related to workmanship. Lengthy tests were run on a large sample to determine if the relays continually degraded with repetitive shocks. A few failed initially, but it was not until about the 45 test that others began to fail, at which time the entire sample began to show degradation. (Lest there be any misunderstanding, we do not wish to even hint that enough data exists to propose shock

as a standardized screening test!)

4. Of the 55 FA test failures, 14 resulted in redesign. Although it does not show in the data, 10 of these 14 tests had not been preceded by the TA test. The remaining four failures were identical on each of the four flight units of one subsystem. The nature of the failure was very subtle, and it is felt by the subsystem personnel that the same anomaly should have been seen during the TA test. Discounting these four FA failures then, all 10 of the 10 FA environmental test failures which resulted in redesign had one thing in common—TA testing had not yet occurred. To an organization which produces many copies of each product, this may not seem a very profound conclusion. But an organization which produces one prototype and a few flight units on a short schedule is continually beset to expedite one unit at the expense of others. To such an organization, the price of subordinating the TA testing may be many redesigns, very late in the project.

5. Vacuum tubes (photomultiplier and vidicon types specifically) represent a very vulnerable vibration soft-spot. Our problems were eventually overcome with much special design effort; the tube applications were subject to severe function demands, and many possible fixes would have compromised the functional performance. Neither of the final fixes (ruggedized structure in one application, isolation in the other) would have been difficult in the initial design. Therefore, we conclude that every vacuum-tube application should be viewed from the very outset as a vibration problem amenable to solution.

6. The 10 workmanship errors detected during TA environmental testing indicate the need for a pre-TA test, which could be identical to the FA test, to fortify the qualification concept.

7. The failures detected during system-level environmental testing illustrate certain unavoidable deficiencies in the assembly-level testing: Some units, such as cables, are not amenable to assembly-level tests; the assembly-level test may be inadequate for units located in an environmental extreme (e.g., a cantilever); and degradation can occur between assembly-level FA testing and incorporation into a spacecraft. We definitely advocate system-level environmental testing.

We believe that we can identify events which contribute to the success of Mariner 4 to date, as we write, and others which could have jeopardized success. We believe further that the administrative and technical activities described here will be essential to success in planetary projects with fixed launch dates.

Assuring Quality and Reliability for Mariner 4

By RICHARD A. WELNICK and FRANK H. WRIGHT
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A hallmark of the modern engineering development, thorough and painstaking work accompanied every step of the Mariner Mars Project to assure the quality and reliability of hardware

Many and various constraints of the Mariner Mars 1964 Project, as described by Jack James on page 34, gave a special edge to its quality-assurance and reliability effort. Launches to Mars do not come often.

Project Management established quality and reliability representation to control its detailed QA and R operations. This article for the most part describes its work—a thorough and painstaking investigation of all hardware. Nothing new was invented for the work. Proven techniques were applied to monitor and control the quality and reliability of the Mariner test and flight spacecraft. A grasp of what this entails will serve every aerospace engineer and planner well.

Quality-Assurance (QA) Program. To meet quality-control and inspection needs, a senior QA engineer was assigned to each hardware-fabrication area. Each of these senior engineers supervised a permanently assigned group of inspectors. This organization enabled QA personnel to become acquainted early with the engineering functions and peculiarities of MM64 hardware and packaging concepts. It also facilitated training inspection personnel, since their learning curve could closely parallel the design, development, manufacture, and test of flight hardware. This approach also made possible intimate knowledge of the peculiarities of vendors and their manufacturing and inspection systems and methods.

JPL resident inspectors were assigned to vendors according to the volume of work, criticality role, and QA ability of each manufacturer. Senior QA engineers monitored them through periodic visits. Constant contact resulted in a rapid indoctrination of vendors to JPL ways, and made them aware of the quality level sought.

This field operation was supported by a parts-oriented QA group and a mechanical inspection and metrology group, both located at JPL. And, as a final checkpoint, an independent inspection group provided redundant inspection at the JPL Spacecraft Assembly Facility (SAF), since all hardware delivered, installed, or removed from the spacecraft was controlled and reinspected there.

Quality Systems and Procedures. At the beginning of MM64, a group was formed to unify and standardize procedures and forms and to generate QA basic policy statements. It concerned itself chiefly with the Vendor Surveys (VS), the Quality Assurance Procedure (QAP), and the Quality Assurance Instruction (QAI).

QAPS are JPL-wide policy documents that delineate the guidelines to be followed in all areas under QA discipline. They are based on JPL packaging and process-engineering specifications. QAIs are hardware-oriented documents submitted by the vendor and approved by cognizant JPL QA groups. Over 100 QAIs and Flow Plans were released for use on MM64 hardware. VSs measure the vendor's ability to control the quality of his hardware. Conducted independently of JPL design personnel, but at their request, the VSs enabled JPL procurement, design, and QA representatives to doublecheck the caliber of a supplier. Some 200 VSs were made for MM64.

In-process Inspection. In-process inspection was the most important single action in controlling and measuring the quality level of MM64 hardware.

A QAI and a Flow Plan constituted the basic guideline documents for in-process inspection. They showed where, when, and how individual pieces of hardware were to be inspected and accepted, and by whom.

Acceptance criteria were listed in the QAI for each inspection point, and specific JPL workmanship specifications were referenced when applicable. The QAI and Flow Plan specified points and types of inspection for both JPL and vendors. The QAI resulted in 1500 specified JPL inspection points during the in-process phase of hardware manufacture, plus 256 final JPL inspection points.

Documenting the history of Mariner hardware was a major responsibility of the QA organization. All inspections were documented and record

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copies forwarded to a documentation center. After the Mariner 3 and 4 launches, an audit of the documentation center revealed that the predetermined controlled inspections of the QAIs resulted in the performance of 25,400 inspections of electronic hardware, not including mechanical or part inspections.

Filed in the documentation center are also photographs of each subsystem at time of delivery to the Spacecraft Assembly Facility. QA documentation, filed by serialized subsystem and spacecraft, forms a history of each Mariner spacecraft and its integral parts.

Spacecraft-Equipment Certification at SAF. This process traced the history of the hardware being submitted for assembly, and confirmed and substantiated its condition at receipt. The certification process consisted of a documentation review, an inspection, color photographs of the hardware, disposition of any open items, and sign-off by design and QA engineers. This redundant, toll-gate inspection by QA personnel served to monitor the efficiency of the in-process inspection process and verified the validity of the criteria being used for acceptance during prior inspections.

Quality Assurance at Cape Kennedy (AFETR). Inspectors monitored packing of all spacecraft and support hardware for shipment to AFETR. To provide proper QA coverage at the launch site, a QA supervisor, three mechanical inspectors, and three QA spacecraft inspectors were sent to Cape Kennedy. A controlled-access stockroom was established, and all movement of hardware to or from the Cape was QA-controlled. All packages were closed with tamper-proof seals and were opened only under inspection surveillance. As each spacecraft arrived at AFETR, all pieces of hardware were inspected for shipment damage and humidity effects. After acceptance, complete 100% control of all hardware was imposed, and round-the-clock surveillance of the spacecraft was conducted by QA. All movement of hardware to or from the flight spacecraft was documented, and "spacecraft status" logs were issued after each major event. These rigorous controls continued right up to the launchings.

Post-launch QA effort consisted of monitoring proper storage of spare Mariner spacecraft, flight spares and preparing final spacecraft-status logs.

Reliability Program. The reliability responsibilities for MM64 were concentrated in several areas, such as Design Groups, subcontractor reliability organizations, and the MM64 Project Office, with centralizing authority in a monitor organization. The main tasks of this monitor group covered surveillance and coordination, including attendance at design reviews; development and maintenance of a Problem and Failure Reporting System; a continuous audit of the reliability status of hardware; and monitoring information flow among subsystem-fabrication groups, subsystem and system test-complexes, the space-flight operations area, the launch-vehicle area, and system management.

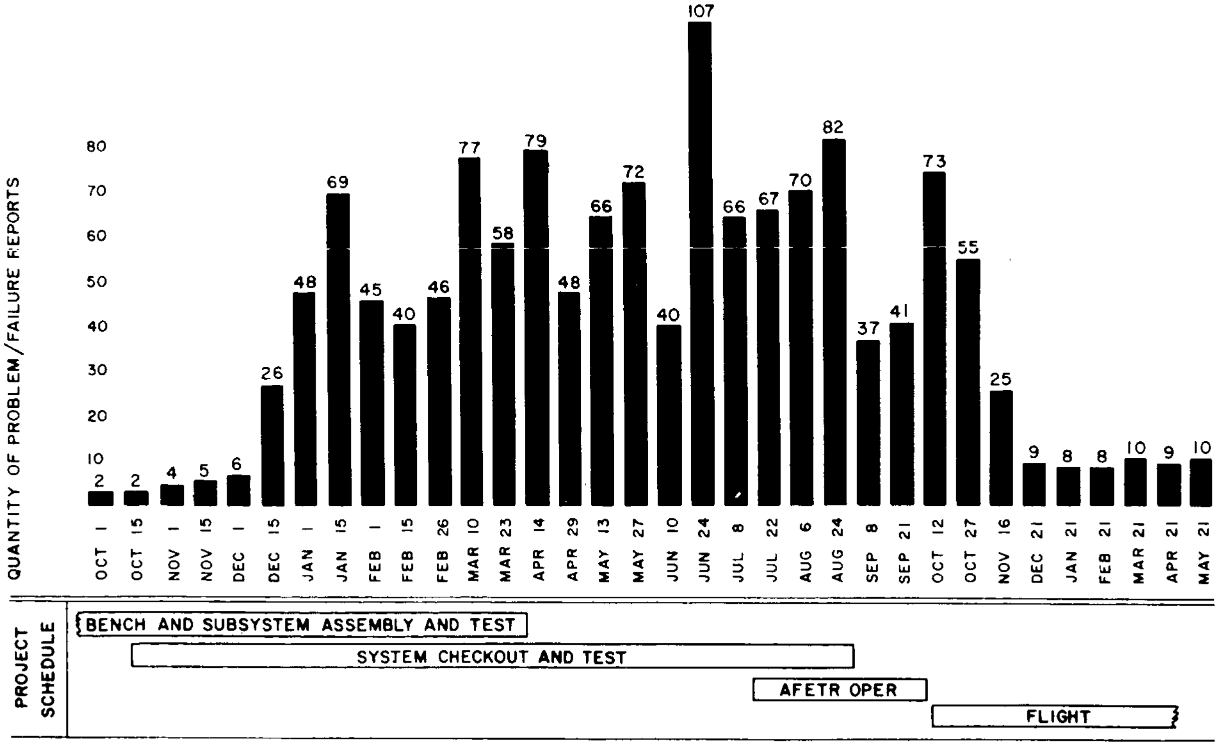
Problem Failure Reporting. JPL's Problem and Failure Reporting (PFR) System for Mariner was planned as a closed loop to give effective notice, distribution, analysis, and corrective action of all reported anomalies. Since MM64 did not involve the quantities of hardware and paperwork associated with a full-scale production program, it was decided not to automate the paper flow, but instead to keep reporting more personal. Anyone could originate a PFR who believed a problem or failure existed. PFRs were written at JPL, at Cape Kennedy, and, during the actual flight, at

RELIABILITY TARGET—MARINER SUBSYSTEM PARTS

This chart will give the reader an idea of the parts distribution on the Mariner spacecraft. The listing includes all the major electronic and electromagnetic part types, but does not give a count of such parts as solar cells and connectors.

