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Voyager

FINAL TECHNICAL REPORT TASK B

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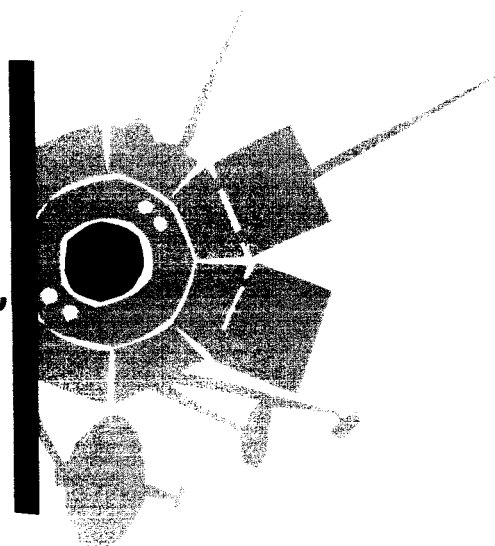
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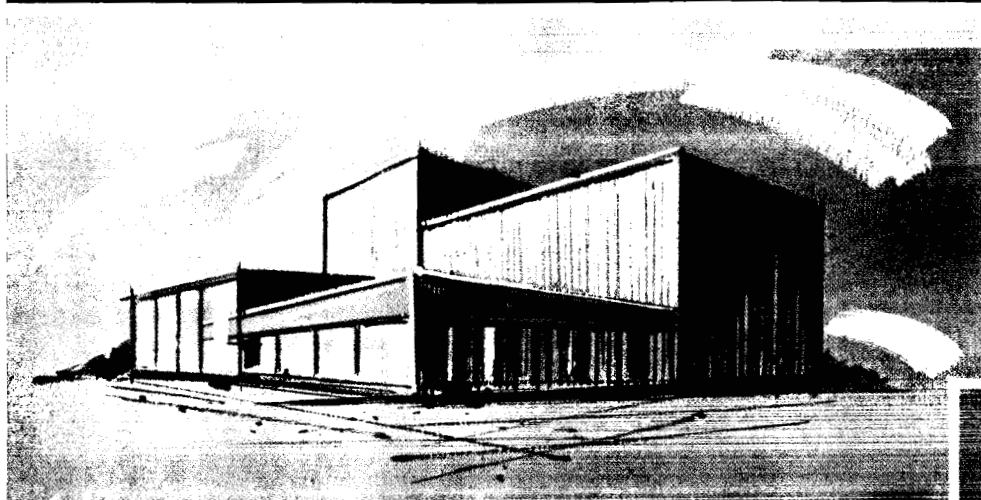
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MANAGEMENT AND
TECHNICAL HIGHLIGHTS

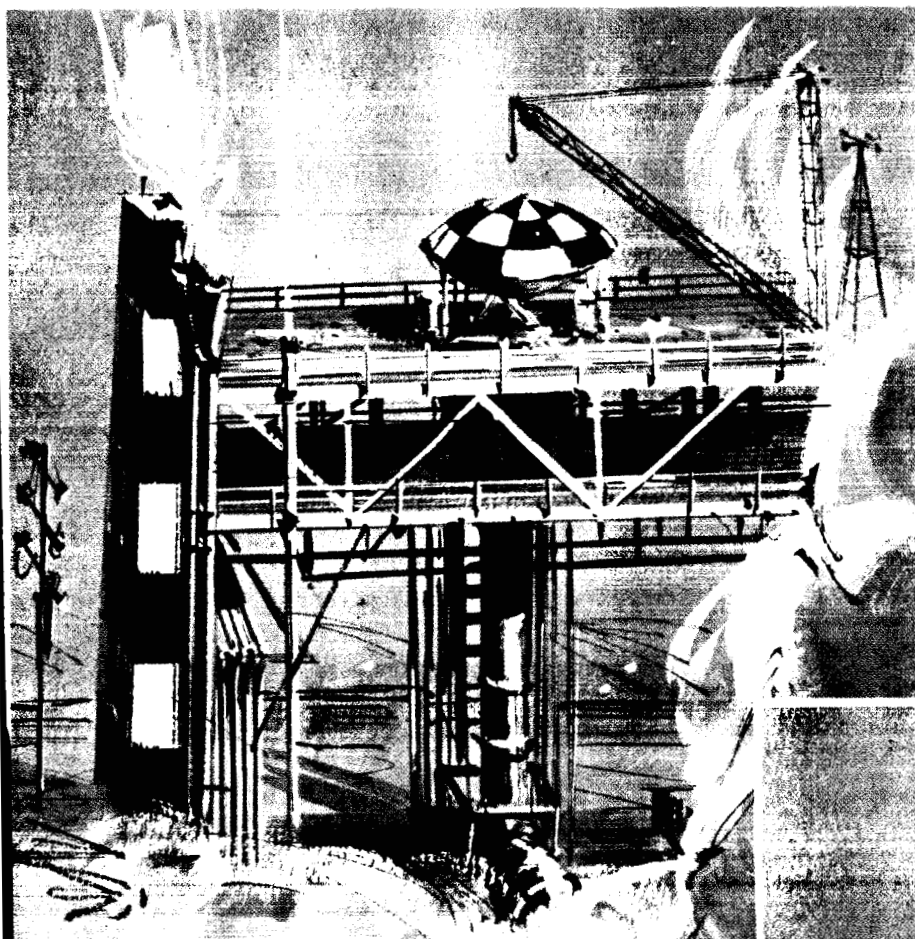
THE BOEING COMPANY • SPACE DIVISION • SEATTLE, WASHINGTON

VOYAGER

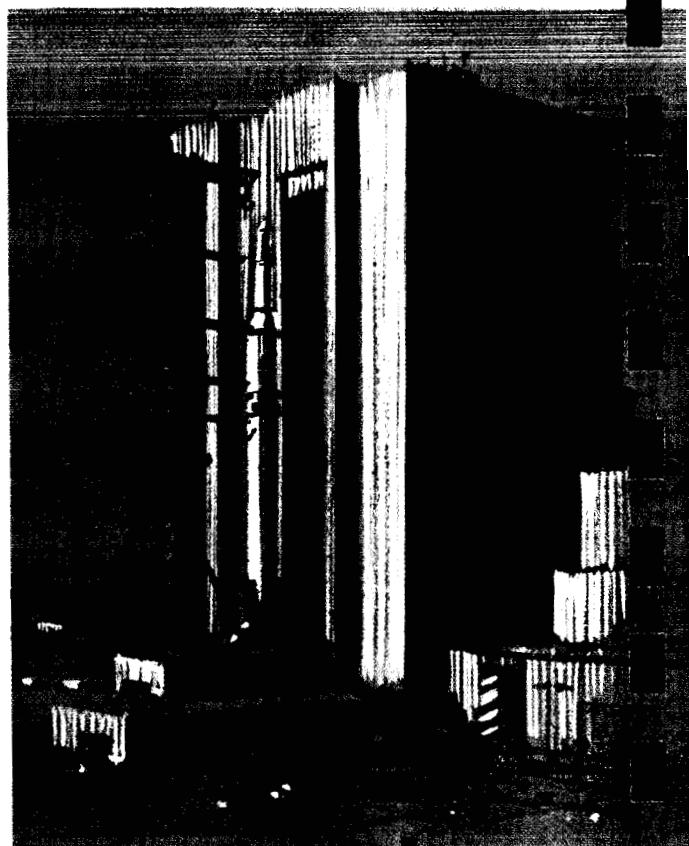




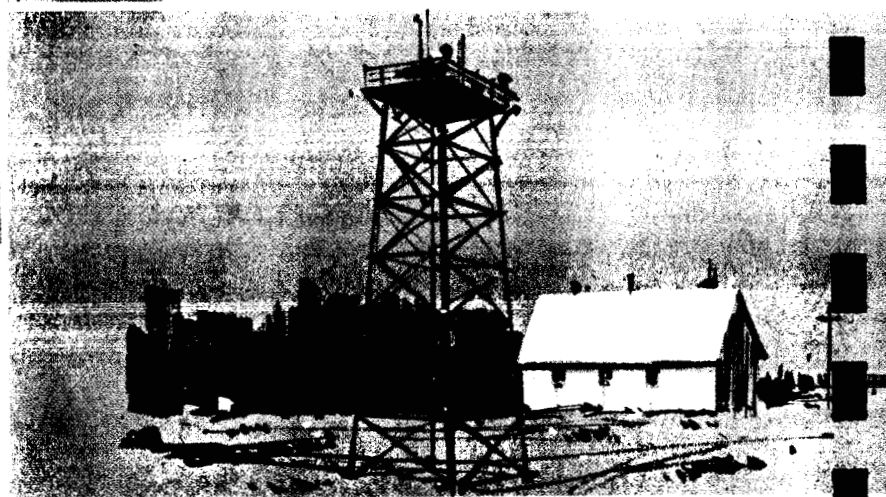
KENT, WASH.



BOARDMAN, ORE.



CAPE KENNEDY, FLA.



TULALIP, WASH.

NASA CR71514

THE **BOEING** COMPANY

AEROSPACE GROUP • P. O. BOX 3707 • SEATTLE, WASHINGTON 98124

January 21, 1966

IN REPLY REFER TO

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena, California

Gentlemen:

The Boeing Company is pleased to submit the technical reports of the work accomplished under Voyager Phase 1A, Task B. Together with the reports of Task A, they represent to us a substantial contribution to our understanding of the objectives of the Voyager Project. As a corollary, it is believed they will demonstrate to you a dedication for, and a capability to perform, those tasks so important to fulfilling the Spacecraft Contractor's responsibilities.

The recently announced delay in the Voyager Program will test the dedication of all parties concerned. Despite our disappointment, we will not let this temporary setback deter our proceeding on a rational basis to be ready when funding levels again allow the program to proceed. It is important to note that the Task B documentation has been submitted as if no change had occurred in the Voyager Program. It should be recognized that corporate and group commitments contained in the documentation, in the areas of facilities and personnel, will be reconsidered when the Voyager program proceeds. At that time, Boeing will update and reaffirm the resources necessary to support the Voyager program.

Because of the cancellation of the Phase 1B, Part II Request for Proposal, we have chosen to highlight some of our management philosophy and organization rationale in a summary document, D2-82709-00. To place this in perspective, we have also summarized the salient features of the spacecraft design. Further, we have postulated some advanced missions, using the 1971 design, for further exploration of the solar system. This latter item is the basis for part of our continuing Voyager work.

Little more remains to be said except to restate that the Voyager Spacecraft System represents to us, more than a new product objective; it is an opportunity to participate in the extension of scientific knowledge in the universe and to contribute to national prestige.

Very truly yours,

THE BOEING COMPANY

Lysle A. Wood

Lysle A. Wood
Group Vice President-Aerospace

SPACE DIVISION
GENERAL MANAGER



GEORGE H. STONER
VICE-PRESIDENT

AUTOPILOT AND
ATTITUDE REFERENCE

R. R. MUELLER
PROJECT MANAGER
AUTONETICS



VOYAGER PROJECT MANAGER
EDWIN G. CZARNECKI



ELECTRICAL POWER



C. I. CUMMINGS
PROJECT MANAGER
ELECTRO-OPTICAL SYSTEMS

TELECOMMUNICATIONS

G. O. MOORE
PROJECT MANAGER
PHILCO



The Boeing Company Voyager Management Team

INTRODUCTION

This document presents a management and technical overview of Boeing's efforts on Jet Propulsion Laboratory (JPL) Contract 95111, Phase IA, Task B. In recognition of the importance of understanding the management task that Boeing hopes to undertake in support of JPL, the first section is addressed to the rationale and management philosophy leading to a projectized organization, *under NAS 7-100*.

The Task B study was directed at defining a preferred Spacecraft System design. To establish a frame of reference for the trade studies leading to this design, a postulated Voyager 1971 mission to Mars was established. Extensive trade studies and analyses lead to a preferred design that has been submitted in the following documents.

Volume A Preferred Design for Flight
(D2-82709-6) Spacecraft and Hardware Sub-
 systems

Volume B Design for the Operational Sup-
(D2-82709-7) port Equipment

Volume C Alternate Designs Considered
(D2-82709-8) for Spacecraft Propulsion Sys-
 tems

The technical highlights of these documents are summarized to acquaint the reader as rapidly as possible with the important elements.

The prime theme of the Voyager Spacecraft System is attainment of a high probability of mission success and the conservatism required to ensure that success. It has been carefully noted that the priority of JPL's competing characteristics is:

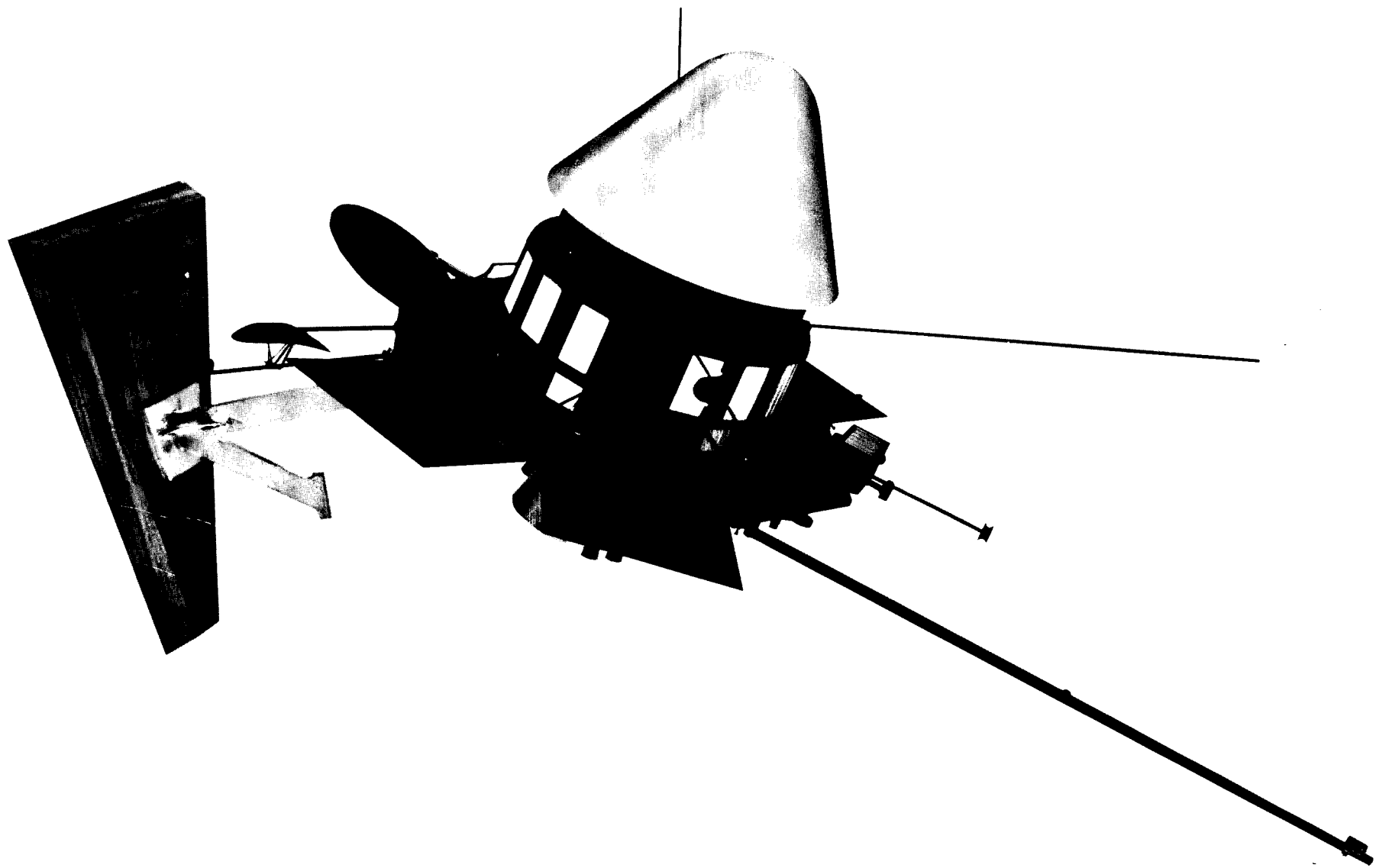
- 1) Probability of success;
- 2) Performance of mission objectives;
- 3) Cost savings;
- 4) Contribution to subsequent missions;
- 5) Additional 1971 mission capability.

The priority ordering of these characteristics requires a close marriage of management disciplines with proven design concepts to accomplish the tasks assigned to the spacecraft system contractor. Highest emphasis for the 1971 mission has been placed on the probability of success, followed by performance of mission objectives and cost savings. Our approach was to select a preferred design based on the first three competing characteristics, and then to determine the extent to which that design could be used for subsequent missions as well as for additional 1971 mission capability. In this way, it is felt that a maximum probability of success for the 1971 mission to Mars was obtained.

A section of this document contains a discussion of the utility of the preferred design when considering expanded missions to other planets. A detailed examination of these data with JPL could lead to a reallocation of payload weight resources, providing a higher probability of success for the 1971 mission to Mars as well as further extending the capability of the design concept to other planets.

Additional technical considerations which preliminary investigation indicated to be desirable in later program phases were discovered during Task B. These are currently in work and will be available, as appropriate, to support future work.

This document is concluded by a brief discussion of the Boeing facilities which will be available for the Voyager Program. Highlighted is the new Boeing Space Center at Kent, Washington, which will house personnel supporting JPL's Voyager Project.



BOEING-SPACE DIVISION

MANAGEMENT HIGHLIGHTS

The Boeing Company is firmly committed to the nation's space program, dedicated to NASA's goals and objectives, and postured to participate as a member of the NASA/JPL team to accomplish successful Voyager missions in 1971 and succeeding years. In executing its responsibilities on the Voyager Spacecraft System, Boeing will apply a philosophy of complete responsiveness to JPL's requirements. The success and growth of The Boeing Company, and in particular, its Aerospace Group, have been largely the result of responsiveness to customer requirements. The experience gained has resulted in a customer-oriented Voyager Systems Branch organization, capable and desirous of responding to JPL's requirements.

The Boeing Company, recognizing the importance of the management tasks associated with the Voyager Spacecraft Systems Project, will highlight its management philosophy and organizational approach in this section. The company practices a decentralized form of management, based on the principle that decision-making authority should be delegated as close as practicable to the operating level. Consistent with this practice, the Space Division of the Aerospace Group has been assigned the task of preparing for and implementing spacecraft system design, development, launch, and mission support activities, which are an integral part of the Voyager Project. These activities will be performed under the direction of Mr. E. G. Czarnecki as manager of the Voyager Systems Branch within the Space Division.

As indicated on the following pages, the full support of corporate resources to Mr. Czarnecki and thus to NASA/JPL has been given by:

- Mr. William M. Allen, President, The Boeing Company
- Mr. J. L. Atwood, President, North American Aviation, Inc.
- Mr. R. O. Fickes, President, Philco Corporation
- Mr. J. C. Wilson, President, Xerox Corporation

It is proposed that an executive council representing the above firms will meet once during,

and again immediately following, Phase IB, and approximately quarterly during Phase II to review progress and to bring top-level attention to major policy and resource problems. Meetings of this council will be rotated among the Boeing team members' facilities so that the council can obtain first-hand information on the progress of the tasks being performed. Boeing has participated in a highly successful precedent for such a council in the Minuteman program.

At Mr. Czarnecki's request, these major sub-contractors have followed Boeing's lead by (1) emphasizing the authority of their Voyager project managers; (2) organizing their Voyager team, insofar as practicable, to parallel the Boeing Voyager organization; and (3) improving their corporate management involvement in Voyager activities by placing their Voyager project managers high within their organizational structure.

To provide perspective to the management environment within which the Voyager Project will be accomplished, it is desirable to discuss in a summary manner the organization of Boeing's Corporate Headquarters, the Aerospace Group, and the Space Division. This discussion will review the organizational structure that has been developed to apply the management philosophy, techniques, and controls required to integrate the company's business base.

CORPORATE HEADQUARTERS

The president guides and controls the business as a whole. The corporate staffs assist the president in establishing common corporate policy baselines for the operating organizations. In addition, they provide advice, counsel, and functional assistance to the president, the groups, and the divisions.

AEROSPACE GROUP

The organization of the Aerospace Group within The Boeing Company is shown in Figure 2-1. The president of the company has delegated substantial responsibility and decision-making authority for operations and program management to the group vice president — Aerospace, consonant with the principle of decentralized man-

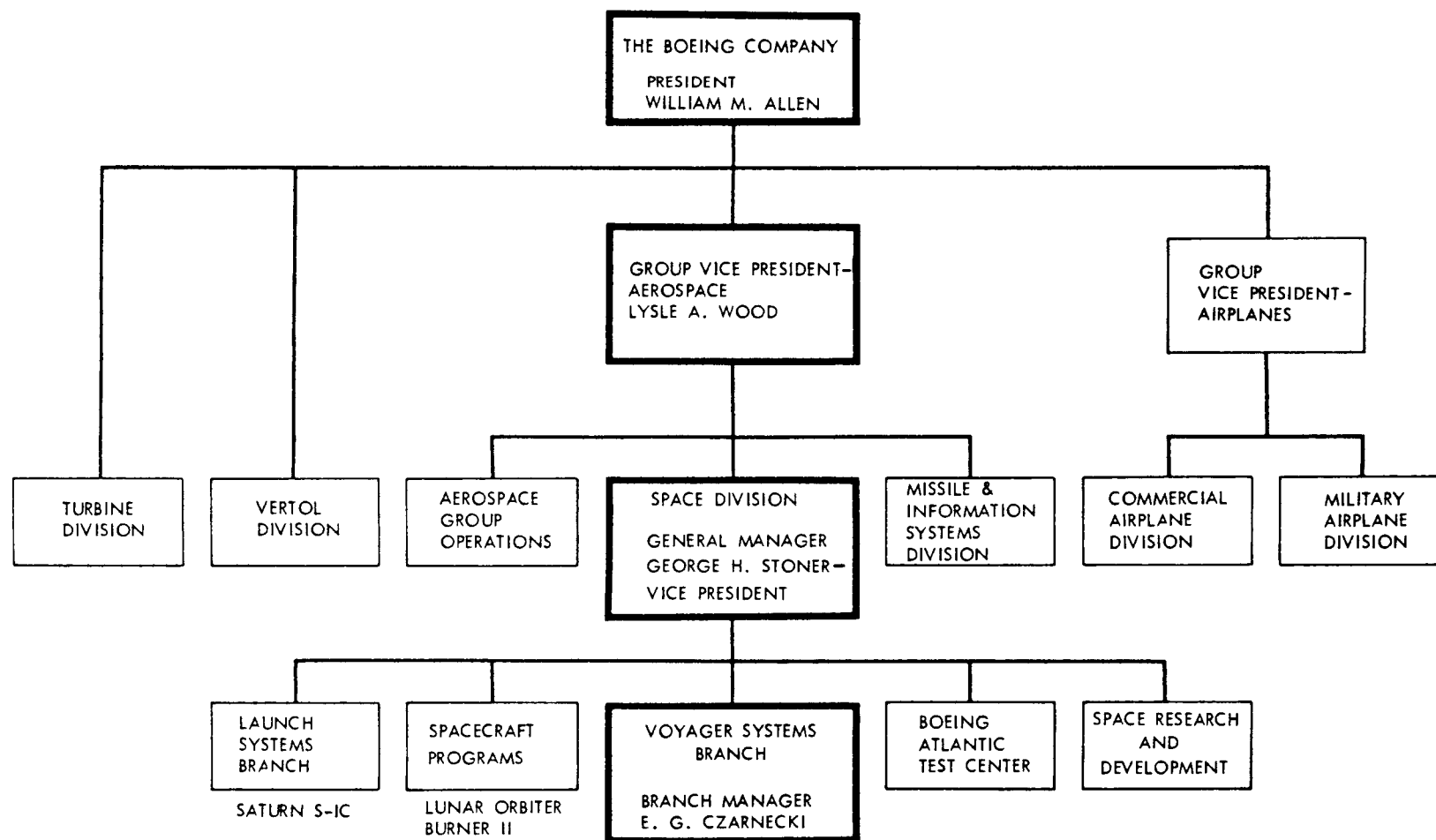


Figure 2-1: Voyager Organization Reporting Relationship



WILLIAM M. ALLEN
PRESIDENT

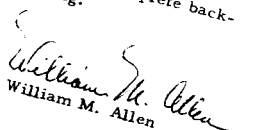
THE BOEING COMPANY
SEATTLE, WASHINGTON 98124

Dear Lysle and George:

I note with considerable interest and personal pleasure that we are again preparing to submit our Company studies of Voyager to JPL. Since the earlier studies, I feel that together we have accomplished in large measure the restructuring of the aerospace elements of the Company to better focus our attention on the challenge of space. Relocation of the Space Division this spring to our new Kent facility is, it seems to me, a significant milestone in our determination to provide the customer with the very best. Further, I understand the specific plans being developed by Mr. Czarnecki for the Voyager Branch will result in a strong, responsive project organization.

Mr. J. L. Atwood, President of North American Aviation;
Mr. J. C. Wilson, President of Xerox Corporation; and
Mr. R. O. Fickes, President of Philco, have again tendered their personal support to our continuing association. Additionally, they have seen fit to organize and staff their Voyager teams much along the lines we have.

You both have my confidence as well as my complete backing in this important national undertaking.


William M. Allen

L. A. Wood
G. H. Stoner



NORTH AMERICAN AVIATION, INC.
GENERAL OFFICES • 1700 EAST IMPERIAL HIGHWAY • EL SEGUNDO, CALIFORNIA
OFFICE OF THE PRESIDENT

December 16, 1965

Mr. William M. Allen
President, The Boeing Company
P. O. Box 3707
Seattle, Washington

Dear Mr. Allen:

I wish to reaffirm my personal interest and support of our Autonetics Division in its continuing participation in Boeing's Voyager Spacecraft Program.

We recognize the Voyager Program objectives of unmanned interplanetary exploration as one of the most ambitious in the next decade. We are proud to provide a Voyager Program team and organization that has the specific technical and managerial skills necessary to execute our responsibilities reliably, on schedule, and within cost.

I am pleased that the long and successful working relationship established on the Minuteman Program between our two companies can be continued on Voyager. Be assured of our full cooperation with the Boeing Company, the Jet Propulsion Laboratory, and the NASA in the accomplishment of this significant program.

Sincerely yours,

J. L. Atwood
J. L. Atwood
President

JLA:oc



PHILCO CORPORATION
A SUBSIDIARY OF *Ford Motor Company*

OFFICE OF THE PRESIDENT

PHILADELPHIA 34
PENNSYLVANIA

December 15, 1965

Mr. William M. Allen
The Boeing Company
Post Office Box 3707
Seattle, Washington

Dear Mr. Allen:

I wish to commend the Boeing Company for holding the recent conference in Seattle which brought together the general management of the Boeing Voyager Team. This meeting did much to insure a complete understanding at the management level of the common technical and management problems we must address in order to fulfill the goals of the Voyager Project.

The program has been reviewed by me and by key members of my staff, and we can assure you that the facilities, manpower, and skills required of Philco will be available to accomplish this most important project. As a minimum, I intend to review the Philco effort on a quarterly basis to insure that this commitment is maintained throughout the program.

Sincerely,

R. O. Fickes
Robert O. Fickes
President

/mg



XEROX
CORPORATION

OFFICE OF THE PRESIDENT

December 7, 1965

Dear Mr. Allen:

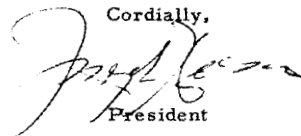
We continue to appreciate and enjoy, more than I can say, our work with Boeing on the Voyager program and are pleased with the opportunity to participate in Task-B of Phase 1-A. The more we learn about the Voyager program and its potential value and interest to the United States space program the more excited we become with the prospect of being a part of the team which supports NASA in this important undertaking.

The Xerox acquisition of EOS had as an included objective work in space systems activities of the Voyager type. We, of course, will commit all appropriate resources.

We recognize that there is more study and detailed planning required on the part of the entire team in preparation for entering into the Phase 1-B study and Phase-2 hardware program. We are preparing to do all that is necessary to help achieve the success we all desire.

Best personal regards, and warmest wishes for a most happy holiday.

Cordially,



President

JCWilson/mp

Mr. William M. Allen, President
Boeing Corporation
P. O. Box 3707
Seattle, Washington 98124

ROCHESTER, NEW YORK 14603

BOEING-SPACE DIVISION

agement. This provides the president the necessary control, through individual accountability, for decisions that are keyed to both the corporation and the customer, and provides for the direct line control of the resources necessary to assure performance of contracted commitments.

The specific role of Aerospace Group headquarters is to integrate and regulate critical plans, resources, and activities to assure that contractual requirements are met at minimum cost; that all resources are utilized as fully as economically feasible; and that appropriate standards are developed, maintained, and applied within the group. The Aerospace Group headquarters control of group activities is exercised through personal involvement in key decisions and through the review and analysis of operating results.

Closely allied with the responsibilities reserved to Aerospace Group headquarters are a number of tasks performed for the group vice president — Aerospace, by individuals designated as functional executives. These functional executives serve as extensions of the group vice president's office in assuring overall effective operation of elements designated within their respective areas of functional jurisdiction. The functional executive role was created to assure the group vice president — Aerospace, the operating divisions, the program managers, and the customers that appropriate standards and the best functional methodology are employed and achieve functional excellence.

This philosophy of management assures NASA/JPL that the full support and capability of The Boeing Company stands behind all programs and projects.

SPACE DIVISION

The Boeing Company has assigned to the Space Division, facilities in the new Boeing Space Center at Kent, Washington, to support activities of the Voyager Systems Branch in the conduct of the Voyager Project. The Space Center, a \$43 million investment commitment, currently includes a space environmental simulator, space-flight simulator, microelectronics laboratory, and a space materials and processes laboratory. Construction of office facilities is under way with occupancy scheduled for early 1966. In addition, the headquarters of the Space Division will be located at the Space Center.

The organization of the Space Division is also shown in Figure 2-1. The division pursues the

development, test, and production technologies required for launch and space programs. A basic policy employed in the Space Division requires that each program or project manager be assigned direct line authority over those functions required to effectively accomplish his assignment. This policy results in the responsibility for all essential activities such as system engineering, design engineering, subsystem and system test, major fabrication and final assembly, project planning and control, and product assurance being assigned to the project manager.

The Space Division has been assigned primary responsibility for the design, development, test, launch, and mission support activities related to the Voyager Spacecraft. Within the Space Division, the Voyager Systems Branch has been delegated these responsibilities.

This branch organization will be the point of contact between other Boeing organizations, major subcontractors, and NASA/JPL for all communications related to Boeing's part in the Voyager Project.

VOYAGER SYSTEMS BRANCH

The Voyager Systems Branch is shown in Figure 2-2. Mr. E. G. Czarnecki has been designated the branch manager. His responsibilities include the fulfillment of commitments assigned to his branch, including the creation and operation of an organization to accomplish this assignment. The complete Voyager engineering, manufacturing, procurement, contracting, and testing job is under the control of Mr. Czarnecki. He will obtain support from other Aerospace Group organizations and other organizations of The Boeing Company as required for performance of the Voyager Project. Additionally, Mr. Czarnecki will maintain effective working relationships with NASA/JPL, other contractors involved in the Voyager Project as required, and his subcontract team.

MANAGEMENT PHILOSOPHY

Management planning has been conducted in parallel with technical studies since Boeing's involvement in the Voyager Spacecraft System Program in 1963. The planning started with a careful review of all the technical and managerial objectives of the total Voyager Program. This review resulted in an appreciation of the total job to be done by the National Aeronautics and Space Administration (NASA), the project management responsibility of the Jet Propulsion Laboratory (JPL), and the individual assignments

BOEING-SPACE DIVISION

of the several systems contractors, including their major suppliers and subcontractors. Figure 2-3 reflects Boeing's understanding of its role as the Spacecraft System contractor in relation to the broader responsibilities of JPL and NASA.

Boeing recognizes JPL's philosophy relating to conservative design and simplicity of interfaces between subsystems. Equally important, the Spacecraft Bus is not an end in itself. The Voyager mission, Figure 2-4, is to transport the Flight Capsule and the science payload to Mars to fulfill the mission objectives. As requested by JPL, Boeing will adapt the Spacecraft System to flights to Mars and other planets subsequent to the 1971 mission.

Within the framework of the understanding of the program relationships described above, Boeing has structured a management philosophy based on the following precepts:

- Understand what it is the customer wants done;
- Understand how the customer wants it done.

These precepts were studied to gain the required understanding; then, the background of management experience gained over many years of successful programs conducted with the gov-

ernment and industry was examined to draw upon those specific elements that particularly serve the needs of the NASA/JPL Voyager Program. These programs, including Lunar Orbiter, Saturn S-IC Stage, Saturn V Mission Support, and Minuteman have involved several customers and have varied widely in basic objectives, detail requirements, dollar volume, and schedules.

A major factor in organization planning is the maintenance of continuity in both management and technical documentation through all phases of the Voyager Project. Figure 2-5 illustrates the interrelationships among these documentation requirements.

In properly organizing to meet these requirements, Boeing desires to take advantage of JPL's Ranger and Mariner experience and technology. To do so, focal points will be established for detail contacts in all functional areas. In addition, the following steps will be taken:

- 1) Key Boeing personnel will be located at JPL for the first 3 months of Phase IB to work with JPL in the development of the spacecraft system. This period will be highlighted by a Systems Requirement Review (SSR) to ensure that proper definition of performance/design requirements have occurred.

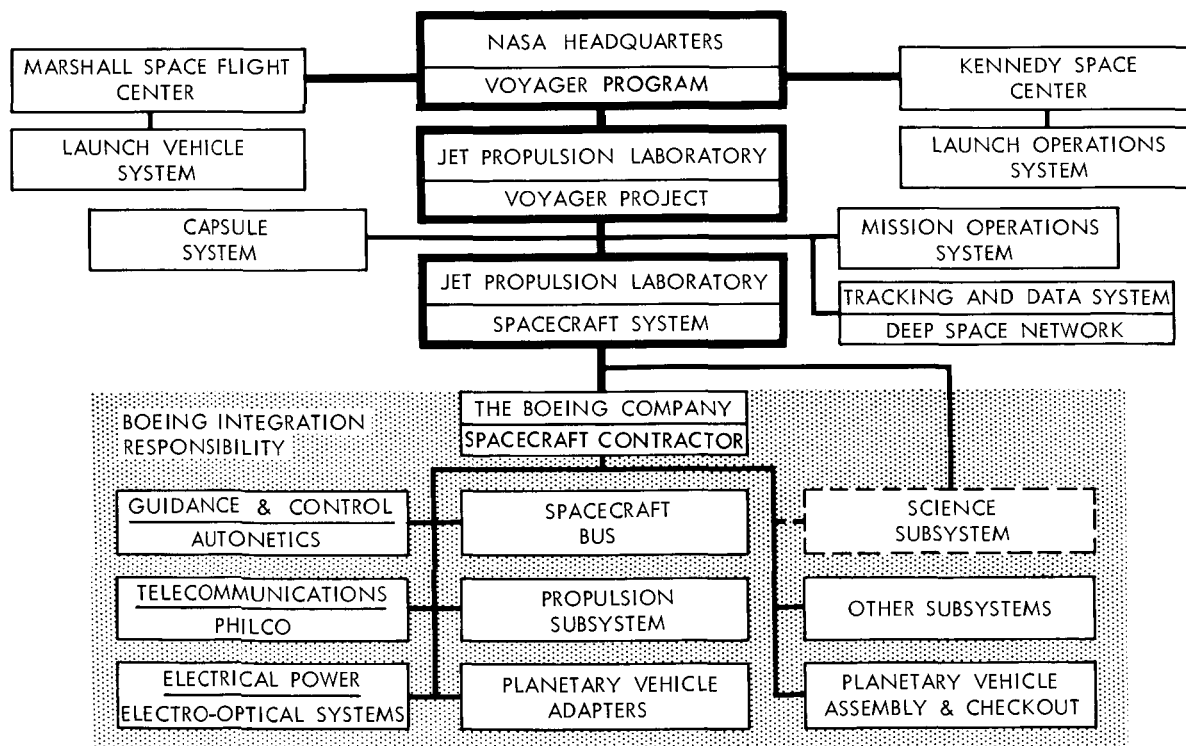
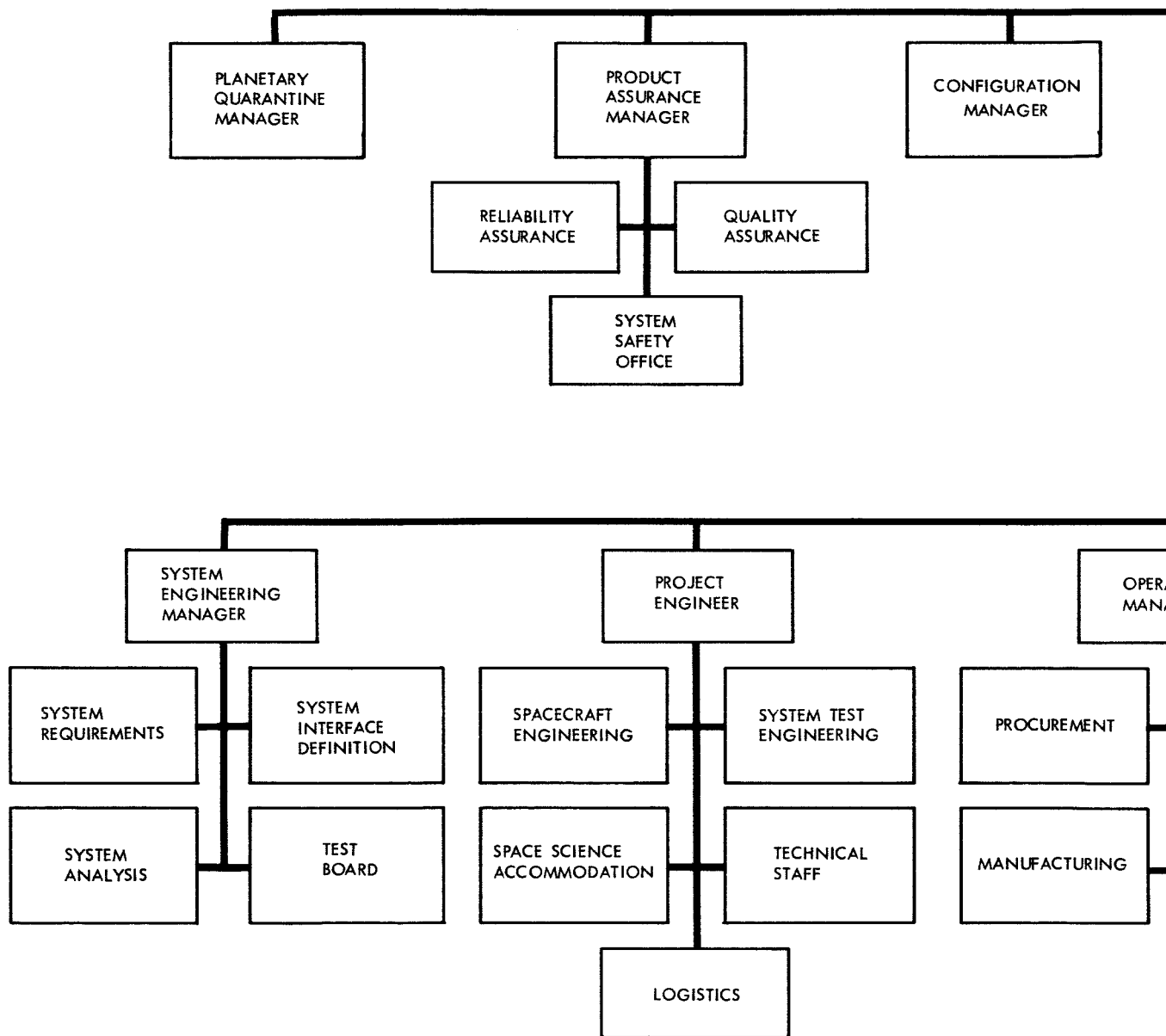


Figure 2-3: Voyager Project Management Task



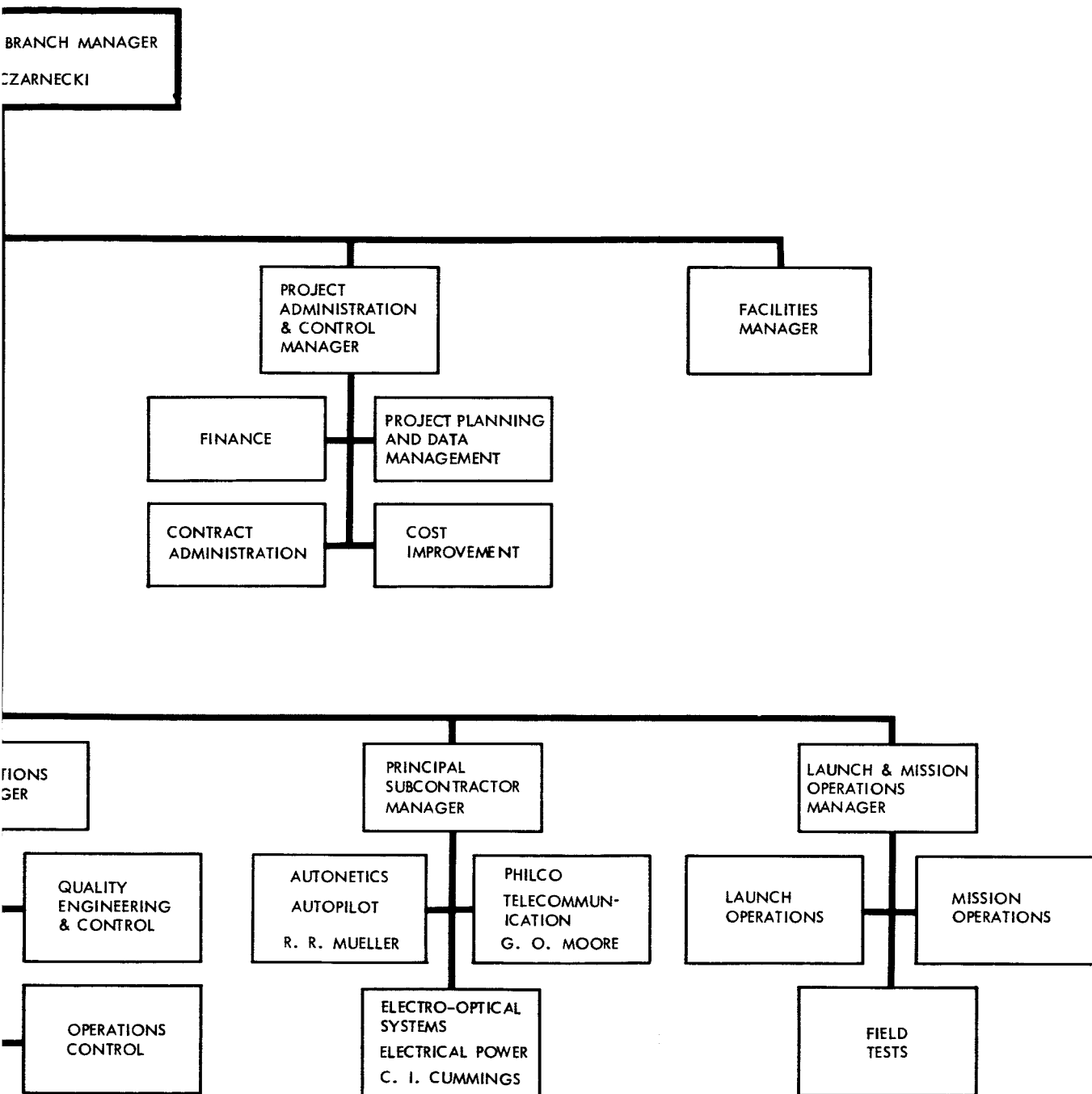


Figure 2-2: Boeing Voyager Organization

BOEING-SPACE DIVISION

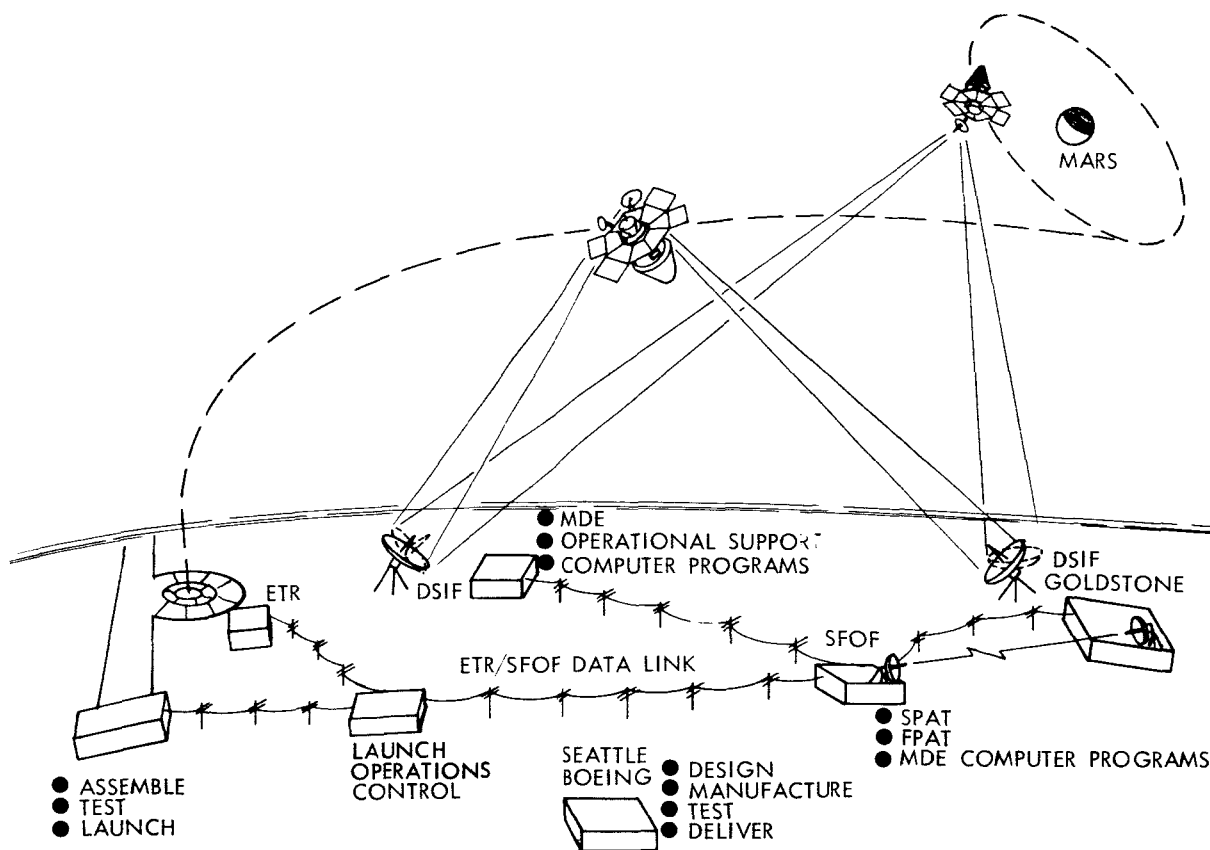


Figure 2-4: Voyager Mission

- 2) During the last 4 months of Phase IB, a JPL team will be located at Boeing to ensure the direction in which JPL desires the program to progress. This period will be highlighted by the conduct of the Preliminary Design Review at the end of the fifth month of Phase IB when Part I's of CEI specifications will be submitted for JPL approval. Throughout the program, the planned focal point relationships between JPL and Boeing will be as shown in Figure 2-6.

SPACECRAFT SYSTEM TASK

Each new program has features common to past programs and adds some new features peculiar to the objectives of the new program. JPL has structured the Voyager Project on the basis of taking a logical step forward in the unmanned scientific exploration of the solar system, starting with knowledge and experience gained from the Ranger and Mariner programs.

This firm basis consists, in part, of using space-proven design concepts, and the same hardware wherever the requirements are matched. To this, JPL has added design simplicity, calculated

conservatism, and has formulated an approach to achievement of reliability goals required to ensure a high probability of success. Careful selection of redundancy options by JPL establishes a design approach for ensuring that no single failure mode of an electronic or electric part or component will have a catastrophic effect on the mission. JPL has also provided criteria in the form of competing characteristics for the selection of the preferred design approach. It is these criteria that tie the technical aspects of the program firmly to the management task. These criteria are:

- Probability of success;
- Performance of mission objectives;
- Cost savings;
- Contribution to subsequent missions;
- Additional 1971 mission capability.

The determination of the critical activities that must be recognized and accommodated within the Voyager organization structure is a key element of the management task. The key considerations have been identified as:

BOEING-SPACE DIVISION

- Assurance of Success — The 9-month unmanned flight to Mars followed by a 6-month orbit duration requires system-wide management effort to assure a high probability of mission success. Costs of the Voyager system coupled with national prestige require that success must be achieved on the maiden flight;
- Extensive Testing — Extensive ground testing is required to identify every potential failure mode and to establish realistic trend data for a scientific research program such as Voyager involving only a few flight articles every 2 years;
- Launch Opportunity — The unalterable launch opportunity, occurring only at 25-month intervals, requires deliberate establishment of schedule margins;
- Planetary Quarantine — It is necessary to determine those activities that may result in biological contamination of Mars, and to develop methods of preventing contamination within the required probability;
- Global Concurrent Activities — The nature of a long-term interplanetary space program requires continuous and concurrent use of all the facilities associated with the world-wide deep-space network. This results in the need for considerable technical and management redundancy in the spacecraft system team;
- Program Duration — The long duration and cyclic nature of the Voyager interplanetary scientific research program requires effective long-term use of resources, not only with respect to technical personnel but also in the area of materials as well.

The management task is clearly to structure an organization and work package in support of JPL to provide a balance between the drive toward a specific launch date while providing the flexibility of design in recognition of the varying nature of scientific payloads.

MANAGEMENT FLEXIBILITY

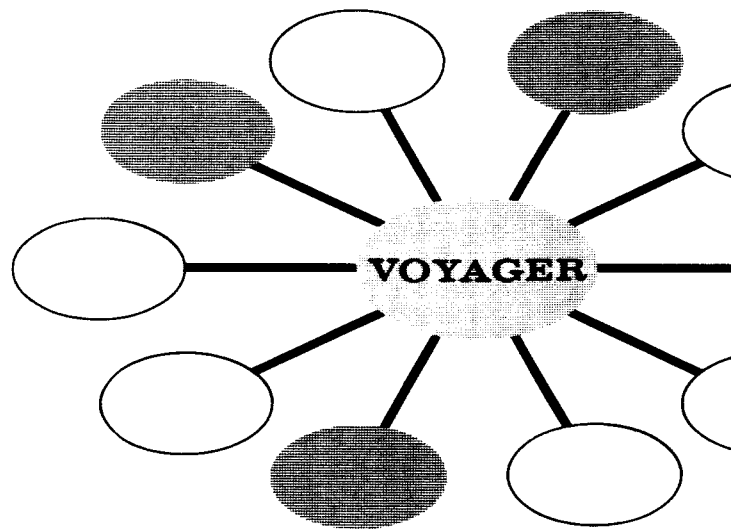
Having identified what is to be done, management next looks to background and experience for selection of those techniques, available and proven, upon which to build the techniques necessary to satisfy the requirements of the program. It is the recognition of the intricacies of the evolving programs that has provided us with the management flexibility to assure a balance of people, time, and facilities to accomplish our assigned tasks.

The Voyager Program will take a large but well-calculated step beyond the Mariner IV program. The logical, conservative steps taken by JPL, coupled with an intensive ground test program, are recognized as major factors in assuring mission success. Boeing's contribution to JPL's team lies in the understanding of the evaluation of technical requirements coupled with the capability to focus technical and management attention on the key problems of long-duration, unattended space flight preceded by an orderly, controlled countdown to an unalterable launch date.

The Boeing Company's history of providing assistance to the government in programs of national scope and importance has brought us the management and technical experience required to support NASA/JPL in the Voyager Project. Each of these programs has had its own peculiar requirements; each has led to a management approach answering the question of how we will accomplish the job.

Management flexibility and top-quality products have been our prime goals. Flexibility and quality are closely related to achievement of reliability as evidenced by Boeing's experience over the history of evolution from airplanes to space vehicles. Achieving reliability during this evolution is directly related to the knowledge and understanding of the customer's desires. Boeing's long experience on major airframe contracts has demonstrated that an airplane can be maintained daily, if desired. A cost-effective approach is to design the basic airframe for 50,000 to 80,000 hours of life, and provide planned replacement for items such as landing gear, which should not be designed for the same life as the basic structure. This is true because if the airplane were to be designed for this long a life with no maintenance, it would be too heavy, and the customer's profits would fall. Years of experience have allowed us to identify which parts should be replaced, and what their replacement schedule should be. The airplane is then cost effective, but at the same time has high reliability for its design mission. The reliability is attained, in this case, by a combination of conservative design of basic structure and a design for planned replacement. This is one example of a quality product Boeing has produced by recognizing that it is tailored to a particular requirement.

Evolution to missile systems, such as Minuteman, brought forth another key management approach to the achievement of reliability in that program. The need for minimizing ground maintenance for long periods of time, followed by



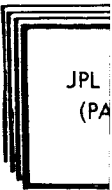
PHASE IA
TASK B AWARD

JPL
STATEMENT
OF WORK
(REVISED)

- SYSTEM SPEC
- ENVIRONMENT DOCUMENT
- PRELIMINARY MISSION
DESCRIPTION OCT. 1965



- WBS
- BUDGET
- REFERENCE
- POLICY
- PROGRAM



- DATA MODEL
(PRELIMINARY)
- CONFIGURATION
- GUIDELINES

PHASE IA TASK B COMPLETE
 PHASE IB RFP PART I COMPLETE
 JAN. 31, 1966

- BASELINE CONFIG. (PRELIM)
- SYSTEMS CONCEPT DATA
- ANALYSES & TRADES
- FUNCTIONAL FLOW DIAG.
- SOURCE DOCUMENTS (REFINED)

- VOL C PROPULSION
- VOL B OSE
- VOL A PHASE IA TASK B
 FINAL TECHNICAL REPORT

JPL

- DESIGN DATA
- MISSION GUIDELINES

- VOL III PHASE IB (PART I)
 MANAGEMENT PLANS

T & SCHEDULE
 NCE DATA
 & STRATEGY
 AM DIRECTIVES

- WBS
- REFERENCE DATA
- PRACTICES & PROCEDURES
- PROPOSAL POLICY &
 STRATEGY PROGRAM
 DIRECTIVES

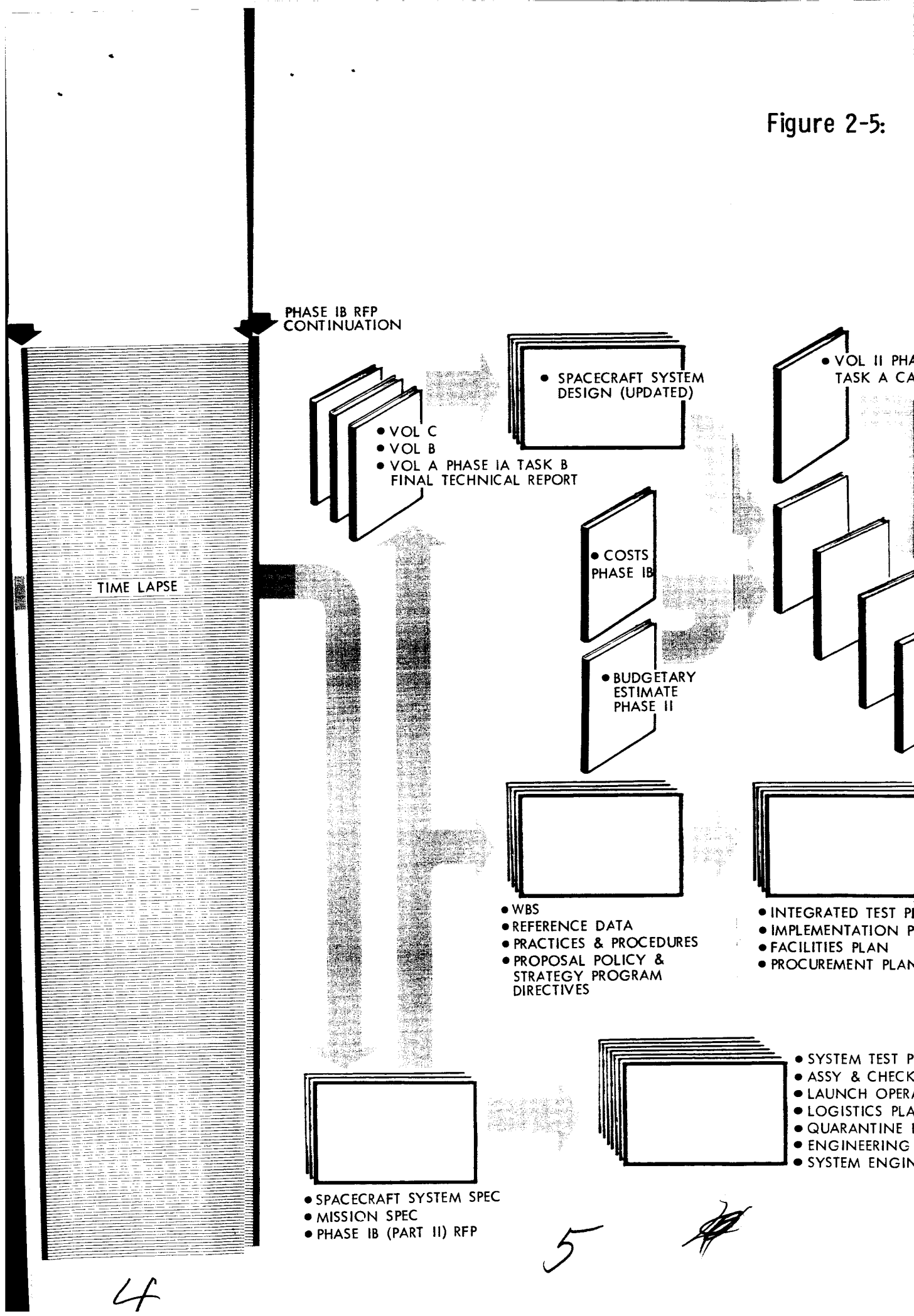
- OTHER INFORMATION
- MANUFACTURING PLAN (PRELIMINARY)
- RELIABILITY PLAN (PRELIMINARY)
- QUALITY PROGRAM PLAN (PRELIMINARY)
- DATA MANAGEMENT PLAN
- PROJECT CONTROL PLAN
- CONFIG. MANAGEMENT PLAN
- SAFETY PLAN
- ORGANIZATION PLAN

PHASE IB
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MANAGEMENT PLAN
 (PRELIMINARY)
 MANAGEMENT MANUAL (PRELIMINARY)
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3

Figure 2-5:



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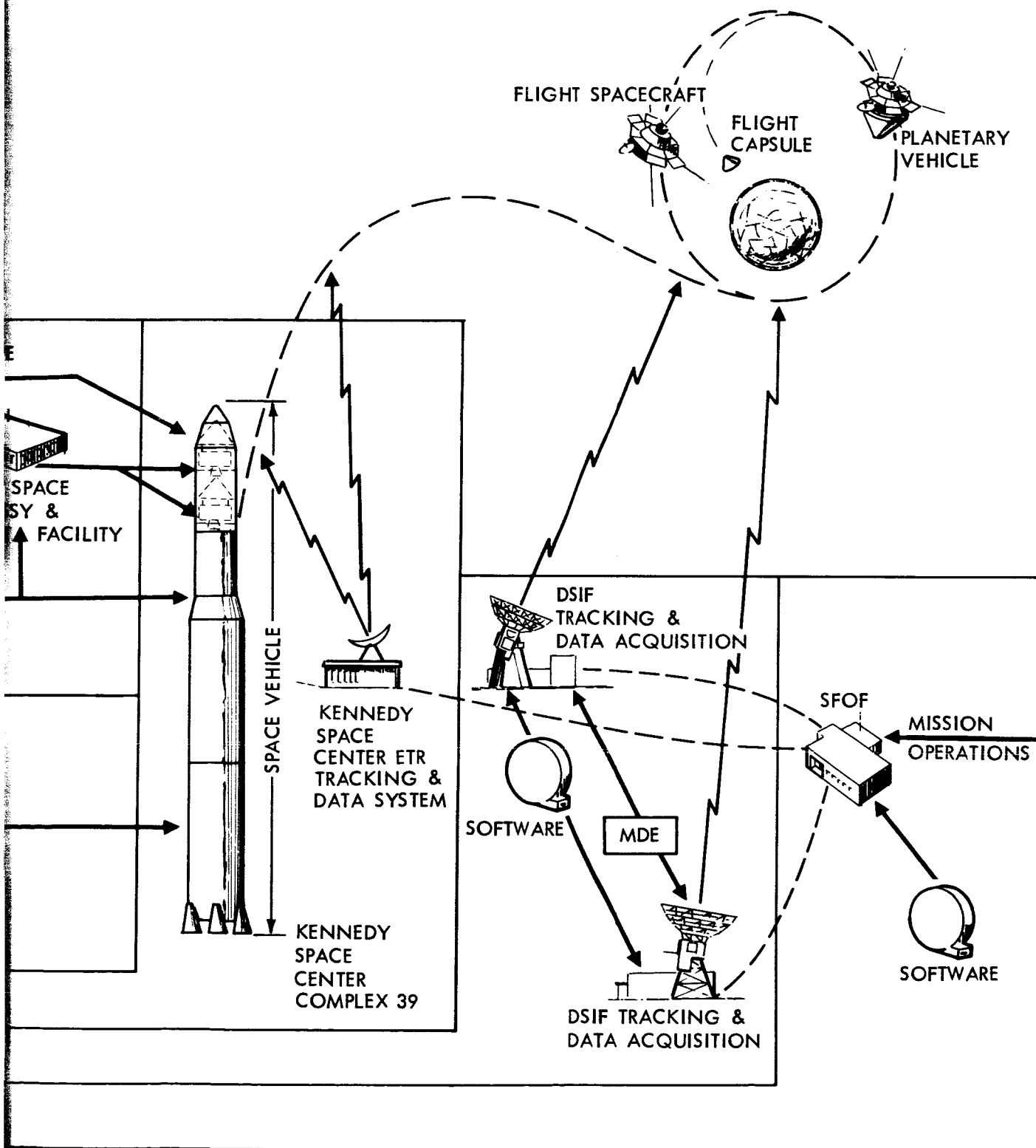
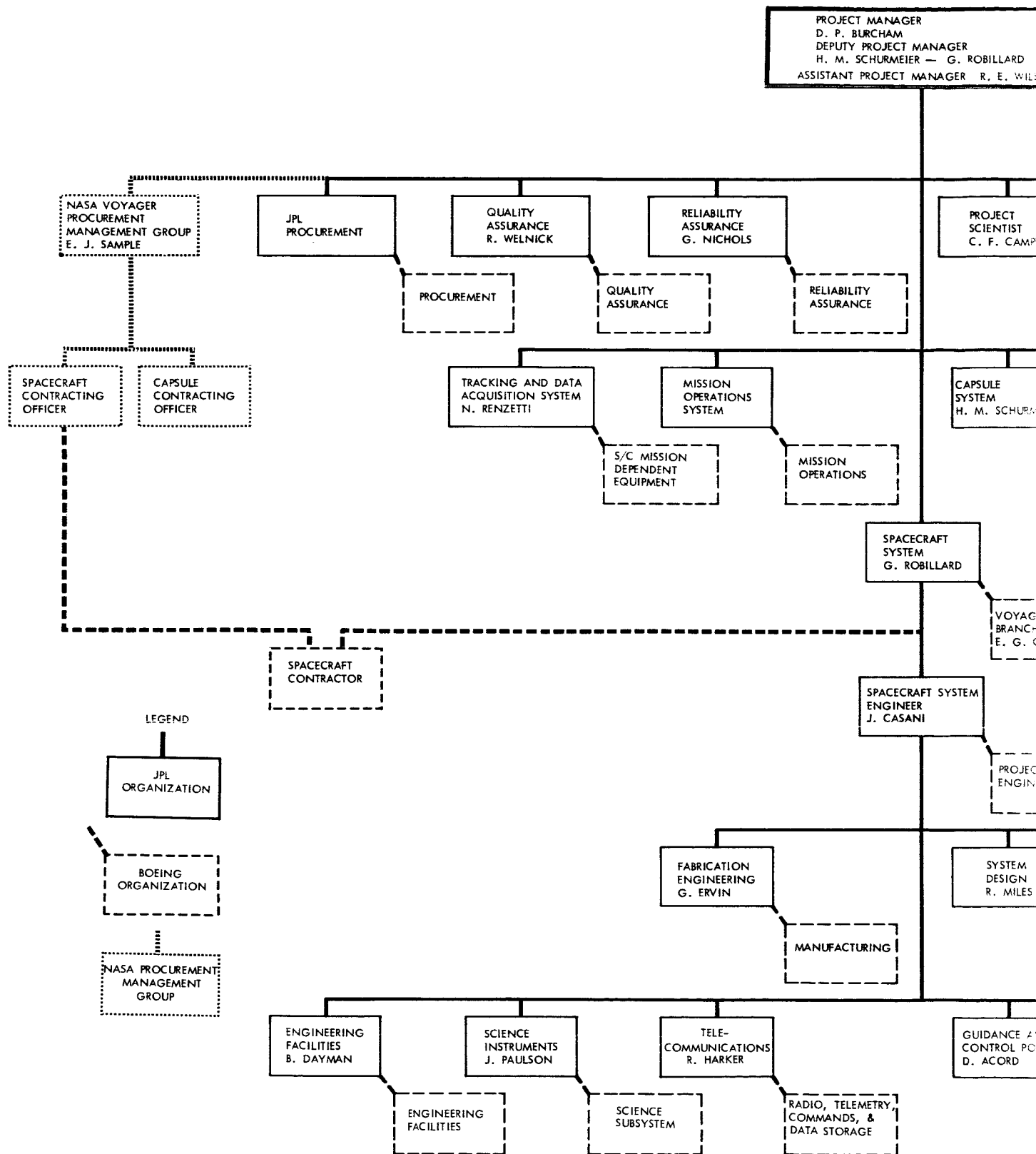


Figure 2-7: Mission System Elements



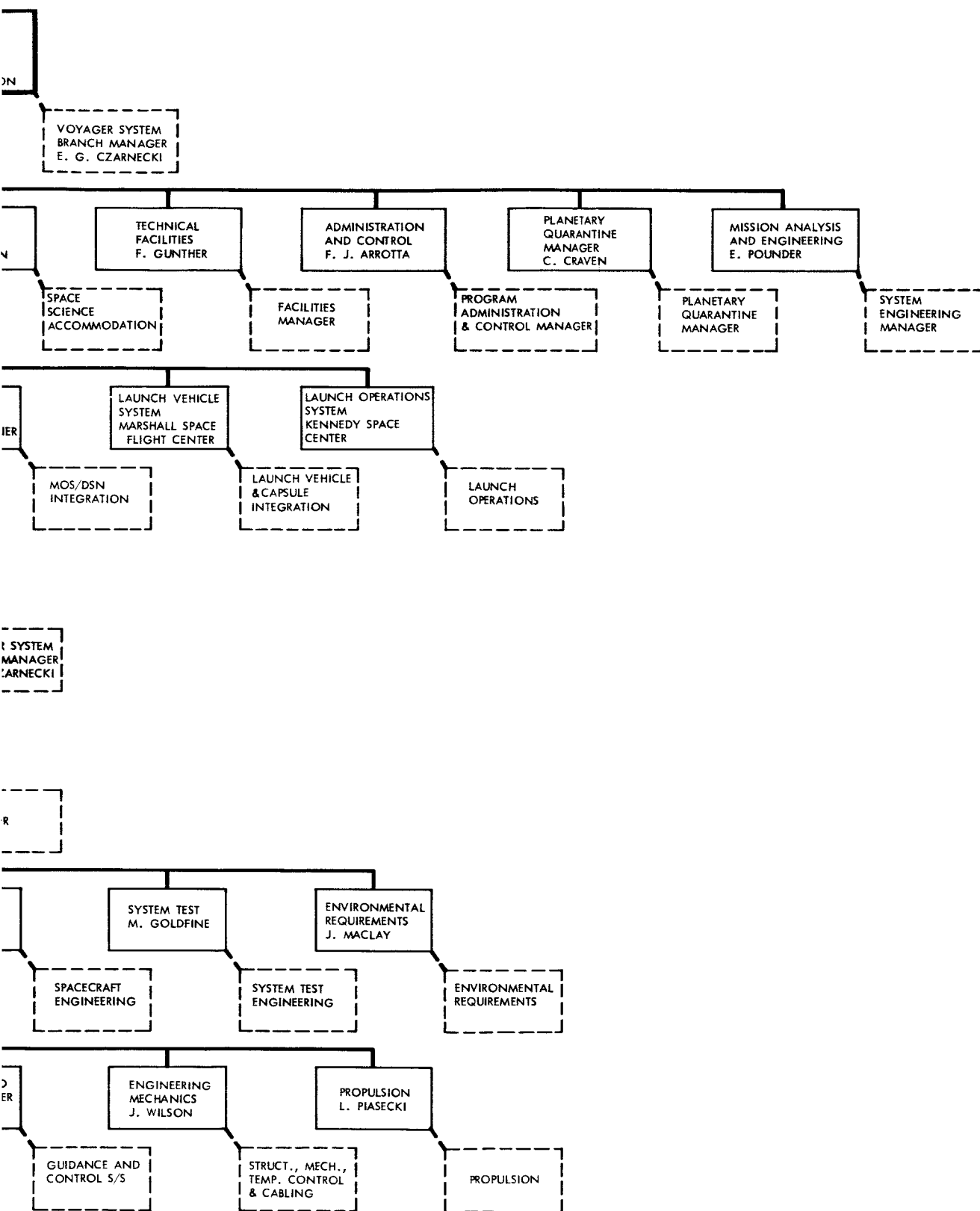


Figure 2-6: Voyager Project Focal Point Relationships

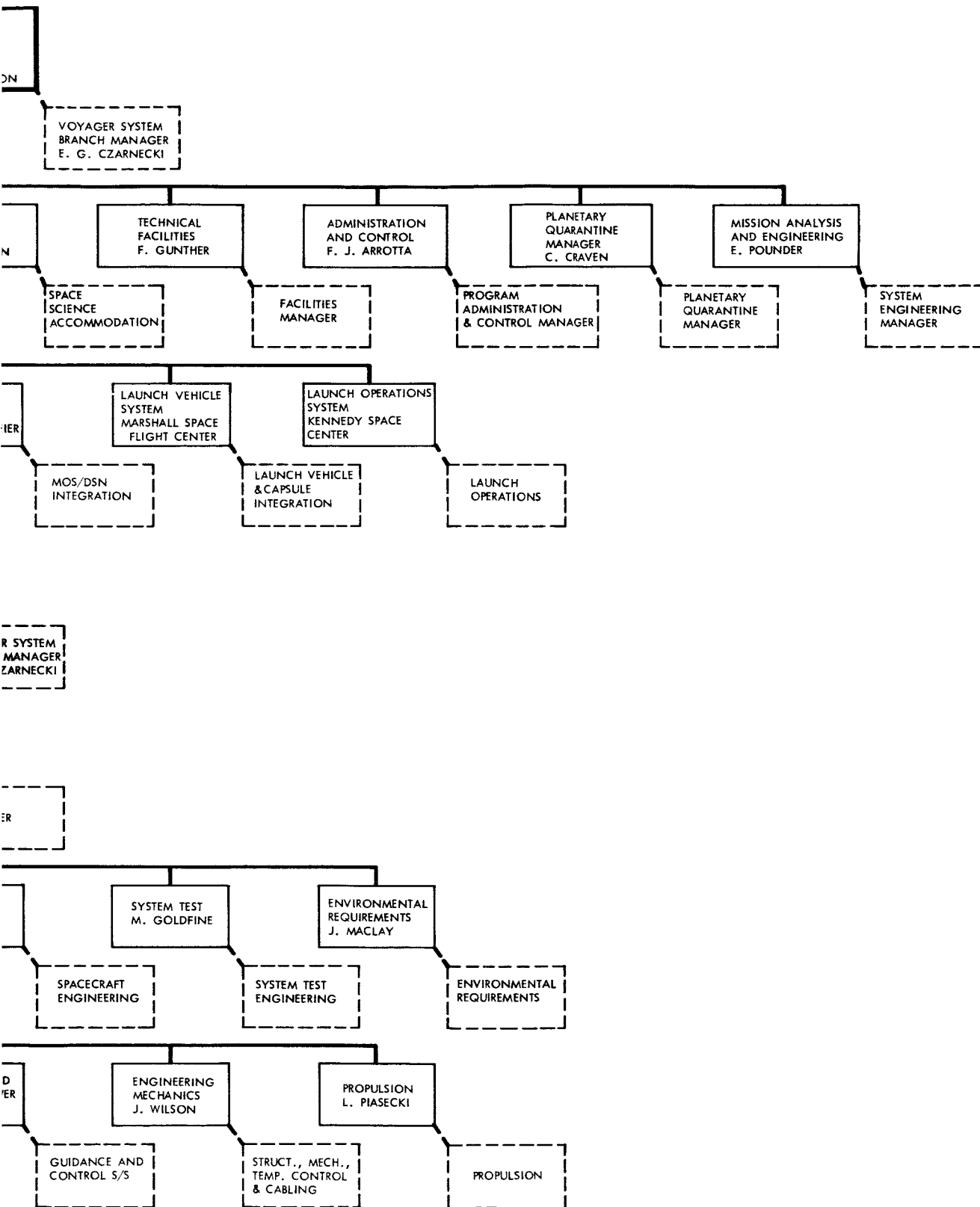


Figure 2-6: Voyager Project Focal Point Relationships

BOEING-SPACE DIVISION

quick-reaction, very short flight times, is nearly the reverse of airplane design approach. The concept here is to design for minimum maintenance, that is, go into the silo only when the missile does not meet its ready-status. Further, the flight reliability must be very high, but the powered flight time is very short. The requirements are entirely different from those for airplanes, but quality and reliability are equally important.

The next step was to convert from short-time, Earth-constrained flight, to long-time flight such as that for the Lunar Orbiter. In this case, the reliability requirement is high when considering the long-time, unattended flight through space. Redundancy in function is mandatory to achieve the reliability goal. Throughout this thread from airplanes to space vehicles, the key item that is changing is the decreasing time available for human maintenance. This means that a space program, such as Voyager, must be structured on the firm criterion that reliability is achieved by understanding every mode of failure of the hardware planned for use. There can be no such thing as "random failure" on this program.

Exhaustive ground testing is required to know exactly what the hardware will do for flight periods up to a year in a space environment. This is the essence of trend data for Voyager hardware. Trend data, taken from ground tests, establishes knowledge of the slow, but known, degradation of performance as the design life of the hardware is being expended.

The achievement of the above capability has been attained through careful attention to program requirements, detail attention to the reliability aspects of the design process, and the structuring of extensive proof test programs prior to commitment to launch. The planning for the spacecraft system contractor's support to JPL has been based on careful attention to mission requirements for the spacecraft and on the interfaces with the other elements of the project. This is depicted in Figure 2-7.

The achievement of success in the Voyager Program is keyed to an intensive ground test program to uncover possible failure modes of any equipment. To the extent practical, the flight hardware has preplanned safety measures preprogrammed on board. Redundancy of function is a top design requirement. Many trade studies have been completed that relate the effectiveness of alternate modes with reliability achieved. All this has been accomplished keeping in mind that probability of success for the 1971 mission is

the highest competing characteristic. Achievement of the highest probability of success possible is constrained by the Spacecraft Bus weight allocation of 2100 pounds. Extensive studies have been made to determine where the best payoff comes in allocating weight to subsystem redundancies.

Having achieved the best possible reliability within the weight constraint, attention turns to the ground test program required to establish confidence that the predicted reliability has been achieved. This is the basis for the integrated test program. Integration in this sense means that the total test program, from development, breadboard, type approval, flight acceptance, and combined systems test, has been examined in detail to establish levels (parts, components, end items) and sequences of all tests. The kinds of tests for each level and sequence have been identified and the required flow times established. Similarly, the reliability, development, and test requirements for operational support equipment that is vital to successful launch and mission conduct have been critically examined. This information is the prime basis for the program schedule shown in Figure 2-8.

There are several important management aspects of this schedule. There is a built-in 3-1/2-month reserve at Cape Kennedy. All Boeing activities are keyed to this date, which is 6 months prior to launch. There is a minimum of concurrency of major ground tests to ensure that data from one test can be effective on subsequent tests. The key to attainment of this objective is early initiation of hardware releases.

ORGANIZATION CONCEPT

Having described Boeing's understanding of the task to be undertaken for JPL and the management flexibility to plan for it, it is important to demonstrate how we will accomplish the task. The key Boeing management objectives as the spacecraft system contractor are to provide:

- 1) The maximum possible space-design background and training to our people prior to commitment to hardware;
- 2) A management scheme that provides for two competent people in each of several key positions;
- 3) The highest degree of reliability and quality that is possible within program constraints;
- 4) Knowledge of every known source of hardware failure within the constraints of time and money. No hardware will be used until

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the reasons for test failures have been discovered. "Random failure" will not be accepted as an explanation;

- 5) A spacecraft design that is flexible in its accommodation of potentially varying experiments.

With the above objectives in mind, and with a clear understanding of the task to be undertaken, it is time to discuss organization features and concepts.

A key feature is that the organization should provide a separation of functions to achieve checks and balances. For example, the total system requirements are gathered under one head: System Engineering in the Boeing organization. This organization head has to look to JPL, and inwardly to our organization, to carry out the joint task of establishing mutually agreed-upon requirements. To ensure that the total engineering team understands the requirements, he solicits active participation of all affected Boeing organizations in the generation of requirements from the top level down, and then prepares the statement of work to cover the tasks necessary to accomplish the requirements.

Once the requirements are established, the implementation to fill them begins in the project engineering organization. Here the design, analysis, development and breadboard testing, and the creation of drawings is accomplished to implement physical hardware in accordance with the requirements. The technical staff performs analyses of the design to ensure integrity and overall product excellence. A check and balance is achieved here in that the signature of a technical staff engineer on the same drawing as signed by the designer means that an independent technical check has been accomplished and that the design will perform as intended.

The drawings are then released to the fabrication and assembly shops and the physical hardware is created. Another check and balance of this process is obtained by use of quality control personnel who inspect the hardware against the drawings. The hardware then is returned to engineering, which conducts all testing at the subsystem and system level. These engineers are the same as those who designed the hardware and are called cognizant engineers. This is the formation of the test team, which will be made up from personnel responsible for the subsystem testing. In this way, knowledge and experience are carried throughout the total test program.

A check and balance is obtained here by the use of a test board, which is part of system engineering. One of the prime jobs of the test board is to certify that the test results are valid. The test team comes under the cognizance of the Launch and Mission Operations organization for outplant tests. Boeing has developed trained personnel in field test operations involving major interfacing elements such as boosters, test ranges, telemetry systems, launch complexes, and control centers. Launch and Mission Operations is the manager of test activities. Highly qualified personnel are used from the same test team that was used inplant.

Throughout the whole procedure, from the generation of requirements through hardware design, ground test, launch, mission operations, and recovery of data, we have established under the Product Assurance organization, product assurance (PA) engineers. These are highly qualified design/technical experts who follow functional areas such as electric power and propulsion throughout the whole process. Since they are not under pressure as designers, fabricators, or testers, we can look to these PA engineers to check all aspects of all operations that occur that directly affect the attainment of quality and, therefore, maximum probability of success. The signature of a PA engineer means that he has observed a particular action and agrees that all associated quality aspects have been accomplished. The product assurance manager has direct access to the program manager to report any action that could injure the quality of the product. Evidence of their contribution to the end item they are following is a terse summary of all the planning and events the hardware has gone through, along with all the major decisions and underlying rationale. This is our prime method of check and balance to assure quality, and therefore reliability.

The Project Administration and Control office serves the total Voyager Systems Branch organization rather than being broken up to serve individual elements. It provides planning, control, and reporting functions along with cost collection and contract administration. This office also contains the total data management function to assure that all pertinent data may be obtained by any authorized person in a rapid and efficient manner.