Subsystem	Transistors	Resistors	Capacitors	Diodes	Transformers and Coils	Relays	Totals
Radio	119	632	553	220	62	4	1590
Command	296	737	342	702	46		2123
Power	119	375	137	220	44	10	905
Central computer and sequencer	266	1061	256	718	15	28	2344
Data encoder	969	3456	1617	1490	113	8	7653
Attitude control	173	751	388	435	28	36	1811
Pyrotechnic	2	102	28	60	6	8	206
Thermal		52					52
Video storage	181	636	201	266	12	4	1300
Data automation[a]	939	3280	620	3825	2738[b]		11,402
Cosmic ray telescope	193	469	257	253	1		1173
Cosmic dust detector	117	359	139	70	13		698
Trapped-radiation detector	46	134	56	40	4		280
Ion chamber	20	207	68	40	3		338
Scan	65	206	82	104	15	7	479
Plasma probe	243	521	137	375	6		1282
Helium magnetometer	104	376	205	79	20	5	789
Television	254	919	341	412	10	3	1939
							36,364

[a]Includes pulse-width converters. [b]Includes 2640 memory cores.



This Problem Failure Report (PFR) record on MM64 indicates the exacting concern for detail in the program—almost 1500 PFRs were written on test, flight, and operational-support equipment.

JPL Space Flight Operations Facility.

The PFR System hinged on the relationship among each subsystem's cognizant engineer (individual responsible for analysis and corrective action on each PFR), his Section Manager, and the Mariner Spacecraft Project Engineer, who had final authority for evaluating analysis and corrective action on the spacecraft system before PFR closeout. This interaction permitted the cognizant engineer to take the degree of corrective action on his subsystem that he felt necessary, such as having parts-failure analysis performed or starting a design-change action. At the same time, by his sign-off responsibility, the cognizant engineer's Section Manager was made aware of the problems or failures occurring on the subsystems under his control. In making a final sign-off, the Project Engineer incorporated his knowledge of the spacecraft—how each PFR, failure analysis, or design change might affect the total system.

Environmental Testing. Although conducted by another group (see page 42), environmental testing was an integral part of the reliability program.

Two classes of tests were most significant—Type Approval (TA) and Flight Acceptance (FA), both done at the subsystem and system level as well as flight-configuration hardware. The TA is a kind of worst-case test to verify design. TA testing demonstrates that an assembly can satisfactorily operate in worse environment than expected; it is not specifically intended to be destructive. To pass TA, equipment could suffer no performance degradation. TA testing was composed of such environments as handling shock, transportation vibration, explosive atmosphere, humidity, RF interference, shock, static acceleration, and vibration. After successful testing, TA hardware was put on a life-test program; it was not intended for flight use.

FA testing was used on MM64 as a final checkpoint to verify fabrication techniques. The adequacy of the equipment design was established by TA tests; FA tests confirmed workmanship. They were composed of environmental and performance tests similar to TA tests but at nominal levels. The combination of these two levels of environmental testing proved very successful, especially for indicating and eliminating design deficiencies and quality problems.

In all, there were 805 TA subsystem tests and 846 subsystem FA tests performed on MM64 hardware.

Reliability Prediction and Failure-modes Analysis. Jack James comments on this work in his article (see page 34) and cites Planning Research Corp. (PRC) reports. Most of the design changes recommended by PRC concerned increasing subsystem reliability through redundancy. Its reports discussed typical failure modes, design interfaces, and success probabilities for each functional unit.

Design Review. Starting in the spring of 1963, design review was made on all the major spacecraft subsystems, on hardware designed and fabricated at JPL, and at subcontractor plants. The Mariner Spacecraft System Manager chaired the Design Review Board. Typical design-review techniques were used, such as check lists and circuit analyses.

Parts-reliability Program. The reliability margins of the Mariner spacecraft were greatly increased by the JPL parts-screening program. The original requirement was for 100% electronic and electromechanical parts screening for all qualification-test, flight, and spare spacecraft hardware. This was relaxed somewhat in October 1963 due to schedule slippage; of the some 31,000 electronic and electromechanical parts on a single Mariner spacecraft, approximately 95% were screened. For the whole project, some 350,000 parts were screened. The process involved burn-in and parameter-drift screening, although some parts received more-sophisticated attention. Normal burn-in included 168 hr at the manufacturer's rated conditions for each part. Drift screening involved taking each part and testing it under normal conditions against rigid allowable-drift tolerance parameters, derived from parts-qualification data. Evidence of this program's value exists in the successful elimination of large numbers of marginal parts and in the failure-free operation of most of the Mariner 4 equipment during its many months of cruise operations, in interplanetary deep-space missions.

Failure Analysis. Failure analysis was performed on approximately 110 individual parts at JPL during the course of MM64. Failure analysis was also conducted at the plants of major subsystem subcontractors.

Many electronic parts caused problems, and necessitated various failure investigations. For example, a typical laboratory investigation of a failed transistor included thorough electrical characterization at several temperatures, hermetic-seal tests, analysis of gases within the case, electron-microprobe analysis of certain metallic materials, and metallurgical sectioning to examine internal structure. A typical investigation not only required the talents of specialists in electronics, but also specialists in chemistry, metallurgy, and physics.

MM64 made use of a preferred-parts list based on extensive JPL qualification and use data. The list covers the electronic parts for all JPL electronic equipment requiring high reliability—equipment supplied by subcontractors as well as fabricated at JPL. The list involved preparing 487 detailed part specifications.

Concluding Remarks. Presently, reliability analysis is being performed on testing and failure data collected on the Mariner program, both during ground-based activities and the actual flight. These studies are intended to supply information for future JPL space efforts, especially Surveyor and Voyager.

Mariner 4 flight path to Mars

In a planetary mission giving the same order of accuracy as Ranger, this spacecraft demonstrates again the high performance of Earth-based radio guidance coupled with a single-impulse midcourse correction

By N. R. HAYNES, J. R. MICHEL, G. W. NULL, and R. K. SLOAN
Jet Propulsion Laboratory

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On Nov. 28, 1964, the Mariner 4 spacecraft was launched from Cape Kennedy toward Mars by an Atlas/Agena launch vehicle, its primary purpose being to obtain scientific information about this planet.

Mariner 4 carries TV equipment, field and particle experiments, and an "occultation" experiment. The TV equipment has been devised to return 21 pictures of Mars with much higher resolution than Earth-based telescopes give. The field and particle experiments are intended to measure Mars' magnetic field and any trapped radiation associated with it. The occultation experiment will attempt to measure pressure and density in Mars' atmosphere through observation of changes in a radio signal passing through the planet's atmosphere on its way back to Earth.

As we write, the spacecraft is performing satisfactorily, and is heading for a close approach to Mars on July 14, 1965—about 5700 mi. from the surface. This article concerns the choice of point of closest approach, or aiming point, that optimizes data return from scientific instruments and the midcourse guidance and orbit-determination methods for achieving it.

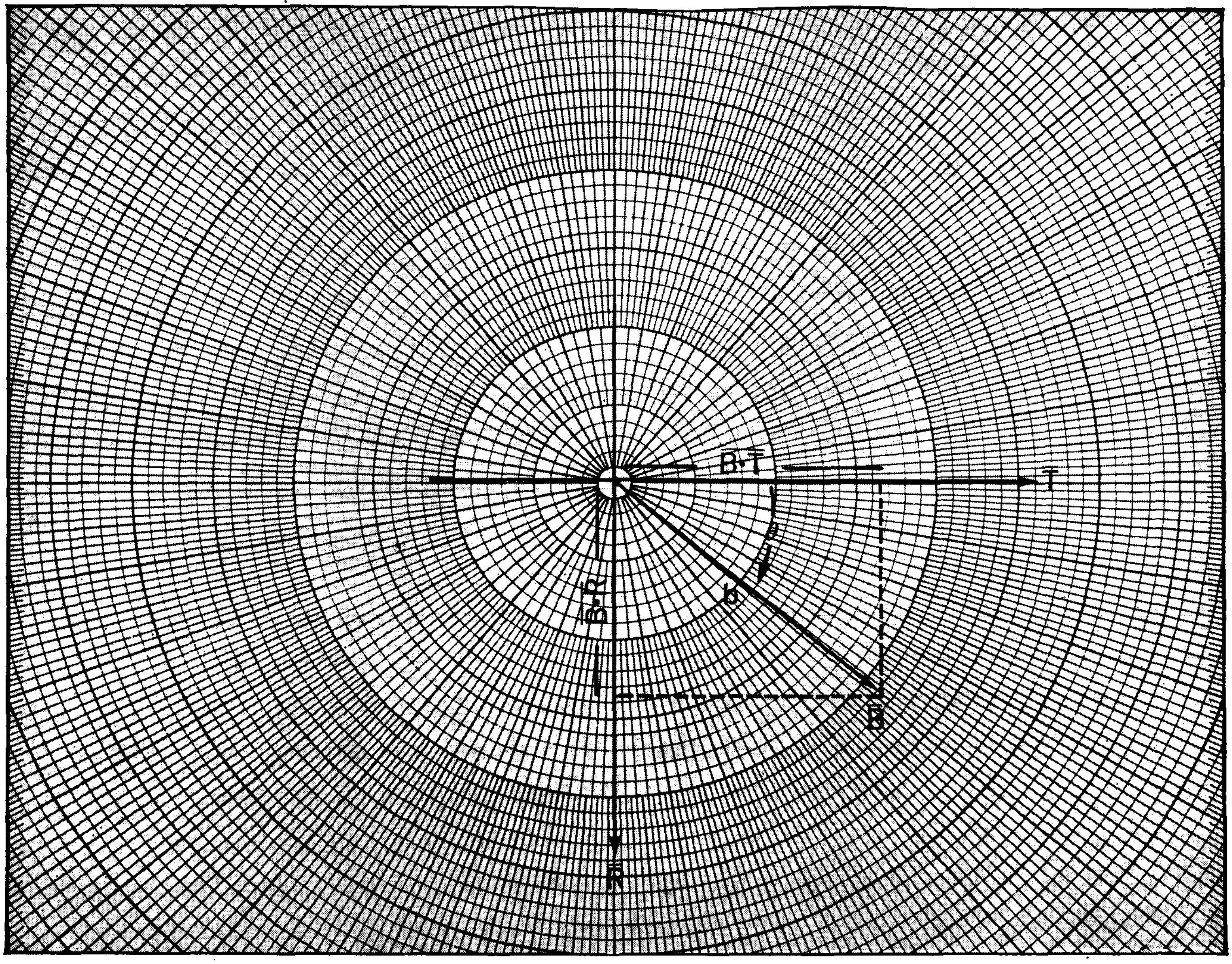
We can begin by defining the aiming-point coordinate system. The aiming-point plane (B plane) is perpendicular to the direction of the approach asymptote of the near-Mars hyperbola and passes through the center of the planet.[6] Within this plane, the aiming point, called the "impact parameter" ($\mathbf{B}$), is defined by the value of the perpendicular distance from the center of the planet to the incoming asymptote. The sketch on page 29 illustrates this geometry. The $\mathbf{T}$ axis is parallel to the ecliptic and $\mathbf{R} = \mathbf{S} \times \mathbf{T}$ where $\mathbf{S}$ = unit vector in the direction of the approach asymptote. The aiming point is then specified by either $\mathbf{B}\cdot\mathbf{T}$ and $\mathbf{B}\cdot\mathbf{R}$ or b and θ, as illustrated.

Operational Constraints and Scientific Value. For a given pair of launch and arrival days, the direction of the incoming asymptote of the areocentric trajectory is approximately constant. This fact permits the operational constraints and scientific value to be represented by contours on the aiming-point plane. Three parameters could be specified independently for each day. Chosen were the time of arrival, distance of the incoming asymptote from the center of Mars, (the impact parameter, b), and an angle specifying the azimuthal position of incoming asymptote on aiming point plane, θ.

As a first step, the time the spacecraft would arrive at Mars was set for a period when the 100-kw transmitter at the Goldstone Tracking Station could "see" the encounter and send commands if necessary at critical stages of the flyby. The time choice determined the longitudinal region of Mars to be viewed by the TV camera and affected some of the operational procedures for assuring a good occultation experiment. Fortunately, the objectives of both the TV and occultation experiments could be met using the particular time selected, and thus the aiming-point selection could be made by specifying the two values, b and θ.

Operational constraints from the mutual interaction between spacecraft and trajectory design specify certain prohibited regions in the aiming-point plane. Piercing these regions causes total mission loss or seriously degrades mission objectives. The chart on page 30 shows the prohibited

NORMAN R. HAYNES (top left) a systems analysis project engineer at JPL for Mariner 4, also heads operations of the flight-path analysis team that determines and corrects Mariner 4 trajectory. **JOHN R. MICHEL** (top right) a mathematician–engineer in the space guidance group of JPL's Systems Div., did Mariner 4 preflight guidance analysis and assisted post-launch maneuver operations. **GEORGE W. NULL** (below left) works in the orbit determination group, performing orbit analyses on the Mariner 2 Venus probe and Mariner 4 Mars probe. **RICHARD K. SLOAN** (below right) serves as project scientist on Mariner 4, and has performed payload definition and other studies for the Ranger, Voyager, and Mariner projects.



The impact parameter, or point of closest approach, is defined by the perpendicular distance from the center of the planet to the incoming asymptote of the space vehicle.

areas in dash line and the boundaries of the individual constraints. The line $CPM = 36$ deg bounds the region within which the angle Canopus-Spacecraft-Mars becomes less than 36 deg at some time. In this region, the Canopus tracker might follow stray light from Mars and roll the spacecraft, causing loss of communication to Earth. The line $SPM = 0$ indicates the region within which the Sun would be hidden by Mars, causing loss of power and undesirable thermal transients. The line $IMP = 10^{-4}$ surrounding the planet bounds the region for which the *a priori* probability of impacting Mars is greater than 1 in 10,000. That is, if a single maneuver were performed to attempt a passage within this region, the midcourse execution errors and trajectory uncertainties would give a probability of impact greater than 10^{-4} and violate the NASA contamination constraint.

Contours in the aiming-point plane express values for the scientific experiments. In each case, the experiment value ranges from 1.0—for trajectories where the chances of fulfilling the experiment objectives are expected to be greatest—to zero, where either no data or no significant data can be expected. The gradations between the extremes indicate the expected partial success. The contours for the fields and particles experiments are time-independent. The television contours depend on arrival date and time, and the occultation contours depend on both launch date and arrival date.

Mariner 4's magnetometer, trapped radiation detector (TRD), ionization-chamber (ION), and cosmic-ray-telescope (CRT) experiments all require the spacecraft to pass within Mars' magnetopause, which has uncertain size. However, similitude arguments plus an assumption of the largest possible magnetic moment of Mars (one-tenth that of Earth) give a rough indication of its maximum extent. The value of each of these experiments increases rapidly with decreasing distance to the planet and can be related to the detectability of an increasingly smaller value for the ratio of Mars' dipole strength to Earth's. The following equations give an approximate relation between the value of these experiments and the aiming point described by b and θ:

Value of magnetometer

$$= 1 - \frac{1}{S}\left(\frac{b}{33{,}400}\right)^3$$

Value of the TRD, ION, and CRT

$$= 1 - S^3\left(\frac{b}{33{,}400}\right)^3$$

With b in kilometers:

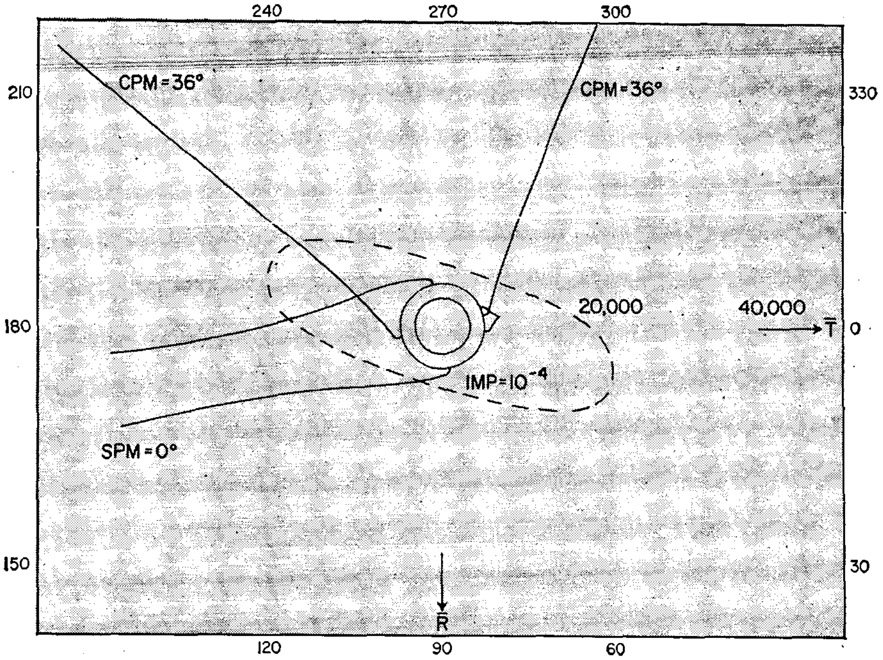
$$S = \sqrt{1 + 3\sin^2\theta}$$

The TV-value contours have been constructed by combining the effects of several television experiment parameters according to the equation

$$v_{TV} = v_R v_B v_F \; (0.5\, v_T + 0.5\, v_{SF})$$

The quantity v_R decreases where the the spacecraft–planet distance during

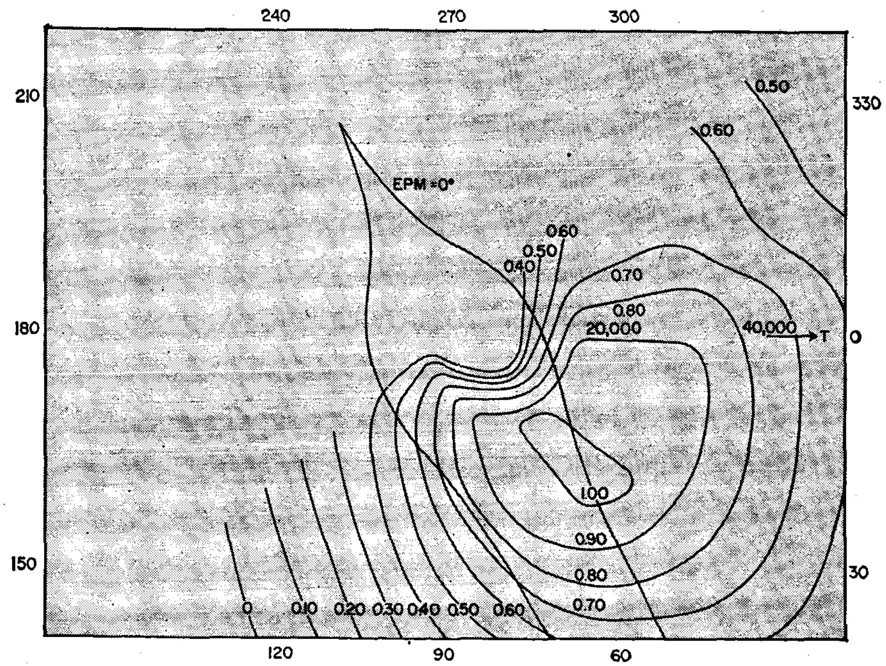
SPACECRAFT CONSTRAINTS AT THE AIMING POINT



the TV encounter sequence is greater than 30,000 km, expressing the decrease in TV-experiment value as distance degrades resolution. The quantity v_B expresses the degradation of picture quality caused by blur; as the spacecraft–planet distance decreases below about 16,000 km, v_B decreases. It also depends on the angle between the camera axis and the spacecraft velocity vector during TV encounter. The quantity v_F accounts for the fact that, as the frame size decreases with decreasing distance to the planet, it will become difficult to correlate TV-picture information with known surface features on the planet.

The quantity v_T depends on whether terminator coverage will be obtained. For θ values greater than about 60 deg, the locus of pictures across the planet foreshortens enough to exclude the terminator. However, v_T does not fall off abruptly at this point because pictures near the terminator are also expected to show surface irregularities, although with less certainty.