The Configuration Management office is another key element in the assurance of product integrity. It also can be viewed as a check and balance in that change control is separated from any other functional organization so that an objective view

PHASE IB AWARD

SYSTEM REQUIREMENTS REVIEW

- TIME LINES
- SCHEMATIC DIAGRAMS
- TRADE-STUDY SUMMARIES
- RAS
- FUNCTIONAL FLOW DIAGRAMS

- CEI DETAIL SPI PRELIMINARY I
- CEI CRITICAL (PART I) PRELIA

- DESIGN DATA (UPDATED)
- REQUIREMENT DOCUMENT (UPDATED)

MOS SUPPORT
(PRELIMINARY PLANNING)

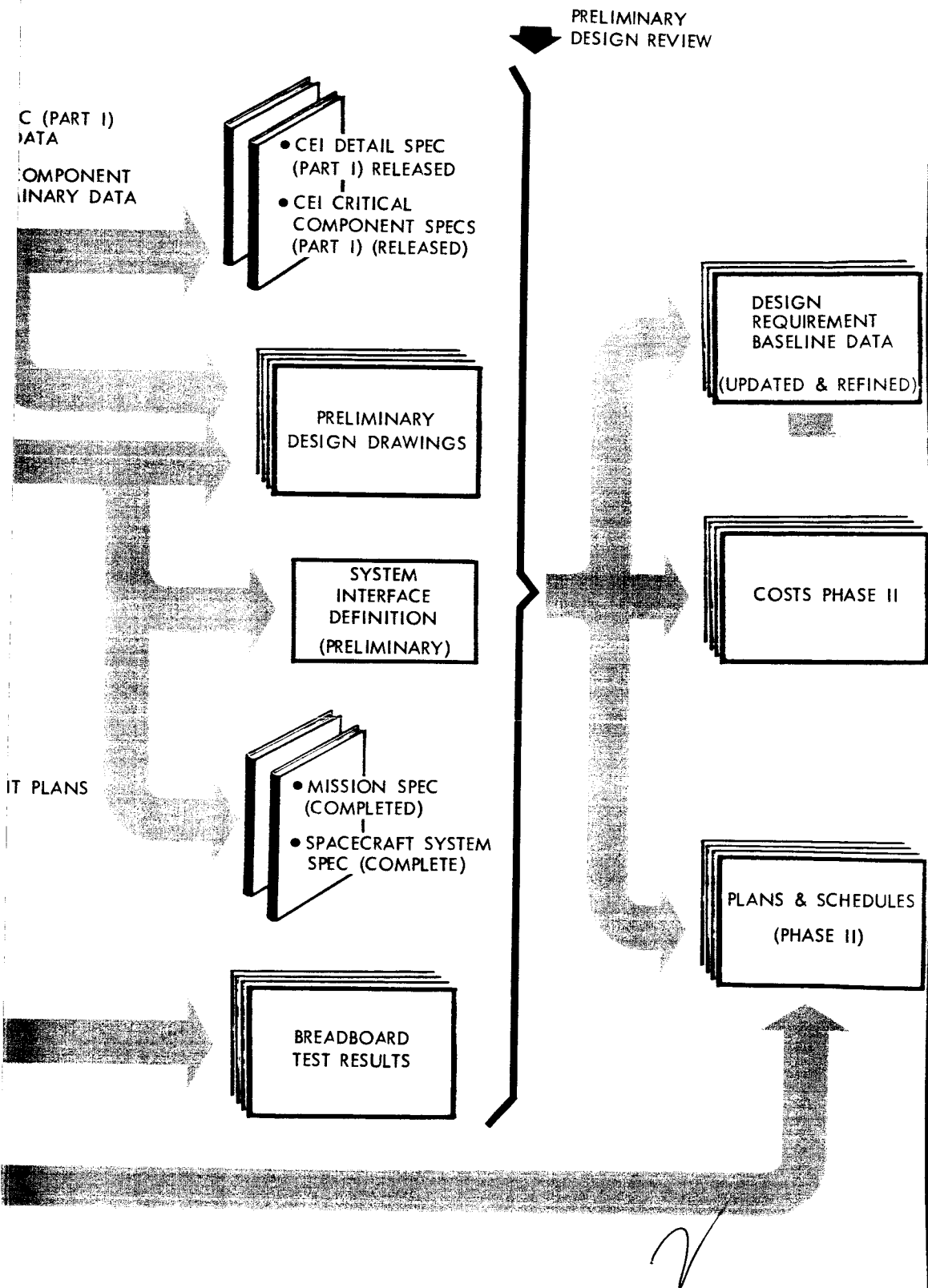
SPACECRAFT
SYSTEM SPEC
(UPDATED)

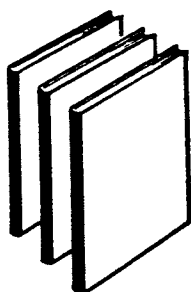
- VOL III
- PHASE IB (PART II) MANAGEMENT PLANS (UPDATED)

BREADBOARD DEVELOPMENT

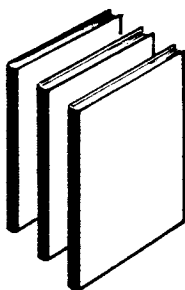
- SPACECRAFT SYSTEM SPEC
- MISSION SPEC
- PHASE IB STATEMENT OF WORK

- WBS
- BUDGET & SCHEDULES
- REFERENCE DATA
- PROGRAM POLICY & STRATEGY PROGRAM DIRECTIVES

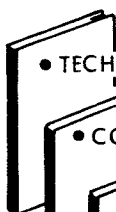




- INTERFACE REQUIREMENTS
- DESIGN DATA (UPDATED)
- REQUIREMENT DOCUMENT (UPDATED)



- SPACECRAFT SYSTEM SPEC
- CEI DETAIL SPEC (PART I)
- CEI CRITICAL COMPONENT SPEC (PART I)



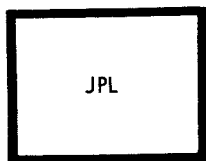
- TECHNICAL PROPOSAL



- COST ELEMENTS



- PHASE II PROPOSAL
PLANS & SCHEDULES



PHASE II AWARD

DETAIL DESIGN
DRAWINGS

- DESIGN DATA (UPDATED)
- REQUIREMENT DOCUMENT (UPDATED)

CEI
SPECS
PART II

- PLANS
- BUDGET & SCHEDULES
- REFERENCE DATA
- PROGRAM POLICY
- PROGRAM DIRECTIVES

- CEI SPECS (DETAIL & CRITICAL COMPONENT)
- MISSION SPEC
- SPACECRAFT SYSTEM SPEC
- PHASE II STATEMENT OF WORK

SY
CO
DC

CRITICAL
DESIGN
REVIEW



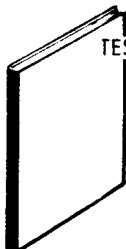
FIRST ARTICLE
COMPLIANCE
INSPECTION



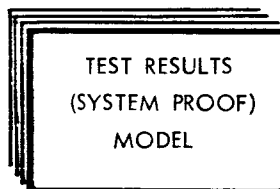
TEST RESULTS

TEST MODELS:

- STRUCTURAL
- THERMAL CONTROL
- PROPELLANT INTERACTION
- SEPARATION



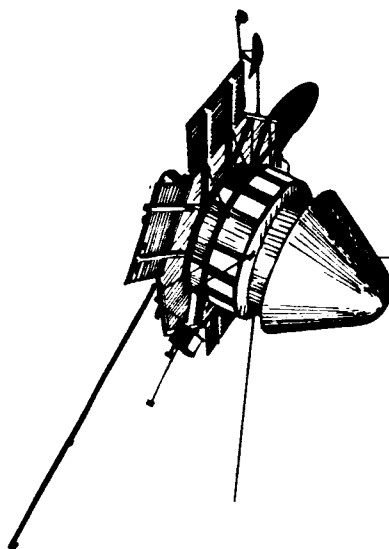
TEST RESULTS
(SYSTEM PROOF)
MODEL



MISSION OPERATIONS

- PRELAUNCH
- LAUNCH
- MISSION SUPPORT
- PERFORMANCE ANALYSIS
- DATA REPORT

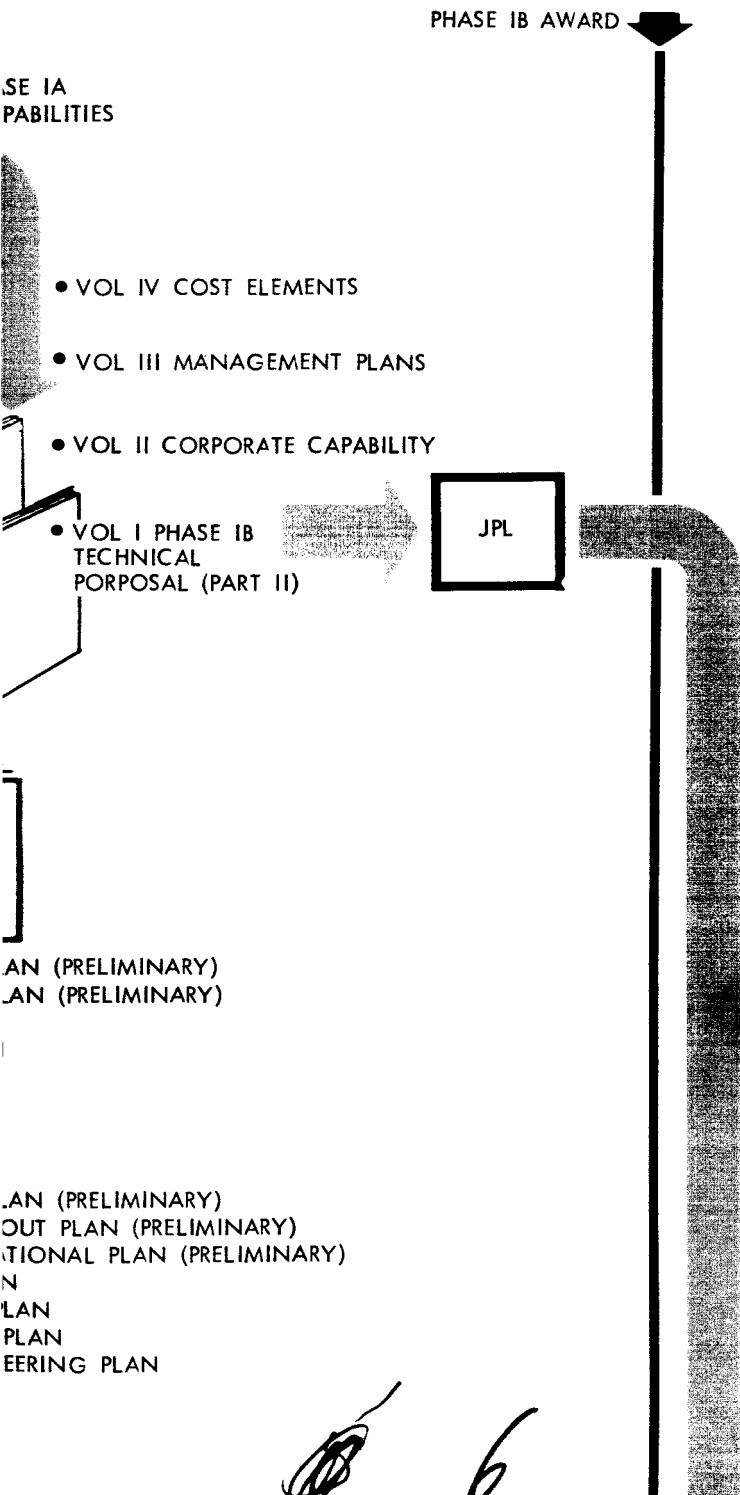
SYSTEM INTERFACE
CONTROL (DATA &
DOCUMENTATION)

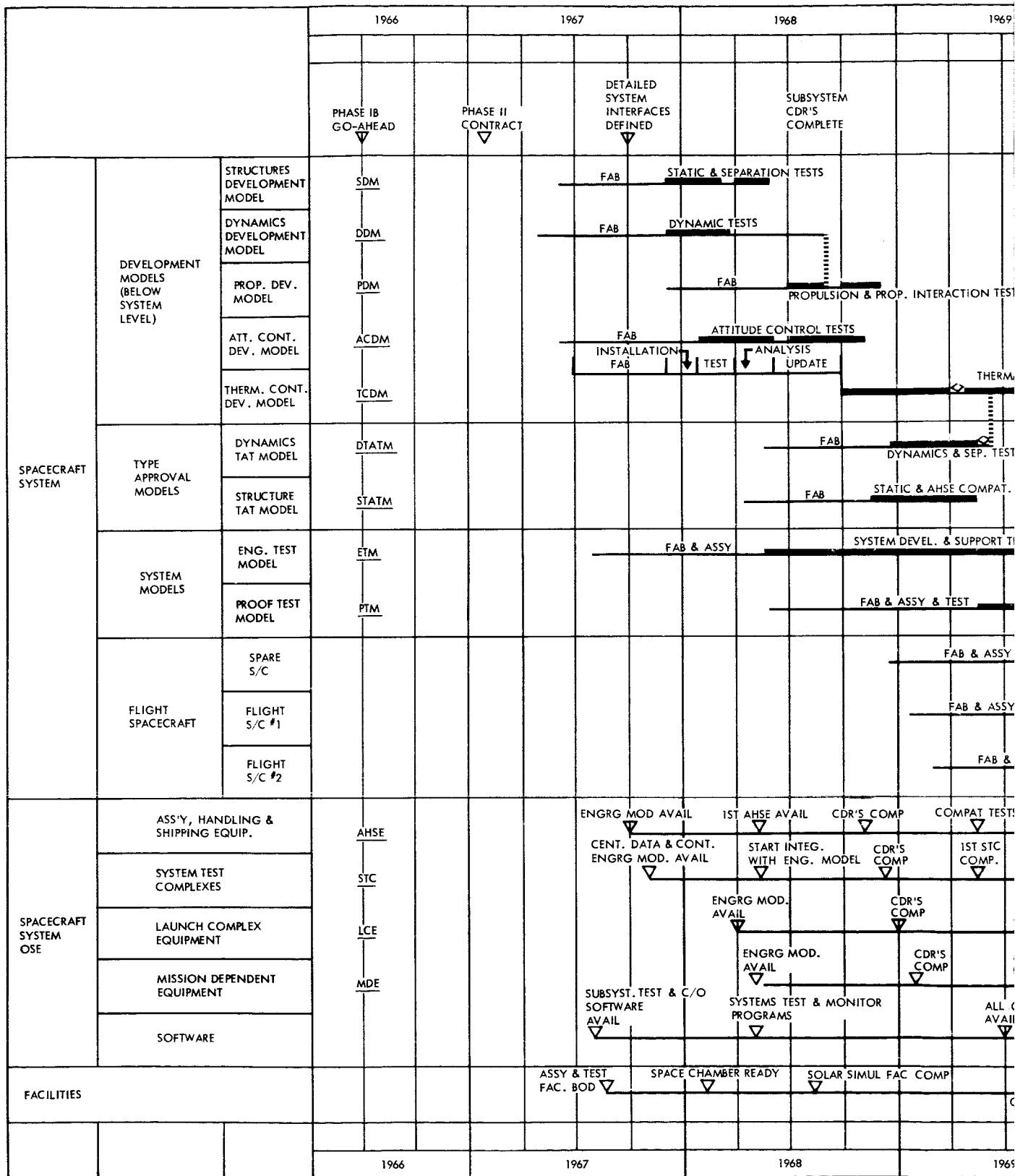


5



Voyager Spacecraft End-Item Development





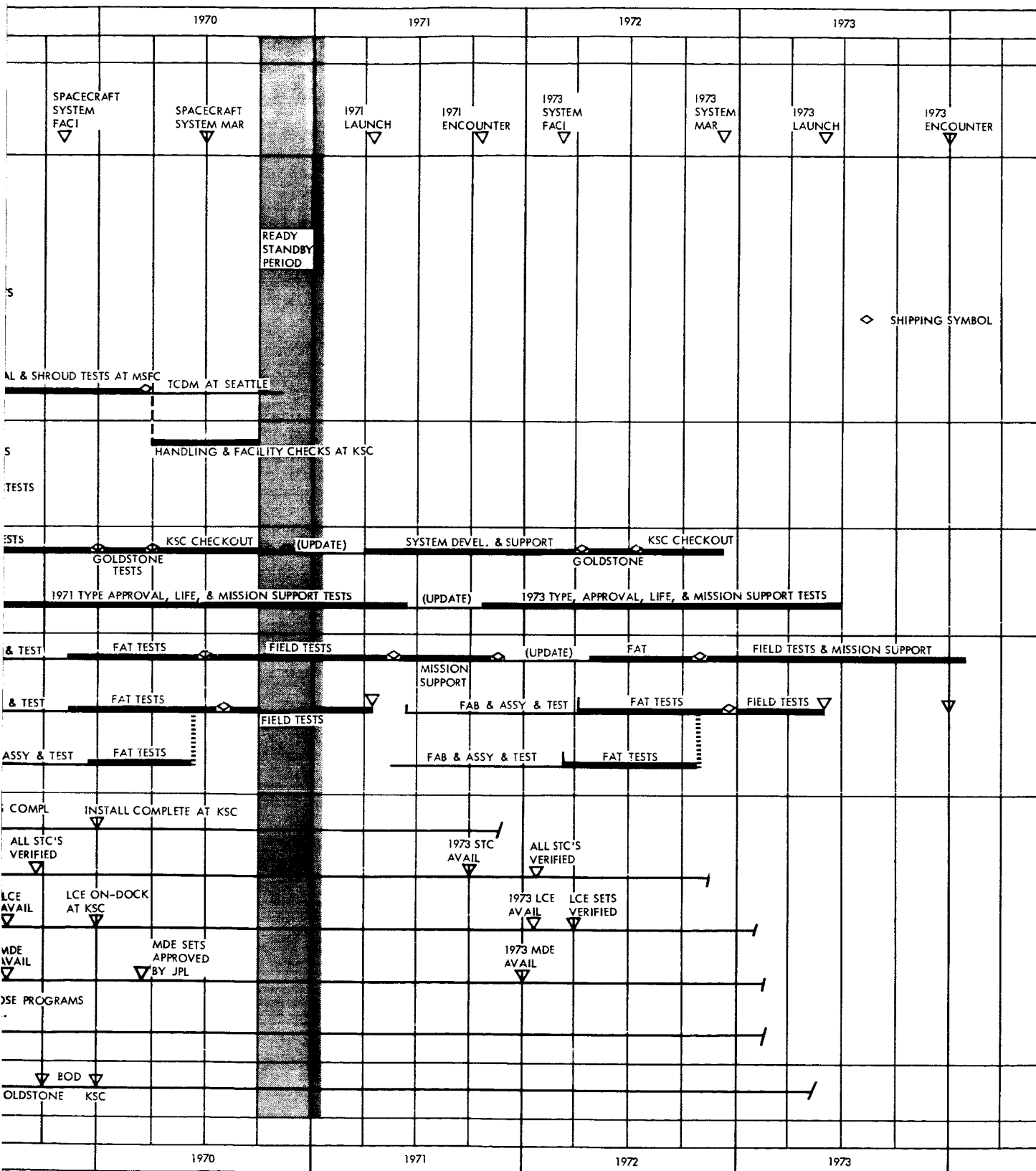


Figure 2-8: VOYAGER SPACECRAFT PROGRAM

BOEING-SPACE DIVISION

can be taken of all changes. This office also plans for the way we will identify all products so that visibility to all levels may be obtained.

The Principal Subcontractor Management office has been established to provide a direct line of control and communication between Boeing and its principal subcontractors on the Voyager Spacecraft System Project. This action is one that provides for delegation of certain management elements of the branch manager's responsibility to an office with exclusive focus on the management of the principal subcontractors.

Boeing will continue, subject to NASA/JPL approval, to extend its management and technical capabilities in depth by association with the following specialists: Autonetics, North American Aviation, subcontractor for the autopilot and attitude reference; Philco, Western Development Laboratories, subcontractor for the telecommunication subsystem; and Electro-Optical Systems, Inc., subcontractor for the electrical power subsystem. Electro-Optical Systems and Philco have been associated with Boeing on the Voyager Project since late in 1964. Autonetics, North American Aviation, joined the Boeing team in mid-1965. There has been a continuous and free exchange of information resulting in a mutual understanding on statements of work, interfaces,

and modes of operation. Rapid effective implementation of their efforts is possible upon receipt of contract award. The Principal Subcontract Management office will be responsible for administration of major subcontracts. A subcontract management team will be located at the subcontractor's facility and includes technical, procurement, and quality control experts. This team, dedicated to the delivery of a quality product on time and within cost, possesses a direct link with the technical areas on the project. These major subcontractors have top visibility in the Boeing Voyager organization by reporting to a principal subcontract manager within the framework of the subcontract management loop shown in Figure 2-9.

Effective communications is a prime management tool. The geographical dispersion of the Voyager Program participants requires that a coordinated communications network be established between the participants. Boeing's experience with other widespread programs has resulted in the development of the network shown in Figure 2-10, which will be in operation in time for service to the Voyager Program.

It is planned to retain the concept of the Technical Review Board as set forth in previous Boeing documentation on the Voyager Spacecraft Project. This group of scientific and technical experts

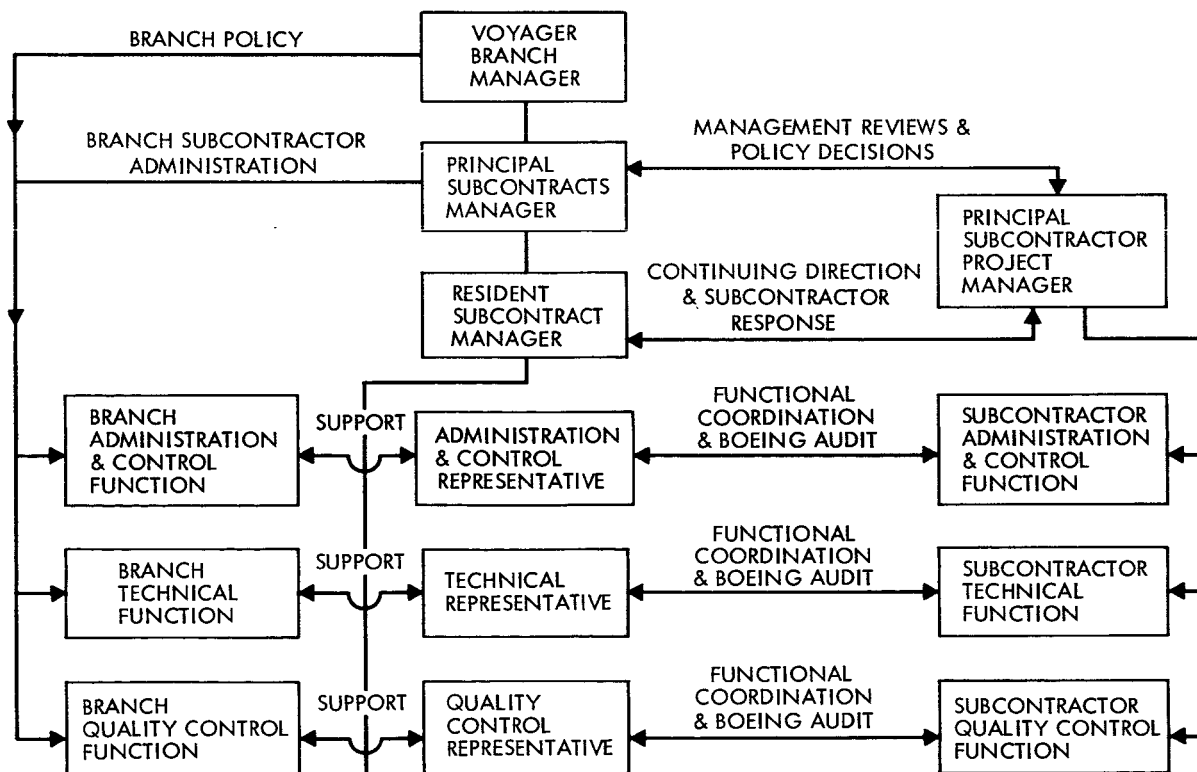


Figure 2-9: Boeing and Principal Subcontractor Management Loop

BOEING-SPACE DIVISION

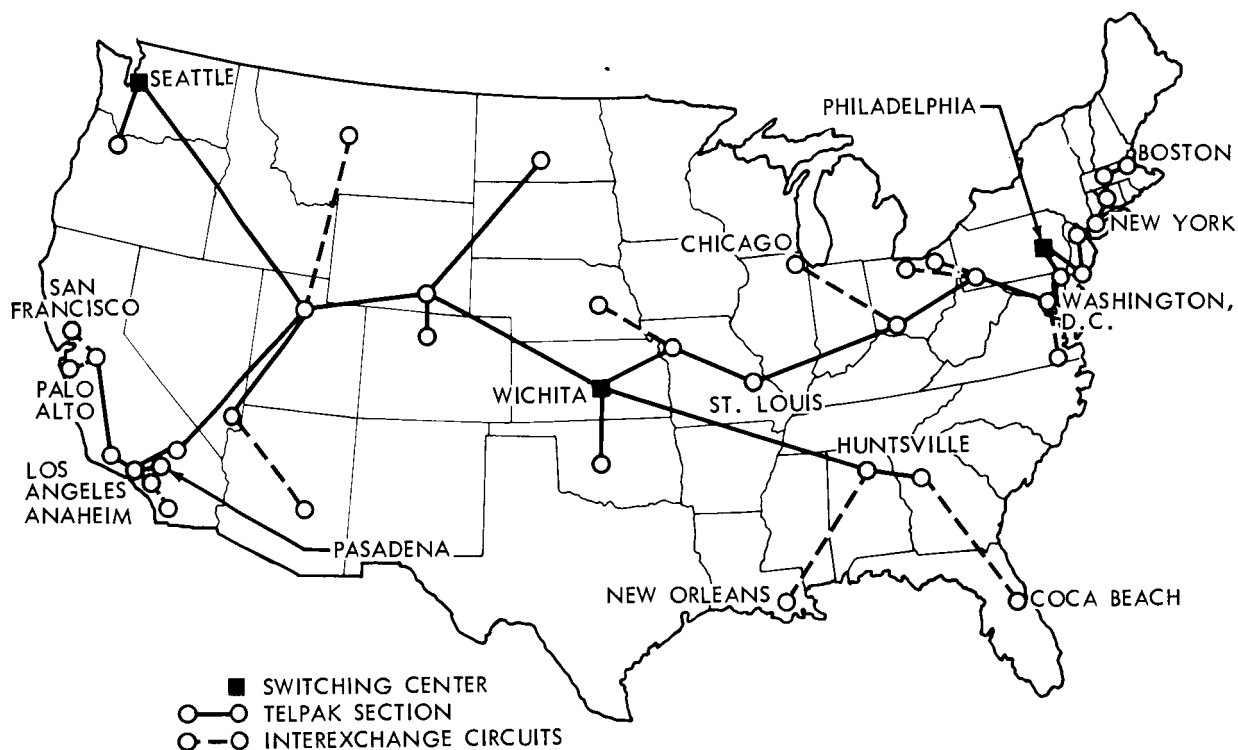


Figure 2-10: Boeing Administrative Communications Network — September 1966

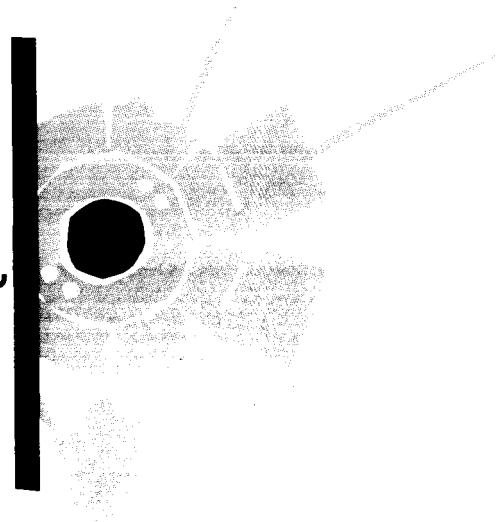
has been selected for their technical competence and demonstrated ability. They will be called upon individually and as a group to contribute to particularly complex problems. The membership of the Technical Review Board is shown in Figure 2-11.

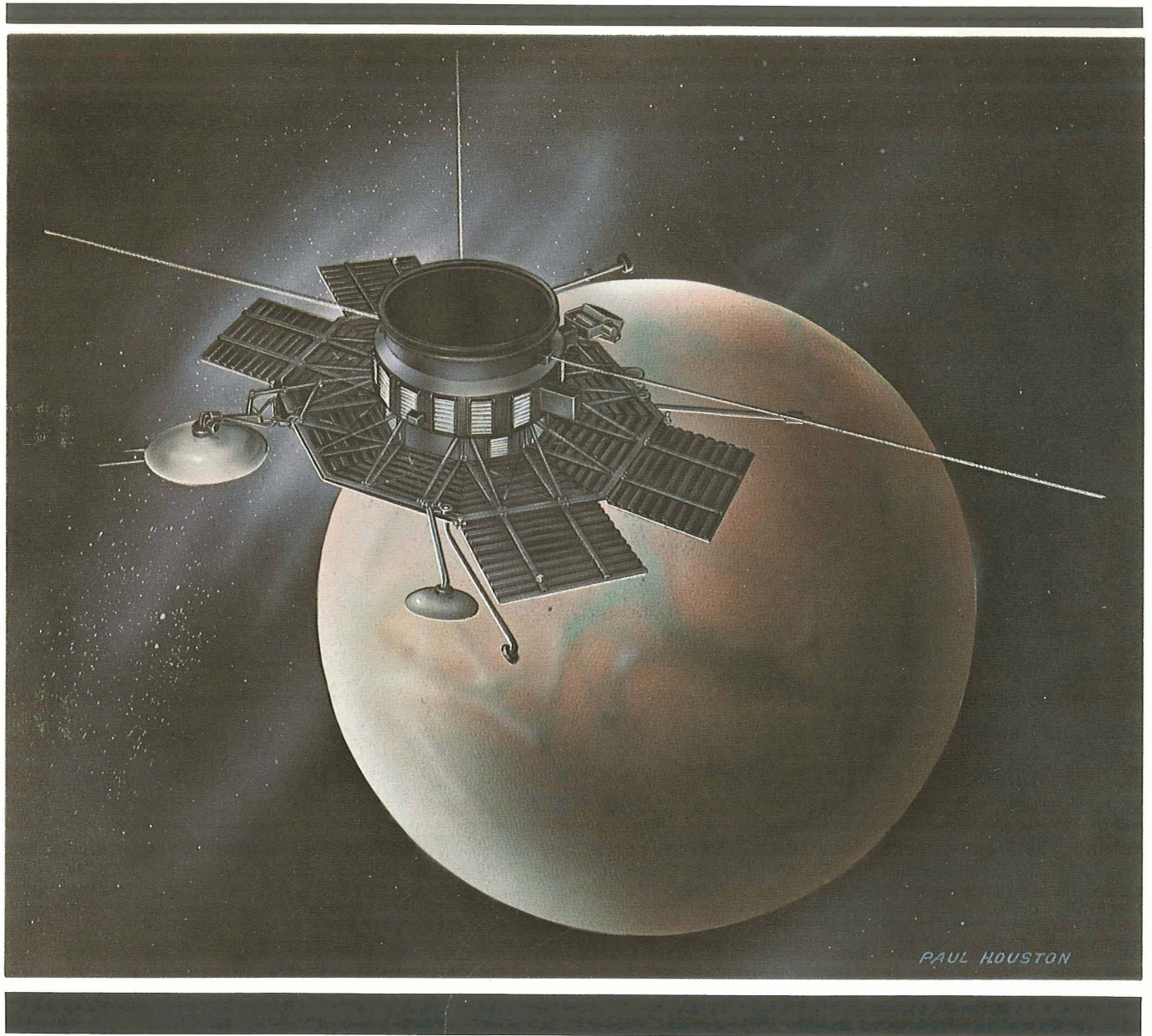
The foregoing are the highlights and rationale for the selection of the organization structure for the Boeing Voyager Spacecraft System team. The Boeing Company and the selected team members feel confident of successfully performing the assigned management tasks.

NAME	TITLE OR POSITION	AFFILIATION
E. G. Czarnecki (Chairman)	Voyager Branch Manager	Space Division
Dr. L. W. Ball	Director of Product Assurance	Boeing Aerospace Group
Dr. D. P. Chandler	Chief Scientist	Astrionics Division, NAA
Dr. W. A. Dwyer	Systems Analysis	Boeing Aerospace Group
Dr. W. A. Hane	Chief Scientist	Boeing Aerospace Group
G. L. Hollingsworth	Director	Boeing Scientific Research Laboratory
Dr. F. Proschan	Visiting Professor at University of California (Berkeley)	Boeing Scientific Research Laboratory
Dr. H. L. Richter	Corporate Area Technical Specialist	Electro-Optical Systems, Inc.
Dr. Otto Schwede	Director Technical Staff	Philco WDL
G. H. Stoner	Vice President General Manager	Space Division

Figure 2-11: Technical Review Board

VOYAGER





PAUL HOUSTON

TECHICAL HIGHLIGHTS

This section discusses the technical highlights of the preferred design evolved during the Task B study effort. It is divided into five major subsections:

- Preferred Flight Spacecraft Configuration;
- Subsystem Design;
- Mission Versatility;
- Planetary Quarantine;
- Operational Support Equipment.

The configuration and subsystem sections discuss design features that enhance reliability and performance and the extent to which the design takes advantage of Ranger, Mariner, and other space program experience.

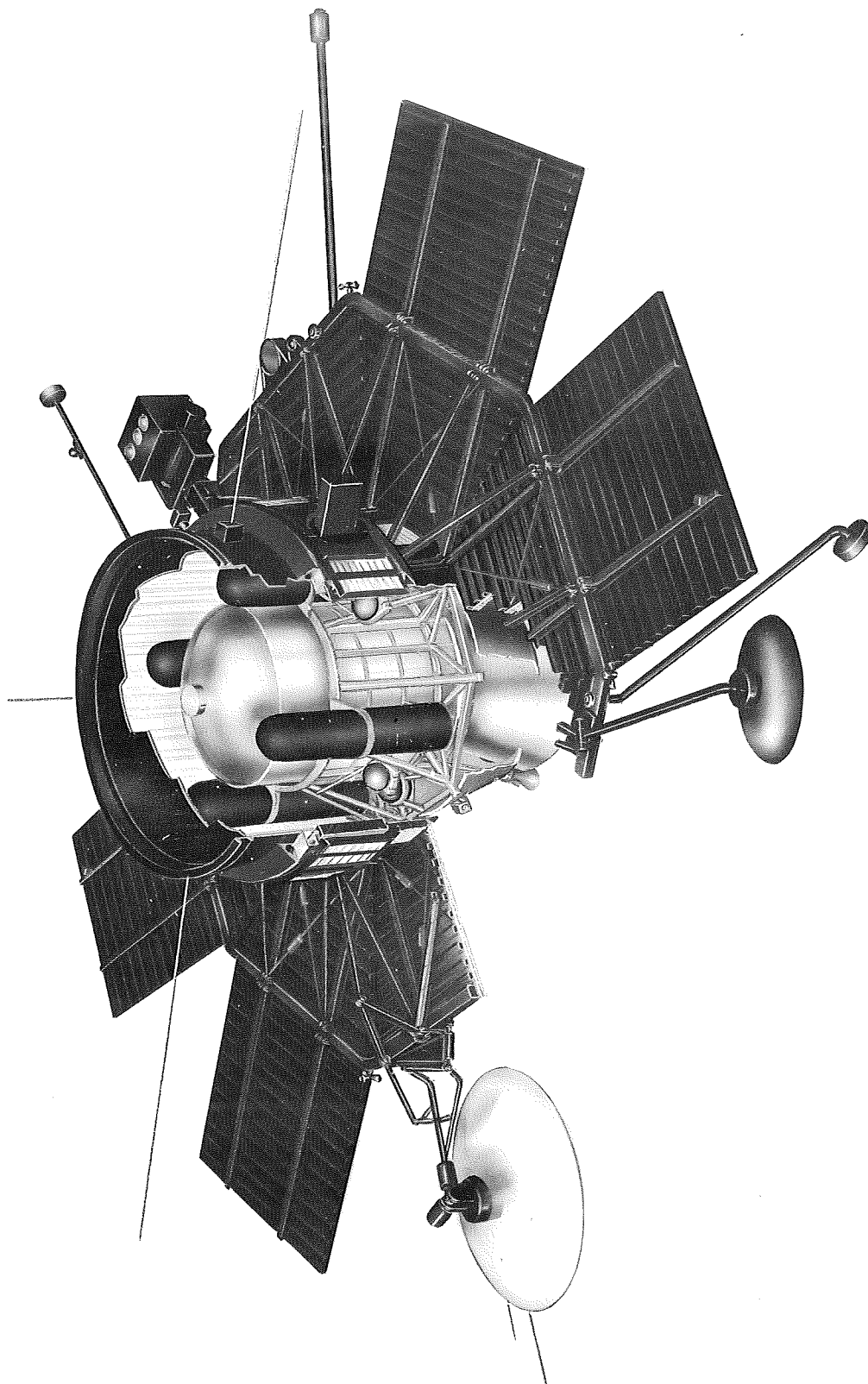
It is recognized that the primary purpose of the Voyager Program is recovery of scientific data. Therefore, the preferred design represents a responsive solution to the requirements imposed by the Flight Capsule and Science Subsystem experiments. It is further recognized that scientific experiments will change as the program evolves. To ensure experiment flexibility, Task B studies were conducted to relate postulated Science Subsystem experiments to spacecraft accommodation requirements. Boeing is prepared to discuss the design impact of varying science payloads during Phase IB to ensure that the Flight Spacecraft meets JPL scientific objectives.

The preferred spacecraft is responsive to the requirements in the JPL mission description and system specification documents. It incorporates the results of trades performed during Task B. It also serves as the basis for the schedules shown in the management proposal.

The preferred design emphasizes the JPL requirement that no single failure will cause an effect that is catastrophic to the mission. A design that meets this requirement will also improve the probability of success, which is the primary JPL competing characteristic. To further enhance this primary design characteristic, proven space hardware and design approaches from the Ranger and Mariner programs are used wherever possible.

Many decisions regarding trades between weight and reliability were made during the Task B study. Major reliability features that resulted from these design efforts are included in the following subsystem discussions. In recognition of the JPL desire to provide at least a 400-pound Science Subsystem payload, a concerted effort was made to stay within the JPL 2100-pound weight allocation for the Spacecraft Bus. During Phase IB, some of the reliability versus weight trade decisions will be reviewed with JPL to decide if certain weight allocations should be changed.

To improve cost effectiveness of the preferred design, requirements for the 1973, 1975, and 1977 missions, as well as the 1971 missions, have been satisfied if possible. As an example, it has been possible to design the propulsion module to meet all mission requirements from 1971 through 1977 without redesign and without exceeding the 15,000-pound JPL weight allocation. Additional growth possibilities that might be desirable have been identified but require weight reallocation. These possibilities will be discussed with JPL during Phase IB.



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PREFERRED FLIGHT SPACECRAFT CONFIGURATION

This section describes the 1971 Voyager spacecraft configuration displayed on the facing page, its desirable features, and some of the considerations that affected its choice.

More than 40 spacecraft configurations were evaluated. This evaluation emphasized features that enhanced overall configuration design, such as reduced spacecraft length, improved sensor views, reduced solar panel temperatures, and design flexibility. Four candidates using the selected propulsion concept were further evaluated, and a preferred design combining most of the desirable features of each was selected. Its most desirable features are summarized below.

- Modularization of subsystems permits complete checkout before installation in the spacecraft.
- Convenient access to the electronic equipment assemblies and propulsion subsystems results in ease of installation, maintenance, and testing, thereby enhancing spacecraft reliability.
- A Flight Spacecraft 50 inches shorter than the JPL-allotted dynamic envelope provides either room for growth or a 20-percent reduction in ultimate compressive boost loads due to axial and bending loads at the critical launch vehicle station.

- Unobstructed fields of view are provided during the entire 6-month mission for the antennas, science scan platform, guidance scan platform, communication antennas, and Canopus sensor with the largest Flight Capsule defined by JPL.

- Electrical power growth potential can be provided by either increasing the size of the deployable segments of the basic 316-square-foot solar array or by adding deployable segments without changing the general arrangement of the Flight Spacecraft.

- Capability exists to grow from a 6.5-foot-diameter to an 8- by 12-foot antenna with no effect on spacecraft length, if increased data rates are required.

- Capability exists in the propulsion system to accomplish all Mars missions from 1971 through 1977 without any modification.

- A 2.39-kilometer-per-second ΔV capability exists for the 1971 mission, which allows a wide range of Mars orbits. This results in periapsis altitudes as low as 400 kilometers and orbit periods as low as 2.8 hours. A more detailed discussion of orbit flexibility is contained in the mission versatility subsection, which follows.

The preferred design, shown in Figure 3-1, provides a cylindrical 120-inch-diameter semimonocoque primary structure that extends from the Planetary Vehicle Adapter interface to the Flight

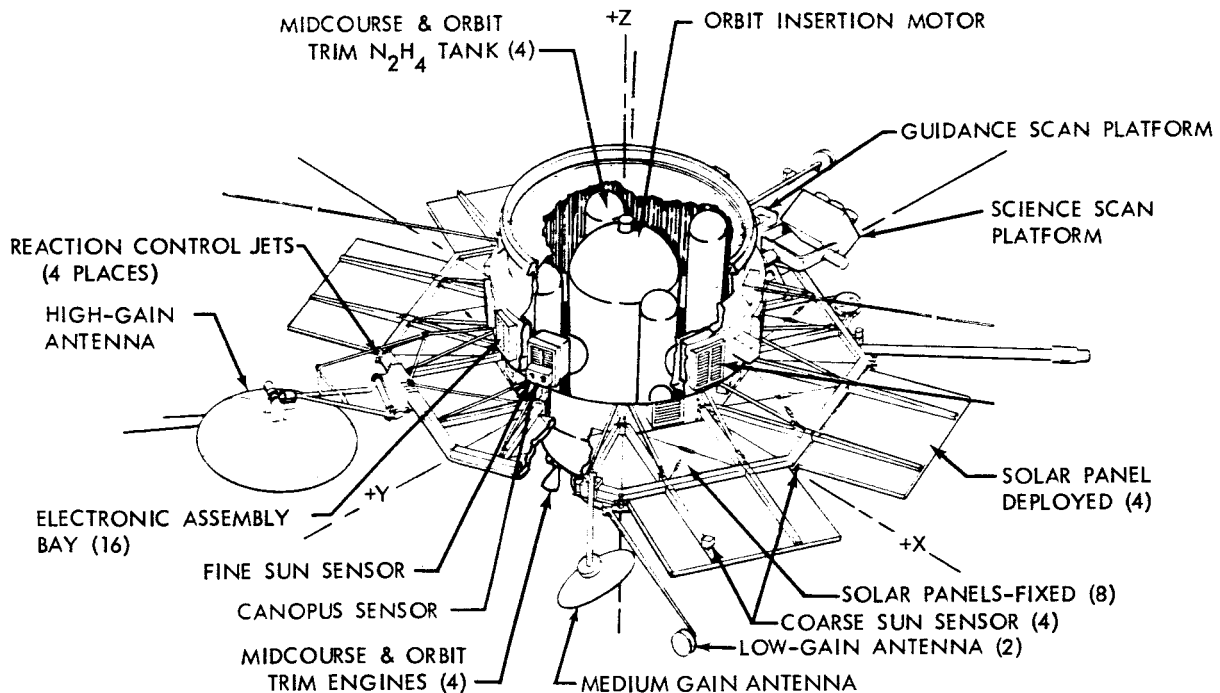
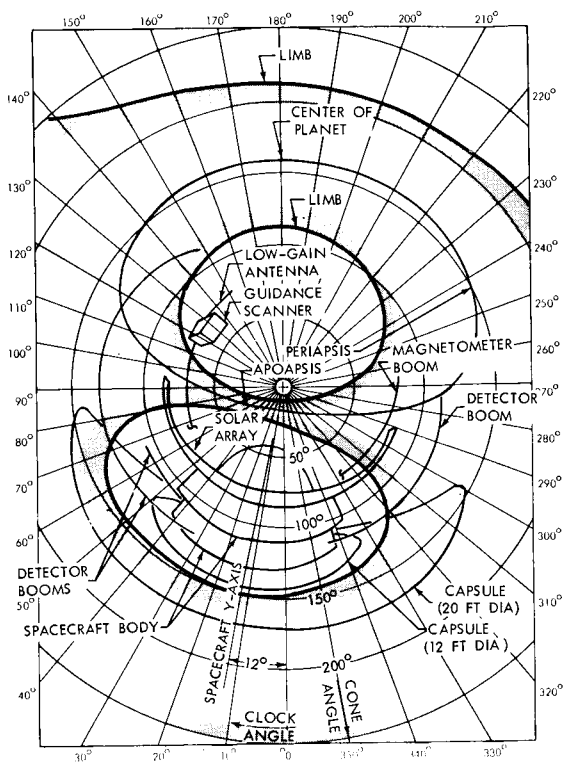
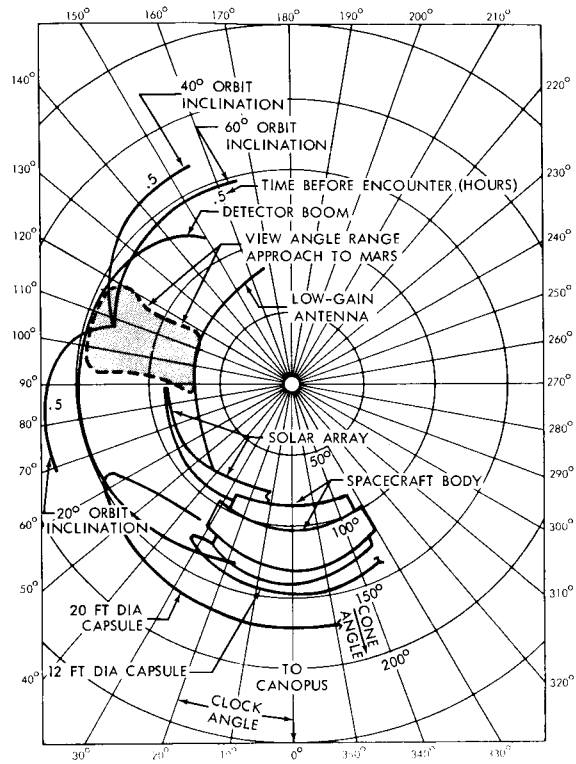


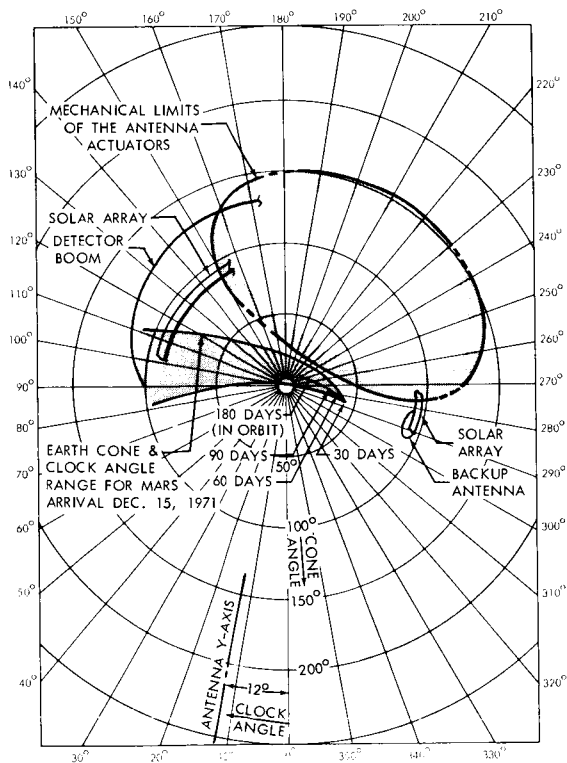
Figure 3-1: Preferred Spacecraft Design



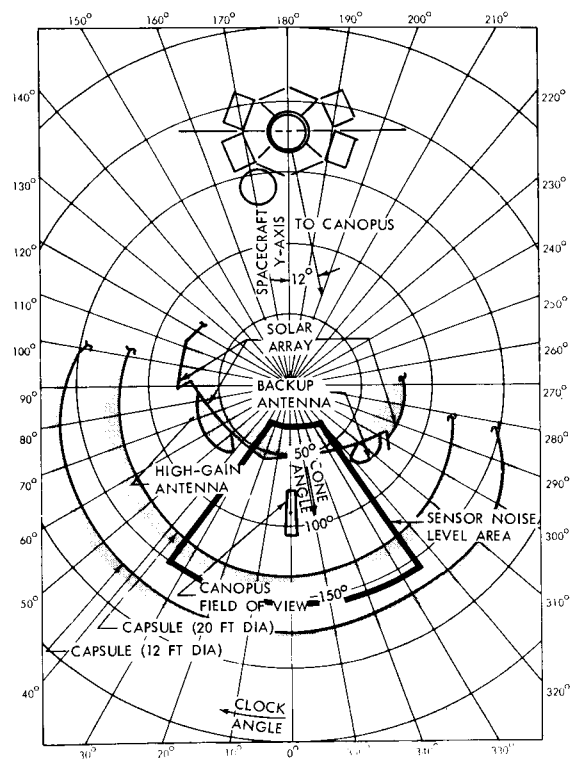
Science Scanner Shadowgraph
(Toward Sun) 1st Day Of Orbit



Guidance Scanner Shadowgraph
(Toward Sun)



High-Gain Antenna Shadowgraph
(Toward Sun)



Canopus Sensor Shadowgraph
Voyager Trajectories (Toward Sun)

Figure 3-2:

View Angles

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Capsule interface. The Planetary Vehicle Adapter, a semimonocoque, closed-section shell, supports the Planetary Vehicle at the base of the 120-inch-diameter shell. The adapter carries these loads to the nose fairing. Separation of the Planetary Vehicle from the adapter occurs at the base of the 120-inch-diameter shell.