The quantity v_{SF} indicates the expected coverage of various surface features and the bright–dark transitions between them. This quantity is time-dependent, since the rotation of the planet changes the surface feature presented to the TV camera. The chart at bottom of this page shows a typical set of TV-value contours. These contour lines cover only half of the aiming-point plane because the planetary scan cannot point at Mars in the other half.



TV value contours and Earth occultation region.

The occultation-experiment value contours obviously fall entirely within the Earth occultation region projected on the aiming-point plane (labeled $EPM = 0$ in the chart just cited). Within the occultation region two types of value degradation can occur. First, the spacecraft passes close to the edge of the planet during occultation, and the magnitude of doppler velocity change will be decreased then. The region for this will be relatively small, permitting the occultation value to be approximated by a discontinuous change from one to zero. Second, with increasing miss distance, the spacecraft signal will be increasingly attenuated by differential refraction in the planet's atmosphere.

To select the final aiming point, it would normally be necessary to compromise among the demands of the individual experiments by weighing each against some value and plotting composite contours to indicate the optimum aiming point for the mission as a whole. It fortuitously turned out, however, that the final aiming-point selection for Mariner 4 was almost independent of weighting, i.e., the three experiments called for approximately the same aiming point. The aiming point finally selected was $b = 12{,}000$ km and $\theta = 60$ deg. As a rule of thumb, the distance of closest approach is approximately 2000 km less than b. The nominal trajectory was therefore designed to pass the planet at about 10,000 km from the center, or about 6600 km from closest approach to the surface.

As it was extremely unlikely, however, that the exact nominal aiming point could be achieved with one midcourse maneuver, it was necessary to decide how much error in the aiming point could be tolerated before a second midcourse correction.

Previous experience with Ranger and Mariner 2 had led to a high confidence in the ability to perform a midcourse correction maneuver successfully. Consequently, it was decided that a second midcourse maneuver would be attempted unless the trajectory following the first maneuver passed through Region III, as indicated in the chart on page 31, bounded by TV and occultation $FOM \geq 0.6$. This region—somewhat larger than Region II, the prime aiming one —represents the tolerance acceptable in the first maneuver to avoid the use of a second.

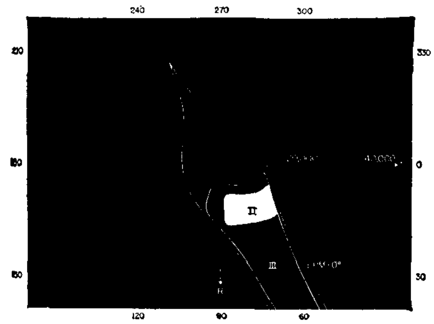
Midcourse-Guidance and Orbit-Determination Results. To determine the probability that the spacecraft could pass through the prime aiming region, it was necessary to analyze the accuracy of the midcourse-guidance system and the associated radio tracking and orbit-determination processes.

Preflight analysis indicated that the errors in executing the midcourse maneuver depended on three factors:

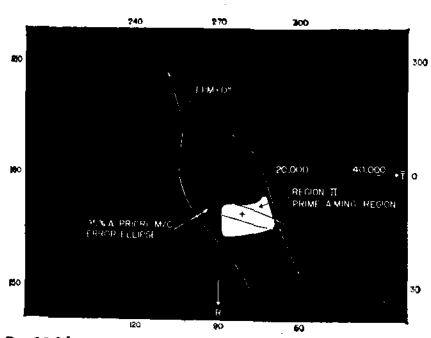
1. The error in executing the midcourse maneuver was approximately proportional to the terminal error to be corrected, and hence directly depended on the injection error from the Atlas/Agena launch vehicle. The ellipses shown in the chart at bottom represent the target dispersion of injection errors and the capability of the full midcourse-maneuver velocity increment to null these errors. Preflight analysis of injection errors and fuel loading has been covered in JPL reports.[1-3]

2. The error depends on the tolerances in the turns required by the spacecraft to orient the midcourse vector in the proper direction and on the resolution and shut-off errors in controlling magnitude of the maneuver. For simplicity and reliability, Mariner 4 uses a timer shut-off rather than more accurate integrating accelerometer of Mariner 2 and Ranger.

MIDCOURSE MANEUVER ERROR-DISPERSION ELLIPSES

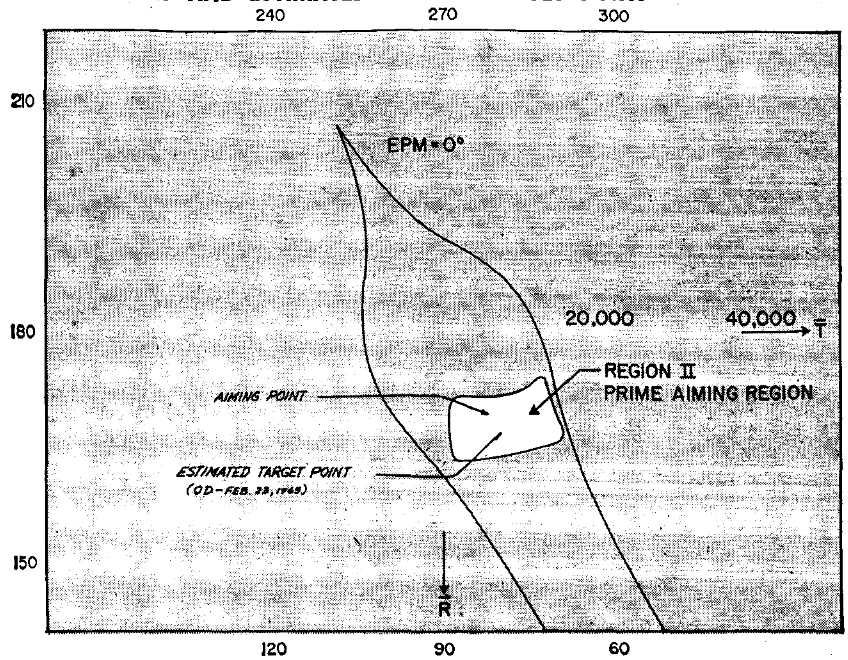


A. Primary.



B. 95% *a priori.*

AIMING POINT AND ESTIMATED ACTUAL TARGET POINT



3. The amount of error also depends on the uncertainty in the determination of the actual trajectory the spacecraft is following. This causes an uncertainty in the amount of miss to be corrected by the midcourse maneuver.

Preflight estimates were that with a 15 m/s correction—corresponding to about 1-σ injection errors—the errors in executing the maneuver would be a 1σ of about 4100 km and that the orbit-determination errors should be below 2250 km after 2 to 3 days of radio tracking. The root mean square of these error sources gives a 1σ error of about 4700 km.

With this error model, it was found that the probability of hitting the prime aiming region was 0.85 with one maneuver. The chart at the top of this page shows the mapping of the spacecraft-maneuver errors into the aiming-point plane, with the ellipse representing the 0.95 probability bound; that is, 95% of a large number of maneuvers should fall within this region.

Following the launch of Mariner 3, it was discovered that the aerodynamic nose fairing failed to separate. It was necessary to make extensive modifications to the nose fairing before Mariner 4 could be launched. The time these modifications took introduced a danger of running outside the launch period. For this reason, it was decided to remove the Agena retrorocket to reduce stage weight and hence extend the launch period a few days. This in turn necessitated biasing the launch aiming point about 600,000 km (to insure that the Agena would maintain a 10^{-4} probability or less of impacting Mars—the purpose of the retrorocket). The new launch aiming point increased the expected size of the maneuver, but it still came within the correction capability of the spacecraft propulsion system.

Tracking Data. The NASA/JPL Deep Space Network (DSN), with stations at Goldstone, Calif., Johannesburg, S.A., and Woomera and Canberra, Australia, has tracked MA-4 since launch (Canberra commenced tracking late in January). The DSN stations transmit a highly stable signal of approximately 2300 Mc to Mariner 4, which multiplies it by a constant and retransmits to Earth. The doppler shift is counted over evenly spaced intervals and divided by the interval to give an average doppler-frequency shift. All the DSN stations have atomic (rubidium) frequency standards for driving station equipment in transmission and reception. These frequency standards are stable to a few parts in 10^{12} over periods of one hour or more. The stations produce tracking data with very small high frequency noise levels and normally no detectable bias due to equipment.

Average doppler shift can be expressed in terms of range rate measured from the tracking station; for S-band stations, 16.5 cps equals 1 m/sec. When the doppler shift is counted over several hundred seconds to reduce cycle roundoff to a negligible amount, the observed high-frequency noise is approximately 0.01 cps—in velocity units, 0.6 mm/sec. Since the spacecraft velocity may range from 3-20 km/sec, the observed the maneuver results can be attributed to this source.

All of the orbit runs listed in the two tables used the JPL Venus-bounce value for the AU: 149,598,500 ± 500 km. Since $\partial B/\partial$ AU = 0.6, the AU uncertainty produces an uncertainty in B of approximately 300 km. The accuracy of Mariner 4's AU solution should improve to less than 1000 km before encounter and with encounter tracking to 500 km or less.

Conclusions. Mariner 4 should pass well inside the prime aiming zone, according to the current trajectory predictions, which the chart on page 32 gives. The deviation from the nominal aiming point is currently estimated to be 2400 km, approximately 1000 km of which was caused by the change in the solar-pressure constant of the spacecraft.

Here we might indicate the accuracy achieved to date by Earth-based radio guidance in the Ranger and Mariner series. Mariner 4, Mariner 2, and Ranger 6, 7, 8, and 9 maneuvers, in terms of miss from the nominal aiming point, gave these results:

Mission	ΔB, km
Ranger 6	36
Ranger 7	25
Ranger 8	27
Ranger 9	7
Mariner 2	17,800
Mariner 4	2430

As a point of interest, a 1 m/s velocity increment in the most sensitive direction can change a lunar trajectory about 200 km, a 1964 Mars trajectory 20,000 km, and the 1962 Venus trajectory about 10,000 km.

It can be seen from the table that the Ranger series and Mariner 4 achieved results of about the same order. The Mariner 4 flight has again demonstrated the high accuracy that can be achieved by Earth-based radio guidance coupled with a single-impulse midcourse correction.

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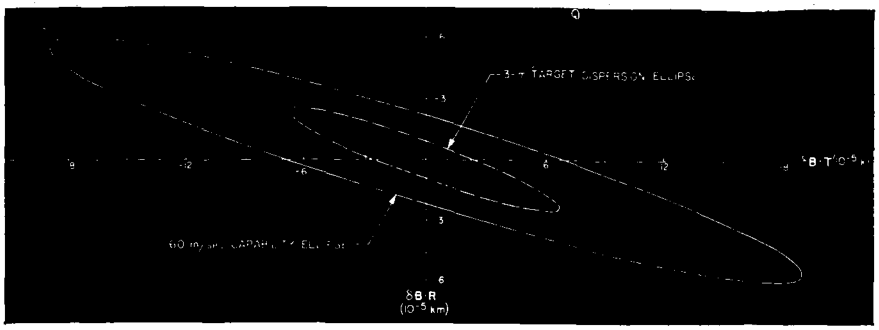
INJECTION DISPERSION VS. MIDCOURSE VELOCITY CAPABILITY



noise represents computer noise for the single precision computer program (ODP) currently in use.

Methods Used in Data Analysis.[4-5] The orbit-determination method is based on the fact that computed values of the doppler observables can be obtained by numerically integrating the equations of motion of the spacecraft and by taking into account the observation time, station coordinates, and speed of light. Partial derivatives of observables with respect to initial conditions are also available. These initial conditions can be included in the solution:

1. Position and velocity at an initial epoch before the data being fit.
2. Station locations with respect to the center of the Earth.
3. Dynamical constants that affect the trajectory, including the astronomical unit (AU), the masses of the Earth, Moon, Mars, and Venus, and spacecraft solar-pressure constant (G).

The observable and partial-derivative quantities are used in an iterative-weighted least squares procedure that adjusts the values of the initial conditions to make the weighted sum of squares of residuals between observed and computed data points a minimum. *A priori* information about the initial conditions is also treated as data and forms part of the quantity to be minimized.

Flight Results. The first estimate of the actual spacecraft orbit, made about two hours after launch, indicated that the spacecraft had been injected into an orbit passing about 250,000 km from the planet—well within the correction capability of the midcourse system. In fact, the launch-vehicle errors gave dispersions which tended to cancel rather than enlarge the initial intentional bias, a fortunate circumstance.

The table just below indicates the refinement in the pre-midcourse determination of the actual Mariner orbit as additional tracking data became available.

TABLE OF REAL-TIME PRE-MIDCOURSE SOLUTIONS

Injection epoch on Nov. 28, 1964, at $15^h\ 07^m\ 57^s$ GMT. All lengths in kilometers.

Last data point (GMT)	Orbit No.	$B \cdot T_C$	$\sigma B \cdot T_C$	$B \cdot RC$	$\sigma B \cdot RC$	TCA[a] July 17	σTCA, sec	SMAA[b]	THETA,[c] deg
Nov. 28 (15^h45^m)	1	−224616.	31587.	−116215.	12304.	0^h10^m	8780	32270	168.
Nov. 28 (17^h20^m)	2	−231403.	6161.	−118516.	3204.	0^h43^m	2520	6772	24.
Nov. 28 (21^h51^m)	2	−224935.	1122.	−122296.	530.	01^h32^m	505	1128	7.
Nov. 29 (0^h21^m)	4	−224160.	697.	−123342.	735.	01^h31^m	133	953	47.
Nov. 30 (05^h14^m)	5	−225039.	1201.	−121934.	447.	01^h31^m	458	1220	169.
Dec. 4 (07^h45^m)	6	−224503.	1040.	−121700.	415.	01^h27^m	458	1079	164.

[a]Time of closest approach.
[b]SMAA: The largest eigenvalue of uncertainties in the *B* plane.
[c]THETA: Angle measured counterclockwise from the *T* axis to SMAA.

Orbits 1-4 were solved for only initial position and velocity assuming other error sources negligible. This is the normal procedure during the first part of a mission to gain computing speed and solution stability, at the expense of a slight inaccuracy in the solution and solution statistics. Orbits 5 and 6 were solved for position, velocity, station locations, and dynamic constants including the solar-pressure coefficient.

The primary tool used to find required midcourse conditions was the Midcourse Maneuver Operations Program, which used as input the latest trajectory estimates and certain mission and spacecraft parameters. The maneuver was computed for execution on any one of seven days in early December 1964. The study showed that all constraints would be satisfied on each of the days and that the maneuver magnitude slowly decreased as a function of time.

The maneuver was executed on December 5 at 6:35 a.m. PST. The spacecraft was commanded to pitch 39.23 deg, roll 156.08 deg, and then to impart a velocity magnitude of 16.7 m/sec. All systems operated well.

The Mariner 4 maneuver used only about 19% of the available fuel; and compared to preflight statistical analysis, the maneuver magnitude was nearly a 1σ case, in spite of the nearly 2σ injection error.

Following the execution of the maneuver, the orbit redetermination process began. The table at bottom gives results for several orbits computed after the maneuver.

In Orbits 6 and 7, it was possible to solve for the solar-pressure constant to about 2% accuracy, the solution differing from pre-midcourse estimates by about 5%. Two vanes mounted at the end of the spacecraft solar panels to aid in attitude control of the spacecraft failed to deploy properly and are currently about 20 deg away from the position assumed before the maneuver. This appreciably increased the effective area of the spacecraft facing the Sun and probably caused the 5% deviation. Since a 5% solar-pressure change corresponds to a $\Delta B \cong 1000$ km, at least a portion of the error in

TABLE OF REAL TIME POST-MIDCOURSE SOLUTIONS

The maneuver occurred Dec. 5, 1964, at approximately $16^h\ 07^m$ GMT.

Last data point (GMT)	Orbit No.	$B \cdot TC$	$\sigma B \cdot T_C$	$B \cdot RC$	$\sigma B \cdot RC$	TCA[a] July 15, 1965	σTCA, sec	SMAA	THETA
Maneuver Aiming Point		(6007)		(10401)		(01^h47^m)			
Dec. 6 (01^h08^m)[b,c]	1	7237	13381	10555	6964	01^h13^m	2262	13672	13.
Dec. 7 (0^h05^m)[b,c]	2	6471	2106	9409	1604	01^h11^m	532	2194	221.
Dec. 14 (22^h)[c]	3	6118	1374	11822	713	01^h15^m	783	1456	159.
Dec. 21 (05^h)[c]	4	6266	1118	11822	522	01^h14^m	594	1177	161.
Jan. 3 (23^h)[c]	5	6784	860	11972	502	01^h11^m	513	949	153.
Jan. 23	6	7411	450	12303	441	01^h03^m	386	587	134.
Feb. 25	7	7434	320	12519	210	01^h03^m	161	349	161.

[a]Time of closest approach.
[b]Computations used information from pre-midcourse about position at time of command, while succeeding orbits do not.
[c]Computations used *a priori* information about spacecraft solar-pressure constant, while others do not.

The Mariner 4 Occultation Experiment

By ARVYDAS J. KLIORE, DAN L. CAIN, GERALD S. LEVY, VON R. ESHLEMAN,
FRANK D. DRAKE, and GUNNAR FJELDBO

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The Mariner 4 Occultation Experiment

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If successful, it will provide a fairly complete estimate of the Martian atmosphere and ionosphere, including data necessary for the design of spacecraft to land on the planet's surface

Improving our knowledge of the atmosphere and ionosphere of Mars has long been an important scientific objective of astronomers and other investigators. Recently, the technological value of this knowledge has been greatly enhanced: More accurate information on the physical properties of the Martian atmosphere will be needed to design survivable landing capsules. These capsules will carry perhaps the most important experiments of planetary exploration—those in search of extraterrestrial life.

The present knowledge of such atmospheric properties as surface pressure and scale height is quite inexact. The surface pressure, as deduced from recent spectroscopic observations,[1-3] is thought to lie between 10 and 25 millibars, in contrast to the 85-millibar figure previously derived from Rayleigh-scattering measurements. The vertical structure of the atmosphere, including the properties of the troposphere and the scale height in the stratosphere, are not accessible to direct Earth-based measurement, and can therefore only be estimated on the basis of assumptions of atmospheric constituents and temperatures. Likewise, the properties of the Martian ionosphere have been open only to postulation of models based in turn on the estimated structure of the Martian upper atmosphere. Current models indicate that the peak electron density might be between 10^{11} and 2×10^{13} el./m^3.[4]

Direct measurement by means of an atmospheric entry capsule[5] would provide most of the lacking information. However, such capsules are unlikely to be flown earlier than 1971, or 1969 at the earliest, and would probably come too late to provide information for design of lander capsules for the early biological experiments. Thus, the only opportunity to improve significantly knowledge of the atmosphere and ionosphere of Mars before these years will be offered by the occultation experiment to be performed this month as the Mariner IV spacecraft passes Mars on July 14.

Description of Experiment. Approximately one hour after its closest approach to Mars, the Mariner IV spacecraft will be occulted by the limb of the planet and will remain in occultation for