An assembly bay located midway on the cylindrical primary structure houses electronic and electrical equipment associated with the subsystems. The electronic assemblies are mounted to the exterior surface of the primary structure and are normally shaded by the solar panels. Each assembly is handled as an entity and is located around the periphery as dictated by the requirements of thermal balance, center of gravity, cable coupling, and magnetic fields. This arrangement provides for maximum accessibility for installation, maintenance, repair, inspection, and test. The electronic assembly bay provides for sixteen standard, separate electronic assemblies 16 inches wide, 32 inches high, and 8.5 inches deep, resulting in a total usable volume of 38 cubic feet. Fourteen assemblies are used in the preferred design, leaving two for future growth. Standard radiator plates and thermal control louvers are mounted to the exterior surface of each assembly.

The propulsion module consists of a modified second-stage Wing VI Minuteman solid-propellant orbit-insertion motor and a hydrazine monopropellant system for midcourse and orbit-trim maneuvers. The propulsion subsystem module is within the 120-inch-diameter spacecraft primary structure and is attached to the base of the Spacecraft Bus. The orbit-insertion motor is oriented with the nozzle in the direction of the Sun along the vehicle longitudinal axis. The midcourse and orbit-trim engines are located in pairs 45 degrees from the X and Y axes. Four monopropellant tanks are located symmetrically about the principal control axes to maintain center-of-gravity control. Expulsion of propellant is achieved by nitrogen stored in two spherical titanium tanks. The propulsion subsystem module can be assembled, bench-checked, leak-tested, or test-fired as a complete unit before installation in the spacecraft.

The modular reaction control system uses cold nitrogen gas with coupled jet nozzles to provide attitude control. The jet nozzles (0.125-pound-thrust pitch and yaw, and 0.035-pound-thrust roll) are located at a 234-inch diameter. Four spherical tanks, located symmetrically about the longitudinal axis, supply the cold gas to the reaction control subsystem, which is completely redun-

dant. Each half-section can control the vehicle independently if a component fails in the other section.

Solar panels that provide 316 square feet of gross panel area are attached at the base of the primary body structure. This location results in excellent backface panel heat radiation characteristics and provides thermal compatibility with the largest Flight Capsule defined by JPL. The solar array is made up of eight fixed trapezoidal panels (178 square feet) and four deployable panels (138 square feet).

A major configuration effort was made during Task B to locate the Flight Spacecraft sensors to provide fields of view without interference from the Planetary Vehicle. Figure 3-2 shows the resulting fields of view for the preferred design and relates them to the desired targets and to spacecraft obstructions in terms of cone and clock angles. The resulting sensor locations are shown in Figure 3-1.

Table 3-1 summarizes weights of the various elements of the Planetary Vehicle. In all cases, the current weight estimates are less than the JPL allocation. The difference between the current weight estimate and the weight allocation is shown as a contingency. A 182-pound contingency for the Spacecraft Bus and a 158-pound contingency for Spacecraft Propulsion Subsystem inerters is considered adequate for weight growth considering the state of development of the preferred design and the amount of existing equipment involved. The 650-pound contingency on the Planetary Vehicle Adapter is sufficient to allow either design for the 1977 mission weights or reallocation to other system elements.

Table 3-2 summarizes Boeing-assigned reliability allocations and an assessment of design reliabilities achieved based on 1 month of orbital operation. The reliability allocations were derived by revising the Task A allocation considering additional equipment functions and introduction of alternate or redundant equipment.

The Flight Spacecraft reliability assessment exceeds the allocation. Known improvements investigated during Task B, but not integrated into the preferred design, could increase the reliability of the Flight Spacecraft from 0.72 to 0.82 but also entail a 153-pound weight increase (see Section 3.11 of Volume A). Further consideration will be given to these and newly identified reliability improvements during Phase IB.

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Table 3-1: Preferred System Weight Summary (Pounds)

	ALLOCATED	CURRENT	CONTINGENCY
CAPSULE SYSTEM	2000 *	2000*	0
SPACECRAFT BUS	2100	1918	182
POWER		542	
GUIDANCE AND CONTROL		266	
RADIO		163	
TELEMETRY		57	
DATA STORAGE		108	
COMMAND		31	
COMPUTING AND SEQUENCING		45	
PYROTECHNIC		34	
TEMPERATURE CONTROL		80	
STRUCTURAL AND MECHANICAL		408	
CABLING		184	
SPACECRAFT PROPULSION	15,000	14,842	158
MIDCOURSE AND ORBIT TRIM PROPULSION		3962	
ORBIT INSERTION PROPULSION		10,400	
STRUCTURE		337	
THERMAL CONTROL		118	
CABLING AND POWER CONDITIONING		25	
SPACECRAFT SCIENCE	400 *	400*	0*
(PLANETARY VEHICLE)	(19,500)	(19,160)	(340)
PLANETARY VEHICLE ADAPTER	1500	850	650
	(21,000)	(20,010)	(990)

* FOR REFERENCE ONLY

In meeting the JPL objective that no single electrical or electronic failure will cause a catastrophic mission effect, the preferred design has made maximum use of redundancy modes, which require neither sensing nor switching, and alter-

nate paths, which require no additional redundant equipment. For the Task B study, a catastrophic mission effect was defined as one that would prevent the acquisition and transmission to Earth of any scientific data by the spacecraft during Mars orbital operations.

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Table 3-2: Preferred System Reliability Evaluation Summary

SYSTEM ELEMENT	ALLOCATED RELIABILITY	ASSESSED RELIABILITY
Power Subsystem	0.980	0.9932
Computing and Sequencing Subsystem	0.970	0.9914
Command Subsystem	0.970	
Guidance and Control Subsystem	0.994	0.9945
Radio Subsystem	0.970	0.9820
Telemetry Subsystem	0.990	0.9978
Data Storage Subsystem	0.940	0.9490
Structural and Mechanical Subsystem	0.998	0.9984
Pyrotechnics Subsystem	0.999	0.9999
Temperature Control Subsystem	0.996	0.9990
Cabling Subsystem	0.999	0.9960
Spacecraft Bus	0.821	0.9045
Planetary Vehicle Adapter	0.999	0.9990
Propulsion Subsystem	0.995	0.9962
Science Subsystem (six primary experiments)*	0.800	0.8001
Reliability Contingency	0.950	
Flight Spacecraft	0.621	0.7202
Flight Spacecraft Performance	0.993	0.9930
No Meteoroid Damage	0.990	0.9900
Flight Spacecraft P_S (Total)	0.611	0.7080
P_S (at least 1 out of 2 spacecraft)	0.849	0.9147
Spacecraft OSE (Mission Critical)	0.990	0.990
Launch Vehicle P_S	0.900	0.900
Launch on Time	0.990	0.990
MOS and TDS (Mission Critical) P_S	0.980	0.980
P_S (Mission Total)	0.734	0.7907

*Boeing evaluation of experiments

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SUBSYSTEM DESIGN

This section summarizes the elements, functional description, design conditions, and technical highlights of the major subsystems. Because they are related, the radio, command, data storage, and telemetry subsystems are discussed together under "Telecommunications." Figure 3-3 shows the major subsystem elements of the Flight Spacecraft.

POWER SUBSYSTEM

The power subsystem shown in Figure 3-4 consists of an array of eight fixed and four deployable solar cell panels, three silver-cadmium batteries, three battery chargers, three series switching d.c. regulators, three 2400-cps inverters, two 400-cps inverters, failure-sensing circuits, power-switching and control equipment, and two precision oscillators with countdown registers.

The power subsystem provides electrical power to operate the spacecraft during all phases of the mission and controls power during spacecraft

test and prelaunch activities. Power is distributed in the form of 50-volt, 2400-cps, single-phase a.c. and 400-cps, single-phase a.c. Unregulated 37- to 100-volt d.c. power is supplied to a limited number of selected loads. The solar cell array provides power during the on-Sun portion of the mission. Three 38-cell silver-cadmium batteries, rated 2720 watt-hours total, provide electrical power during off-Sun operations. Each battery has its own charger. Three identical 175-watt pulse-width-modulated series switching regulators convert the raw d.c. power to 35 ± 0.35 volts d.c. Each regulator supplies a 150-watt single-phase inverter that produces 2400-cps square-wave a.c. power for one of the three distribution buses. Redundant precision oscillators and countdown registers provide 460.8-, 72-, 19.2-, 2.4-, and 0.4-kc 4.5-volt square-wave frequencies. A power-switching and logic unit commands the transfer from external power to on-board spacecraft power and automatic switching from solar array to battery power. A power control unit switches the 400-cps power and provides overload protection circuits for raw d.c. circuits.

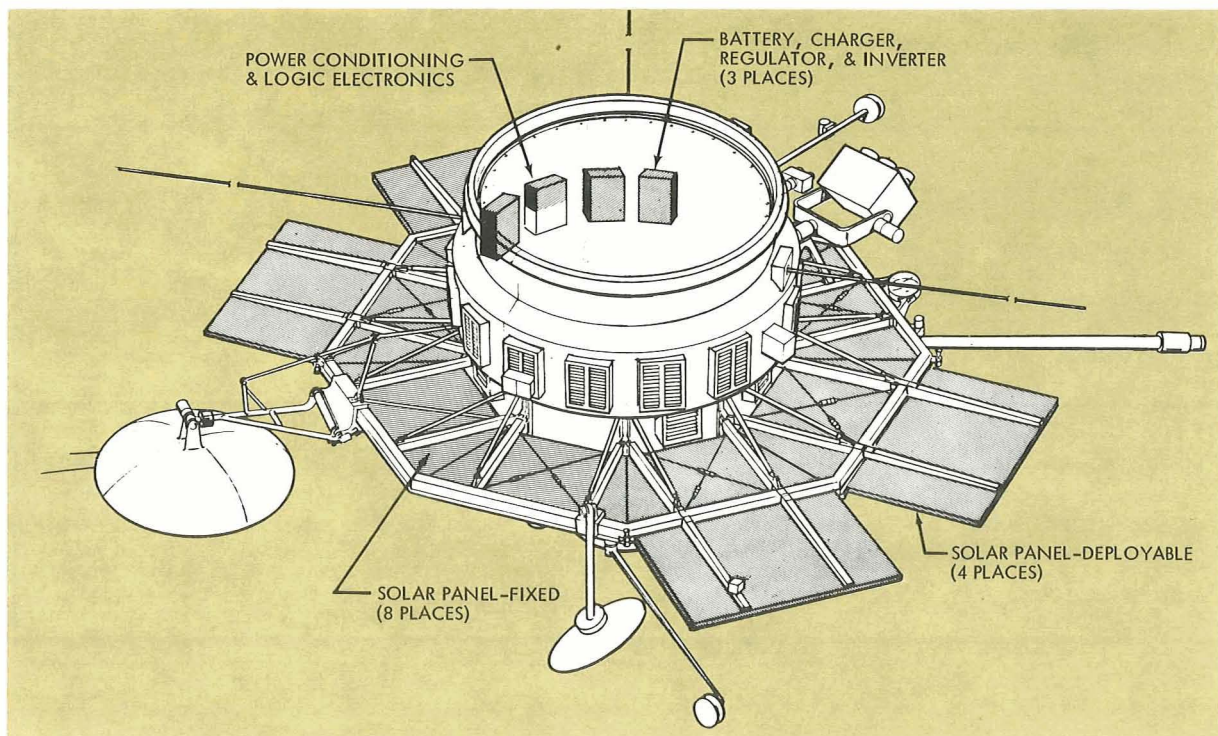
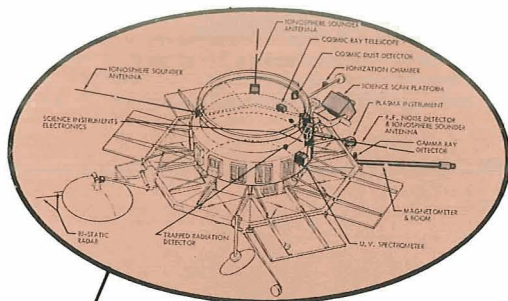
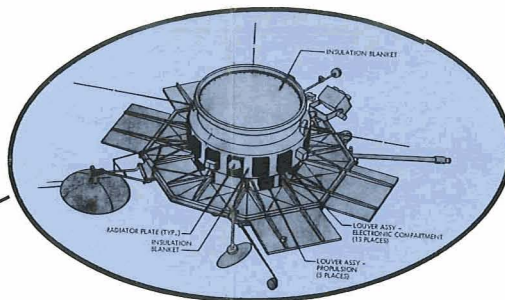


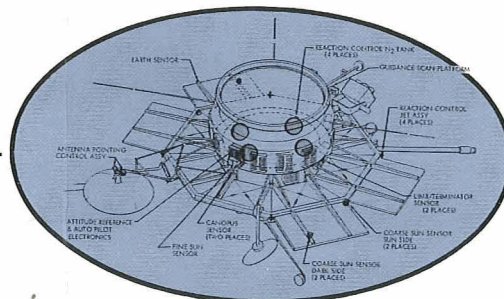
Figure 3-4: Power Subsystem



GFE
SCIENCE SUBSYSTEM

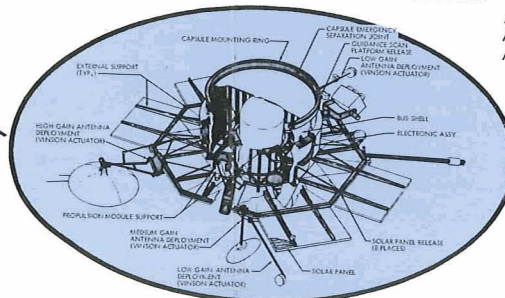


BOEING
TEMPERATURE CONTROL
SUBSYSTEM

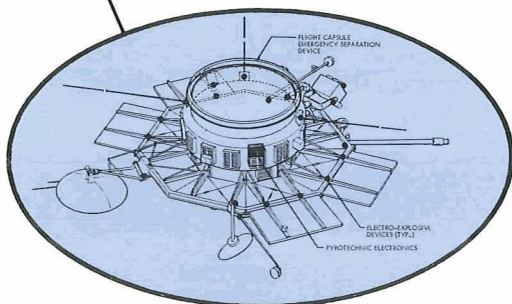


BOEING
GUIDANCE & CONTROL
SUBSYSTEM

AUTONETICS
AUTOPILOT AND
ATTITUDE REFERENCE



BOEING
STRUCTURAL & MECHANICAL
SUBSYSTEM



BOEING
PYROTECHNICS SUBSYSTEM

craft Subsystem

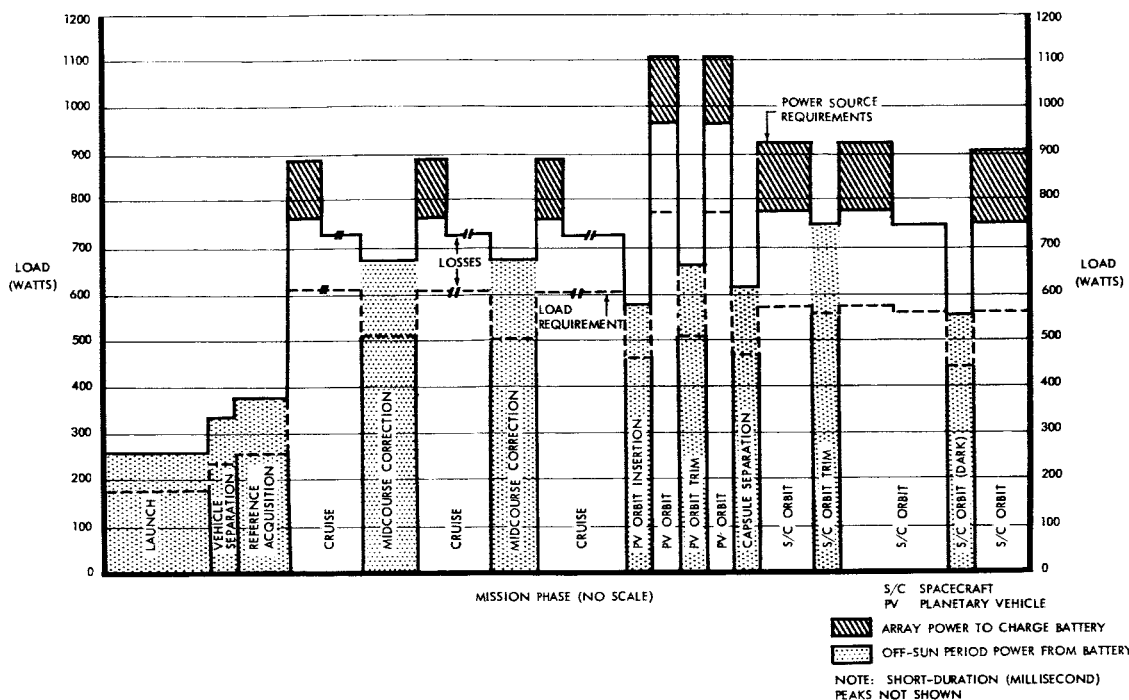


Figure 3-5: Load Profile

The power subsystem design is based on the electrical load profile shown in Figure 3-5. The solar panel array size (316 square feet) was established by the design goal for 6 months in Mars orbit and the estimated 908-watt load at the end of the mission. The battery size was determined by the longest probable off-Sun operating period, which was 3.7 hours occultation in a 24-hour orbit.

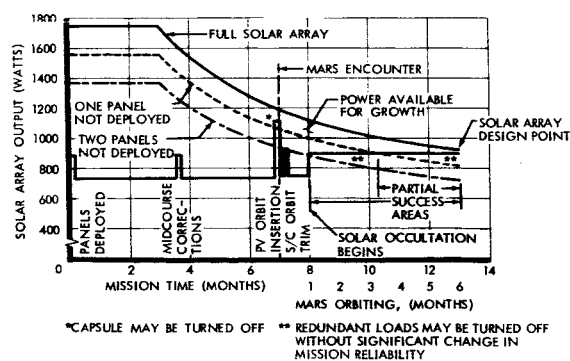


Figure 3-6: Solar-Array Power Capability and Load Requirements

The solar array provides 10-percent redundancy of solar cells so that cell failures can occur without endangering mission success. Figure 3-6 compares the array requirements established by the estimated load profile and the performance

capability, considering the effect of solar intensity variation and solar array degradation with time. As shown by the curve, adequate solar array area has been provided so that, if one solar panel fails to deploy, the mission can be conducted with no degradation in performance up through 3.5 months in Mars orbit. In addition, if a serious loss of solar array or battery power occurs, the ability exists to disconnect spacecraft redundant loads by ground command and continue the mission.

Additional features were included in the design to achieve high reliability:

- A three-battery configuration was selected which allows successful mission completion with only two batteries operating. Each battery charger has failure-sensing and switching capability to disconnect the charger if a battery or charger fails;
- Each electrical load has a redundant bus so that a fault of one bus or a fault in any load connected to that bus will not affect the mission;
- Automatic overload protection and failure sensing and switching prevent load faults or a fault within a regulator or inverter associated with one bus from affecting power quality on the remaining two buses; regulator power can be restored by ground command after the fault has been cleared;

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- Two redundant, raw d.c. circuits furnish power to equipment requiring such power;
- A diode logic network is arranged so that spacecraft battery power cannot be used by the capsule and so that a short-circuit failure of a battery charger will disable only a third of the solar array and will not directly consume battery power.

The solar array with N on P cells, corrugated aluminum substrate, and deployment devices is similar to the Mariner C solar array. Sealed silver-cadmium batteries were used on the Explorer II series spacecraft and the last OGO satellite, although without the high overcharge capability of the Voyager design. Power-conditioning electronics are in most cases similar to Ranger and Mariner equipment.

COMPUTING AND SEQUENCING SUBSYSTEM

The computing and sequencing (C&S) subsystem is a cycled, special-purpose, programmable digital computer consisting of redundant data processors, each containing a 1024-word magnetic core memory, an interprocessor control unit, and redundant input-output signal conditioning. The location of this equipment is shown in Figure 3-7. Figure 3-8 is a simplified block diagram of the C&S that identifies major input-output interfaces.

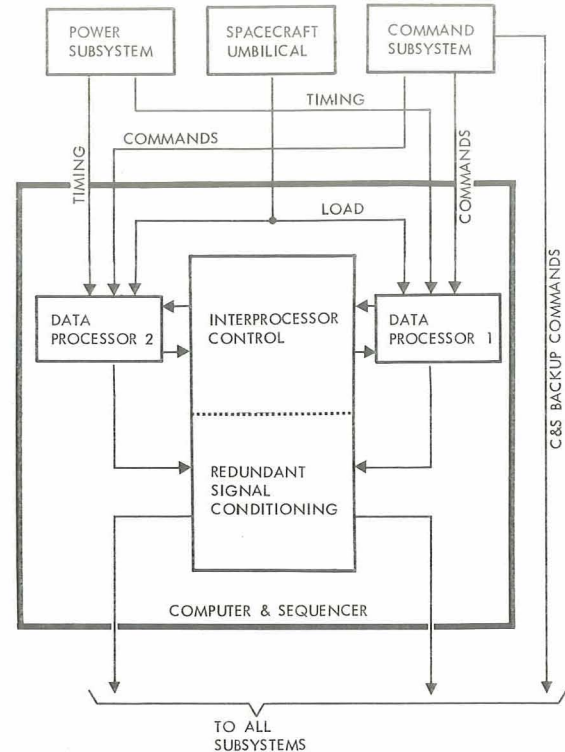


Figure 3-8: Computing and Sequencing Subsystem Interfaces

The primary function of the C&S is to control the sequencing of all spacecraft events that must occur as a function of time during the mission.

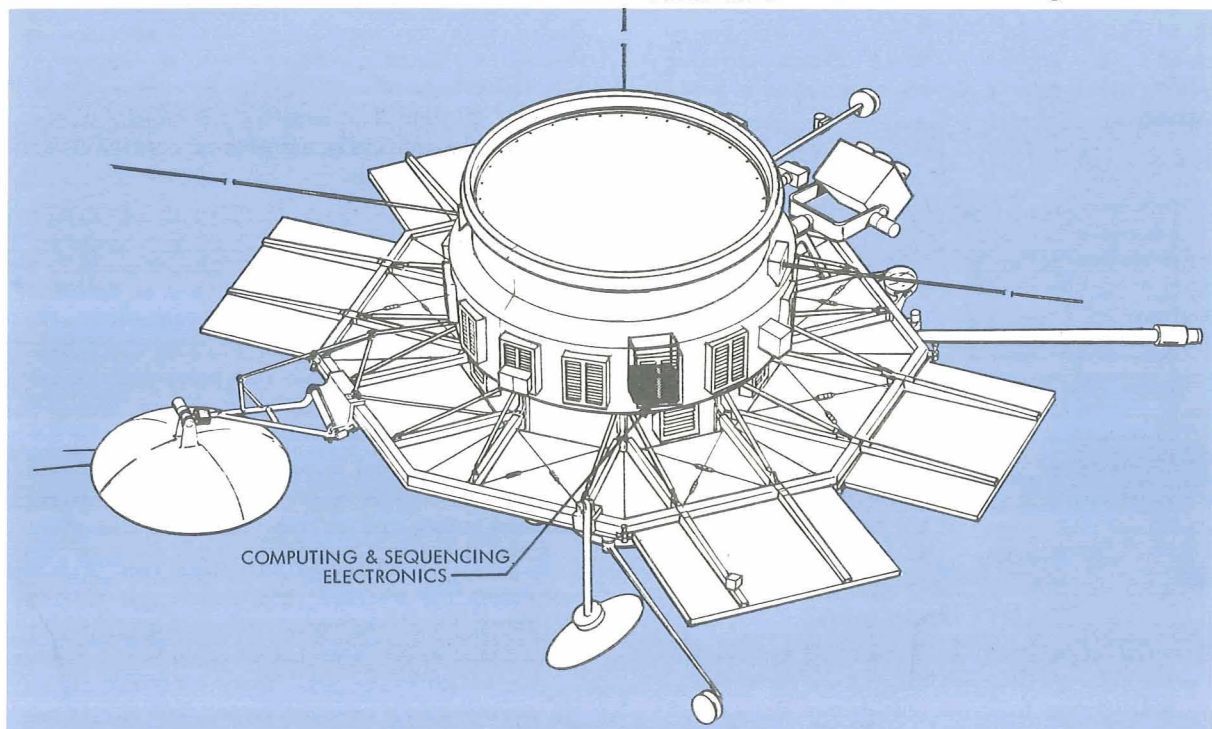


Figure 3-7: Computing and Sequencing Subsystem

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Additional sequencing is performed as a function of on-board condition or event sensing (i.e., terminator crossing or Canopus occultation). It provides automatic event control from launch through end of mission without ground command unless required by trajectory biasing or dispersions. Provision is made for ground updating or override capability by the command subsystem.

More than 170 output signals to the spacecraft subsystems and capsule are provided for fully automatic flight control. These signals initiate on-board events, provide magnitude and polarity for attitude control and provide for data transmission to Earth. These signals are generated by the stored flight program using internal data such as time and input signals from the command and other subsystems.

Spacecraft time is accurate to within ± 20 seconds for orbit insertion. It can be updated by ground command. The timing for spacecraft control is computed from several base times (e.g., time from launch, time of Mars arrival, and time from an event).

The C&S will not fail to issue a correct command in the event of any single electronic part failure. Redundant processors, operating in a synchronous parallel mode with interprocessor data transfer and cross-checking, provide the most efficient method of preventing mission loss from a single failure. Of the several methods considered, this

technique provided the least part count, weight, and power consumption.

The preplanned program routines required to execute mission events are loaded into the computer before launch. The program can be readily modified on the ground or in flight to accommodate off-nominal conditions or changes in mission plans. The programs include subroutines for error detection and correction by cross-comparison of data and past-event checking to prevent false commands. All output commands are issued as a.c. signals to permit d.c. isolation of signal paths to other subsystems.

The C&S is capable of reconstituting its total state and all of its outputs in the event of a temporary power failure. This is achieved by power failure sensing circuitry that generates an interrupt signal to the computer. Upon receipt of the interrupt signal, the computer transfers to a shutdown routine, which stores all of the information to regenerate the state of the computer and its outputs when power is reapplied.

The 1024 (27 bit) word memories are adequate for current mission requirements and provide a 30-percent reserve for growth and future missions.

GUIDANCE AND CONTROL SUBSYSTEM

Major elements of the guidance and control (G&C) subsystem are shown in Figure 3-9 and include the attitude references, autopilot, reaction-

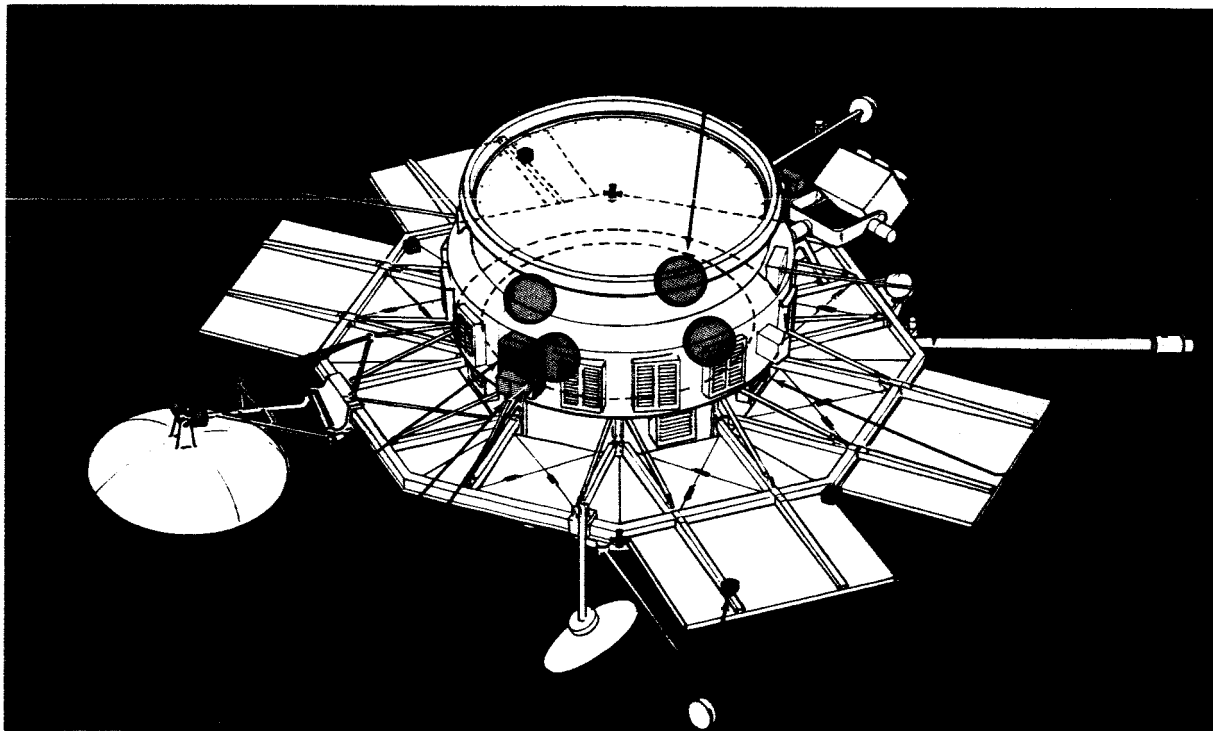


Figure 3-9: Guidance and Control Subsystem

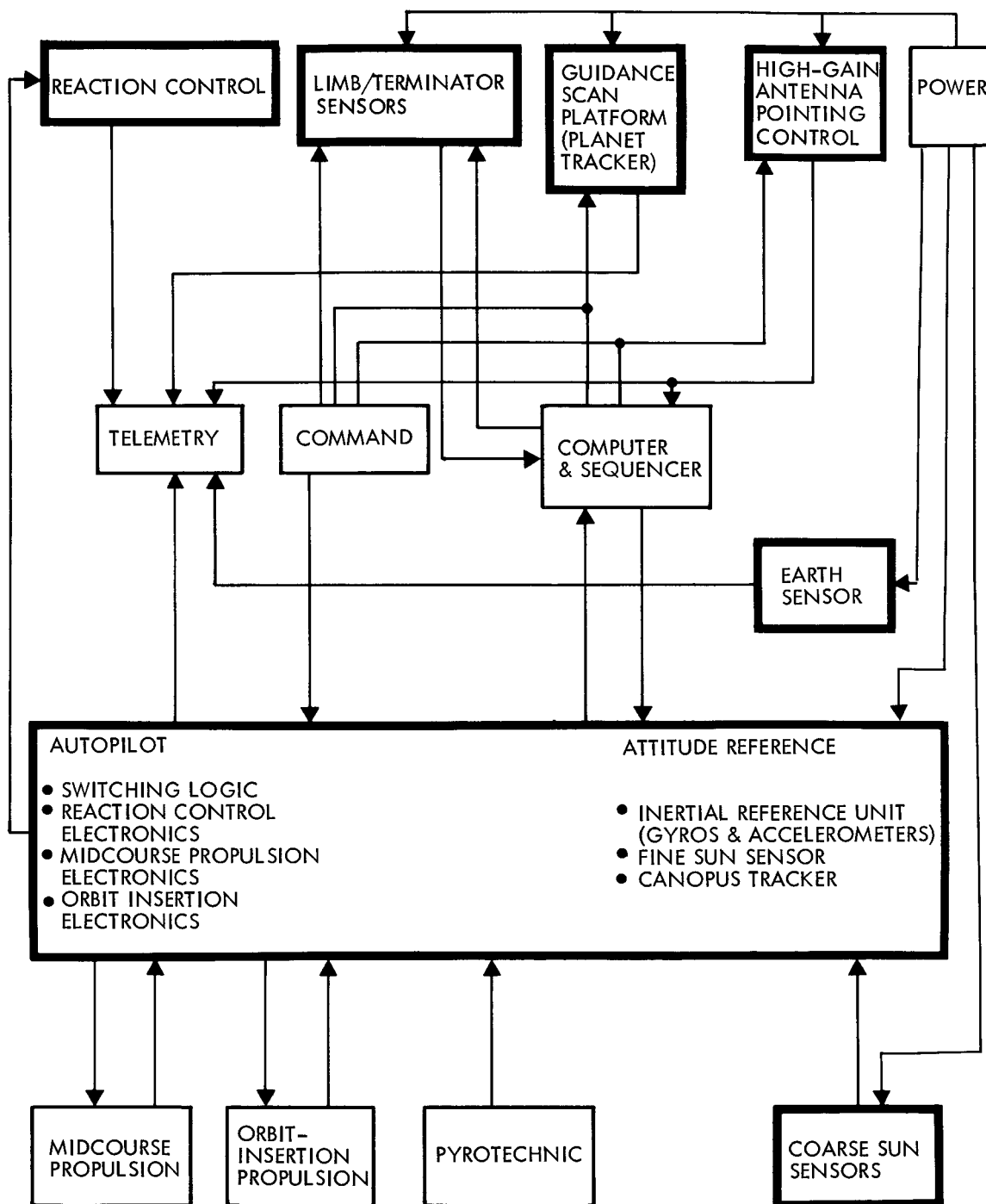


Figure 3-10: Guidance and Control Subsystem and Prime Interfaces

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control equipment, high-gain-antenna pointing controllers, guidance scan platform assembly, accelerometers, and the limb and terminator detector.

The most important task of the G&C subsystem is maintaining accurate and reliable attitude stabilization during cruise, maneuvers, orbit, and thrusting. Significant interfaces between the G&C elements and other subsystems are shown in Figure 3-10.

The thrust levels and minimum on-time of the reaction control jets and the dead band of the autopilot were designed to satisfy the requirement that attitude be maintained within ± 0.3 degree and that the attitude rates not exceed 0.007 degree per second during cruise conditions. The ± 0.3 -degree error is the attitude control allocation of the overall ± 0.94 -degree high-gain-antenna pointing error allowance. The 0.007-degree-per-second angular rate is the design objective for the science scan platform. Maneuvering accuracies, autopilot gains, and various alignment tolerances are established by the requirement to maintain ΔV pointing error within 0.5 degree (one sigma).

Use of the proven nitrogen cold-gas concept of reaction control for Voyager is possible because of efficient operation. This efficiency is achieved by basing the design on low operation rates for maneuver and limit cycle control. All Voyager attitude-control elements are redundant because failure of a single-thread attitude-control system would cause catastrophic mission loss. With this redundancy, reaction-control capabilities of the G&C subsystem are greater than normal mission requirements. If no open thruster failures occur, sufficient nitrogen quantities are provided to permit active attitude control to be maintained for 4 years.

The cumulative error in G&C elements is within the limits required for pointing the high-gain antenna, science and guidance scan platforms, and thrust vector control. Antenna pointing error contributes less than a 1-decibel loss to telemetry transmission. Contributors to this error include antenna-position control, attitude sensing, spacecraft stabilization, and structural deflection. In addition, gyro drift during a 3.7-hour Sun occultation are compatible with antenna- and scan-platform pointing requirements.

Fully automatic operation of the G&C subsystem is achieved by a combination of internal logic and time-oriented commands generated by the com-

puter and sequencer (C&S). If C&S command capability is lost, the command subsystem can perform all G&C sequencing. The command subsystem also can provide failure override commands to those G&C elements that employ malfunction detection equipment.

Design of the G&C subsystem features components of known performance:

- Redundant Sun sensor based on the cadmium-sulfide detectors used on Mariner C;
- Three modified Minuteman free-rotor gyros that provide dual redundancy in all three axes;
- Two JPL/Barnes Canopus sensors similar to those used on Mariner C;
- Analog autopilot electronics that, with reaction-control and thrust vectoring actuators, provide cooperative multichannel redundancy;
- Two independent cold-gas reaction-control systems identical in concept to those used on Ranger and Mariner;
- A Bell Model DVM IIIB accelerometer similar to that used on Mariner II;
- Mariner C planet sensor with modified optics;
- Limb and terminator detector based on the cadmium-sulfide detectors used on Mariner C;
- Hermetically sealed harmonic-drive positioners for the high-gain antenna and guidance scan platform.

TELECOMMUNICATIONS

Voyager telecommunications include the radio, data storage, command, and telemetry subsystems. Components of these subsystems are shown in Figure 3-11.

The radio subsystem is a fully redundant, conventional S-band system comprised of low-gain omnidirectional, fixed medium-gain, and gimbaled high-gain antennas; preamplifiers and S-band receivers; 1-watt launch transmitter; and exciters for 50-watt traveling-wave-tube (TWT) power amplifiers. Planetary ranging and doppler tracking electronics are also part of this subsystem. The data-storage components include seven recorders with a total capacity of more than 3.8×10^8 bits and input-output data buffers. The command subsystem includes redundant command detectors and decoders, and selection logic. Major elements of the telemetry subsystem are biphase

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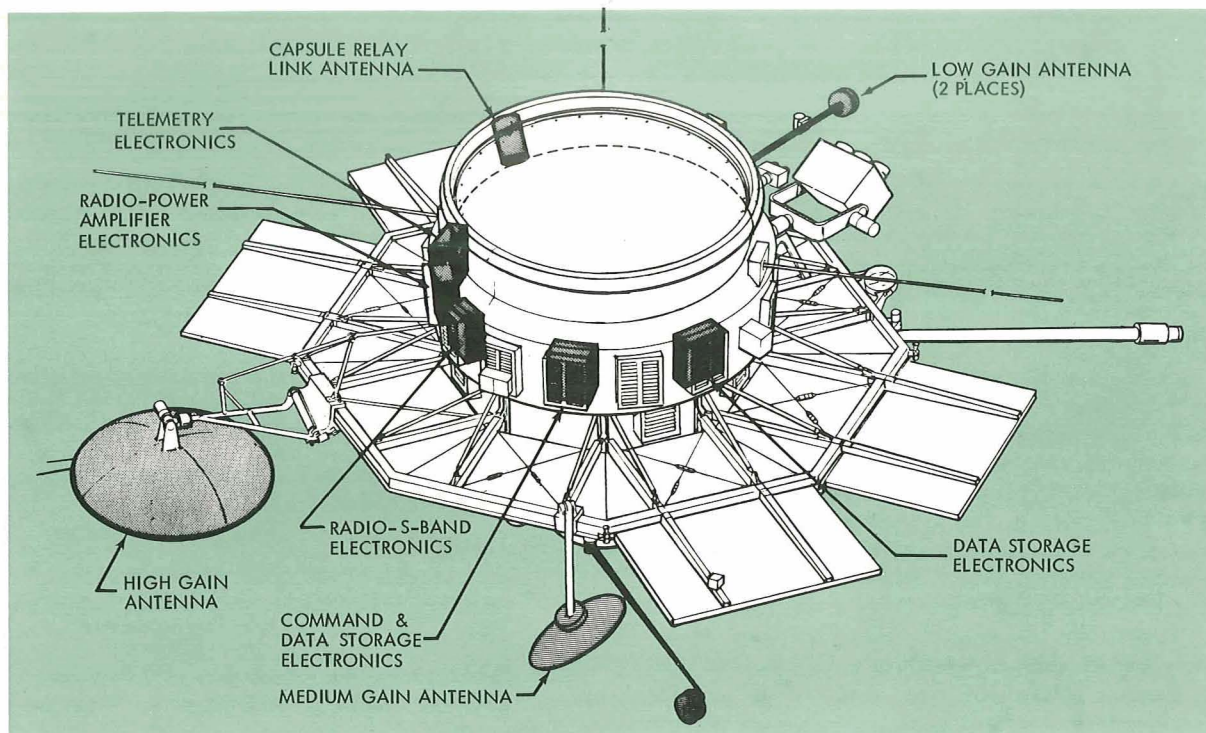


Figure 3-11: Telecommunications

modulators, a pseudo-noise sync generator, multiplexers, encoders, and a master digital programmer.

Telecommunication functions are to:

- Provide a coherent radio-frequency carrier signal for doppler tracking of the spacecraft by DSN stations;
- Provide storage, processing, and transmission of engineering and scientific data;
- Receive and process ground commands for the C&S and other spacecraft subsystems;
- Receive and transmit a planetary ranging code;
- Receive, process, and transmit capsule data.

Figure 3-12 shows the functional relationships in the telecommunications system and its interfaces with other subsystems.

The preferred telecommunication design was selected to fulfill the requirements of "JPL Mission Description and Spacecraft General Specification." These requirements were further clarified during this study. The principal design considerations are enumerated below.

- Telecommunication data-handling capacity to accommodate:

- Planetary science data stored during sunlit periapsis (approximately 2 hours) and transmitted during the remainder of the orbit. For the first 30 days, the assumed input data rate is 66,700 bits per second for the television and 4800 bits per second for the infrared and ultraviolet data.
- Capsule entry data transmission: the assumed input rate (from the relay radio) is 50,000 to 200,000 bits per second for 40 minutes.
- Spacecraft engineering, capsule engineering, and cruise science data during the mission. The data rate is estimated to be 288 bits per second during the first 30 days in orbit. Approximately 1 hour of storage for this data must be available during maneuvers. Separate storage is required for cruise science data; the assumed rate is 825 bits per second for 20 minutes.
- Command capability is required to completely update the C&S, provide backup for all C&S commands, and provide additional malfunction switching commands. The capability to receive commands during abnormal spacecraft attitude or antenna misorientation is essential. A word-error probability of

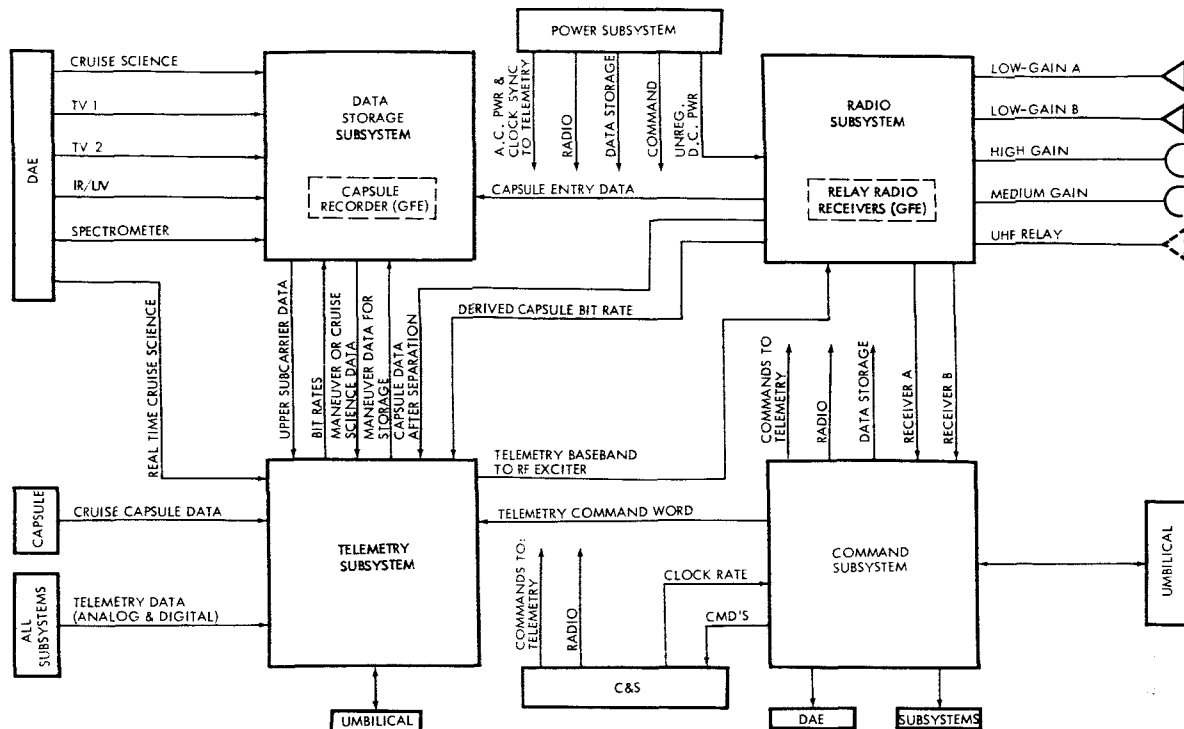


Figure 3-12: Telecommunications Functional Interfaces

less than 10^{-8} is required. The probability of executing a false command is less than 10^{-20} with a command bit error rate of 10^{-5} .

- Command message formats and verification techniques, tracking and ranging methods, and telemetry data are compatible with DSN capability defined in EPD 283. Maximum telemetry data-transmission rate is 7500 bits per second from either Flight Spacecraft with growth potential to 15,000 bits per second.
- Worst-case communication-link margins are used to establish subsystem design. The permissible bit error rate is 5 in 10^3 bits for the telemetry link and 1 in 10^5 bits for command transmission.

As proposed in Task A, (16,5) biorthogonal coding of the telemetry bit stream has been incorporated. About 2 decibels of improvement in overall link gain is achieved by this technique, with negligible effect on reliability. This permits the high-gain-antenna diameter to be reduced from 8 to 6.5 feet for the same bit rate and saves about 15 pounds in structure. This weight saving is allocated to improve probability of mission success.

Complete redundancy has been incorporated in all critical functional paths. This includes dual

rf exciters and power amplifiers, two low-gain antennas, each connected to separate receivers, and the Mariner antenna as a backup to the high-gain antenna. The telemetry subsystem contains redundant circuits. The data-storage subsystem contains at least two of each type of recorder; should a recorder fail, its function can be time-shared on another recorder, with some reduction in capacity for both functions. The backup assignment for the recorders is presented in Table 3-3.

Table 3-3: Recording Capability

RECORDER PRIME FUNCTION	CAPACITY	BACKUP FOR
TV 1	1.2×10^8	Selection is by the DAE
TV 2	1.2×10^8	
Capsule Relay	1.2×10^8	TV
IR Scanner	1.0×10^7	IR-UV Spectrometer
IR-UV Spectrometer	1.0×10^7	IR Scanner
Fields and Particles	1.0×10^6	Spacecraft Data
Spacecraft Data	1.0×10^6	Fields and Particles

The data storage, telemetry, and radio subsystems are designed to provide a balanced capability for acquisition, storage, and transmission of planetary science data on each orbit. At the assumed planetary science data-acquisition rate, the data storage requirement is slightly more than 3×10^8 bits. The two 1.2×10^8 bit, parallel,

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reel-to-reel tape recorders are provided primarily for this purpose. The remaining capacity is provided by the 1.2×10^8 bit capsule recorder, which is available after capsule descent and capsule data playback. At a 7200-bit-per-second data transmission rate, these recorders can be emptied on each nominal orbit. This leaves 288 bits per second for the lower subcarrier channel to transmit engineering and low-rate scientific data.

System versatility is supplied by five basic data modes augmented by the submodes of equipment combinations shown in Table 3-4. Low-noise tunnel-diode preamplifiers in the radio subsystem allow reception of ground commands from the 100-kw DSIF transmitter and 85-foot DSIF antennas through dual low-gain-antenna receiver channels during the entire mission.

Based on assumed data requirements, telemetry modes have been designed to provide optimum performance for each mission phase. However, the system design is highly flexible, and a number of variations in the basic design are possible.

- The third 1.2×10^8 bit capsule recorder can be held in reserve as a redundant backup unit. A data rate of 4800 bits per second will then transmit the active storage capacity in one 14-hour orbit.

- At 4800 bits per second, biorthogonal encoding can be eliminated or the high-gain-antenna diameter can be reduced to 5 feet. Either change will support 1200 bits per second at end of mission.
- Sufficient link margin is available in the proposed 6.5-foot-diameter high-gain antenna, 50-watt traveling-wave tube, and 210-foot DSIF antenna configuration to support a 15,000-bit-per-second mode for at least 20 days after encounter if it is needed to accommodate scientific requirements or a low-period orbit.
- Additional modes with intermediate data rates between 7200 and 1200 bits per second can be included to increase total data output at the end of the mission.
- The TV-1 and TV-2 recorders can be used as backup to the capsule recorder. Either parallel or serial operation, or a combination of these can be programmed to satisfy capsule data requirements.

These and other possible variations will be explored with JPL during Phase IB.

STRUCTURAL AND MECHANICAL SUBSYSTEM

The structural and mechanical subsystem includes the 120-inch-diameter cylindrical primary struc-

Table 3-4: Telemetry-Mode-Link Performance Capability

MISSION PHASE	MODE (UPPER/LOWER CHANNEL - BPS)	TRANSMITTER POWER (WATTS)	SPACECRAFT ANTENNA	GROUND ANTENNA (DIA-FT)	PERFORMANCE LIMIT	
					GREYOUT (WORST CASE)	BLACKOUT (NOMINAL)
Launch	M1 (0/80)	1W	Nose Fairing	4 (Sta 71)	3.4 min.	>> Line of sight
Launch	M1 (0/80)	1W	LGA	4 (Sta 71)	> Line of sight	--
Acquisition	M5 (Carrier only)	1W	LGA	2 x 2	Launch + 9 hr	Launch + 12 hr
Acquisition	M1 (0/80)	1W	LGA	2 x 2	Launch + 65 min.	Launch + 100 min.
Early Cruise	M1 (0/80)	1W	LGA	85	Launch + 13 days	Launch + 20 days
Early Cruise	M1 (0/80)	50W	LGA	85	Launch + 81 days	Launch + 104 days
Mid-Cruise	M1 (0/80)	50W	LGA	210	Launch + 148 days	Launch + 176 days
Maneuver	M5 (Carrier only)	50W	LGA	210	Launch + 188 days	ENC. + 9 mo.
Late Cruise	M1 (0/80)	50W	HGA	210	>> ENC. + 6 mo.	--
Early Orbit	M2 (7200/288)	50W	HGA	210	ENC. + 2.9 mo.	>> ENC. + 6 mo.
Late Orbit	M3 (1200/60)	50W	HGA	210	>> ENC. + 6 mo.	--
Early Orbit	M3 (1200/60)	50W	MGA	210	ENC. + 3 1/2 mo.	ENC. + 4 2/3 mo.
Late Orbit	M1 (0/80)	50W	MGA	210	ENC. + 5 1/2 mo.	> ENC. + 6 mo.
Early Orbit	M1 (0/80)	1W	HGA	210	ENC. + 3.9 mo.	>> ENC. + 6 mo.
Early Orbit	M4 (0/1-5/8)	50W	LGA	210	ENC. + 2.7 mo.	> ENC. + 6 mo.
Late Orbit	M4 (0/1-5/8)	50W	MGA	210	> ENC. + 6 mo.	--
Late Orbit	M4 (0/1-5/8)	1W	HGA	210	>> ENC. + 6 mo.	--

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ture section between the Flight Capsule and the Planetary Vehicle Adapter; external structure that supports solar panels, antennas, and Science Subsystem components; Flight Capsule emergency separation mechanism; Planetary Vehicle separation mechanisms; and mechanisms for stowing, deploying, and locking extendable appendages. A separate structure attached to the base of the spacecraft supports the modular propulsion subsystem. The major structural and mechanical elements are shown in Figure 3-13.

The subsystem provides all structural support, deployment, and separation functions on the spacecraft. In addition, the structure serves as a thermal mass and heat transfer medium during transient thermal conditions, and provides meteoroid protection for critical components. Magnesium bumpers 0.01-inch thick spaced 1 inch from the primary structural shell above and below the electronic equipment assembly bay protect propulsion module components from meteoroids. Louvers, radiator plates, and equipment mounting frames provide meteoroid protection for the electronic assemblies. Barriers attached to insulation baffles and on either end of the spacecraft cylinder complete the meteoroid protection function.

The present JPL steady-state acceleration requirements specify a +5.6 g's axial combined with a ± 1.0 g lateral or 0 g axial combined with

a ± 1.0 -g lateral limit boost acceleration. These conditions were modified for the preferred design by conservative dynamic considerations to give +5.6 g's axial combined with ± 2.0 g's lateral and a tensile -2.8 g's axial combined with a ± 2.0 -g lateral limit acceleration. The majority of the structure elements are sized by the +5.6-g axial and 2.0-g lateral load conditions. A safety factor of 1.25 was used. The critical masses are the 2000-pound Flight Capsule and the 15,000-pound propulsion subsystem. The Flight Capsule center of gravity was assumed conservatively to be 70 inches forward of the separation plane. Appendage stiffnesses were designed to provide a minimum 4-cps first bending frequency when stowed. The design minimum frequency for the orbit-insertion condition was 2 cps. Structure enclosing electronic assembly bays and propulsion components had the additional requirement of providing a 0.99 probability of no meteoroid penetration for 6 months in Mars orbit.

A Planetary Vehicle plus Adapter modal analysis was performed with 100 nodes and 300 structural members. The important lower frequencies were 9.6 cps for Flight Spacecraft axial response, 9.9 cps for Planetary Vehicle bending, and 21.2 cps for propulsion module axial response. A Planetary Vehicle dynamic analysis for longitudinal loads yielded accelerations of only +1.6 g's for S-IC engine ignition, +1.55 g's for launch release, and +5.6 to -0.9 g's for S-IC cutoff. The

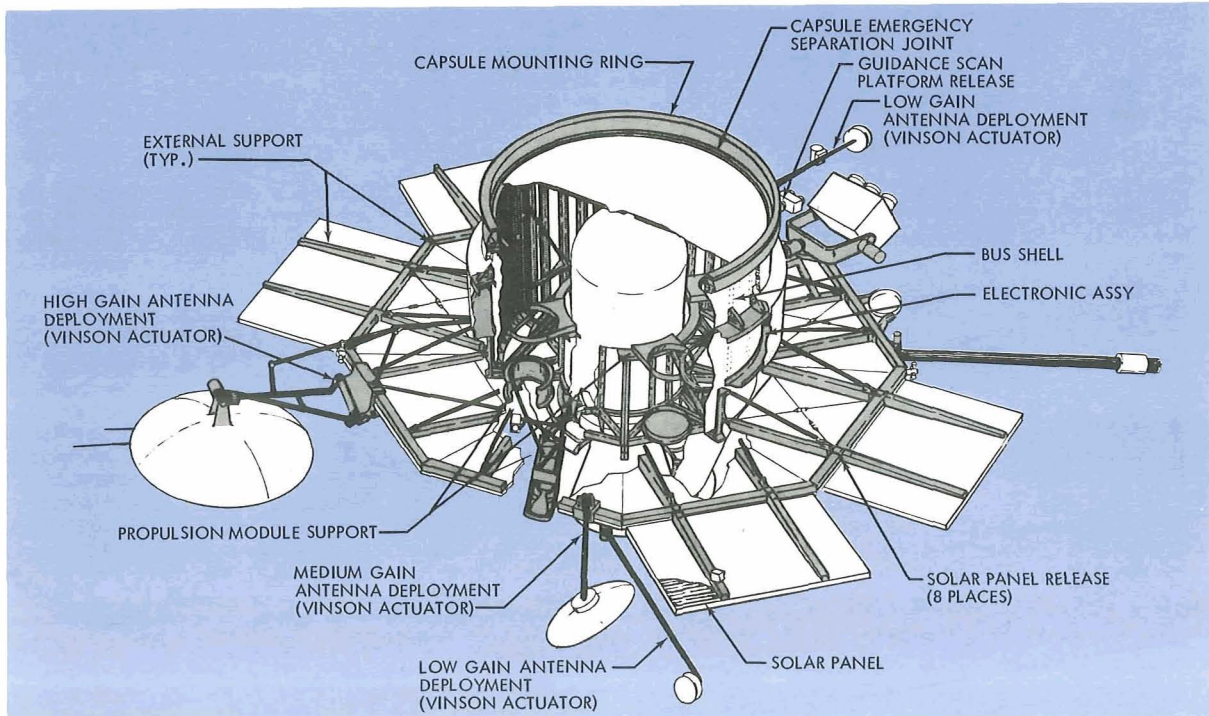


Figure 3-13: Structural and Mechanical Subsystem

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conclusion is that the axial dynamic load conditions used for the design of the spacecraft structure are conservative.

The cylindrical primary structure uses conventional stringer-and-frame-stiffened skin. External structure is truss construction. Magnesium, aluminum, and a limited amount of titanium alloys are used to minimize spacecraft weight and meet magnetic field and thermal requirements.

Mechanisms were chosen from existing, space-proven devices of the type used on Ranger and Mariner spacecraft. Oil-damped spring actuators were successfully employed on Mariner B and, although heavier than other competitive actuators, were selected because of their deployment reliability. Similarly, the V-block band separation scheme was selected because this concept has functioned reliably on Mariner C and other spacecraft.

PYROTECHNIC SUBSYSTEM

The pyrotechnic subsystem consists of the pyrotechnic power switching unit, pyrotechnic arming switch, separation-initiated timer, electroexplosive devices (EED's), and associated cabling. Figure 3-14 shows the location of these elements.

The pyrotechnic subsystem initiates (1) deployment of solar panels, booms, antennas, and platforms; (2) jettisoning of protective covers; (3) Flight Capsule separation, propellant subsystem valve actuation, and ignition of the orbit-insertion motor; and (4) release of the biological barrier.

Electroexplosive devices such as pin pullers, pyrotechnic valves, and rocket motor igniters in the pyrotechnic subsystem will meet the requirements of AFETRM 127-1. Pyrotechnic subsystem safety is further provided by the safe-and-arm device, which deprives the pyrotechnic subsystem of electrical power until spacecraft separation from the S-IVB. In addition, inadvertent ignition of the solid-propellant motor is prevented by the pyrotechnic arming switch which is activated just prior to orbit insertion.

The pyrotechnic arming switch and the separation-initiated timer are the same as on Mariner. Energy storage capacitors and solid-state switches follow the basic design concept of the Mariner spacecraft series.

Pyrotechnic subsystem reliability requires that functions occur when commanded and also that they do not occur prematurely. Firing upon command is achieved by separate paths from the

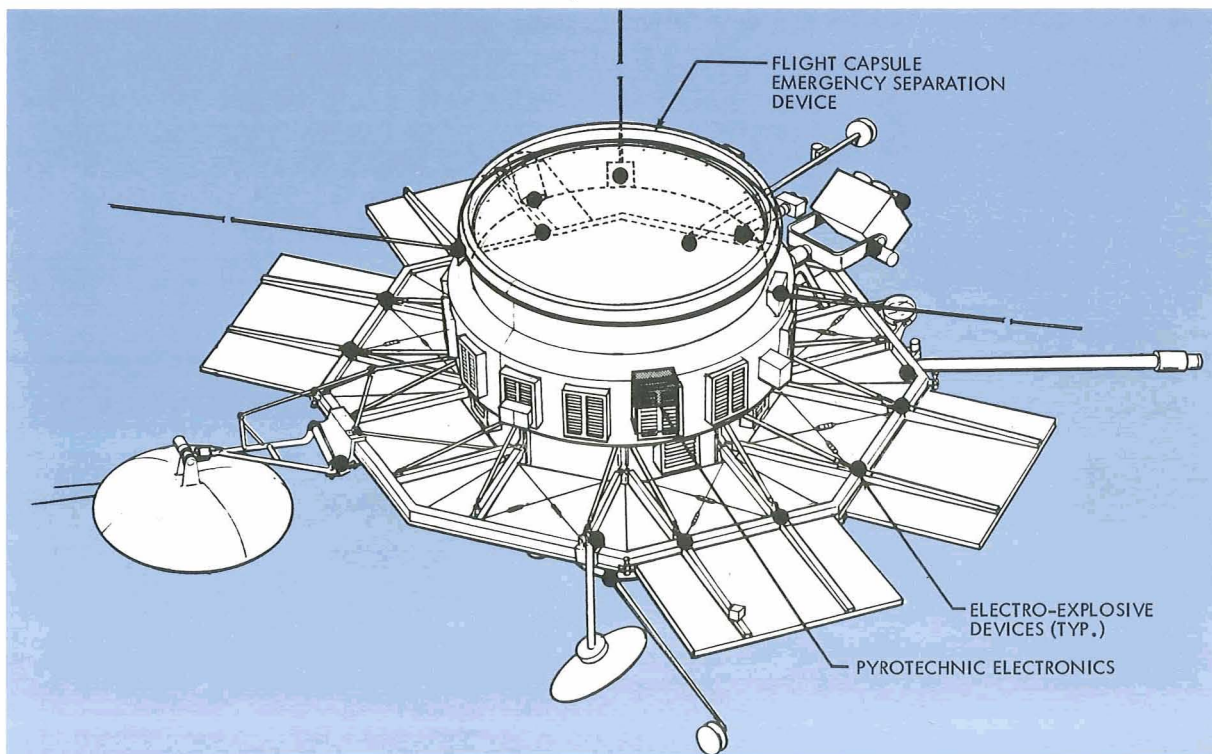


Figure 3-14: Pyrotechnic Subsystem

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input signal to each of two bridgewires and thus to each electroexplosive device. For functions that are not time-critical, both paths are activated by a single signal from the computing and sequencing subsystem or from the command subsystem. For time-critical functions, the redundant paths are activated by separate computing and sequencing subsystem signals that occur simultaneously or by one command subsystem signal that feeds both paths.

To prevent premature firing of time-critical functions, each driver has a two-section series output that will not fire when any single component failure occurs. Premature firing of functions that are not time-critical is prevented by using a single-channel circuit that cannot fail ON. In addition, power cannot be applied to the pyrotechnic subsystem until after Planetary Vehicle separation from the launch vehicle. In summary, no single failure will inhibit a command function or cause premature firing.

TEMPERATURE CONTROL SUBSYSTEM

The Voyager temperature control concept divides the spacecraft into three thermally isolated groupings, each with a common thermal approach. The groupings are the electronic assembly bay equipment, the propulsion module, and the external equipment. The identifiable

temperature control components used are bimetal actuated louvers, radiator plates, solar absorber plates, thermal ties, solar-engine heat shield, multilayer thermal shields, and surface finishes and coatings. Figure 3-15 shows the location of temperature control equipment.

The major design considerations of the thermal control subsystem are associated with the propulsion module and electronic assembly bay. Propulsion module internal equipment must be maintained between 75 and 110°F during all mission phases. The solar-engine heat shield design is dictated primarily by the requirement to maintain the internal temperature below 110°F during orbit-insertion engine burn. The design of the propulsion module louver assembly is established by the requirement to maintain the temperature below 110°F near Earth with a solar input of 130 watts per square foot. The requirement to maintain the temperature above 75°F at Mars with a solar input of 57 watts per square foot dictates the design of the propulsion module solar absorbers.

The electronic assembly bay equipment temperature must be maintained between 50 and 80°F during the most severe steady-state conditions and between 14 and 116°F during the worst thermal transient conditions. The worst steady-state conditions design the electronic assembly

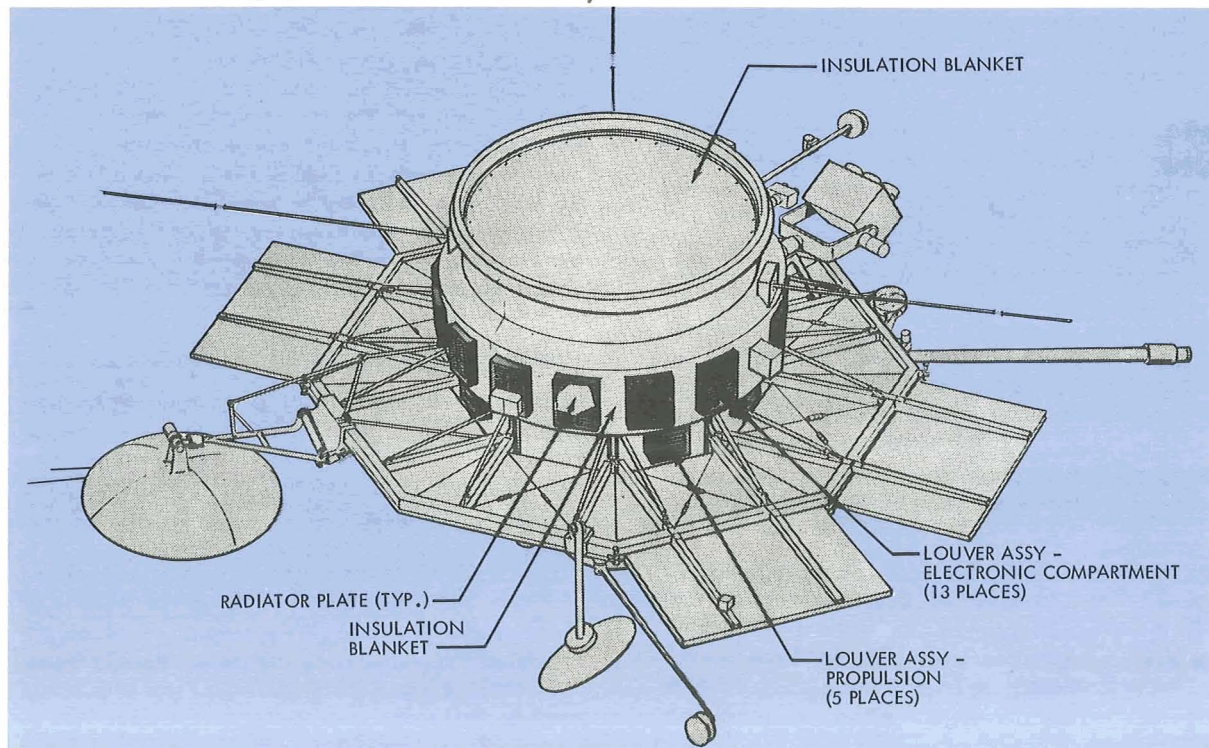


Figure 3-15: Temperature Control Subsystem

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bay louver-radiator assembly. These conditions are experienced near Earth where the solar input is 130 watts per square foot and the maximum internal heat generation for a single assembly is 80 watts. The worst transient conditions are experienced by the aft Planetary Vehicle. These conditions arise from the operating equipment internal heat and the combination of events beginning with the launch phase and ending with solar acquisition by the aft Planetary Vehicle. Heat loads during these periods include 65 watts per square foot from the nose fairing during launch, 40 watts per square foot from the nose fairing during Earth orbit, and 130 watts per square foot from the Sun prior to solar acquisition. These worst transient conditions and the 115°F maximum allowable equipment temperature determine the minimum allowable thermal capacitance of each electronic assembly module.

Temperature control of the external equipment, such as booms, solar panels, and gimbal actuators, is accomplished by thermal isolation and proper selection of heaters in the mechanisms as required, material diffusivity, and surface treatment (α/ϵ).

The Voyager temperature control subsystem uses the louver design technique from Mariner C. On the basis of recommendations from the Mariner postflight evaluations by JPL, louver assembly blade stops, which maintain maximum heat dissipating performance throughout the mission, have been incorporated on the preferred design.

During the cruise and orbiting phases, the effects of solar influences on the electronic assemblies have been minimized by using the propulsion module as a thermal buffer. This is accomplished by a combination of the long thermal path between the propulsion module and the electronic assemblies, and proper selection of insulation, mountings, and thermal coatings. The thermal isolation of the propulsion module allows for major revisions to the propulsion module without affecting the temperature control subsystem for the electronic assembly bay and permits the majority of the spacecraft thermal development tests to be performed without the propulsion subsystem.

The preferred design includes a margin for louver failures. Ten percent of the louver blades can fail in one extreme position without perturbing the equipment temperature outside of the operating range. Twenty percent can fail, half in each extreme position, without affecting predicted performance of operating equipment.

Thermal coupling performance in conjunction with judicious location of internal equipment assemblies in the preferred design for 1971 allow thermal control of the following extreme situations:

- Assemblies with power densities 33 percent higher than the design maximum of 23 watts per square foot;
- Assemblies with thermal capacitance ($M \times C_p$) 20 percent below the design minimum of 3.8 watt-hours per °F.

These provisions will allow late revisions or additions to subsystems with a minimum change to the preferred design.

The following growth provisions, inherent in the temperature control preferred design, are available for subsequent Mars missions:

- Increase in capsule diameter from 12 to 20 feet without redesign;
- Increase in internal heat load of 20 percent with the 20-foot capsule without redesign;
- Increase in internal heat load of 100 percent with the 20-foot capsule with minor changes such as the addition of a highly reflective coating to the bottom of the Flight Capsule.

PROPULSION SUBSYSTEM

The preferred spacecraft propulsion subsystem contains a midcourse and orbit-trim propulsion unit and an orbit-insertion propulsion unit. These two units are combined into a single propulsion module as shown in Figure 3-16. This configuration exceeds all requirements for the 1971 and 1973 missions and requires no modification in hardware or propellant loading to accomplish the assumed 1975 and 1977 missions.

Midcourse and Orbit-Trim Propulsion

This unit consists of four restartable hydrazine engines with jet-vane thrust vectoring, valving and feed system, four hydrazine propellant tanks with positive-expulsion bladders, and a nitrogen pressurization unit.

The unit provides the impulse for trajectory corrections while the spacecraft is in transit to Mars, supplements the ΔV provision of the orbit-insertion unit, and corrects or trims the Mars orbit when it has been achieved. Each of the four engines produces 200 pounds of thrust and is regulated-pressure-fed and radiation cooled.

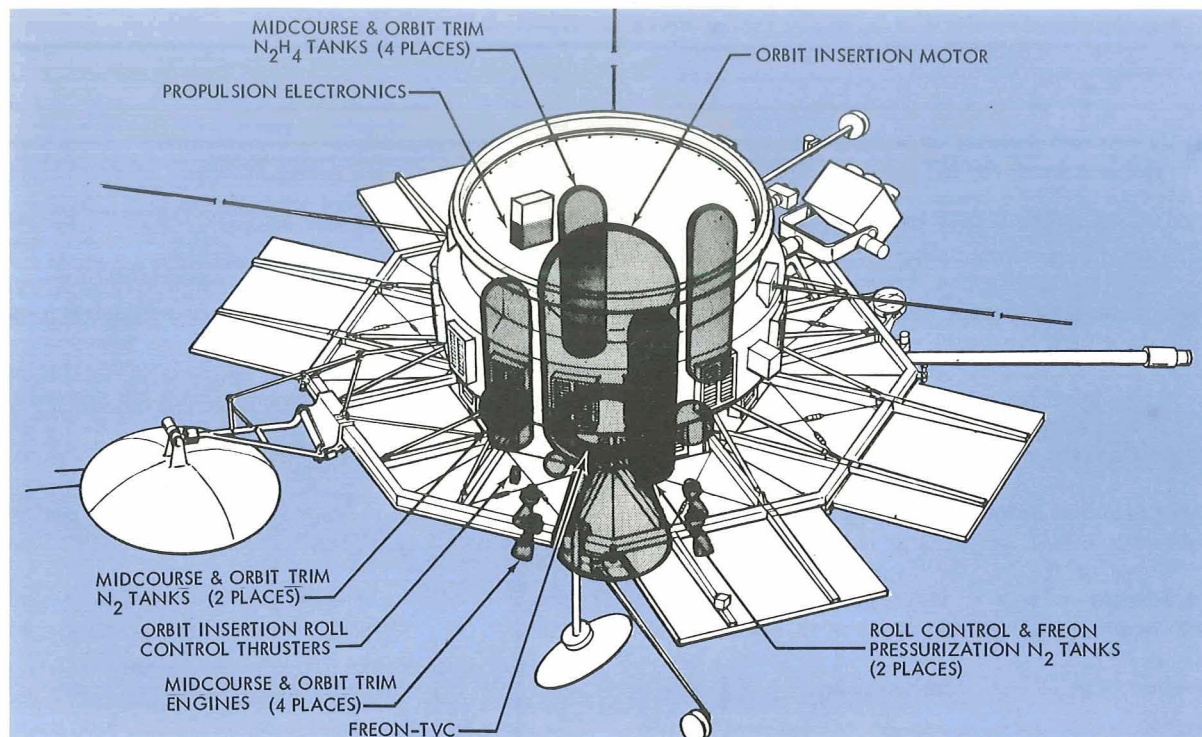


Figure 3-16: Propulsion Subsystem

Multiple restart capability is provided by a spontaneous-decomposition catalyst. The engines are mounted well aft and are canted at 13 degrees to the roll axis. Thrust vector control is accomplished by jet vanes. Maximum effective thrust vector angular deflection is ± 5 degrees. Elastomeric bladders ensure positive expulsion of hydrazine. Regulated nitrogen gas pressurizes the hydrazine tanks.

The midcourse and orbit-trim propulsion unit was designed for the 1977 mission and can provide impulse for 200 meters per second of midcourse correction and 100 meters per second of orbit trim with a 3500-pound Flight Spacecraft and 10,000-pound Flight Capsule.

Midcourse and orbit-trim propulsion unit reliability is achieved by designing for an engine-out condition and by incorporating series-parallel redundancy of valves and regulators to provide functional redundancy without a malfunction detection system. The unit also has the same functional characteristics as the Mariner monopropellant system.

Orbit-Insertion Propulsion

This unit consists of a modified Minuteman Wing VI second-stage motor, liquid-freon secondary injection unit, nitrogen pressurization unit, roll control nozzles, and an adapter ring for attachment to the midcourse and orbit-trim unit.

The propulsion module provides sufficient impulse to achieve satisfactory Mars orbit from 1971 through 1977. A forward-mounted sealed pyrogen igniter initiates solid-motor burn. Pitch and yaw thrust vector control during motor burning is achieved by secondary injection of freon 114B2. An effective thrust vector angular deflection in excess of the ± 2 -degree requirement is provided. Roll thrust vector control is provided by 6-pound-thrust nitrogen thrusters with quad-redundant valves.

The orbit-insertion unit was sized in conjunction with the midcourse and orbit-trim unit as shown in Table 3-5. The orbit-insertion unit was sized

Table 3-5: Propulsion Subsystem ΔV Capability

MISSION		ΔV CAPABILITY (METERS/SEC)			
Year	Capsule Weight (lbs)	Midcourse (N_2H_4)	Orbit Insertion (Solid)	Orbit Insertion (N_2H_4)	Orbit Trim (N_2H_4)
71-73	2000	200	2050	337	100
	3000	200	1910	267	100
75-77	8000	200	1350	40	100
	10000	200	1230	0	100

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by determining the weight margin remaining after subtracting the weight of a midcourse and orbit-trim propulsion unit that meets the 1977 design conditions and the weight of the propulsion module insulation and support structure from the total propulsion subsystem allocation of 15,000 pounds. Sizing the orbit-insertion unit in this manner results in exceeding the 2200-meter-per-second 1971 orbit-insertion velocity increment design goal with a 2000-pound Flight Capsule and the 2000-meter-per-second minimum requirement orbit-insertion velocity increment for a 3000-pound Flight Capsule. The unit also provides for an orbit-insertion velocity increment in excess of 1200 meters per second for the 10,000-pound Flight Capsule of the 1977 mission.

The preferred orbit-insertion unit contributes to achieving a high probability of mission success by virtue of the high demonstrated reliability of the Minuteman motor. Only changes on the motor that improve probability of mission success or are required to meet mission requirements have been adopted. These changes are:

- Decrease the motor length to achieve a lower acceleration during motor burn;
- Increase the nozzle length to reduce plume radiant heating to the solar panels and improve specific impulse;
- Use nitrogen as the freon pressurant because it is easier to regulate and provides a more favorable thermal environment than the Minuteman hot gas generator.
- Change the freon tank material to titanium because it has more favorable magnetic properties.

MISSION VERSATILITY

The preferred spacecraft design has a significant degree of inherent and built-in mission flexibility and versatility. The 1971 design provides that:

- A wide range of transit trajectories and Mars orbits can be achieved for the 1971 mission;
- The spacecraft science payload can be selected at a relatively late date with minimum effect on the spacecraft design;
- Orbital maneuvers can be conducted, beyond the basic orbit-trim capability, to modify the nominal Mars orbit;
- Subsequent Mars missions can be performed with only minor modification to the 1971 spacecraft.

A typical nominal transit trajectory is shown in Figure 3-17. Also shown is the basic 1971 trajectory design chart reflecting potential trajectories consistent with the spacecraft capability and JPL constraints. The maximum practical launch period is about 93 days. Within this range, numerous trajectory sets can be selected to satisfy the 45-day minimum launch period. For example, 45-day launch periods can be specified that will provide a fixed arrival date (fixed communication distance), provide a constant arrival velocity between 3.0 and 4.5 kilometers per second, provide a constant trip time, or coincide with specific calendar dates to preclude interference with other NASA programs and DSN scheduling.

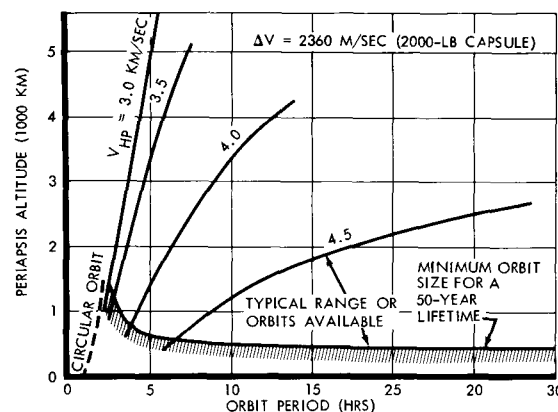
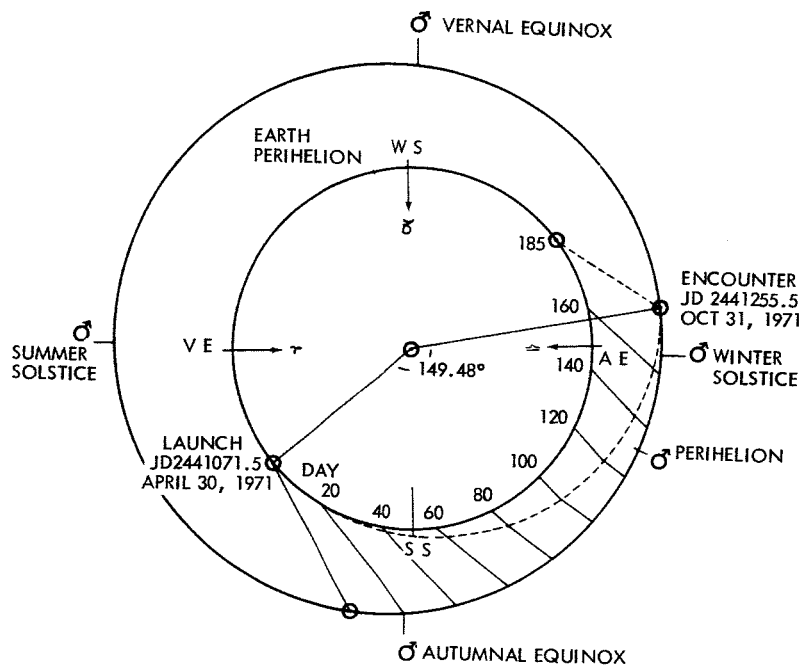


Figure 3-18: 1971 Mars Orbit Capability

Figure 3-18 shows the variation in Mars orbit size available with the preferred spacecraft design propulsion capability. At the highest planetary approach velocity (VHP) of 4.5 kilometers per second, an orbit period as low as 9 hours with a periapsis altitude of 1000 kilometers can be achieved. Longer period orbits are also available. As time progresses from the opening of the launch period, lower arrival velocities can be achieved, thus increasing the range of Mars orbits available. In mid-December, at an approach velocity of 3.0 kilometers per second, orbits with periods as low as 2.8 hours can safely be selected. This velocity also allows position of the Mars orbit periapsis in the orbit plane to be located within ± 100 degrees of the hyperbolic approach periapsis. The available orbit inclinations vary from nearly equatorial to polar. Because prolonged interference with the Canopus seeker is not desirable, polar orbits require that periapsis be low and near the north pole.