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Kliore


Cain

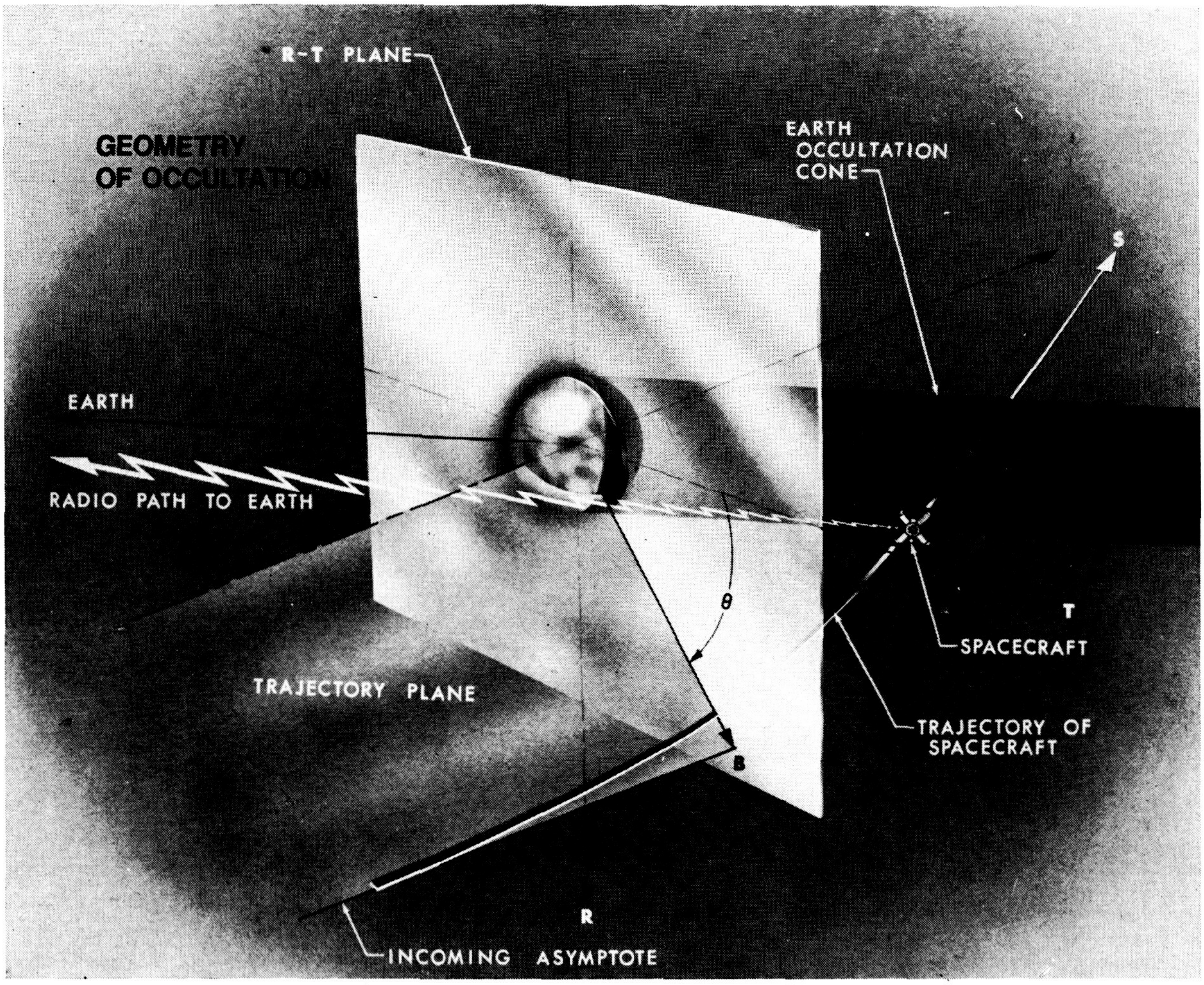

Levy


Eshleman


Drake


Fjeldbo

GEOMETRY OF OCCULTATION



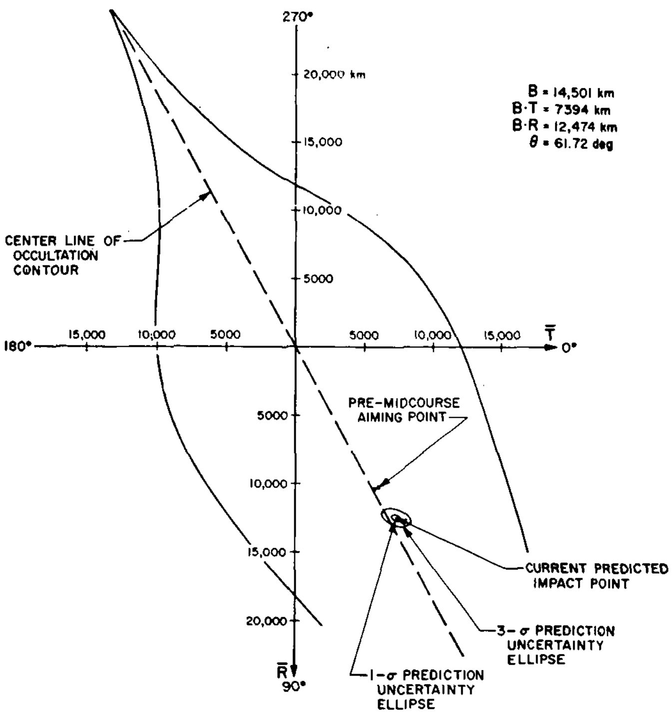
approximately 50 min. Immediately before and immediately following occultation, the spacecraft's 2300-Mc tracking and telemetry signal will traverse the atmosphere and ionosphere of Mars. The changes caused in the frequency, phase, and amplitude of the spacecraft's signal by passage through these media will constitute the raw data of the experiment.

The idea of using a trajectory that would cause the spacecraft to be occulted by Mars was first advanced by investigators from Stanford Univ. primarily in connection with a proposed two frequency radio progagation measurement of the Martian ionosphere. Later, investigators from the Jet Propulsion Laboratory, aware of the refractive effects of the Earth's atmosphere on the phase and frequency of spacecraft tracking signals, concluded that changes in the phase, or frequency, of the doppler tracking signal caused by passage through the neutral atmosphere of Mars could be accurately measured and used to infer some of its physical properties.

This conclusion was based on the demonstrated precision of the radio-tracking technique and the accuracy of the equipment of the NASA/JPL Deep Space Instrumentation Facility (DSIF). Analysis of tracking data from the flights of Mariner II to Venus and the Ranger series of lunar spacecraft has provided significant improvements in the knowledge of the Earth-Venus mass ratio, the masses of Venus and the Moon, the Astronomical Unit, and other physical constants of the solar system.[6,7] The planetary radar experiments[8] have also produced significant results, as well as important advancements in the state of the art of space-communication systems.

The present occultation experiment was proposed in the spring of 1964, and was subsequently accepted as an experiment on Mariner IV, with the authors comprising the investigator team. It is interesting that only a change in the planned trajectory was needed to accommodate this additional planetary experiment. Several of the team members had studied in detail the expected phase and frequency perturbations to the tracking signals due to the neutral atmosphere of Mars, and the feasibility of making the required measurements.[9] Other team members had included consideration of the expected changes in signal amplitude and in limb diffraction oscillations due to

TRAJECTORY IMPACT-POINT IN THE OCCULTATION ZONE



the atmosphere, as well as possible ionospheric perturbations to the signals.[10-13]

A pictorial view of the geometry of occultation appears on page 73. The trajectory of the spacecraft near Mars is shown in relation to the Earth-occultation region, a nearly cylindrical cone with the viewing station as the apex. The $\bar{R}$–$\bar{T}$ plane is normal to the direction of the incoming asymptote $\bar{S}$, and the vector $\bar{B}$ locates the trajectory impact-point in the $\bar{R}$–$\bar{T}$ plane.

The chart at the top shows the location of the currently predicted (March 6, 1965) trajectory impact-point in the $\bar{R}$–$\bar{T}$ plane. The cross indicates the pre-midcourse-correction aiming point, and the ellipses show the boundaries of the 1-σ and 3-σ orbit-determination uncertainties. The accuracy of the midcourse trajectory-correction maneuver is quite remarkable, as the total "miss" is only about 2500 km. Moreover, the predicted trajectory impact-point is so close to the center-line of the occultation contour, that the projection of the flight path, as seen from the Earth, will pass within about 300 km from the center of Mars.

As the spacecraft approaches the occultation region, the presence of an ionosphere and atmosphere will first cause the velocity of propagation of the radio signal to change from that in free space, owing to the non-unity effective index of refraction of the ionospheric and atmospheric medium. Secondly, the radial gradient in the effective index of refraction will cause the radio beam to be refracted slightly from a straight line path. Both of these effects will cause the phase path of the signal at any time, t, to differ from what would be observed in the absence of an atmosphere and ionosphere by the amount,

$$\Delta r(t) = \int_{\Gamma(t)} n\,ds - R(t)$$

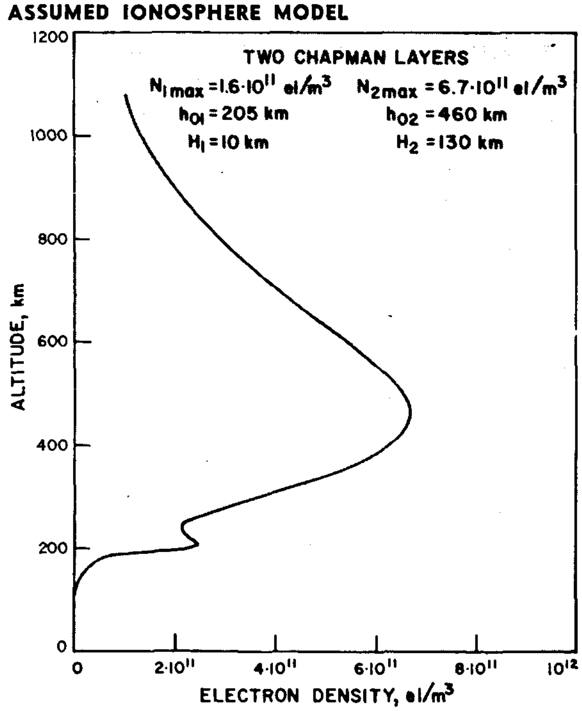
where n = index of refraction, $\Gamma(t)$ = actual path taken by the refracted ray from the tracking station on Earth to the spacecraft at time t, and $R(t)$ = straightline path to the spacecraft at time t.

Thus, if the geometry of the spacecraft trajectory and the spatial characteristics of the index of refraction are known, the phase change can be computed readily. Conversely, if the geometry and the amount of phase change are known at any given time, the spatial characteristics of the index of refraction, and hence of the atmosphere and ionosphere, can be inferred by a process of inverting the above equation or by model-fitting.[9, 10, 13]

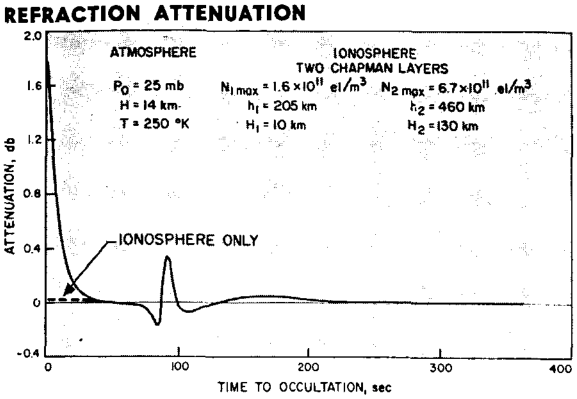
After encounter, the trajectory of the spacecraft relative to Mars at the time of occultation will be known very precisely. It is conservatively estimated that, by using current orbit-determination techniques applied to two-way doppler tracking data taken for 10 days before and 10 days after encounter, the range rate of the spacecraft will be determined with an accuracy of 0.0015 m/sec. Thus, any significant deviation of the received doppler from predictions based on orbit determination will be caused by atmospheric and ionospheric phase effects.