The preferred spacecraft was designed to accomplish subsequent Mars missions to the greatest extent practical within JPL-imposed require-



Typical Transit Trajectory

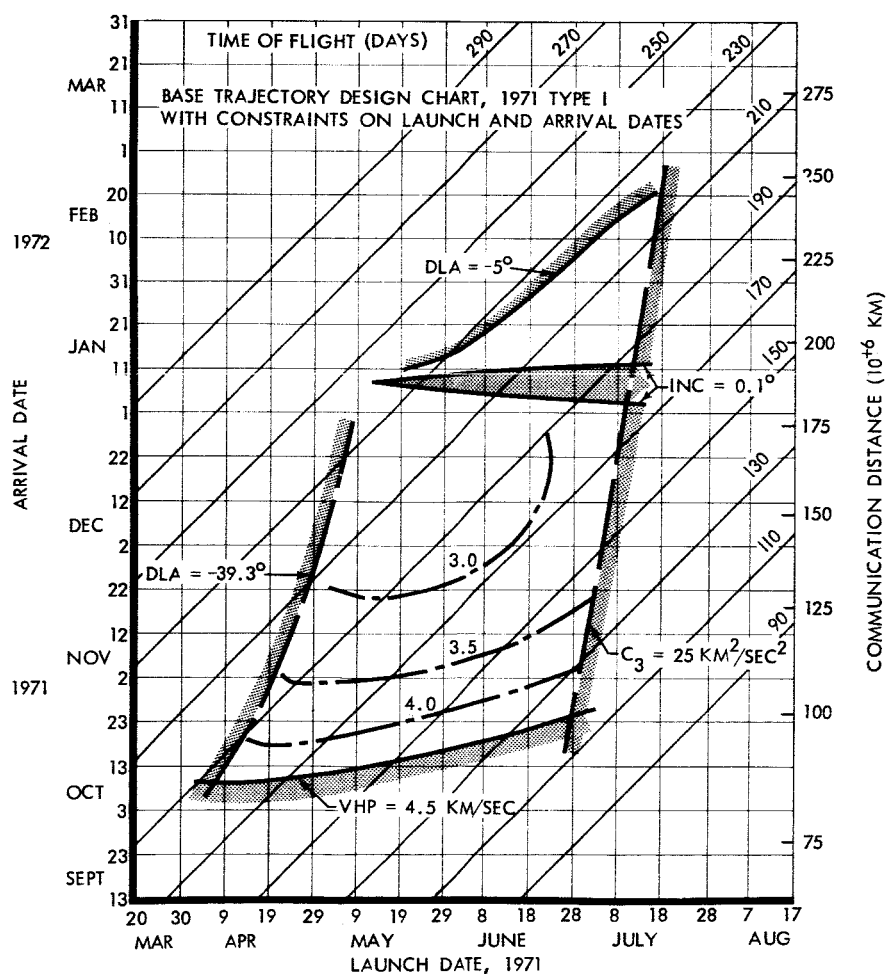


Figure 3-17: Typical Transit Trajectory

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ments and constraints. The propulsion subsystem accommodates missions subsequent to 1971. This results in a midcourse and orbit-trim maneuver capability in excess of the 1971 specified requirements; the amount depends on arrival velocity. This excess capability can be used to modify the spacecraft orbit and enhance scientific observations or possibly prevent orbit decay, if atmospheric density exceeds current estimates.

PLANETARY QUARANTINE

The Boeing Phase IA Task A study concluded that compliance with the planetary quarantine constraint would require sterilization of the propulsion and attitude control subsystems to preclude contamination by exhaust products. It also concluded that the microbiological burden on the spacecraft surfaces might require reduction by either heat treatment or ethylene oxide because of the probability of contamination of Mars by ejecta resulting from meteoroid impacts on the spacecraft. During Phase IA Task B, studies were performed to re-evaluate planetary quarantine problems; these considered the concept of a single Saturn V launch vehicle with two Planetary Vehicles and revised Mars environmental prediction data. Changes during Task B are summarized below.

Following are the principal considerations that resulted in these changes.

- **Sporicidal Effects of Propellants** — Tests were performed to determine the sporicidal effects of propellants. Results indicate that hydrazine monopropellant is sporicidal, and that aluminized polybutadiene solid propellant is not.

- **Ejecta from Exhaust Products** — Fourteen firing tests were performed on solid-propellant engines to determine if viable organisms survive in the exhaust. Natural propellants inoculated with bacterial spores were tested. During two of the six tests with inoculated propellants, microorganisms were recovered. However, the techniques used in handling and culturing samples from these particular tests cast doubt on the validity of the results. Because there were no microorganisms recovered on any of the other tests using inoculated or natural propellants, it has been concluded that the probability of survival is low.
- **Live Ejecta from Meteoroid Impact** — A series of tests was conducted to determine whether viable organisms will survive meteoroid impact. As a result, it was established that live organisms will be spalled off by meteoroid impacts.
- **Reduction in Meteoroid Flux** — According to the JPL "Preliminary Voyager Environmental Predictions" document, a four-order-of-magnitude reduction in meteoroid flux in the vicinity of Mars from the environment specified in Task A can be expected. This reduction results in fewer spacecraft ejecta from meteoroid impacts than previously calculated.
- **Emissivity of Spores** — Additional tests have shown that the emissivity value of spores approximates 0.2 compared to the previous value estimated as 0.5 to 1.0. This dictated a re-evaluation of the thermodynamic heat pulse that can be expected on entry into the Martian atmosphere and reduced the probability of entry survival of particles from the estimates in Task A.

Table 3-6: Task A vs Task B Planetary Quarantine Comparison

TASK A	TASK B	REASON
Sterilize midcourse correction and orbit-trim engine.	No sterilization required	Tests show that the hydrazine monopropellant is self-sterilizing.
Sterilize orbit-injection engine	Sterilize only TVC system	Tests indicate low probability of organisms surviving in the exhaust.
Study decontamination further	No decontamination	Analysis indicates that the effect of ultraviolet, reduced meteoroid flux, thermodynamic heat pulse, and lowered probability of spore survival on Mars, will probably reduce contamination to an acceptable level.
Sterilize reaction control system	No change	

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- **Solar Radiation** — Recent data in the literature has indicated that ultraviolet radiation with wavelengths between 2000 and 3000 angstroms reaches the Martian surface with little atmospheric attenuation. Ultraviolet energy at this wavelength is lethal to most microorganisms. Because little work has been done to determine the effect of ultraviolet and photoreactivation by white light in a space environment, experiments were conducted during Task B in which viable organisms were subjected to ultraviolet irradiation coincidentally with white light. The tests resulted in an estimate of ultraviolet lethality of 97 to 99 percent both in space and on the Martian surface.

- **Initial Microbial Loading of Spacecraft** — An investigation was conducted to determine the initial microbial load of a spacecraft. Based on recently reported studies of clean-room microbial loadings, it was estimated that the original load estimate of 10^9 microorganisms per spacecraft can be reduced to 10^7 .

A new functional analysis was performed to identify all mission events that could result in contamination of Mars. Event equations based on this functional analysis were developed and translated into unconditional probability equations to arrive at the allocations shown in Figure 3-19. This is based on the total contamination constraint of 10^{-4} . As a result of analysis to date, it appears that the 10^{-4} planetary quarantine constraint can be met by proper trajectory control, sterilization of the internal surfaces of the orbit-insertion thrust vector control system, sterilization of the internal surfaces of the reaction control system, and aseptic loading of nitrogen and freon. Additional analysis and tests must be conducted during Phase IB to substantiate these findings. These tests will simulate as closely as possible the real situation that will be encountered with respect to materials, subsystems, and environmental conditions.

tered with respect to materials, subsystems, and environmental conditions.

OPERATIONAL SUPPORT EQUIPMENT (OSE)

Flight Spacecraft OSE includes subsystem OSE; the System Test Complex (STC); Launch Complex Equipment (LCE); Mission-Dependent Equipment (MDE); Assembly, Handling, and Shipping Equipment (AHSE); and associated software. The function of this equipment is to support subsystem or system operations during assembly, test, launch, and the mission. To fulfill this function and to provide a smooth transition from subsystem bench testing through evaluation of the spacecraft as it performs its mission, OSE is designed around a concept that provides:

- A building-block approach wherein identical test hardware is used for various test levels;
- A unified software design that permits a standard set of programs and procedures with selected elements at all levels of test.

Subsystem OSE that is used for spacecraft subsystem type-approval tests (TAT) and flight-approval tests (FAT) is designed by the same individuals who designed the Flight Spacecraft subsystems. In addition, appropriate parts of these same test sets are used as integral portions of the STC and LCE where they are under the control of the test or launch director through a central data and control system. In this manner, subsystem through system tests are performed using identical test equipment designs from TAT at the manufacturers facility through flight readiness testing at the launch pad.

When the spacecraft has been assembled, a team of engineers and technicians follows it from initial testing through launch. This approach will significantly enhance the effectiveness of the Voyager system. The STC is designed for air and road transportation so it can accompany the spacecraft through its various test locations to liftoff at Launch Complex 39. The STC equipment consists of the subsystem OSE mentioned above, adapters for each set, a central data and control system (CDCS), a voice communication system, and flight hardware simulators. By using a Scientific Data Systems 920 (or 930) computer, the CDCS provides the capability to manually or automatically control all spacecraft testing, to process and store all test data, to provide automatic computer control of spacecraft sequencing, and to automatically display test data and status.

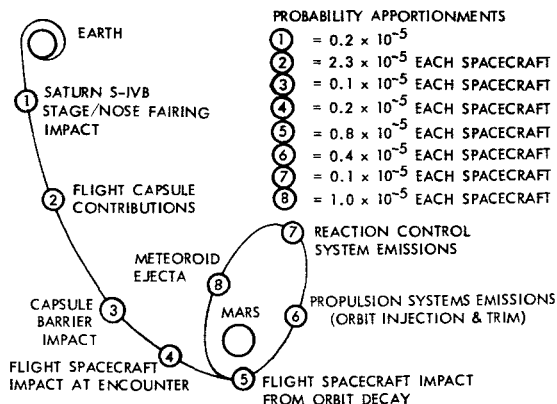


Figure 3-19: Possible Contamination Events

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Figure 3-20 shows a scale model of the STC in a simulated test setup. The model provides visibility of packaging and setup details to allow effective planning for test system transportation and facility requirements.

The LCE makes extensive use of the STC to determine the operational readiness of the Planetary Vehicle, to monitor status during final countdown and to condition the vehicle for launch. The LCE thus consists of the CDCS and the subsystem OSE adapters from the STC and adds to them signal conditioning devices, remote displays, and recording and control equipment necessitated by the geographical separation of LCE elements and personnel. Communication with the Planetary Vehicle is accomplished through the rf link and umbilicals. Figure 3-21 is a concept of the LCE. Each LCE can check out and monitor two Plane-

tary Vehicles by integrating two STC's with the other equipment needed.

Unified software design begins with bench-level tests of the individual subsystems. Software design is parallel to hardware design to permit a standard set of programs and procedures for generating tapes for the subsystem test set tape readers and for the data processing and trend analysis. The same subsystem software then provides the basis for preparing STC and LCE software for system-level testing. Common hardware makes common software possible and meaningful. For example, by using the same subsystem test sets at all levels of testing and identical CDCS with the STC and LCE, the system test engineer can accurately correlate his test data with data from the subsystem test at the point of manufacture.

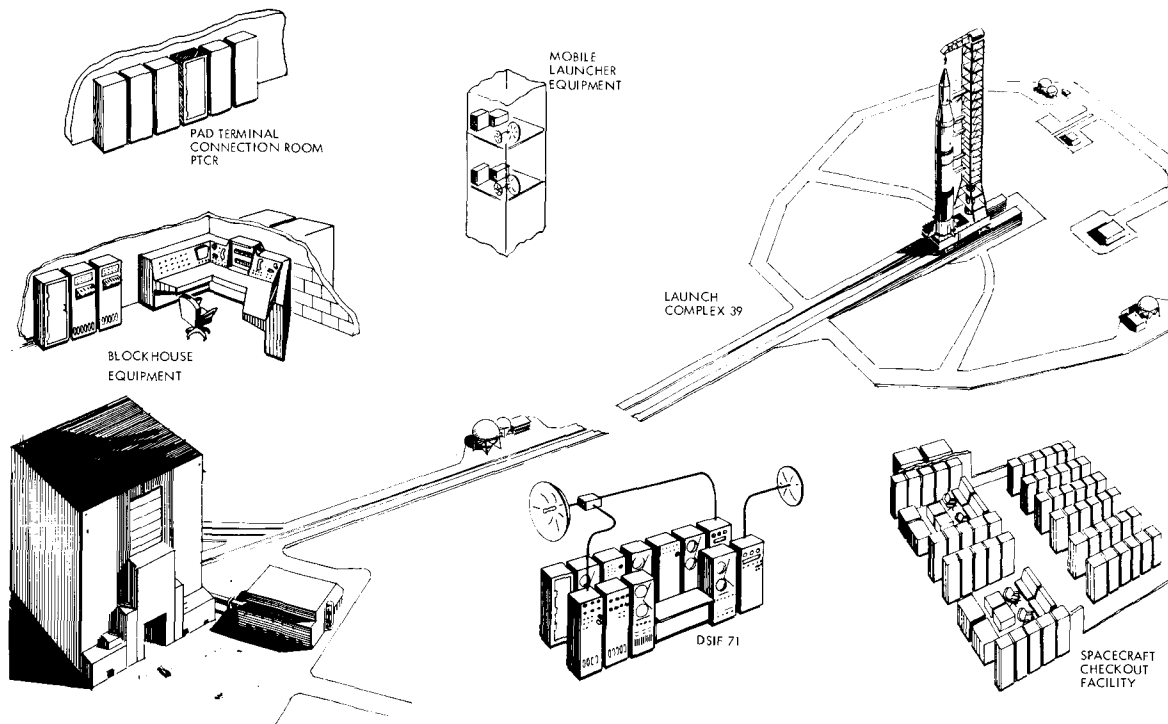


Figure 3-21: Principal Elements of the LCE

VIEW LOOKING DOWN

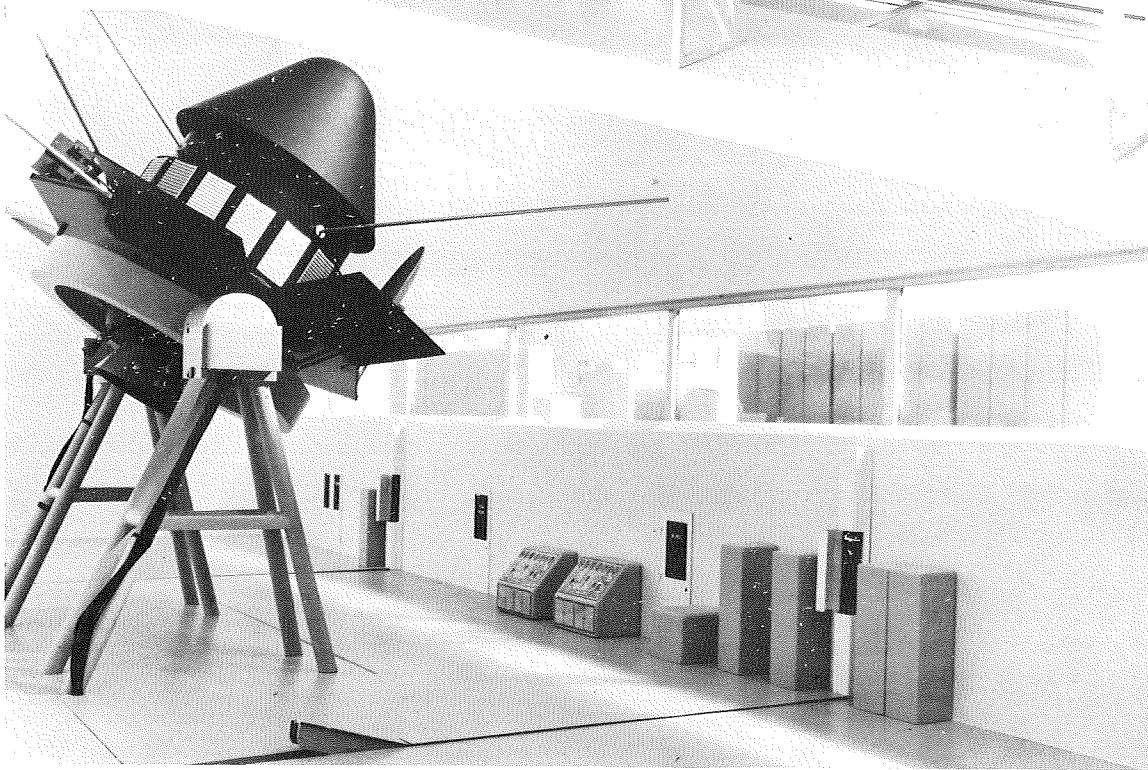
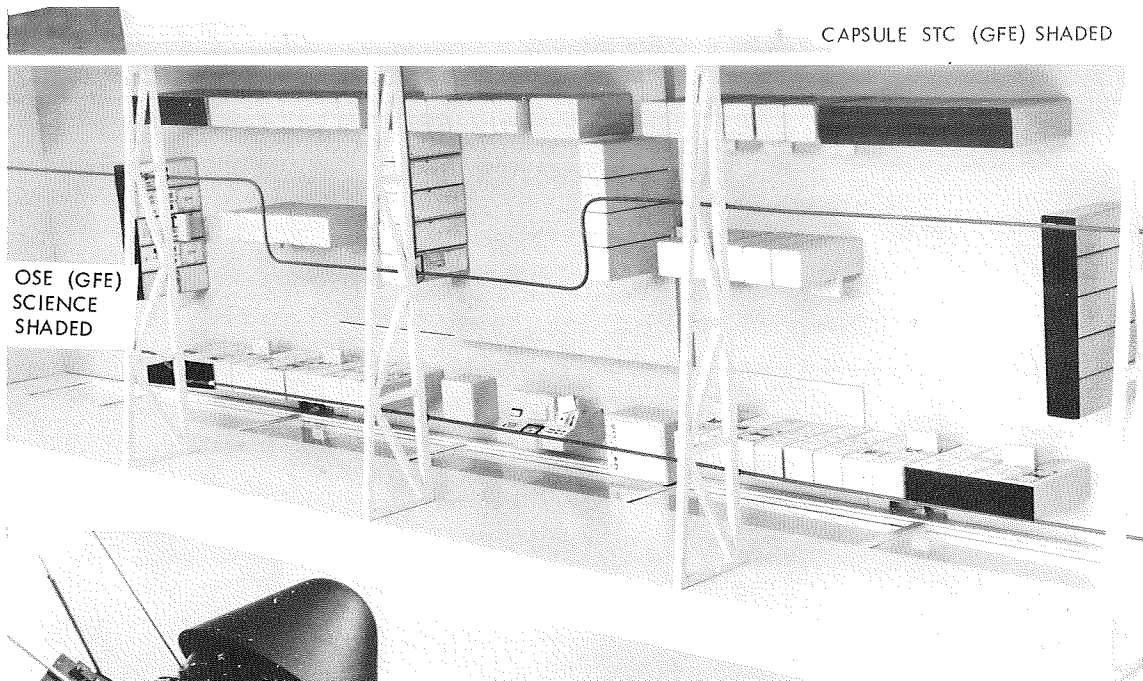
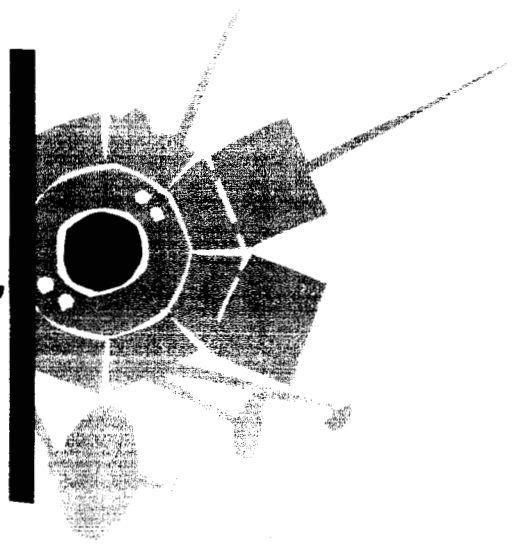
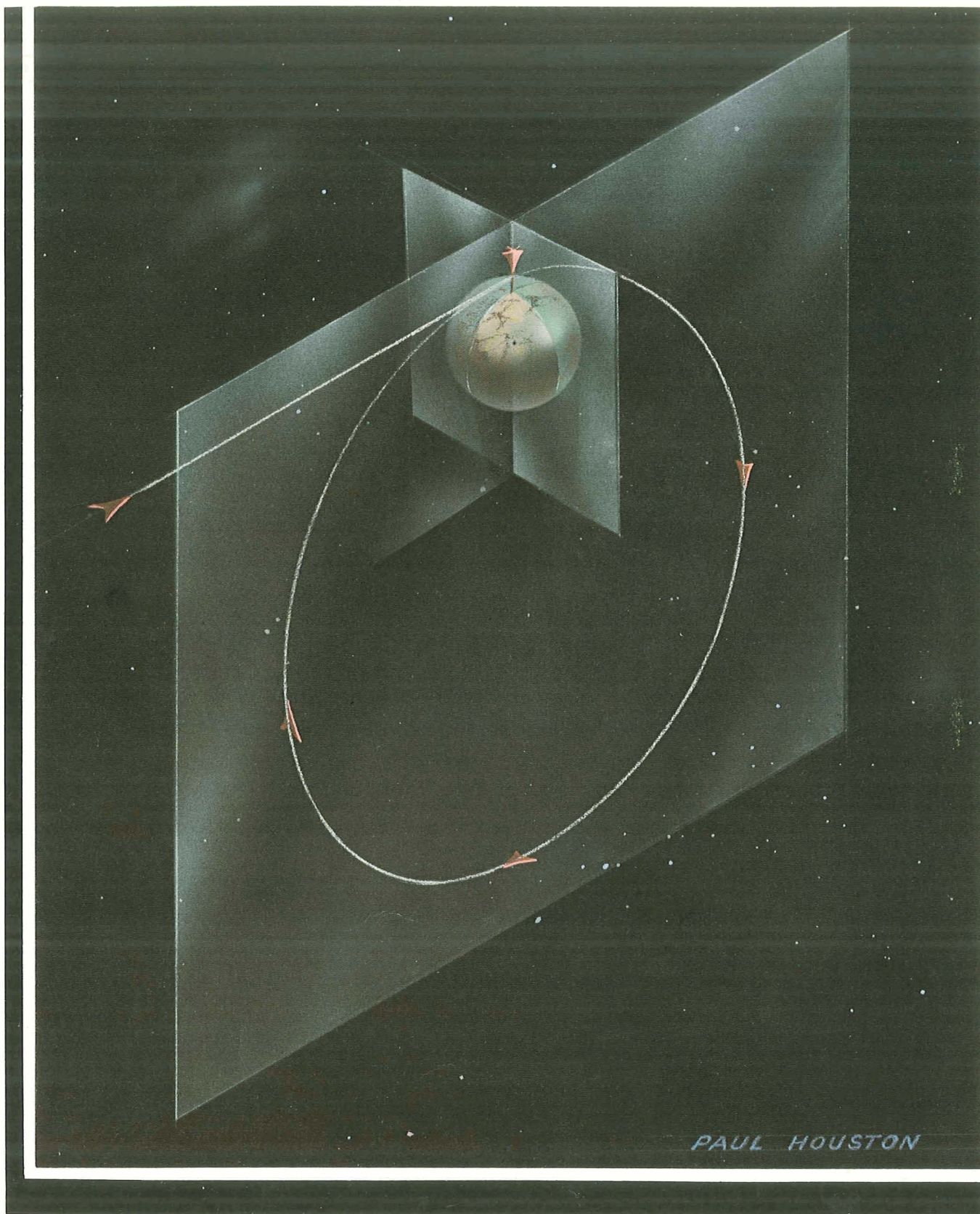


Figure 3-20: System Test Complex And Equipment

VOYAGER





PAUL HOUSTON

EXPANDED VOYAGER MISSIONS

The "Voyager 1971 Preliminary Mission Description" defines the Voyager Program objective as that of scientific exploration of the solar system. It is further stated, relative to the Mars mission, that an orderly program of continually improving knowledge in science and technology is required to achieve the mission objectives. This same philosophy, when applied to the overall Voyager Program plan, emphasizes the need to understand the various elements of this total program. With even a cursory program outline, it is possible to assess the improvement in knowledge necessary to evolve the entire program.

The objective here then is to determine the basic technological requirements for subsequent missions, both to Mars and to other planets, and to compare these with the 1971 mission system capability. This comparison provides information concerning: (1) the degree to which the 1971 system might be adapted to other missions, and (2) areas that require technological development.

The use of the three-stage Saturn V launch vehicle provides, for the first time, the capability for including targets other than Venus and Mars for detailed observations. Whereas the Mariner program was handicapped due to available booster capability, the powerful Saturn V can cover the range of missions extending from flybys of the more distant planets to close-in Sun probes. The regions now open for mission considerations are illustrated in Figure 4-1, which shows potential space-vehicle weight versus booster-injection-energy requirements.

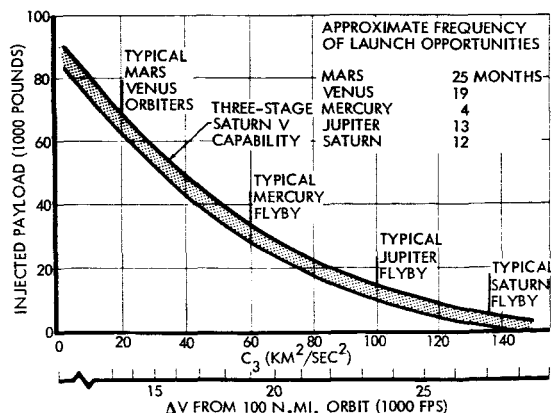


Figure 4-1: Saturn V Payload Capability for Planetary Missions

A number of advanced missions have been postulated, beginning with the 1973 mission to Mars. Each mission has been analyzed to determine general science objectives and to establish typical flight profiles. On this basis, system and subsystem performance requirements are estimated and compared with the 1971 system capability.

Boeing recognizes that, although the following analysis is not a part of the Phase IA contract, it is important because it illustrates the flexibility of the 1971 spacecraft within the requirements and constraints defined by JPL. As the 1971 Mars program evolves into Phase IB it may be desirable to improve the system adaptability; if so, these types of data will facilitate timely consideration of the many factors involved. This effort has been, of necessity, limited. Boeing recommends that more detailed analyses be conducted, either before or early in Phase IB, so that information on contributions to subsequent missions is available before system design commitments.

VOYAGER PROGRAM SCIENTIFIC OBJECTIVES

In general, all science objectives and, consequently, specific experiments are included under one of five categories:

- Interplanetary particles and fields,
- Solar physics,
- Planetary atmospheres,
- Planetology,
- Bioscience.

The exploration of neighboring planets is considered to be the immediate objective because NASA has stipulated that primary emphasis will be placed on acquiring scientific information concerning the origin, evolution, and nature of life, and the application of this information to an understanding of terrestrial life.

In this light then, the principal scientific objectives will be oriented to provide data of the

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Table 4-1: Representative Mars Mission Sequences

	YEAR	MISSION MODE	SYSTEM WEIGHT			TRAJECTORY TRANSFER TYPE	LAUNCH AZIMUTH (Degrees East of North)	MAX C_3 (km^2/sec^2)	LAUNCH PERIOD (Days)	FLIGHT TIME (Days)	COMM. DIST. AT ENCOUNTER (10^6 km)	MAX VHP (km/sec)	ORBIT INSERTION ΔV (km/sec)	ORBIT PARAMETERS	
			FLIGHT SPACECRAFT BUS (Pounds)	FLIGHT CAPSULE (Pounds)	PROPULSION MODULE (Pounds)									PERIAPSIS ALTITUDE (km)	ORBIT PERIOD (Hours)
NOMINAL PROGRAM	1973	Orbiter/Lander	2500	3000	15000	I	60-115	19	45	190-198	153-205	3.4	≈ 2.2	1000-2000	4-20
	1975	Orbiter/Lander	3500	10000	15000	II	60-115	25	45	300-328	308-352	3.0	≈ 1.2	600	6-20
	1977	Orbiter/Lander	3500	10000	15000	II	60-115	25	45	295-328	275-310	3.0	≈ 1.2	600	6-20
INDIVIDUAL ALTERNATIVES	1975	Orbiter/Lander	3500	10000	15000	I	35-120	25	20	210-245	255-275	3.0	≈ 1.2	600	6-20
	1977	Orbiter/Lander	3500	10000	15000	I	35-120	25	20	232-255	250-270	3.0	≈ 1.2	600	6-20

1 Capability of Voyager Propulsion Module

2 C_3 limit for mission planning purposes (JPL mission description, dated October 15, 1965)

3 For a 8000 lb. Flight Capsule, orbit insertion ΔV capability is increased to 1.4 km/sec which provides greater flexibility in periapsis altitude selection.

4 Solid engine plus augmentation from the liquid engines

following types:

- Planet Vicinity — Radiation, magnetic fields, diameter and geometric oblateness, mass, gravity, and meteoric matter;
- Planet Atmosphere — Composition, density, pressure, temperature, aerosols, meteorology, and ionosphere;
- Planet Lithosphere — Soil, depth, composition, topography, surface-radiation balance, and seismic characteristics;
- Planet Biosphere — Life forms;
- Planet Hydrosphere — If any.

In addition, interplanetary characteristics will be investigated, including magnetic fields, solar plasma, meteoric matter, interplanetary gas, cosmic rays, and solar coronas.

REPRESENTATIVE MISSION PARAMETERS

Pertinent mission parameters associated with postulated flights to Mars in 1973, 1975, and 1977, as well as trips to Venus, Jupiter, Mercury, and Saturn, are summarized in Tables 4-1 and 4-2. The next section presents descriptions of and comments on the selected sequence for each of these missions.

Boeing has been using these data to define mission-dependent requirements imposed on the spacecraft system until such time as JPL provides more complete Voyager mission planning.

ADVANCED MISSION DESIGN IMPLICATIONS

As stated previously, the accomplishment of the overall Voyager Program goal presents a variety of space missions and consequently different spacecraft design requirements. A universal spacecraft that could handle all missions is very attractive; however, this concept competes with the policy of design conservatism to enhance mission success. Although Boeing has emphasized design conservatism, it recognizes that a good spacecraft design allows for future growth.

Growth capability of the basic spacecraft design can be measured by postulating advanced missions and then measuring the design requirements against a baseline design. A significant product of such an evaluation is identification of subsystems that will require completely new design approaches (e.g., radioisotope thermoelectric generator units instead of photovoltaic cells for electrical power generation).

Table 4-2: Representative Mission Sequences to Other Planets

PLANET	MISSION MODE	PLANETARY VEHICLE WEIGHT (POUND)	TRAJECTORY TRANSFER TYPE	LAUNCH AZIMUTH (Degrees East of North)	MAX C_3 (km^2/sec^2)	LAUNCH PERIOD (Days)	FLIGHT TIME (Days)	COMMUNICATION DISTANCE AT ENCOUNTER (10^6 km)	MAX VHP (km/sec)	ORBIT INSERTION ΔV (km/sec)	HYPERBOLIC APPROACH DISTANCE (km)	ORBIT PARAMETERS	
												PERIAPSIS ALTITUDE (km)	ORBIT PERIOD (Hours)
Venus	Orbiter/Lander	2 @ 19,500	I	60-115	20	45	146-191	108	4.1	≈ 1.9	8200	8200	8
Jupiter	Flyby/Probes	2 @ 11,450 2 @ 6,850	I	60-115	93	20	650	710	9	Flyby Only	230,000		
Mercury	Flyby/Probes	2 @ 19,000	II	60-115	50	20	110-125	170	15	Flyby Only	—		
Saturn	Flyby/Probes	1 @ 11,200 1 @ 3,650	I	60-115	135	20	1200	1350	9	Flyby Only	Planet Radii		

1 Using the Voyager propulsion module as a kick stage during earth escape.

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The following sections discuss requirements for spacecraft subsystems for each of the postulated advanced missions. Each system was compared to the preferred 1971 Voyager design described in D2-82709-6. Table 4-3 summarizes the results of these analyses.

In general, the 1971 spacecraft preferred design is satisfactory for all postulated subsequent Mars missions through 1977 with structural and guidance and control subsystem modifications required to incorporate heavier Flight Capsule weights. The preferred design is adequate for the postulated Venus mission except that temperature control considerations require modest changes. The more advanced missions (Jupiter, Mercury, and Saturn) necessitate design modifications and the development of new design approaches, particularly in the area of electrical power generation.

1973 MARS MISSION

The 1973 flight will be the first attempt to place an operable science package on the surface of Mars. The 1971 mission is oriented toward obtaining data necessary for a lander design and for proving orbital operations, whereas the 1973 flight is primarily concerned with demonstrating the capability of a module to land and survive on Mars.

The Flight Capsule has an operating-life goal of 2 days on the Martian surface. During this time it will transmit, to Earth, data on surface environmental conditions, soil characteristics, and topography. The Flight Spacecraft will gather environmental and surface data during a 6-month Mars orbit.

Mission Description

The postulated 1973 Mars mission is similar to the 1971 flight in that two Planetary Vehicles are launched by a single booster. Each Planetary Vehicle consists of a Flight Spacecraft and a Flight Capsule. A 45-day launch period using Type I trajectories is assumed, resulting in trip times under 200 days (Table 4-1).

Although slightly heavier than the 1971 configuration, the 1973 Planetary Vehicle can achieve a Mars orbit similar in size and shape to that in 1971. The orbit has an inclination of 35 to 60 degrees, oriented so that the subperiapsis point is located between 20 and 50 degrees north latitude. This orientation is selected in order to observe the wave of darkening, which is in the northern hemisphere during the 1973 oppor-

tunity. The Flight Capsule landing site will also be in the northern hemisphere and, if possible, will correspond to the anticipated 1975 landing site. This latter consideration will be of particular benefit to the development and operational planning associated with the more complex 1975 Flight Capsule lander.

Science Payload

Representative experiments selected for the Flight Spacecraft Bus payload are intended to: (1) extend observations of unique environmental phenomena discovered during the 1971 flight, and (2) survey and select probable sites for subsequent science observations on the Martian surface. In general, the scientific instruments will be the same as for the 1971 mission.

Design Implications

As shown in Table 4-3, the 1971 Voyager design appears adequate to accomplish the 1973 mission objectives and mission sequence defined above. The geometry and weight of the Flight Capsule is the principal factor that would affect the design. Since the structural design of the preferred Spacecraft design is based on carrying a 2000-pound capsule, structural design modifications will be necessary if capsule weight is increased. The increase in structural weight could be as high as 150 pounds for a 10,000-pound capsule or as low as 25 pounds for a 3000-pound capsule. Increased attitude-control thrust levels will require about 25 pounds of additional propellant and inerts.

Any changes in the Science Subsystem instrumentation will require minor revision to the spacecraft cabling subsystem and to the structural mounting of new instruments.

Although the communication distance for the 1973 mission is slightly greater than that for the 1971 mission, the telecommunication subsystem capability meets a 7200-bps science data requirement and is considered adequate. The power, pyrotechnic, and temperature control subsystems, and the propulsion module of the 1971 Voyager are adequate for the 1973 mission.

1975 MARS MISSION

The 1975 flight is envisioned to be primarily oriented toward placing a relatively large payload on the Mars surface to determine the existence and characteristics of life forms.

SUBSYSTEM MISSION	SPACECRAFT BUS					PROPULSION MODULE		SPACE SCIENCE
	POWER SUBSYSTEM	GUIDANCE & CONTROL SUBSYSTEM	TELECOMMUNICATIONS SUBSYSTEM	STRUCTURAL & MECHANICAL SUBSYSTEM	TEMPERATURE CONTROL SUBSYSTEM	MIDCOURSE & ORBIT TRIM SUBSYSTEM	ORBIT INSERTION	SPACE SCIENCE INTERFACE
MARS 1973	• Not affected.	• Slight Increase in propellant & thrust levels to accommodate heavier planetary Vehicle Wt = +25 lbs	• Not affected	• Minor detail changes to accommodate a slightly heavier Flight Capsule. Wt = +25 lbs	• Not affected.	• Not affected. (Reserve propellant can be used to augment the solid engine for orbit insertion)	• Not affected.	• Not affected.
MARS 1975	• Not affected. • Additional power is available by adding to solar panels. System is designed for growth	• Increase in propellant & thrust levels to accommodate heavier Planetary Vehicle. Wt = +500 lbs. Alternate is hot-gas reaction control.	• Minor changes may be required to accommodate Flight Capsule Science Payload.	• Requires modification & strengthening to accommodate larger Flight Capsule. Wt = +150 lbs.	• Not affected.	• Not affected.	• Not affected. (Orbit insertion ΔV capability decreased due to heavier planetary vehicle)	• Generally not affected (depending on NASA/JPL science package).
MARS 1977	• Not affected. • Additional power is easily obtained by adding to solar panel area. System is designed for growth.	• Increase in propellant & thrust levels to accommodate heavier Planetary Vehicle. Wt = +500 lbs. Alternate is hot-gas reaction control.	• Minor changes may be required to accommodate Flight Capsule Science Payload.	• Requires modification & strengthening to accommodate larger Flight Capsule. Wt = +150 lbs.	• Not affected.	• Not affected.	• Not affected. (Orbit insertion ΔV capability decreased due to heavier Planetary Vehicle)	• Generally not affected (depending on NASA/JPL science package).
VENUS	• Present power subsystem has excess capacity. Solar panel can be reduced. • Minor modifications to switching diodes	• Minor changes	• Adequate transmission capability for 7500 bps. • Modify signal conditioner to accommodate revised experiments.	• Add provision for 3 ft. dia. antenna radar science experiment. • Add small antennas to scan platform for science experiments	Modifications required: • Adjust lower stop settings • Equip. compartment packing density • Selective location of equip. • Insertion motor insulation • Minimize structural support heat shorts	• Adequate for 19,500 lb Planetary Vehicle	• More than adequate for 19,500 lb vehicle (by about 5 percent of solid propellant loading)	• Experiments & science subsystem revised due to Venus cloud cover. • Spacecraft science instruments will weigh about 186 lbs, require 365 watts, & will generate about 5.55×10^6 bits per orbit.
						• For a 20,500 lb Planetary Vehicle off load 620 lbs. of N_2H_4	• Adequate	
JUPITER Equipment must be qualified for trip times of 650 days.	• Replace solar panels with a radioisotope power supply.	• Attitude control propellant must be increased.	• Modify signal conditioner to accommodate revised experiments. • Data transmission rate about 300 bps.	• Minor modification to accommodate different science subsystem, radioisotope power source & shielding. • Increase meteoroid shielding.	• Require modification to accommodate radioisotope power supply. • Modify insulation in selected areas.	• Off load 2600 lbs. N_2H_4 and three tanks	• Remove engine for low weight flyby case	• Experiments & instruments revised to measure unique Jupiter characteristics.
						• Off load 2200 lbs N_2H_4	• Use engine as a kick stage during earth escape	
MERCURY	• Replace solar panels with a solar-thermoelectric generator or possibly a radioisotope power supply.	• Minor changes.	• Data transmission rate is 7500 bps • Modify signal conditioner to accommodate revised experiments	• Config. modifications due to temp. control, power & propulsion subsystem changes. • May req. additional rad. protection.	• Complete thermal redesign required.	• Off load 1560 lbs N_2H_4 and two tanks	• Remove engine. (Not required for flyby)	• Experiments & instruments revised to measure unique Mercury characteristics.
SATURN Equipment must be qualified for trip times of 3 years.	• Replace solar panels with a radioisotope power supply.	• Attitude control propellant must be increased.	• Data transmission rate about 100 bps • Modify signal conditioner to accommodate revised experiments	• Modification to accommodate different science subsystem, radioisotope power source & shielding. • Increase meteoroid shielding.	• Modify insulation in selected areas. • Require modifications to accommodate radioisotope power supply	• Off load 3030 lbs N_2H_4 and three tanks	• Remove engine for low weight flyby case	• Experiments & instruments revised. Similar to Jupiter requirements.
						• Off load 2700 lbs N_2H_4	• Use engine as a kick stage during earth escape	

Table 4-3: Voyager Spacecraft Advanced Missions Adaptation Study

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The Flight Capsule will have an operating-life goal on the surface of Mars of 2 months, during which time it will transmit information directly to Earth. The Spacecraft Bus will gather environmental and surface data from Mars orbit and will serve as a backup communication relay station for a period of 6 months.

Mission Description

The basic mission mode for 1975 is the same as for previous Mars missions. Two Planetary Vehicles are launched on a single booster; each Planetary Vehicle will consist of a Flight Spacecraft and Flight Capsule.

Selecting the type of trans-Mars trajectory to be used for 1975 is more difficult than for the 1973 opportunity. Both Type I and Type II trajectories are considered (Table 4-1). The Type I is favorable from the standpoint of lower communication distance at arrival and shorter trip time, whereas Type II is characterized by longer launch periods and more favorable launch azimuths. Another factor favorable to the Type I trajectory is the heliocentric position of Earth and Mars at arrival. The communication time between orbit insertion and occultation of Earth by the Sun is 4.5 to 5 months for Type I and only 2 to 3.5 months for Type II. Assuming that a short launch period (20 days) is acceptable, the Type I appears to be the preferred trajectory.

The Flight Capsule is taken into Mars orbit and then deorbited and injected along the desired entry trajectory. This mode of operation was selected over direct entry to obtain better control of the atmosphere entry angle. If preceding missions improve confidence in the direct-entry approach, this mode of operation might be more desirable, depending on the useful weight desired in orbit versus the weight desired on the surface.

Selecting the Mars orbit for 1975 will depend ultimately on results of preceding missions. If the wave of darkening continues to be of primary interest, observations will generally be conducted in the northern hemisphere where the maximum contrast in the wave of darkening and the recession of the polar cap are in progress. Typical orbit inclinations are 70 to 90 degrees, with the subperiapsis point located between 40 and 85 degrees north latitude. The heavier 1975 Planetary Vehicle reduces the insertion ΔV capability of the 15,000-pound propulsion module to about 1.2 kilometers per second. To enter the shorter-period orbits (about 12 hours) throughout a 45-day launch window with this reduced

insertion ΔV , the nominal periapsis altitude is reduced to 600 kilometers, which is not unreasonable following successful 1971 and/or 1973 missions. Depending on VHP, orbit periods can range from 6 to 20 hours.

Science Payload

The science payload for the 1975 flight (Table 4-4) is envisioned to be generally the same as that for the 1971 flight, with the additional possibility of a higher-resolution TV camera system.