The magnitude of these effects has been computed for a simple isothermal model atmosphere, having a surface pressure of 25 millibars, a scale

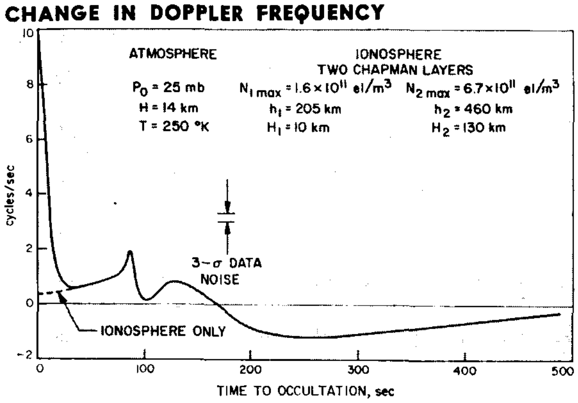
ASSUMED IONOSPHERE MODEL



REFRACTION ATTENUATION



CHANGE IN DOPPLER FREQUENCY



height of 14 km, and the ionosphere described by the graph on page 74. (This graph and the ionosphere portions of the graphs shown on this page are adapted from a Stanford Univ. memorandum of June 6, 1964, by V. R. Eshleman, G. Fjeldbo, O. K. Garriott, and F. L. Smith III.) This ionosphere model is comprised of two Chapman layers with maximum electron densities of 6.7×10^{11} el/m³ at 460 km and 1.6×10^{11} at 205 km. This model is meant to illustrate several possible ionospheric effects. While its exact shape does not result from a consideration of ionization reactions and other characteristics of a particular model atmosphere, the existence of several layers and a peak density on the order of 10^{12} el/m³ is expected from such considerations. Assuming a two-way pass through the ionosphere and atmosphere, the expected change in phase path will be as shown in the graph here at the right. The ionosphere causes a considerable decrease in phase path (about 275 cycles at 165 sec to occultation), which is later partially cancelled by the effect of the neutral atmosphere, acting during approximately the last 30 sec before occultation.

The bottom graph at top shows corresponding change in range-rate doppler. Here the maximum change (about 10 cycles/sec) is caused by the neutral atmosphere, and it occurs during the last 30 sec before occultation.

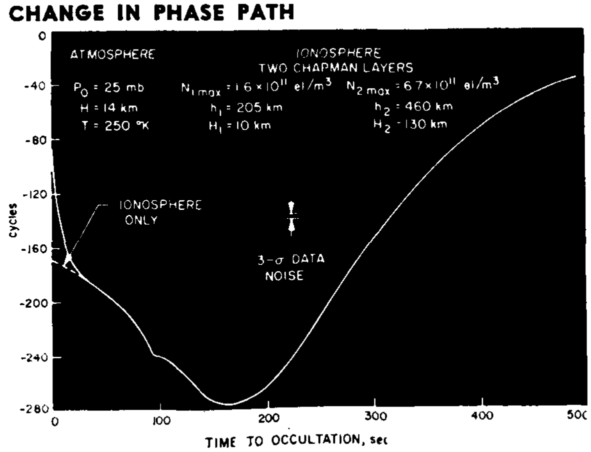
It will be shown in a following section that the current tracking-data noise has a 3-σ magnitude of about 0.3 cps in doppler and less than 2 cycles in phase. This indicates that, even in the case of a very tenuous atmosphere and ionosphere, the effects on phase and doppler will be well within the precision of the doppler-measurement techniques, and should yield not only the near-surface conditions, but also information on the vertical structure of the atmosphere and ionosphere.

The focusing of the waves by the Martian atmosphere will cause changes in the amplitude of the signal transmitted from the spacecraft. The top graph shown at left illustrates the effect. This refraction attenuation would be a serious problem if its magnitude exceeded the available margin of signal level above threshold. However, it is anticipated that the margin available at Mariner IV encounter will be quite adequate to overcome refractive attenuation. In addition, there will be fluctuations in the amplitude in the vicinity of the shadow boundary due to diffraction at the planetary limb, as indicated in the graph on page 77. The atmosphere affects the diffraction pattern by tending to stretch it out, increasing the period of the amplitude oscillations. This "Fresnel stretch" (F in the graph) is also a measure of the atmospheric refractive index at the surface.

Refraction in the atmosphere tends to bend the propagation path around the limb of the planet. This effect is also shown in the graph, where ΔT is the time delay of signal extinction due to atmospheric refraction. The quantity ΔT can be calculated from a measurement of the time that elapses between signal extinction and commencement, assuming that the radius of the planet is known with sufficient accuracy.

Some signal power will also be reflected towards the Earth from the limb of the planet. There is no way of making an exact computation of the reflected power, since one does not know the shape and the electromagnetic properties of the surface at the limb. However, calculations based on a smooth limb show that the reflected part of

CHANGE IN PHASE PATH



the telemetry signal would be too weak to give rise to discrimination problems with the trajectory available for the 1965 occultation. The smooth-limb assumption is probably not very realistic; but a rough limb would usually be expected to reflect even less power towards the Earth than would a smooth limb, since the effective aperture is smaller due to shadowing and because the scattering is no longer coherent. The conclusion should therefore remain the same.

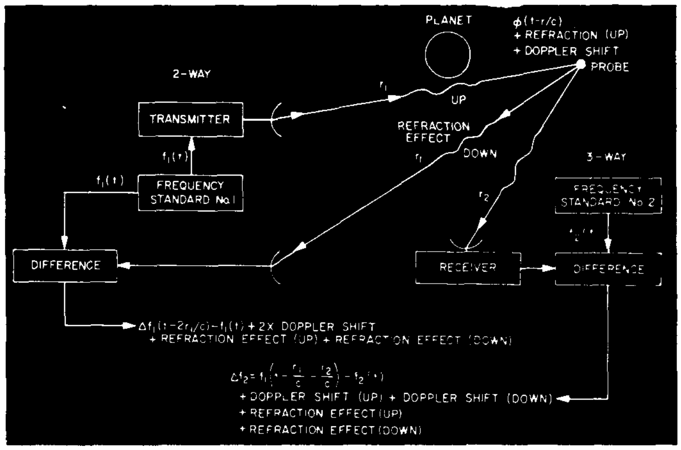
Besides the vertical profile shown in atmospheric-model graph on page 74, there might be important horizontal gradients in the electron-density distribution of the Martian ionosphere, and possibly also dense, sporadic E-type patches of ionization, similar to those found in the Earth's lower ionosphere. If so, one may find that the ionosphere masks the lower neutral atmosphere, making it difficult to determine its physical characteristics near the surface of the planet. However, the rest of the data may be used to calculate the ionospheric electron-density profile, which, again, is related to the neutral constituents at ionospheric heights. In any event, it is expected that the occultation data will yield important new information about the Martian atmosphere.

Instrumentation and Accuracy. The quantities to be measured at the DSIF receiver station during the occultation experiment will be frequency and signal strength as a function of time. These data will then be used to compute path attenuation and phase path change. To understand these measurements, it will be helpful to have a brief description of the DSIF radio system, including the special modifications for the experiment. The block diagram on page 77 depicts the system.

The ground station uses a rubidium standard to drive a frequency synthesizer, which produces a signal of approximately 22 Mc. This signal is then modulated, multiplied 96 times in frequency, amplified, and transmitted to the spacecraft. Its radio system coherently translates the frequency and phase of the approximate 2.11-Gc received signal by the ratio of 240 to 221, yielding a carrier frequency of approximately 2.297 Gc. When the spacecraft receiver is in lock with the ground-station signal, the 2.297 Gc signal is derived from the receiver's voltage-controlled oscillator (VCO), which is phaselocked to the received signal. When no uplink signal is received, however, the signal is derived from a free-running crystal oscillator in the spacecraft. The RF signal is amplified and transmitted from a high-gain spacecraft antenna.

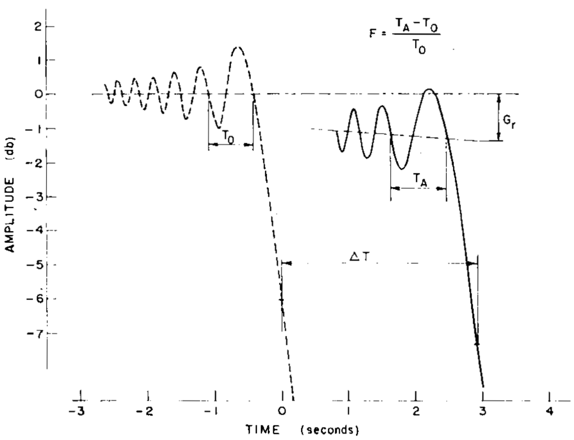
The ground transmitter and receiver system employs an 85-ft parabolic antenna with a Cassegrainian simultaneous-lobing feed. The receiver's front end employs a traveling-wave maser cooled by a closed-cycle helium refrigerator operating at about 4.2 K. After amplification by the maser, the signal is split into two separate receiver channels. The first channel consists of a triple-conversion phaselocked receiver. It is operated in the standard DSIF receiver configuration. This receiver's VCO is kept in phase synchronism with the received signal. By a series of frequency multiplications, divisions, and additions, the transmitter exciter frequency is coherently compared to the receiver VCO to obtain the two-way doppler frequency. The receiver Automatic Gain Control (AGC), which is a received-signal power-level tracking servo. is used to determine received power level. Appropriate AGC voltages will be recorded on magnetic tape, and the doppler count will be digitized. This system will yield frequency and power information in real time. This channel is also used as the sum channel of the pointing system for the simultaneous-lobing antenna.

DOPPLER SYSTEM



The second receiver channel—a manually tuned constant-gain, triple-conversion superheterodyne—is operated in a non-standard configuration. It will simply amplify and frequency-translate the S-band input signal to the audio-frequency region of the spectrum and then record it on magnetic tape. The local-oscillator (LO) signals for this receiver will be derived from the rubidium frequency standard, which will drive a pair of synthesizers. The first synthesizer will produce a signal of approximately 23.4 Mc. This will be multiplied by 96 to produce the first LO at approximately 2.245 Gc. This LO frequency will be periodically stepped to keep the signal in the receiver's pass band. The second and third LOs will be derived from the second synthesizer operating at 19.996 Mc. The output of the third mixer will have a pass band of 1-3 kc, which will be recorded on magnetic tape. The analog information on the magnetic tape will be digitized after the mission for use in a digital computer.