SENSORS AND EQUIPMENT (1)	PURPOSE	WEIGHT (Pounds)	POWER (Watts)	TOTAL BITS/ORBIT	REMARKS
Plasma Probe	Solar flare data during low Sun activity period	7	3	480	Require simultaneous data acquisition
Cosmic Ray Telescope		3	1	380	
R.F. Noise Detector		4	3	500	
Ion Chamber		3	1	240	
Gamma-Ray Detector		3	1	5.4×10^3	
Cosmic Dust Detector	Blue Cloud experiments	2	1	40	
Magnetometer		8	8	50	
UV Spectrometer		25	12	3×10^6	
IR Spectrometer		25	8	5×10^4	
IR Scanner		10	10	1.2×10^7	
TV Cameras	Surface reflectivity & emissivity	40	40	2.9×10^8	

(1) Trapped radiation detector, ionosphere sounder, bistatic radar, and gravimeter are assumed to have satisfied martian mission requirements during the 1971 and 1973 opportunities.

Table 4-4: 1975 Science Payload

Design Implications

Within the assumptions stated above for the flight profile and science payload, no major modifications appear necessary to the 1971 preferred design (see Table 4-3). The reaction control system would have to be modified to accommodate the larger space vehicle. The additional propellant and tankage would amount to 500 pounds if the cold-gas concept is retained. By changing to a hot-gas concept, the system weight would be about the same, or slightly less than the 1973 Mars design.

For the heavier capsule weight, it will be necessary to increase the gage size of the magnesium shell, including stiffeners and the aluminum longerons in the bus primary structure. This would add about 150 pounds to the structure. A 10,000-pound Flight Capsule and a 3500-pound Flight Spacecraft bus will cause about a 380-pound increase in Planetary Vehicle Adapter weight; however, this is still well within the 1500-pound-adapter weight allocation.

No changes are necessary in the telecommunication subsystem if there is no major change in the data transmission requirements of the Science Subsystem aboard either the Flight Spacecraft or Flight Capsule.

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If a Type II transfer trajectory is used for the 1975 mission, the communication distances at Mars encounter are such that the transmission rate will be reduced from a science bit rate of 7200 bps to 1200 bps. The preferred design includes a telemetry mode at this lower data rate. Communication requirements exceeding 1200 bps require a larger high-gain antenna and/or a higher output transmitter. Higher electrical power can be obtained using the built-in growth capability of the power system. The area of the solar-panel array can be increased from 316 square feet to 454 square feet, which increases power output approximately 40 percent. The weight increase of the area change would be 120 pounds.

1977 MARS MISSION

The objectives for the 1977 mission are generally the same as those for the 1975 mission. Possibly more emphasis will be placed on obtaining information on seismic disturbances, Mars surface changes due to erosion mechanisms, and on other planetological phenomena.

From the system and operational standpoint the only changes relative to the 1975 mission are the possible location of the Flight Capsules on the Martian surface and the longer operational goal of 6 months on the surface.

Mission Description

System implementation for the 1977 Mars mission is basically the same as that for the 1975 mission; the only notable difference is the duration of capsule operation.

Both Type I and II trajectories were considered (Table 4-1); however, unlike 1975, there is little difference between these trajectory types in communication distance at arrival or communication time from orbit insertion to Earth occultation by the Sun. Assuming the longer trip time (approximately 60 days) is acceptable, the Type II appears to be the preferred trajectory because of a longer launch period and a more favorable launch azimuth range.

The reference Mars orbit is similar in size and shape to that in 1975. Again, if the wave of darkening is still of primary interest, observations will be made in the northern hemisphere with the subperiapsis point located between 0 and 30 degrees north latitude. Orbit inclinations range from 40 to 60 degrees.

Science Payload

No significant additions to the complement of science equipment, over the 1975 modifications, are envisioned for the orbiting spacecraft. This, of course, depends largely on the findings of preceding flights.

Design Implications

The modifications to the 1971 preferred design discussed above for the 1975 mission apply to the 1977 flight (see Table 4-3).

VENUS MISSION

Specific mission objectives are to verify Mariner II data, and to gather additional information concerning Venus' magnetic and radiation fields, figure, geoid, atmosphere, cloud layer, and surface characteristics.

Mission Description

An orbiter-lander mission, similar to the Mars 1971 mission, has been selected as the baseline Venus mission. Two 19,500-pound Planetary Vehicles are launched on a single Saturn V. Each Planetary Vehicle is placed in an elliptical orbit around Venus from which a capsule-probe is ejected and deorbited by a self-contained retropropulsion system for atmospheric entry. Measurements are taken until systems become inoperative because of excessive temperatures near the surface of Venus. Like the Mars 1971 flight, the capsule is deployed from orbit to provide greater control of entry angles and, consequently, slower descent times.

A 1975 mission was chosen as typical, although it is recognized that prior commitments of the DSN and SFOF for the Mars 1975 flight may not allow another planetary mission during the same time period.

Type II transfers were chosen to reduce the hyperbolic excess velocity at Venus arrival. The transfer orbits are Class II for the first 40 days of launch window and Class I thereafter. Consequently, the trip time varies from 191 days for a May 2nd launch to 146 days for a June 15th launch. This arrival date results in minimum communication distance within DSIF tracking constraints while minimizing the arrival velocity.

The communication distance (Venus to Earth) at arrival is 108×10^6 kilometers, which increases to 170×10^6 kilometers after 60 days (January 8, 1976). A representative orbit with a periapsis

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of 8200 kilometers and an apoapsis of 30,000 kilometers is selected. The periapsis is low enough to satisfy requirements of the scientific payload and high enough to ensure a long lifetime. The apoapsis distance was selected to allow for a survey of the planet's magnetic field and trapped radiation, if any. The orbit injection requirement is 6280 feet per second. The orientation of the approach asymptote allows the periapsis point to be located in sunlight and near the terminator. Orbit period is 8 hours.

As an alternate to using the 1971 Mars Planetary Vehicle weights, it is possible to launch two Planetary Vehicles weighing 20,500 pounds each on the Venus mission discussed above. For this latter case, the full capability of the solid-propellant orbit insertion engine would be used.

Scientific Payload

Primary emphasis has been placed on measurement of the atmosphere and verification of the Mariner II data, with some instruments added to the package for measuring surface features.

Table 4-5 contains a representative list of the Venus Spacecraft Bus scientific instruments, their purpose, and their design parameters. Experiments selected for the capsule-probe payload are intended to obtain information on the atmospheric parameters below the cloud layer and to obtain a better definition of the surface features.

Design Implications

An initial-series mission to Venus — using both a bus and a capsule probe to measure the Venus atmosphere, cloud layer, and surface — can be performed within the communication, propulsive, and electrical-power capabilities of the preferred design. However, the environmental control subsystem must be changed to maintain temperature control.

A comparison of the Mars and Venus missions shows that a Venus-bound spacecraft is exposed to a much higher solar flux, a lower micrometeoroid flux, and a shorter communication distance. Increased solar flux results in more severe thermal, particulate, and ultraviolet radiation environments. Therefore, the electronic subsystems must be more carefully selected or shielded to avoid degradation from high-energy particulate radiation and thermal effects. Coatings, transparencies, fiberglass, lubricants, and resins are typical of the materials that must be more carefully screened and qualified for use on a Venus design. The lower micrometeoroid flux and shorter communication distance are advantages and allow a relaxation of design requirements.

Communications — The Voyager telecommunication system provides ample capability for the Venus mission. It is only necessary to mate the different science instruments to the system.

Table 4-5: Venus Orbiter Scientific Payload

NO. EXPERIMENT	WEIGHT (Pounds)	POWER (Watts)	DATA RATE (Bits/Orbit)	FEATURES OBSERVED
1 Infrared Radiometer	2	2	2.5×10^4	Cloud movement, temperature, pressure
2 Microwave Radiometer	2	2	1.25×10^4	Surface temperature of planet
3 IR-Vis-UV Spectrometer	30	10	0.5×10^4	Composition of clouds and atmosphere
4 Dual-Frequency Radio Transponder	3	10	800	Atmospheric density
5 Television Camera	35	20	5×10^6	Cloud patterns, possibly surface features
6 Solar Radiation Monitor	3	0.5	1000	High-energy particulate radiation
7 Helium Vector Magnetometer	8	8	50	Magnetic fields
8 Optical Polarimeter	10	4	200	Gross density, albedo
9 Plasma Probe	7	3	480	Solar winds
10 Ion Probe	1	1	240	Local ion population
11 Dust and Meteoroid Detector	1	1	40	Local meteoroid population
12 Radar Imaging System and (Radar Altimeter)	80	300	50×10^4 (4000)	Map surface features (Planet figure and orbit perturbations)
13 UHF Receiver	4	3	500	Nonthermal radio emissions
TOTALS	186	364.5	5.55×10^6	

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Propulsion—The preferred propulsion system is more than adequate for the nominal mission. Solid-propellant can be off-loaded by some 5 percent and still meet the orbit-insertion requirements. For Planetary Vehicles weighing 20,500 pounds, 620 pounds of N_2H_4 can be off-loaded from the propulsion module, thereby increasing the useful payload in Venus orbit.

Electrical Power Generation—The solar array can provide over 2500 watts during the envisioned Venus mission. This includes capability near Earth, to Venus, and in Venus orbit. Because the highest estimated source power requirement is 1200 watts during the Sun portion of a Venus orbit, the solar-panel area can be reduced to less than one half its present size, which would save about 120 pounds.

The highest energy requirement for the Venus mission occurs during Sun occultation in orbit. The estimated energy requirement during this period is 1395 watt-hours (including the radar altimeter), which is less than the 2720 watt-hours provided by the batteries for the 1971 Mars mission. The power-conditioning equipment selected for the Voyager Mars mission can meet the presently estimated load requirements if the radar altimeter is supplied from the unregulated bus. It is possible that the switching logic diodes might require minor modification or uprating. The power conditioning equipment is estimated to operate properly at environmental temperatures up to 150°F, which is well above those expected during the Venus mission.

Thermal Control—Thermal design of Voyager is adequate for Venus flights. Only minor design modifications are required to account for the differences in solar flux and planetary radiation; these are reviewed in Table 4-6.

Table 4-6: Ambient Environment Comparison

PLANET	SOLAR RADIATION FLUX INTENSITY (Btu/Hr-Ft ²)	PLANET ALBEDO INTENSITY (Btu/Hr-Ft ²)	THERMAL EMISSION INTENSITY (Btu/Hr-Ft ²)
Venus	849	645	51
Earth	443	155	72
Mars	162-210	29	45

Semiactive temperature control of the equipment section is accomplished by an adjustable louver radiator network. The significant aspect of the Venus mission is that the louver system heat rejection capacity is reduced. Adequate heat rejection can be achieved by an optimum setting

of louver stops and careful attention to the packing density of electronic equipment within the equipment section. Equipment heat rejection is also reduced by planetary albedo when the spacecraft passes close to the sunlit Venus surface during the orbit phase. Careful planning of the Venus orbit periapsis location relative to the subsolar point, selective location of high-heat-producing equipment, and full use of equipment thermal capacitance is required.

External bus equipment operates at much higher temperatures and experiences much larger temperature transients in Venus orbit than in Mars orbit. Solar panel equilibrium temperature is 242°F near Venus compared with 40°F near Mars. Because the convex side of the high-gain antenna is exposed to the Sun for the Venus mission, a low-absorptance finish is required to minimize thermal distortion. Before firing the orbit insertion motor, the solar panels and antennas are precooled by maneuvering off-Sun.

Minor modifications in the temperature control arrangement of the propulsion subsystem are required to account for solar flux differences of the Venus mission. The heat influx is reduced by decreasing the conductance of the absorber plates and the solar engine heat shield. Heat input through the orbit insertion nozzle is minimized during cruise in the present design with a nozzle cover. Thermal capacitance (MC_p) of the orbit-insertion motor (about 850 watt-hours/°F) supplements the nozzle cover in controlling temperature during cruise. The orbit-insertion motor constitutes a thermal problem after it has been fired and returns to thermal equilibrium. Solar radiation is absorbed through the nozzle throat. For an overall wall thermal conductance of 0.01 Btu/hr-°F-ft², the internal equilibrium temperature of the case is 241°F. The resulting heat load on the monopropellant tanks is 63 watts. Louver area must be adjusted to balance the total heat influx at this condition. The midcourse and orbit trim maneuver engines will reach 378°F at Venus (240°F at Earth), which is acceptable for the catalyst beds. Valves are thermally isolated from the engine and operate within their control range of 75 to 110°F. The midcourse engine mounting is insulated to minimize heat influx to the propellant tanks.

JUPITER MISSION

The specific objective of the Jupiter mission is to gather information concerning the planet's magnetic and radiation fields, radio-emission phenomenon, differential rotation at different

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latitudes, atmosphere constituents, cloud layer and surface features, and planet figure and geoid.

Mission Description

The reference mission mode selected for an initial Jupiter flight is a flyby to within 230,000 kilometers of the surface. Although the Saturn V could place a single orbiter spacecraft at Jupiter, the probability of a spacecraft performing satisfactorily in the unknown but suspected high-radiation and magnetic fields surrounding the planet is considered marginal. Therefore, the selected mission plan is to launch two Planetary Vehicles on a single Saturn V booster. Each vehicle consists of a basic flyby bus and a propulsion module that provides 200 meters per second for trajectory correction and time separation biasing maneuvers. A Type I transfer trajectory resulting in a trip time of 650 days was chosen (Table 4-2) in consideration of favorable planet positions for communications and the higher probability of success resulting from shorter trip times. Launch periods of 45 and 20 days were examined. Although the longer period affords a higher success probability, it reduces the booster payload capability from 13,700 pounds to something under 4000 pounds.

With the spacecraft weight available, one or two impact-type Flight Capsules can be included to obtain atmospheric data and to take pictures of the planetary surface itself. The capsule is injected into an entry trajectory prior to planet encounter, and throughout the mission it transmits data at a high rate to the flyby vehicle for subsequent relay to Earth.

In consideration of the flyby miss distance, the hyperbolic approach velocity, the type of science information to be obtained, and the fields about the planet are all important criteria. The relatively high approach velocity at encounter requires either some form of image-motion compensation for the TV scanning equipment or a flyby at a distance where picture blur is not critical. In evaluating the above factors, an approach distance of 230,000 kilometers was chosen.

Scientific Payload

Jupiter is a particularly interesting target for further exploration because it is completely different from the inner planets Mercury, Venus, Earth, and Mars. Although its mass is almost 2.5 times that of all other solar planets combined, its density is less than a fourth of Earth's,

which indicates a completely different composition and internal structure. Table 4-7 contains a representative list of Jupiter flyby scientific instruments, their purpose, and design.

Design Implications

The preferred design will require physical modifications to adapt it to a Jupiter mission, specifically in the area of electrical power generation. To generate the same power level at Jupiter as for the 1971 Mars flight requires about a five-fold increase in solar-panel area because of the reduced solar constant. As an alternate to using more solar-panel area, the use of a radioisotope power system is recommended even though it requires the design of a separate radiator system to dissipate the heat of the energy source.

Communications—The telecommunication distance for the Jupiter flyby varies between 7×10^8 km and 9×10^8 km, about twice the maximum Mars 1971 distance of 4×10^8 km. With the 6.5-foot-diameter (31 db) antenna and 50-watt transmitter in the spacecraft, and the 210-foot (61 db) antenna at the DSIF sites, the communication data rate capability varies between 500 bps at arrival and 300 bps at the end of the mission.

The Voyager storage system includes tape recorders that provide storage of 3.6×10^8 bits; therefore, the experimental data will be recorded at real time on tape and played back at 288 bps (an existing data mode) for transmission to Earth. The only modification to the Voyager telecommunications subsystem is in the signal processors, which is due to different experiments.

Propulsion—The orbit insertion ΔV capability of the 15,000-pound propulsion module is not required because the selected Jupiter mission mode is a flyby. The solid engine and associated components could simply be removed from the basic module, leaving the structure unchanged. It is possible, due to the lighter spacecraft weight, to off-load some of the propellant and inerts of the midcourse and trim propulsion system. Specific changes would include removal of three N_2H_4 tanks and 2600 pounds of maneuvering propellant.

An alternate plan would be to use the fixed solid-propellant loading of the 15,000-pound propulsion module as a kick stage for injection into a trans-Jupiter trajectory. Using this plan, a Planetary Vehicle weight of 11,450 pounds (7950, excluding propulsion) can be obtained as against 6850 pounds (5450, excluding propulsion) for the former case.

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Table 4-7: Jupiter Flyby Scientific Payload

NO.	EXPERIMENT	WEIGHT (Pounds)	POWER (Watts)	TOTAL BITS AT ENCOUNTER	FEATURES OBSERVED
1	Television Imaging Red/Green Filters	30	20	1.2×10^8	Clouds, atmospheric markings, bands, and surface spots
2	UV Imaging (Uvicon)	5	5	1.2×10^8	
3	Decametric Radiometer	35	5	4.8×10^4	Localized sources of radio burst
4	Decimeter Radiometer	15	3	4.8×10^4	Thermal and nonthermal microwave emission origin
5	Helium Vector Magnetometer	8	8	1.2×10^3	Magnetic fields
6	Trapped Radiation Detector	3	0.5	2.4×10^4	Radiation belts
7	Plasma Probe	1	3	5.8×10^3	Solar-charged particles
8	Ion Probe	1	1	5.8×10^3	Ionosphere
9	Dual-frequency Transponder	8	15	1.4×10^5	Atmosphere and ionosphere density
10	Optical Polarimeter	10	4	4.8×10^4	Jovian atmosphere and aerosol content
11	UV-IR Spectrometer	30	10	4.8×10^5	Atmospheric composition
12	IR Scanning Radiometer	2	2	1.2×10^6	Lower atmosphere temperature, possible surface imaging and temperature.
13	Cosmic-Dust Detector	1	1	1×10^2	Micrometeoroid flux in asteroid belt and elsewhere.
14	Cosmic-Ray Detector	3	0.5	2.4×10^4	Cosmic-ray flux variation with solar distance.
TOTALS		152	78.0	2.5×10^8	

Four hour encounter period assumed

Electrical Power Generation— Representative Jupiter mission power requirements dictate a power source capable of supplying over 750 watts during portions of the mission, including the time when the spacecraft emerges from a Jupiter eclipse. The Model 945-8055 solar-array power capability is estimated to decline to 80 watts (± 10 watts) at Jupiter distances; therefore, a total solar-panel area of about 1500 square feet would be needed to supply the required power. This area is more than three times the planned growth capability of the preferred design.

From the standpoint of specific weight and technology status, the most suitable candidate for a 1975 Jupiter mission is a radioisotope thermoelectric generator (RTG) using Pu 238 fuel. One possible power system configuration uses a cluster of five or six 100-watt generators (up-rated and modified SNAP 27 design) conductively coupled to a cylindrical radiator plate.

Thermal Control— The temperature control subsystem for internal bus equipment is satisfactory for Jupiter missions because heat loads are

about the same as for Mars missions. Average heat load is 565 watts with peaks to 748 watts. The environmental control system for the radioisotope power generator includes a separate radiator for temperature control of the energy source. If the isotope system is designed for a usable electrical load of 565 watts and the peak loads are handled by a battery, isotope radiator area is about 39 square feet. This assumes a radiator effectiveness of 0.5 operating at an average temperature of 600°F with an isotope conversion efficiency of 5 percent.

External equipment can be designed to operate colder at Jupiter where the solar flux is 16.4 Btu/hr-ft² or can be heated, if necessary, by radiation coupling with the isotope power source radiator. Minimum temperatures are about the same as during the Mars mission and occur during occultation. Assuming the use of similar thermal coatings, maximum temperatures occur near Earth for both Mars and Jupiter missions.

Propulsion equipment temperature control will be satisfactory out to Mars orbital distances.

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Thereafter, electric heating is required to maintain temperatures above 75°F.

OTHER MISSIONS

Adaptability of the Voyager baseline design has also been considered for flyby missions to Mercury and Saturn.

Mercury Flyby Mission

Specific objectives of a Mercury mission are: temperature measurements of both dark and light faces by using both infrared and microwave radiometers, spectroscopic and polarimetric search for possible atmosphere components, television imaging of parts of the surface, and measurement of solar radiation and particle emission in the vicinity of Mercury.

Mission Description—Although relatively high Planetary Vehicle weights can be realized for Mercury missions, the high approach velocities, insertion ΔV requirements, and resultant high orbit, dictate a near-planet flyby mission rather than an orbiter mission. The postulated mission design considers launching two 19,000-pound Planetary Vehicles (16,710, excluding propulsion). Each vehicle is made up of a Spacecraft Bus, impact-type Flight Capsule(s) for obtaining environment data and/or pictures of the surface, and a propulsion module providing 200 meters per second of midcourse correction and arrival bias ΔV . The data is transmitted at high rates to the flyby spacecraft for subsequent relay to Earth.

A Type II transfer trajectory is used (Table 4-2) instead of a Type I to reduce the launch energy requirements (C_3 of 50 vs 80 km²/sec²), thereby providing substantially higher potential spacecraft weights, and to provide a longer launch period. Trip time and relative approach velocity are not a strong function of trajectory type for a Mercury mission. The hyperbolic approach distance at planet encounter will depend on such factors as the desired probe-entry trajectory and the design of the TV scanning equipment (e.g., image-motion compensation).

Scientific Payload—Instruments on a flyby would include IR and microwave radiometers, a UV-IR spectrometer, a polarimeter, helium vector magnetometer, television camera, and solar radiation and particle monitors. (A soft-landed instrument package, probably near the terminator, would be very useful as a solar monitoring platform, although there would be extreme technical problems in such an operation.)

Design Implications—The communication distance for the Mercury flyby is 1.7×10^8 km. Assuming that adequate environmental control exists, the Voyager communication system will provide results for the Mercury flyby that are similar to the Venus orbiter. Specifically, with a 50-watt transmitter and a 6.5-foot antenna, 7200 bps of science data can be transmitted to Earth at a 5×10^{-3} bit error rate. This data rate, coupled to the Voyager data-storage system, is adequate for transmission to Earth of scientific data.

The 1971 propulsion module is easily altered to accommodate the flyby mission mode. Modifications consist of deleting the solid-propellant orbit-insertion motor and off-loading the midcourse propellant tanks by some 1560 pounds of N₂H₄ plus two tanks because orbit trim is not required.

The Voyager power system is inadequate for the Mercury mission. An estimated 600 to 800 watts of source power is required for a scientific mission to Mercury. The solar flux near Mercury is more than six times the near-Earth flux level. The resulting panel temperature is estimated at 630°F without thermal shielding. The solar array cannot operate above 390°F; hence, it is not suitable for this mission. However, a solar-thermoelectric generator operates favorably at 570°F and is considered the leading candidate for this mission. The power-conditioning equipment must be maintained at temperatures below 150°F in order to perform properly.

The solar intensity at Mercury is approximately 2960 Btu/hr-ft²—more than 10 times that at Mars. Therefore, complete thermal redesign of the preferred 1971 Mars spacecraft will be required.

Saturn Flyby Mission

The general objectives of a mission to Saturn could probably be accomplished by a flyby. Specific objectives would include information concerning the ring structure, magnetic and radiation fields, radio-emission phenomenon, atmosphere constituents, cloud layer and surface features, and the planet figure.

Mission Description—Like the Jupiter mission, a flyby/probe mission mode rather than an orbiter mission is selected for the first Saturn mission because of the suspected extreme radiation environment of Saturn and the high Earth-launch energies involved (C_3 of 135 km²/sec², VHP of 9 km/sec) that limit total Planetary

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Vehicle weight to about 3650 pounds. Because of this weight limitation, only a single Planetary Vehicle will be launched on each booster. Launched within a 20-day launch period and employing a Type I transfer trajectory, typical of the 1975 opportunity, the flyby/probe vehicle will reach Saturn in 1200 days. The probe(s) is released before planet encounter and established on an impact trajectory from which data is transmitted at a high rate to, and recorded by, the flyby vehicle. When free of the anticipated Saturn radiation interference, the flyby vehicle relays the recorded data to Earth.

Scientific Payload—The instruments planned for Jupiter are presumably applicable to Saturn as well, although some simplification may be necessary because of the very long in-transit time and the power required for data return.

Design Implications—The Saturn flyby mission communication distance is 14×10^8 km. At this distance the Voyager telecommunication system is capable of transmitting approximately 100 bps. The Voyager telecommunication system has a data mode of 80 bps, which is adequate. Again, as with Venus, Mercury, and Jupiter flights, all data from the scientific experiments would be recorded at a high rate and transmitted to Earth at a lower rate consistent with system capability. Because the Saturn flyby is not anticipated until the 1980's, advances in the state-of-the-art should increase the transmission data rate.

The Saturn mission is a flyby; therefore, orbit-insertion capability is not required. If the 1971 Mars propulsion module is appropriately revised to delete the solid-engine system and to reduce the midcourse and trim propulsion system, one Planetary Vehicle weighing 3650 pounds (2680, excluding propulsion) can be launched. Required system modifications include removal of the solid engine and off-loading some 3030 pounds of N_2H_4 and three hydrazine tanks.

An alternative for the Saturn flight is to use the 15,000-pound propulsion module for a kick stage as was mentioned for the Jupiter flight. For this case, a single vehicle weighing 11,200 pounds (8200, excluding propulsion) can be launched. Modifications to the propulsion system would only consist of off-loading some 2700 pounds of N_2H_4 .

An estimated 700 to 900 watts of source power is required for a Saturn flyby mission. The low solar flux near Saturn prohibits the use of a photovoltaic conversion system. The mission time (3 years) dictates the system lifetime. A

radioisotope-thermoelectric or -thermionic generator is considered as the leading candidate for this application. The former has been flight tested; hence, it is the most easily adaptable to a Saturn mission. The latter is more efficient but is still in the R&D stage. For a post-1980 mission, it may reach a development status to be considered a serious contender. Important problems associated with conditioning equipment are the long life span required and the equipment maintenance at temperatures above $-60^\circ F$.

The thermal-balance problem presented by the Saturn mission is similar to the Jupiter mission. The solar intensity at Saturn is 4.9 Btu/hr-ft^2 compared to 16.4 Btu/hr-ft^2 for Jupiter. Therefore, external appendages must be designed to accommodate the low equilibrium temperatures or must be heated. Radiation from the isotope power source can be utilized to maintain reasonably high appendage temperatures and/or external bus temperatures.

MARS MISSION SELECTION RATIONALE

Successful performance of the NASA/JPL Mars project depends on establishing a sequence of missions to meet the project objectives. Although each mission is performed separately, a single-thread plan is required; the success of a subsequent mission must be based on the results—operational information and engineering and science data—obtained from the preceding mission. Figure 4-2 presents a typical mission planning flow. As noted, two distinct iterative loops are shown. The engineering loop provides data with which to design more sophisticated spacecraft and plan more complex operational functions. The science loop is based on acquiring or extending purely scientific knowledge. A successful Mars space project depends on obtaining both engineering and science information.

After the mission pattern is established, studies are conducted to develop a spacecraft design (including science experiments and support instruments) and operational concept so the sub-objectives are accomplished with the highest possible probability of success. The term sub-objectives is used in the sense that each mission must also support subsequent missions to achieve the larger project objective.

NASA/JPL has established one of the primary objectives for Mars exploration as the obtaining of information concerning the existence and nature of extraterrestrial life. Although this objective might not be an end in itself, it defines a goal that is technically feasible, scientifically significant, and nationally acceptable.

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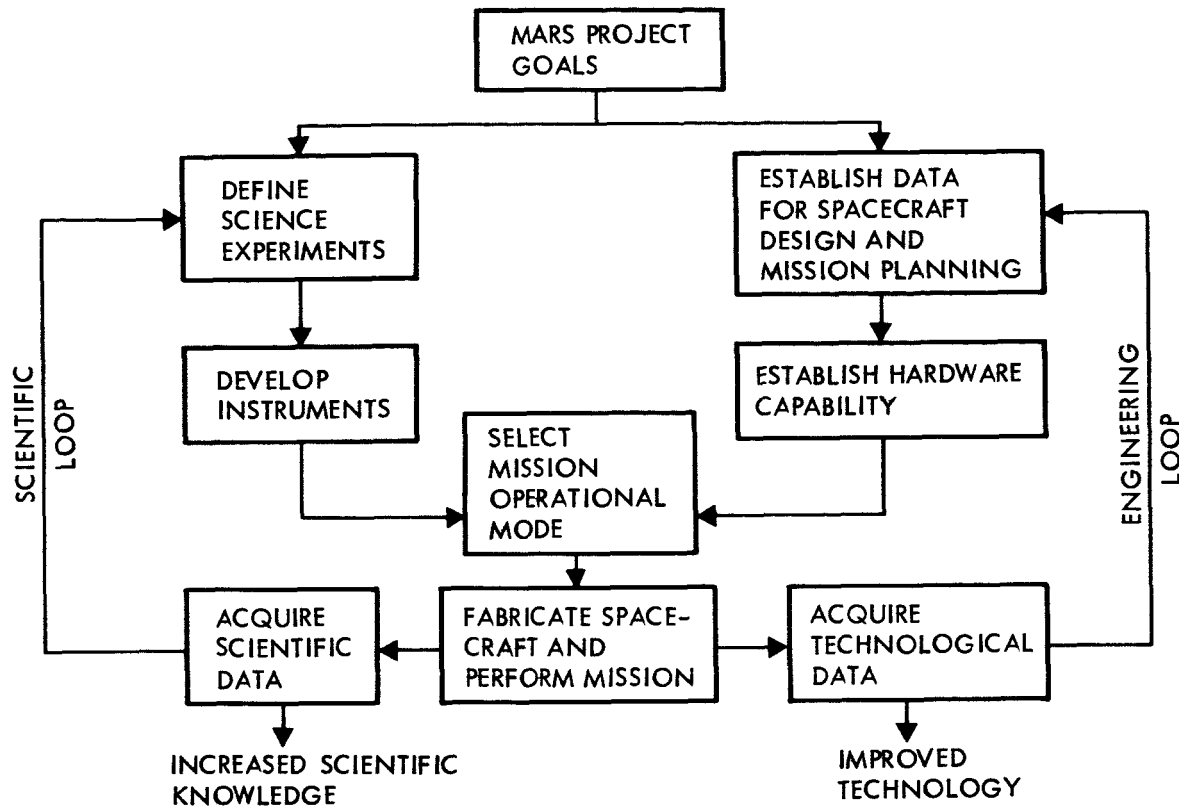


Figure 4-2: Mission Planning Flow

As depicted in Figure 4-2, establishment of data for the final spacecraft design suggests intermediate mission modes that serve to acquire the desired engineering and scientific data. Figure 4-3 presents a comparison between data for spacecraft and mission design against data that can be acquired as a result of conducting various missions, each launched directly from Earth. From the figure, the following generalized conclusions can be made:

- A lander operation is essential to determine the nature of life on Mars;
- An orbiting mission will obtain a large quantity of both preliminary and high-quality data in support of a lander design. Of specific value is extensive topographic mapping and surface weather data;
- An atmospheric entry probe is needed to obtain high-quality atmospheric data that, in turn, is necessary to obtain a high useful weight on the surface of Mars;
- A flyby mission alone will not acquire statistically significant data about the atmosphere or surface characteristics to perform a landing operation for a sophisticated lander

design. However, if instrumented probes are included, the data quality can be significantly improved.

The rationale presented above supports the 1971 Mars mission mode as established by NASA/JPL; that is, use an orbiting mode with a Flight Spacecraft Bus and a Flight Capsule to obtain atmospheric environmental data.

ALTERNATE MISSION MODE FOR 1971

Whereas the NASA/JPL mission design consists of two separate spacecraft (Flight Spacecraft and Flight Capsule), each designed to operate in a specific environment medium, it is possible to consider a single vehicle that would operate initially in Mars orbit and then in the atmosphere. For example, the spacecraft would perform in Mars orbit for 1 to 2 months and then deorbit and inject into a trajectory designed to penetrate the atmosphere. Principal factors to be considered in comparing the two concepts are discussed below with all comparative comments made relative to the NASA/JPL concept.

ENVIRONMENT COMPONENT OR MEASUREMENT	MISSION MODE							
	FLYBY		IMPACT PROBE		ORBITER		LANDER	
	REQUIRE	ACQUIRE	REQUIRE	ACQUIRE	REQUIRE	ACQUIRE	REQUIRE	ACQUIRE
SURFACE MAGNETIC FIELD				△			○	▲
LIFE FORMS (MORPHOLOGY)				△				▲
BLUE HAZE		△		▲		△		▲
CLOUD STRUCTURE AND COM- POSITION		△		△		▲		▲
PRECIPITATION COMPOSITION AND CHARACTERISTICS						△		▲
SEASONAL PHENOMENA		△				▲		▲
TOPOGRAPHICAL DATA		△		▲		▲	○	▲
SOIL TEMPERATURE AND THERMAL CONDUCTIVITY		△				△	○	▲
SOIL COMPOSITION AND DENSITY		△		△		△	○	▲
SOIL BEARING STRENGTH							●	▲
WIND ACTIVITY						▲	●	▲
MARS MAPPING		△				▲	●	▲
ATMOSPHERE DENSITY PROFILE		△	○	▲		▲	●	▲
ATMOSPHERE CONSTITUENTS		△	○	▲		△	●	▲
ATMOSPHERE TEMPERATURE		△		△		△	●	▲
IONOSPHERE ELECTRON DENSITY		△	○	▲	○	▲	○	▲
TRAPPED RADIATION DENSITY FLUX AND DISTRIBUTION		△	○	△	○	▲	○	△
METEOROID DENSITY FLUX AND DISTRIBUTION (MARS)		△	○	△	○	▲	○	△
MARS MAGNETIC FIELD		▲	○	▲	○	▲	○	▲
THERMAL RADIATION		▲		▲	○	▲	○	▲
MARS ALBEDO		▲	○	▲	○	▲	○	▲
MARS GRAVITATIONAL ANOMALIES		△	○	△	○	▲	○	△
MARS MASS AND DIMENSIONS	●	▲	●	▲	●	▲	●	▲
MARS EPHEMERIS	●	▲	●	▲	●	▲	●	▲
METEOROID DENSITY FLUX AND DISTRIBUTION (SPACE)	●	▲	●	▲	●	▲	●	▲
SOLAR CORPUSCULAR RADIATION	●	▲	●	▲	●	▲	●	▲
SOLAR THERMAL RADIATION	●	▲	●	▲	●	▲	●	▲

REQUIRED DATA

● CRITICAL

○ DESIRABLE

ACQUIRED DATA

▲ HIGH QUALITY

△ PRELIMINARY

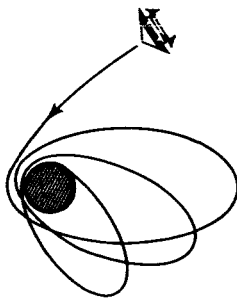
Figure 4-3: Mission Data Required and Acquired

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Probability of Achieving Mission Goals — The probability of obtaining atmospheric data during entry will be degraded somewhat for the single-vehicle concept. With the current dual-vehicle mode, the entry capsule is deployed 3 to 10 days following orbit achievement. The single-vehicle mode requires that the vehicle remain in Mars orbit to perform orbital investigations for an extended time before entry. The probability that an orbiting spacecraft is still functioning (excluding science instruments) after 2 months in orbit is about 5 percent less than after 10 days in orbit. If this reduction in probability of success is unacceptable, the reliability of the spacecraft must be increased at the expense of increased cost and weight.

Data Return — The transmission rate of atmospheric data for the direct versus the relay transmission mode is reduced from 50,000 to 7500 bps, assuming that the Mars-to-Earth communication system designed for the Flight Spacecraft Bus can be used throughout the entry phase. This means that detailed pictures of the Mars surface from low altitudes could not be transmitted if a ballistic entry trajectory is used. However, it is possible, by considering other entry trajectories or propulsion braking during descent, that detailed pictures could be transmitted as well as large quantities of environmental data. To transmit the equivalent amount of data as the Flight Capsule with its 50,000-bps capability requires that the descent through the atmosphere be much slower.

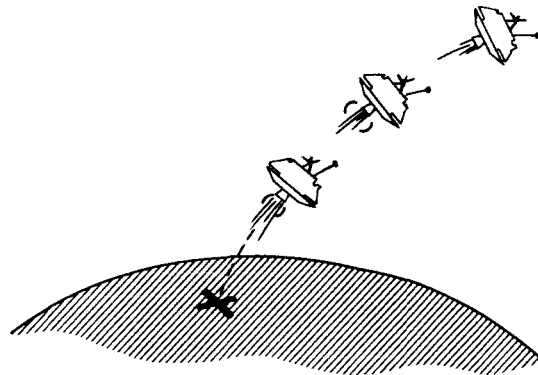
The following sketch represents a trajectory wherein the atmosphere is used to decelerate



the spacecraft. Data would be obtained throughout the descent and transmitted when the spacecraft is in view of Earth. The success of this method of descent would require previous knowledge of the atmospheric density profile. This could be accomplished through the use of precursor probes or possibly by atmospheric sounding instruments from Mars orbit.

If spacecraft weight is not prohibitive, a powered descent could be conducted with retropropulsion

applied to keep the deceleration, aerodynamic, and heating loads within acceptable limits.



System Implications — The most interesting aspect of a single-spacecraft concept is the potential use of a single set of subsystem elements. This in turn suggests:

- The possibility of a more efficient flight system;
- The possibility of lower program costs through a smaller testing program and a reduced number of manufactured parts;
- The possibility of enhancing mission success by eliminating the Flight Capsule separation operation and removing the dependency of the Flight Capsule on a parent spacecraft for such functions as receipt of commands and provision of electrical power during flight storage.

The most attractive concept is the one which will provide the highest useful weight for descent through the atmosphere, assuming that stipulated probability values and data requirements are met. Useful weight is defined as:

$$W_{T \text{ eff}} = W_e - W_{TP} - W_{PS} - W_{TB}$$

where:

$$W_{T \text{ eff}} = \text{effective terminal weight}$$

$$W_e = \text{entry weight}$$

$$W_{TP} = \text{thermal protection weight}$$

$$W_{PS} = \text{primary structure weight}$$

$$W_{TB} = \text{retropropulsion and terminal braking, if any}$$

For the NASA/JPL 1971 Planetary Vehicle mission concept, the weight of the Flight Capsule Bus at entry is about 1700 pounds excluding the

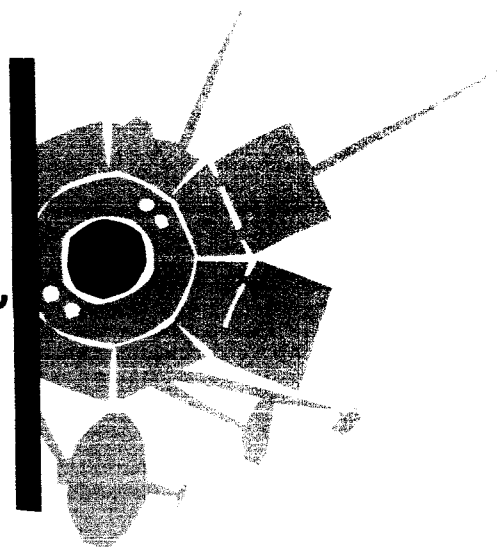
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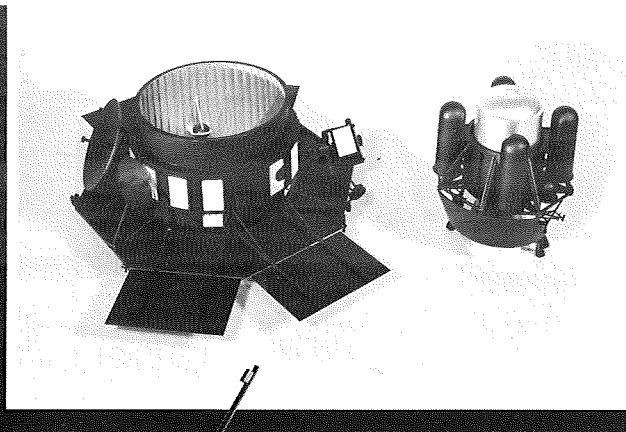
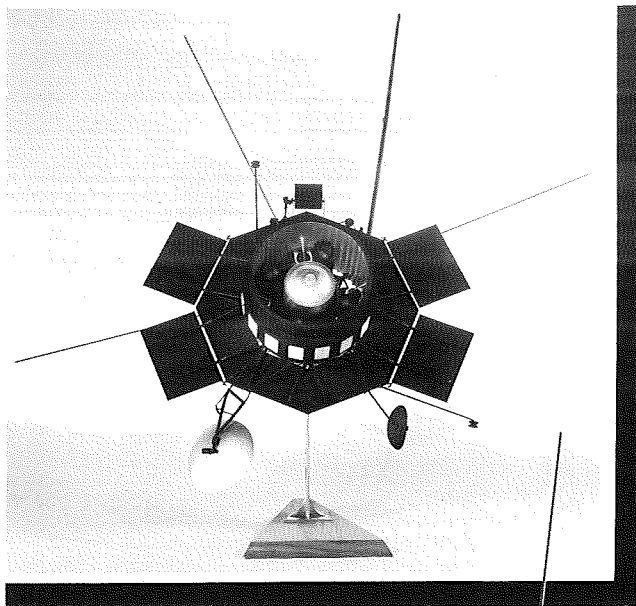
propulsion system. Using all-propulsive braking, as illustrated previously, it is unlikely that a comparable effective terminal weight can be attained for a single-spacecraft weight below 60,000 pounds. Even this would require a two-stage propulsion system and a propulsion system mass fraction of around 0.85. To be competitive with the two-spacecraft concept, it is probable that a combination of propulsive and aerodynamic braking or an all-aerodynamic braking technique must be used. To consider aerodynamic braking, atmospheric characteristics must be known accurately. Since the purpose of the 1971 mission is to obtain atmospheric environment data, implementation of this technique would probably be premature.

The operational concept using a single spacecraft requires that the entire spacecraft be sterilized. Probably the greatest additional sterilization requirement will be for the photovoltaic solar panels. No sterilization problems are anticipated if high-temperature bonding agents are used. One benefit that results from a sterile spacecraft is that, with the lower periapsis altitudes that can be used, higher-resolution pictures of the surface can be obtained.