"FRESNEL STRETCH" EFFECT



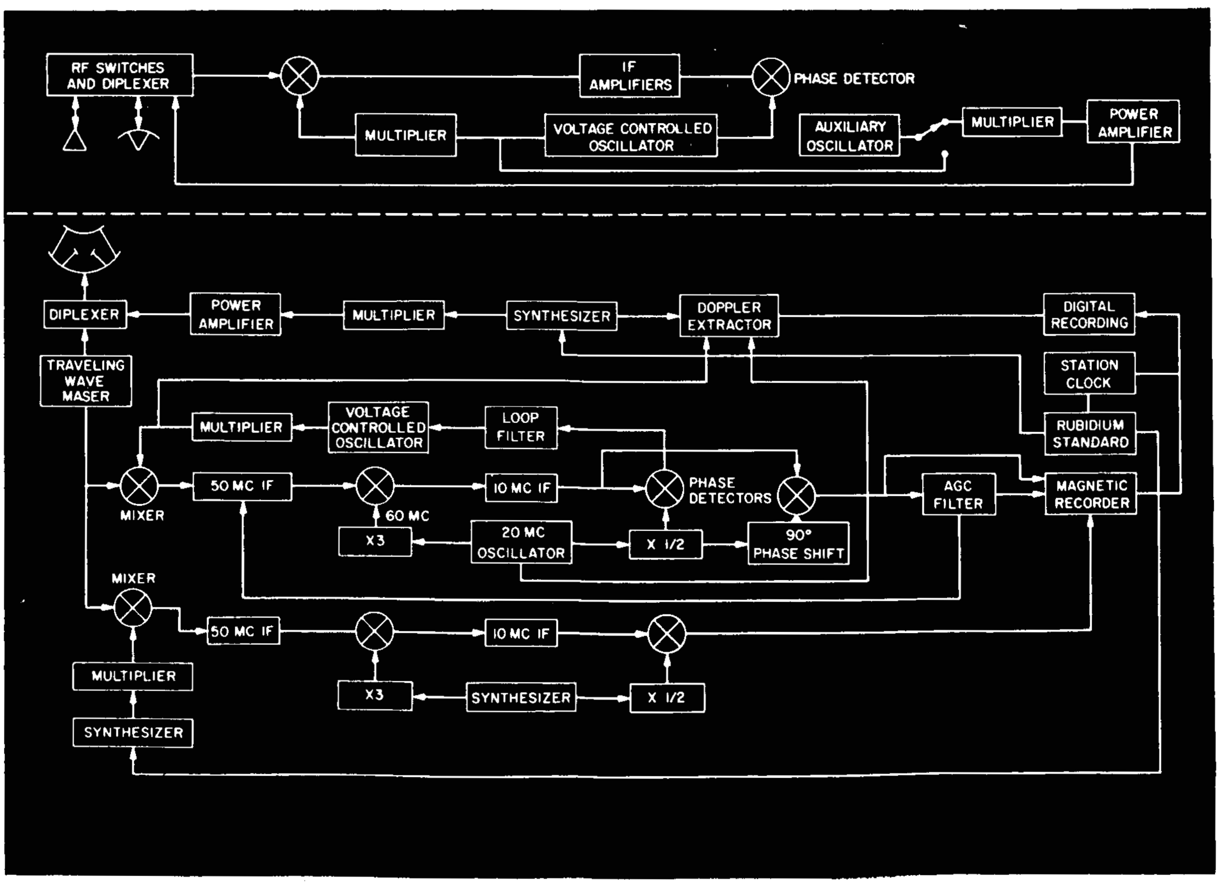
The DSIF doppler system, illustrated schematically on page 76, has the precision to give accurate measurements of phase path changes, from which the atmospheric parameters will be determined; and the changes themselves are expected to be sufficiently large (perhaps 5-20 meters in the atmosphere alone) relative to equipment errors or Earth-atmosphere effects to permit both measurement and interpretation.

The frequency of the standard must be stable during the round-trip transit time of the signal, since any drift will appear as a frequency bias at the time of comparison. That drift will be small has been shown not only by laboratory tests, but also through actual field tests using "passive" monitoring of two-way doppler by a third station. Offsets of frequency, of the order of one part in 10^{10}, are typical; but the short-time variations, which are of interest, are much smaller—less than one part in 10^{11}.

The next contributing error is the unpredictable part of the Earth's tropospheric and ionospheric effects. Unpredictability in both space and time are important. During the period of the round-trip transit time (24 min at occultation), the elevation angle changes by 2.6 deg and the azimuth by 3.6 deg. Thus the up and down rays sample different areas of the Earth's atmosphere. Equally important is the fact that the time difference of 24 min between up and down traversals of the Earth's atmosphere allows temporal changes to take place. These will appear as phase biases in the data. However, it is only the changes in these biases, acting during the time of occultation effects (1 min for the troposphere) that will affect the atmospheric estimates. These changes appear to be small.

Equipment errors, such as phaselock errors and variable time-delays, will also affect the results. Development programs over the past 10 years have reduced equipment errors to a very low level.

RADIO SYSTEM FOR MARINER 4 OCCULTATION EXPERIMENT



The best over-all test for errors comes through comparison of the data with predictions based on the equations of motion of the spacecraft. The sample plot at bottom shows some results of phase-change data (integrated doppler) taken from actual tracking of the Mariner spacecraft on April 2, 1965, when its geocentric distance was about 72×10^6 km. The root mean square (rms) short-period fluctuation is about 2 cm—less than 1% of the expected signal, and the 3-σ fluctuation is less than the two cycles shown in the graph at the bottom of page 75.

The data from which these figures are derived is nondestructively counted doppler, using a sampling interval of 1 sec and multiplying the doppler frequency by a factor of eight before counting to improve resolution. The graph atop of page 79 shows a sample of the doppler residuals (observed minus predicted phase change in one second) resulting from such data. The rms fluctuation is approximately 0.1 cps, leading to a 3-σ noise value of about 0.3 cps, as shown in the second graph on page 75.

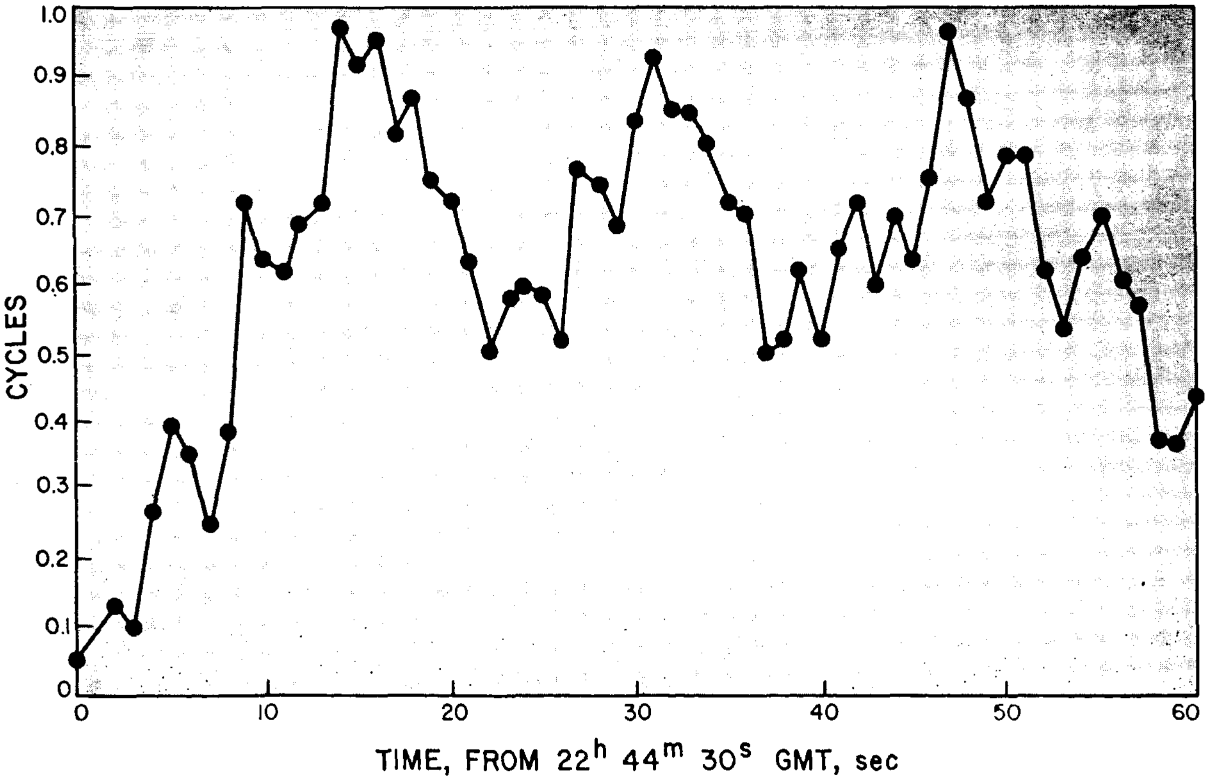
Data Analysis. Analysis will depend somewhat upon the data obtained, and will probably be readjusted and developed during the next few years. The description here concerns procedures to be used in immediate post-flight reduction.

A model-fitting technique will be used to analyze the phase-doppler count data. That is, a complete physical–mathematical model for the data will be assumed, except for the numerical value of certain parameters, which will be estimated. A weighted least-squares estimator will be used. Two separate stages will be involved. First, data will be used to estimate the orbit and mass of Mars, as well as other physical parameters not connected with Mars' atmosphere. Only data unaffected by Mars atmospheric and ionospheric influence will be used in this stage. Second, the Mars atmospheric parameters will be estimated, but the doppler shifts and all other non-atmospheric effects in the data will be obtained from stage one.

For the first stage, current orbit-estimation techniques will be used, but with special care to obtain maximum precision. For the second stage a special digital-computer program is being written. It has the following characteristics:

1. Phase delays at the planet and through the Earth's atmosphere will be computed using ray-tracing techniques. Spherical layering will be assumed in both cases. The models will include constant-temperature-lapse-rate, isothermal, and piecewise-linear refractivity profiles, as well as a Chapman-layer ionosphere.
2. Separate paths will be assumed for up-leg and down-leg, and movement of all bodies during signal transit time will be properly accounted for.
3. Simplified Special Relativity corrections will be made.
4. Current ephemerides of the Earth and Mars will be used.

PHASE-ERROR SAMPLE



formation, the latter available from the scale-height measurements, lead to knowledge of pressure as a function of height.

The pressure, including that at the surface, can be accurately determined even if the scale-height measurements should for some reason fail to give accurately the proportions of nitrogen and argon. This is because the refractivities of argon and nitrogen, at a given pressure and temperature, are virtually identical. So the refractivity data will give an accurate measure of the sum of the partial pressures of each of these gases, even though the relative amounts of each may be unknown.

The most important result will be the determination of the atmospheric density at the surface. Actually, the measurements will give the density at the place on Mars that occults the radio signal. There is a disturbing possibility that this might be, perhaps, a high mountain peak. In analogy to the Earth, this could cause us to observe a density perhaps half that found at most points on the Martian surface. To mitigate this problem, it is very important to observe both the spacecraft immersion and emersion. If both sets of data produce similar results, we will have confidence that the data are valid, and that we are measuring the density near the nominal surface of Mars.

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