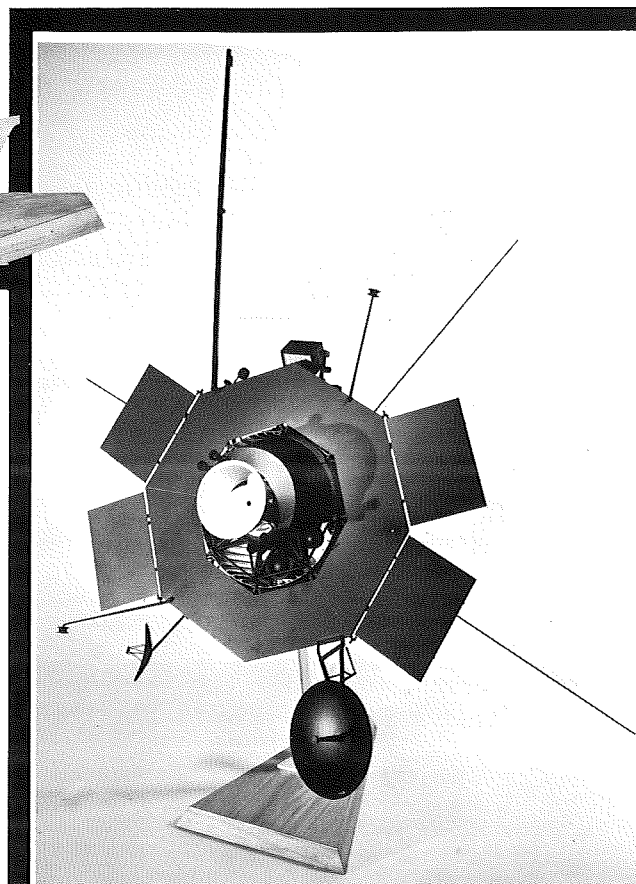
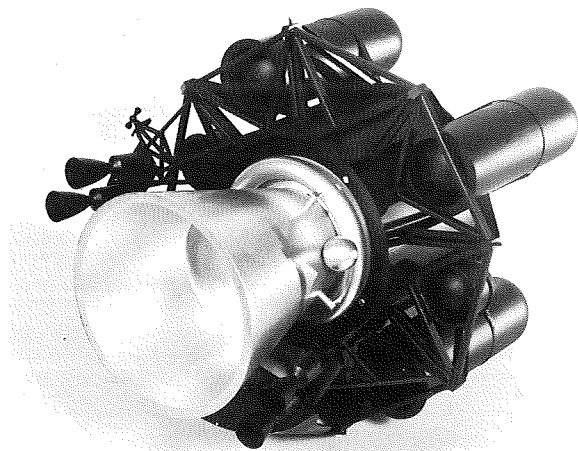
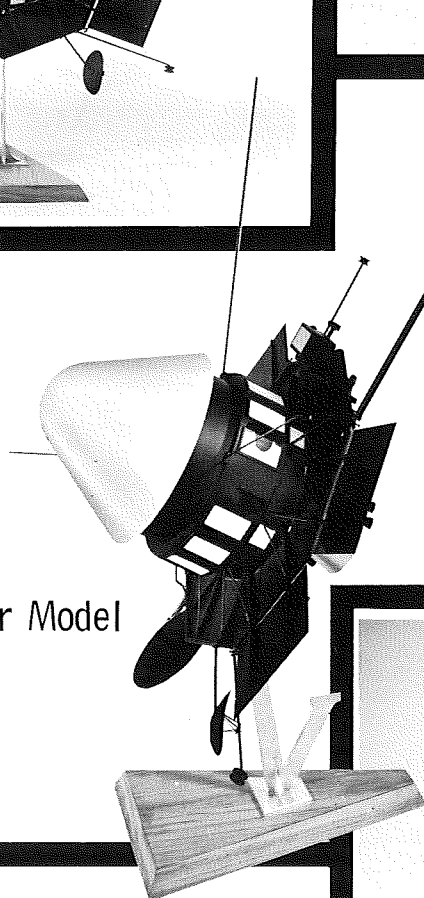
In summary, for the 1971 Mars mission, the alternate mission mode employing a single spacecraft is not considered competitive with the NASA/JPL dual-vehicle concept because of the lower probability of achieving mission goals.

VOYAGER





Voyager Model



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ADDITIONAL TECHNICAL CONSIDERATIONS

Boeing recognizes that the Voyager Program has the ultimate objective of carrying out scientific investigations of the entire solar system by instrumented unmanned spacecraft that will fly by, orbit, or land on the planets. The initial step in achieving this objective is the Mars 1971 mission, which uses systems designed and operated within the priority framework of the competing characteristics:

- Probability of success;
- Performance of mission objectives;
- Cost savings;
- Contributions to subsequent missions;
- Additional 1971 mission capability.

During Task B, Boeing emphasized using the competing characteristics as the basis for conducting trade studies and selecting the preferred spacecraft, including the propulsion subsystem. During the study, several analysis techniques, implementation possibilities, and operational procedures were identified as having an impact on one or more of the competing characteristics. An evaluation of the potential value of each item either requires a more detailed study than was warranted during Task B or necessitates discussions with and approval by JPL.

These topics are briefly discussed below; each is related to the competing characteristic it appears to affect most.

PROBABILITY OF SUCCESS

A carefully planned system simulation, developed along with the spacecraft design, can provide design data and verification, realistic personnel training, and ground testing of command effects.

A Voyager system that has a profound effect on the probability of mission success is the launch vehicle. JPL has requested that, within the mission and system constraints, the length of the spacecraft be minimized. On the presumption that this request had the objective of reducing nose fairing length and therefore, launch vehicle loads, a preliminary launch vehicle load survey was made. This survey indicated that, with

present wind criteria and guidance philosophy, the launch vehicle would experience excessive loads when carrying the Voyager payload. Ways of alleviating this problem that merit detailed investigation are discussed below.

SPACECRAFT SIMULATION CONCEPT

To prepare, test, and integrate a spacecraft design in a cost effective manner, it will be necessary to develop a simulation capability. The simulation concept described here represents a thorough and efficient technique for aiding and verifying the design development and mission conduct.

The simulation concept consists of an integration of facilities, computer programs, and spacecraft subsystems. The simulation center is located at the Boeing Space Center at Kent, Washington. It might prove desirable to have a similar capability at the SFOF. The objective of the simulation concept is to aid in:

- Defining performance requirements;
- Evaluating performance characteristics;
- Isolating problems;
- Predicting future spacecraft status or performance;
- Training and qualifying operations personnel;
- Developing and evaluating operational procedures and aids;
- Evaluating proposed spacecraft commands and reactions;
- Conducting the mission.

To perform the functions necessary to support the simulation objectives, the concept provides:

- Continuous upgrading of simulation characteristics;
- Insertion and use of flight-configuration hardware, increasing gradually from the component level to the complete spacecraft and mission-dependent ground support equipment;
- Provisions for nominal and degraded inputs and outputs for all components and functions.

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Table 5-1 describes the simulation techniques employed in this concept to accomplish the simulation functions and objectives. These techniques are consistent with and make maximum use of Boeing facilities. Boeing simulation facilities at the Boeing Space Center consist of the test data processing center, data system simulator, hybrid computing facility, information system laboratory, and space environment laboratory.

Table 5-1: Voyager Simulation Techniques

PURPOSE	FUNCTIONS	TECHNIQUE
<u>Verify Design Concept</u>	- Evaluate design. - Evaluate OSE-S/C interaction. - Evaluate effect of OSE malfunctions.	- First-level evaluation use digital or analog computer programs. - Second-level evaluation use analog and digital simulators. - OSE components tested with S/C simulators.
• OSE		
• Spacecraft	- Analyze component performance. - Evaluate subsystem compatibility. - Evaluate S/C compatibility. - Define critical parameters.	- Digital or analog component simulation. - Digital/analog system simulation with appropriate feedback.
• Mission Dependent Equipment (MDE)	- Evaluate performance limits. - Evaluate MDE/MIE compatibility. - Evaluate compatibility with S/C & SFOF. - Define critical parameters.	- Digital or analog component simulation. - Integrate into S/C system simulation.
• Mission Dependent Programs (MDP)	- Establish performance tolerances. - Evaluate sensitivity.	- Digital or analog program simulations. - Integrate into S/C system simulation.
<u>Verify Prototype Hardware Performance</u>	- Evaluate performance. - Evaluate compatibility. - Define performance limits. - Define performance degradation.	Insert prototype hardware into simulation stream, replacing its simulator.
<u>Verify System Performance</u>	- Evaluate performance and limits. - Evaluate compatibility. - Evaluate human engineering factors. - Evaluate testing and operating procedures.	Integrate prototype hardware and software systems with MIE and MIP simulators.
<u>Aid Mission Operation</u>	- Predict S/C performance. - Generate alarm limits. - Verify sequence of events. - Evaluate command reaction and response to corrective action.	- Employ simulation developed to verify system performance. - Employ analog or digital simulators developed previously, as required.
<u>Aid Mission Analysis</u>	Attempt to simulate actual performance.	- Simulate mission conditions. - Duplicate observed results by variation of performance parameters. Define hardware status and probable sequence of events.
<u>Aid Future Planning and Development</u>	Provide complete, flexible simulation complex.	Employ and modify as appropriate the existing simulation complex.

The progressive development in this concept of Spacecraft System simulation capability is depicted in Figure 5-1. The simulation begins as a simplified mathematical model that includes all mission phases and elements. As designs develop and more complete information is available, the simulation is upgraded until simulation characteristics closely resemble the final design. By

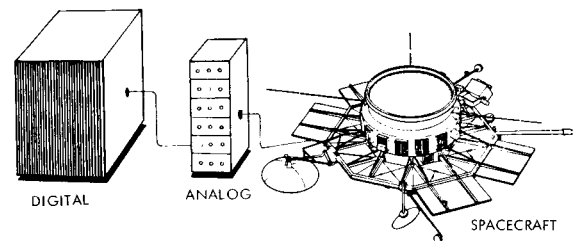
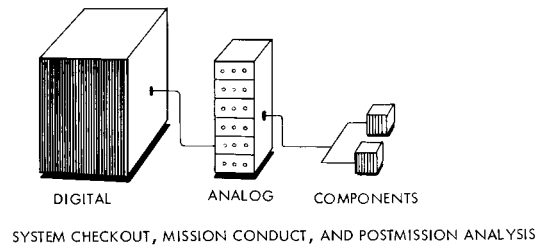
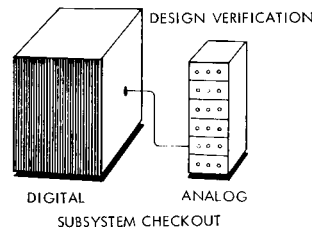


Figure 5-1: Electrical Simulation Development

continually updating and expanding the simulation capability throughout the program, the simulation complex can provide useful data not only for developing mission hardware, but also for planning and evaluating future missions and proposed hardware changes.

STRUCTURAL INTERFACE OF SATURN V AND VOYAGER

Preliminary analyses have been performed to more fully understand the structural interface between the Voyager nose fairing and the Saturn V launch vehicle. Results indicate a potential launch vehicle problem if the present upper-atmosphere wind criteria and control philosophy are retained.

A point-time flight simulation with constant coefficients was performed to obtain inflight vehicle responses and loads for the high-dynamic-pressure region. In the analysis, the length of the cylindrical portion of the Voyager nose fairing was varied, and a constant payload weight of 100,000 pounds uniformly distributed within the nose fairing was maintained. Vehicle simulation included the first free-bending mode. The nominal first-stage boost trajectory for the Saturn V lunar orbital rendezvous mission was used in

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the analysis. A second-order control system with attitude and attitude-rate feedbacks was used. Control gains were chosen for a 0.2-cps pitch frequency and a 0.7 critical damping ratio. Flight-wind criteria were obtained from MTP-AERO-63-8, "Natural Environmental (Climatic) Criteria Guidelines for Use in MSFC Launch Vehicle Development." The current Saturn V upper-atmosphere wind criteria impose a non-directional 95-percent wind profile with a 99-percent wind shear corresponding to the worst month in the year (March) plus an imbedded jet.

At 70 seconds after launch, where the product of dynamic pressure and angle of attack is a maximum $[(q\alpha)_{\text{Max}}]$, ultimate combined loads for several vehicle stations were determined using a safety factor of 1.25 (the requirement for unmanned missions). Loads for one of the critical points — Station 3100 on the S-IVB forward skirt — are shown in Figure 5-2. By current Saturn V ground rules, results indicate an allowable overall payload length of about 42.5 feet. The instrument unit structure is also critical at this length, and other stations become critical as payload length is extended beyond 42.5 feet.

Because the planned Voyager payload will require a length considerably in excess of 42.5 feet, additional analysis is warranted to evaluate measures for extending the allowable payload length. Preliminary analysis indicates that the Saturn V might be capable of accommodating a Voyager payload length of about 82 feet by applying one or both of the following possible measures:

- Relax the environmental criteria to account for seasonal wind-magnitude variations and for wind direction relative to the desired launch azimuths;
- Incorporate load-alleviating control systems or other software changes, such as modified trajectories, in the present Saturn V system.

A thorough load-exceedance probability analysis is needed to define the payload length available from the measures given above. Boeing made extensive analyses of this type for the Dyna-Soar and Minuteman programs, and has available experienced personnel and computer programs. Currently, this capability is being augmented with improved computer programs being developed under a contract with NASA, Houston. Al-

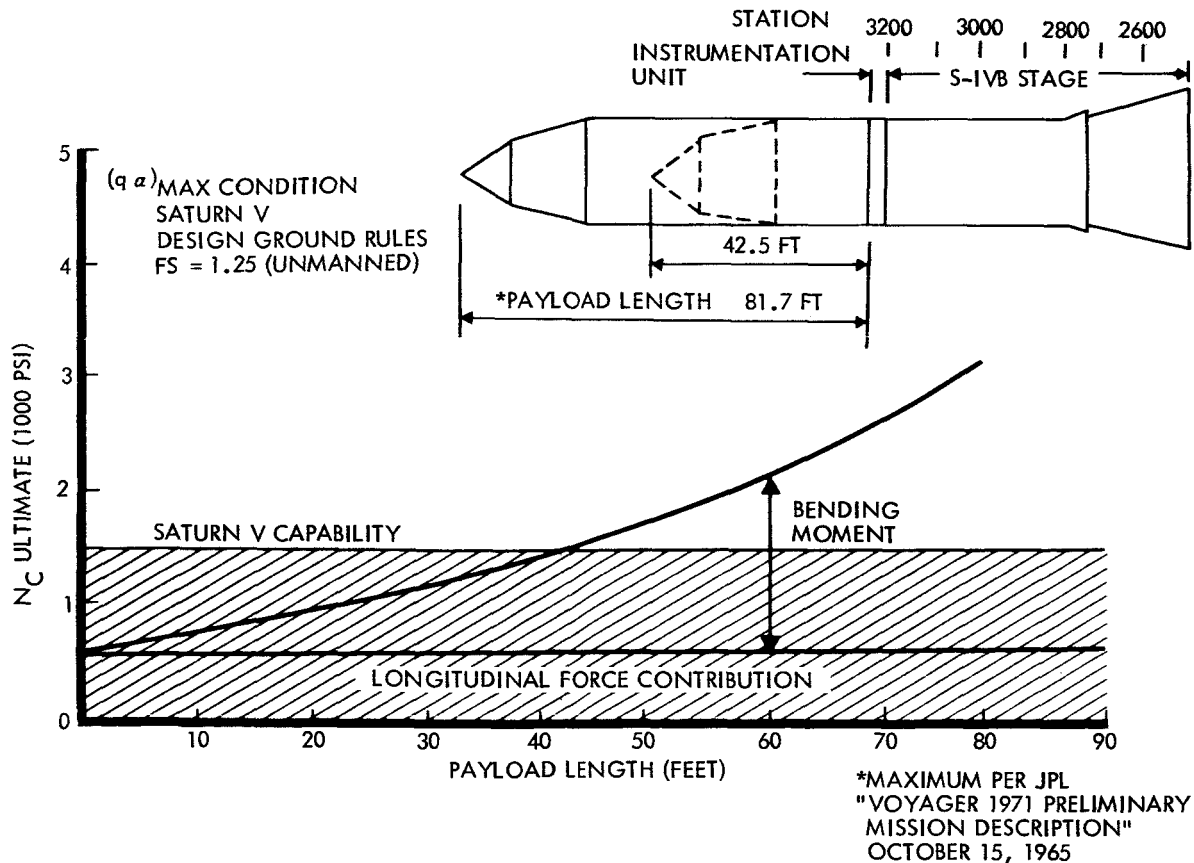


Figure 5-2: Combined Load vs Payload Length

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though these programs for NASA are aimed at Apollo load analysis, they will also be available for Voyager. To cite an example of the detail of one program, the lateral load program includes such factors as nonlinear aerodynamic forces, gust-penetration effects, variation of booster mass with time, and inflight engine malfunction analysis. This program will be operational in February 1966 and will enable completion of a comprehensive booster load analysis in 4 months.

PERFORMANCE OF MISSION OBJECTIVES

Mission objectives are best achieved when the system design emphasis is in balance with the priority of, or value assigned to, each of the subordinate objectives. This suggests the application of optimization processes, two of which are described below. Polyisomax is a general technique that is widely applicable. A second technique attacks the specific problem of allocating available spacecraft resources to best achieve the primary objective of returning significant data from Mars to Earth.

POLYISOMAX

To determine the best mission and hardware design for the Voyager spacecraft, a variety of optimization and boundary-value problems must be solved. Some of these problems can be solved by standard mathematical techniques such as linear programming. However, many problems cannot be solved by standard techniques because they are too complex or lack precise mathematical definition. These problems are usually solved by cross plotting and interpolating the results of a "spray" of calculations or test results. This procedure is both time-consuming and expensive; therefore, a satisfactory rather than an optimum solution is often accepted.

Polyisomax, an abbreviated name for a mathematical technique developed by Boeing, is useful for solving boundary-value problems and optimization problems involving many independent variables. It is reported in "Polynomial Substitution Formulas and Applications for Solving Boundary-Value and Optimization Problems," by Roland Franklin Jagers, presented at the XVIth International Astronautical Congress, Athens, Greece, September 16, 1965. The significant feature of the technique is that it transforms complicated functions into simple polynomials that can be solved by ordinary numerical methods. Each of the parameters represented by the polynomials is assigned a range of values; polyisomax maximizes or minimizes one param-

eter within these ranges. The technique has been successfully used by Boeing to solve launch and space vehicle trajectory and vehicle performance optimization problems, and has demonstrated more than an order of magnitude savings in computer and engineering time.

For illustration, two potential Voyager applications of the polyisomax technique are described below.

Structural Design

In design of the primary and secondary structure, the polyisomax program may be used in conjunction with conventional stress analysis techniques and complex digital programs (such as COSMOS, a Boeing program to determine the elastic characteristics of structures, including stiffness, internal loads, and stress) to obtain minimum-weight structure within the frequency, deflection, stress, and geometric constraints imposed. The program can also be used to determine mounting frequency and spacecraft damping to minimize the spacecraft dynamic loads during boost.

Orbit Selection

The most desirable orbit can be selected using polyisomax by optimizing the total science value function within the constraints imposed by the following orbital relationships:

- Planetary quarantine;
- Long-term perturbation effects on periapsis altitude;
- Value and orbital sensitivity of each science experiment;
- Effect of orbit period on data transmission cycle;
- Effect of launch and arrival date on orbit inclination;
- Location and frequency of ground coverage;
- Capsule-to-spacecraft communication requirements;
- Occultation requirements.

PARTIAL-SUCCESS OPTIMIZATION

A primary objective of the spacecraft system contractor is to optimize the probability of mission success and of partial success in the event of failure. As a result, Boeing has adopted a conservative design approach and has based the design and trade study choices primarily on probability-of-success criteria. This concern

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with success is reflected in the Boeing response to the JPL no-single-catastrophic-failure requirement, and by the selection of highly reliable, space-rated parts and components wherever possible. In addition, a systematic approach has been developed to optimize the probability of partial success. The method or technique discussed in this section defines a technical framework for future partial-success-optimization analyses and trade studies.

A key element in this technique is the establishment of the priority and value of different types of data as a function of mission time. The technique combines the results of reliability, failure mode and failure effect, functional and data priority, and value analyses to assess functional criticality. Criticality indicators, which are mission-time-dependent, define the reliability trades and design improvements to ensure optimum partial mission success. The analysis is accomplished as indicated in Figure 5-3.

Definition of Data Objective

The data objective is determined by combining the following factors:

- Data Quantity—The number of measurements of each type as a function of mission time;
- Data Priority—The desirability of each data type relative to the other types (e.g., the fol-

lowing data priorities might be assigned: capsule science = 50, television = 30, orbital science = 10, spacecraft engineering = 3, capsule engineering = 3, and cruise science = 4);

- Data Value—The relationship between the desirability of a given data type and the number of measurements made (e.g., the hundredth television picture might be considered half as valuable as the first).

When these factors are appropriately combined, the data objective is defined as a function of mission time. To define the factors, Boeing must consult with and obtain agreement from JPL. Figure 5-4 shows the result of this step of the analysis for a mission later than 1971.

Effect of Functional Loss

When a failure causes the spacecraft to lose its capability to perform one or more functions, some part of the data objective becomes unattainable. How important this data loss is to the accomplishment of mission objectives depends on what function is lost and the time of the failure.

To measure the seriousness of a functional failure, the effect of a functional loss is defined as the percentage of the data objective lost. This data loss is inversely related to the time of failure. For example, the results of this type of

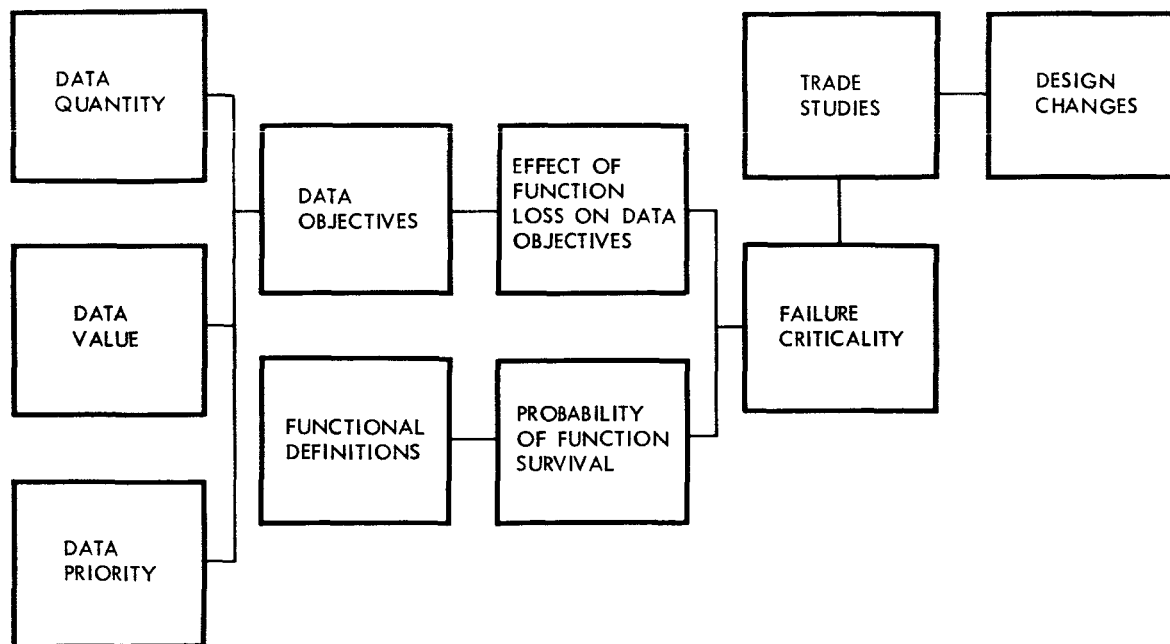


Figure 5-3: Analysis Technique

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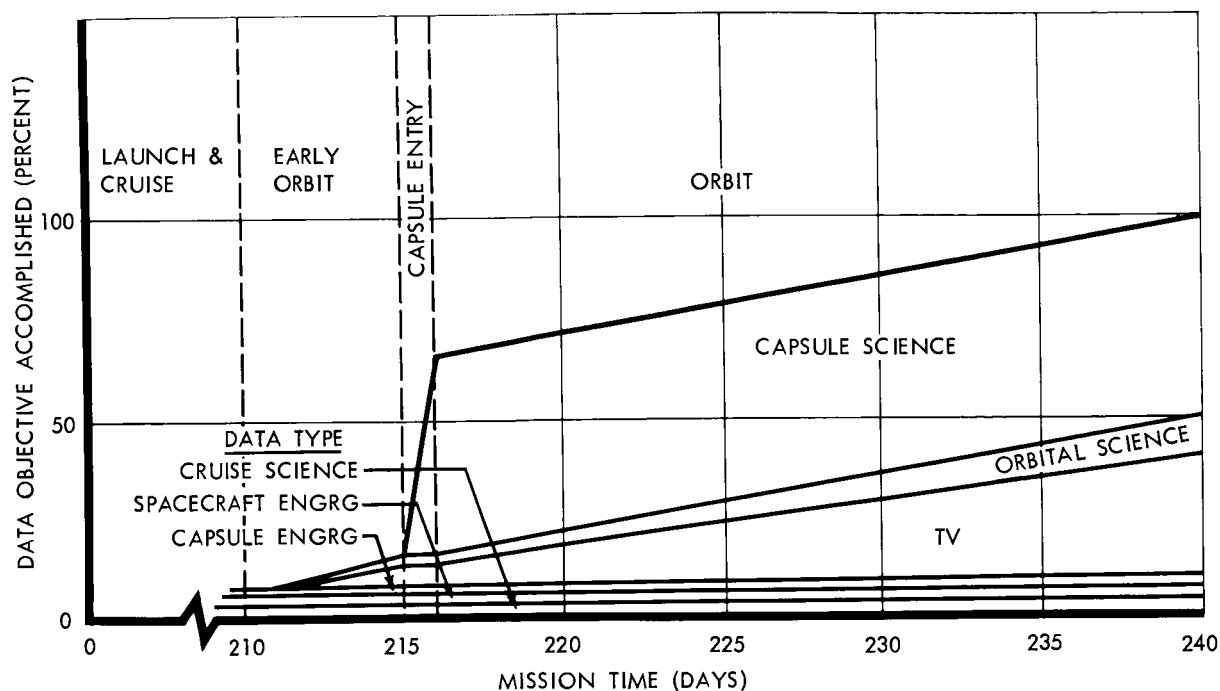
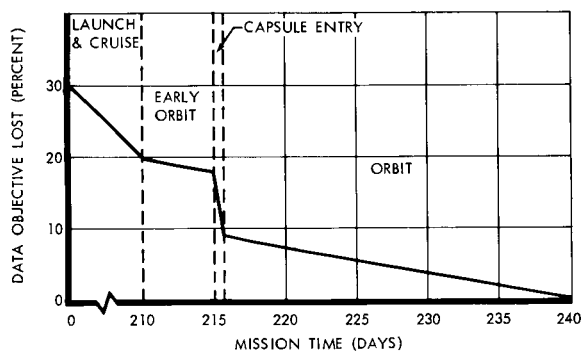


Figure 5-4: Data Objective

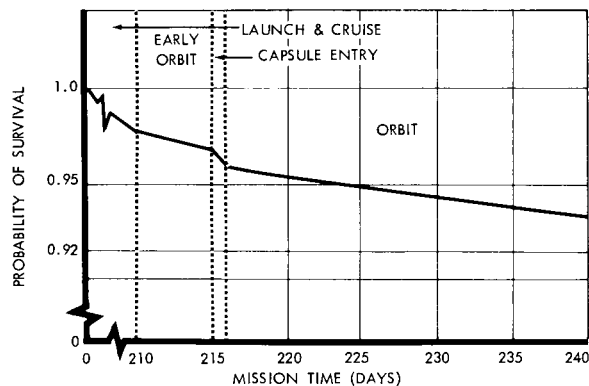
analysis would show that loss of the data encoding function for the gamma-ray detector would be defined to have a small effect on the data objective, whereas a failure of the orientation function for the high-gain antenna would have a large effect. An example failure effect curve for a single function follows.



Effect of Failure — Function X

Probability of Functional Survival

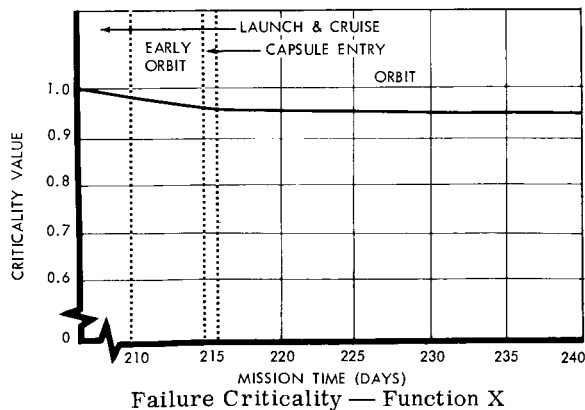
Each spacecraft system function is defined in terms of specific subsystem operations that support the function. Failure modes that would preclude these subsystem operations and their probability of occurrence are defined. By appropriately combining these probability data, the probability of functional survival is determined. The results of this analysis are exemplified in the illustration.



Probability of Survival — Function X

Failure Criticality

Failure criticality is the integrated impact on the data objective of failure effect and probability of

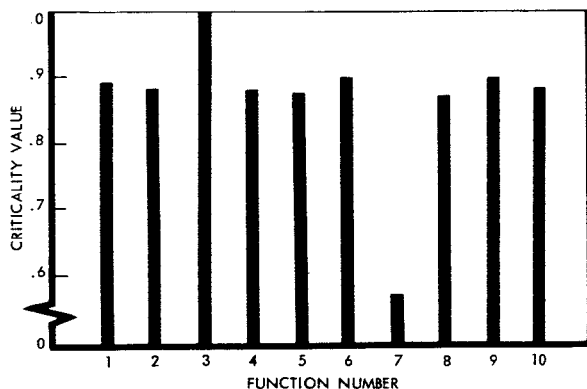


Failure Criticality — Function X

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failure. Criticality is defined as the probability that the function will not be lost, plus the probability that the function will be lost multiplied by the mean percentage of the data objective still attainable in case of failure. This measure of total functional impact on the mission data objective is a function of mission time as illustrated.

The ideal apportionment of unreliability would result in nearly identical failure criticality values for all functions. In this way, the effect of a functional failure upon the mission would be inversely proportional to its probability of occurrence. The illustration is an example criticality comparison for a specific mission time.



Criticality Comparison — Specific Mission Time

In the example, some way should be found to increase the probability of survival of Function 4. If no increase is possible without some sacrifice from one of the other functions, a trade involving survival parameters (e.g., weight or cost) should be considered between Functions 3 and 7. The trade might be accomplished without altering the probability of mission success. If the survival of Function 7 can be enhanced with no degradation to other functions, the probability of mission success would be increased. In either case, the probability of partial success would be increased.

Through this analysis technique, management attention is focused on design areas in which reliability improvement is potentially most fruitful for the accomplishment of mission objectives.

CONTRIBUTION TO SUBSEQUENT MISSIONS

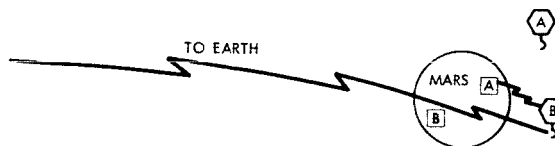
Without compromising the 1971 mission, the preferred spacecraft system was designed so that it will accomplish later missions with minimum changes. Analyses of subsequent mission requirements and objectives that were necessary to achieve this capability resulted in some suggestions for 1971 operations and for possible hardware implementation that might be desirable to

support later missions or improve the chance of their success.

CROSS COMMUNICATION

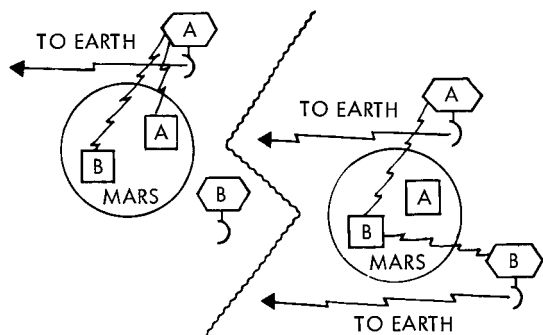
Boeing's desire for successful missions prompts investigations across all interfaces of the Voyager Project. One of the early possibilities for upgrading mission success probability involves the communication link at the capsule and spacecraft interfaces. The provision in each spacecraft of the capability for receiving and retransmitting Mars surface data from both capsules appears to offer a substantial increase in mission success probability and to warrant further study. Mission success is here defined as the return of Mars data from either spacecraft and either capsule. The criterion used for the basic design, for which probability of success numbers are quoted earlier in this document, is defined as the return of Mars data from a single spacecraft and its associated capsule.

With the twin Planetary Vehicle concept in which each vehicle carries a science capsule, 16 possible combinations of spacecraft and capsule failure/no-failure conditions exist. Of these, two offer increased mission success probability if cross-communication capability is available. One of these is the condition in which Spacecraft A and Capsule B operate, Spacecraft B and Capsule A have failed. The second condition is the inverse of the first.



Increased Mission Success Probability

Other combinations offer added data-gathering advantages if cross-communication capability is available, although they do not improve the statistical chances for mission success as defined above. Two of the success-failure combinations would permit reception by one operating spacecraft from both operating capsules. This would mean that data would be available from two different locations on Mars. Two other combinations would involve data reception by two operative spacecraft from a single operative capsule.



Additional Data Gathering Modes

The additional data values from these data-gathering modes warrants further study.

It is proposed that the front end of the spacecraft capsule-data receiver be designed to receive data from either capsule, with the choice commanded from Earth. This capability and the associated increase in probability of returning Mars surface data can be obtained for a small weight and volume penalty. Preliminary calculations indicate that the probability of returning Mars surface data for 1 month in 1975 can be increased by about 15 percent by exploiting this design and operational concept.

With the current Voyager concept, improvement in probability of returning data depends on the year of the mission and will be more useful for later opportunities if capsule life is significantly longer. Further analysis is necessary to determine fully the implications of the concept, particularly in early missions having a shorter capsule life.

1971 MISSION ALTERNATES

The ability to perform subsequent missions can be enhanced by developing a 1971 spacecraft that would readily adapt to subsequent mission requirements. Another aspect, entirely aside from the basic hardware adaptability, is the acquisition in 1971 of basic scientific and engineering knowledge that will enhance the possibility of broadening the scope of subsequent mission planning and development.

The basic difference between the 1971 mission and subsequent Mars missions is the shift in emphasis from orbital and atmospheric investigations to orbital and surface investigations. In subsequent missions, the primary interest will concern capsule operations because it is here that the real search for life begins. An important area wherein the 1971 mission could improve the probability of

mission success for subsequent missions is in reconnoitering potential landing sites.

Plans call for placing the 1971 spacecraft in an orbit with periapsis in the southern hemisphere so that maximum contrast is obtained in the wave of darkening. Arrival dates in 1973 coincide with a shift in the area of interest to the northern hemisphere. Typical 1973 orbits have inclinations of 35 to 60 degrees with periapsis located between 20 and 50 degrees north latitude. Points of interest within this latitude range are not readily observable from the nominal 1971 orbit if periapsis is located near the evening terminator, as anticipated; this is true because darkness and extreme altitudes hinder visual observation. Two mission procedures for the 1971 flight might be considered to improve observations of potential 1973 landing sites in the northern hemisphere. They are:

- Insertion of the second 1971 Planetary Vehicle into an orbit providing coverage of the desired northern latitudes (assuming the success of the first 1971 Planetary Vehicle);
- Provision for additional in-orbit maneuver capability whereby the nominal 1971 orbit can be changed to cover the desired areas.

The first approach would not be desirable if it significantly reduces the probability of achieving the basic 1971 mission objectives.

Relative to in-orbit maneuvering, changes in the line of apsides and orbit inclination might enhance coverage of the areas desired, depending on maneuver capability. With the 1971 periapsis point located near the evening terminator, changes in orbit inclination appear to be the best means of in-orbit maneuvering for obtaining coverage in the northern hemisphere. This would require a large change in the line of apsides and a proportionate impulse capability to obtain satisfactory northern-latitude coverage on the sunlight side of Mars. An initial orbit might have the following characteristics:

Apoapsis altitude	20,000 kilometers
Periapsis altitude	1,000 kilometers
Inclination	40 degrees
Argument of periapsis	310 degrees

If these characteristics are assumed, a ΔV of 1130 meters per second will result in a 35-degree-inclined orbit providing northern-latitude coverage. This correlates with the orbit change that might be desired to observe a potential landing site for the 1973 mission. Preliminary analysis indicates that use of excess Saturn V payload

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capability could provide each Flight Spacecraft with an additional maneuver ΔV of 850 to 1090 meters per second. With the Planetary Vehicle and adapter weight allocation at 42,000 pounds and with an assumed 4000 pounds of nose fairing injected into heliocentric orbit, 8500 pounds can be made available on each Planetary Vehicle; about half will be available for the orbital propulsion itself, the balance needed for increased attitude control, midcourse, and orbit insertion propellant and associated structural changes to accommodate the increased Planetary Vehicle weight.

ADDITIONAL 1971 MISSION CAPABILITY

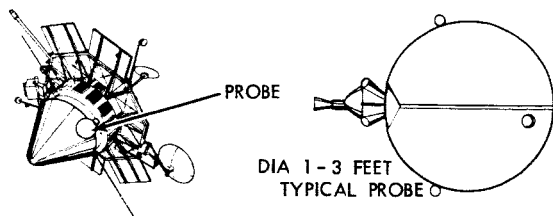
Because the Voyager Program has been reoriented to one of solar system exploration and redirected to the Saturn V booster, it is pertinent to review the 1971 Mars mission from the standpoint of its contributions to achieving overall Voyager goals. Of particular interest are additional capabilities that, if incorporated, would enhance the probability of mission success or allow the accomplishment of additional objectives for the 1971 mission. Several options are available. These include:

- Use of precursor entry probes;
- The incorporation of additional atmospheric probes;
- The provisions for orbital maneuvering capability.

Although these additional capabilities are reasonable from a technical point of view, further study is necessary to determine the effects of their possible incorporation on the success of the basic mission.

PRECURSOR ENTRY PROBE

Probes could be launched from the spacecraft into the Martian atmosphere before the Flight Capsule is deployed; the probes would perform limited atmospheric property measurements. Measurements would be transmitted to Earth directly or via spacecraft relay and analyzed to determine the optimum entry profile for the Flight Capsule. This would increase the probability of achieving a successful Flight Capsule



Precursor Probes

entry by providing data for use in selecting a Flight Capsule trajectory before the capsule is deployed. One or more precursor probes could be installed on each Planetary Vehicle, as illustrated.

The probe shown is typical of one that could be launched before orbit insertion. Measurement techniques include three-axis accelerometers for density and scale height and a mass spectrometer for atmospheric composition. Probe weights could range from about 30 to 160 pounds. Probes could be launched before or after insertion of the Planetary Vehicle into Mars orbit.

ADDITIONAL ATMOSPHERIC PROBES

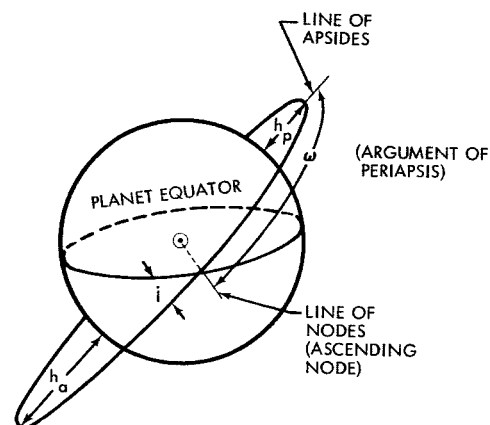
Multiple small probes might be installed on each Planetary Vehicle to evaluate alternate measurement techniques and provide additional data points for statistical verification of critical variables. Such probes might increase the probability of accomplishing mission objectives by providing a degree of backup to the Flight Capsule. Such probes would be similar in design and installation to the precursor probes.

ORBITAL MANEUVER PROPULSION

Orbit maneuvering capability (over the nominal orbit trim) can have a significant effect on achieving basic Mars mission objectives. This capability can be used effectively in two ways:

- Extend the range of high-resolution measurements and observations over a greater area of the planet;
- Change the orbit based on initial observations. This is particularly significant in light of the limited knowledge about the surface of Mars on which orbit selection must be based.

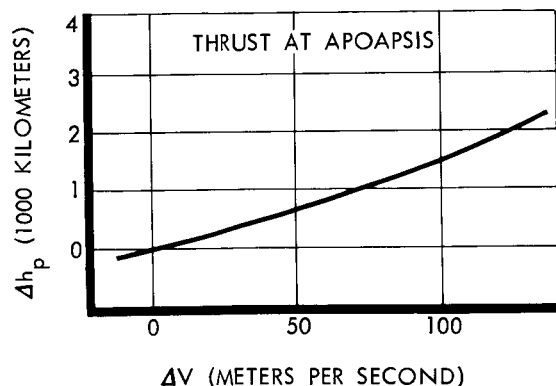
Preliminary analyses have been conducted to determine what impulsive velocity increment will



Typical Orbit Geometry

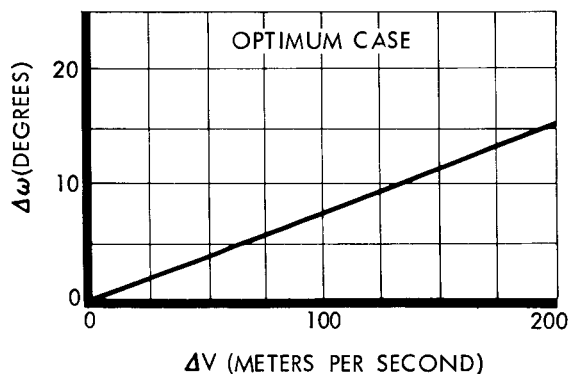
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change various orbital elements. The elements considered were periapsis altitude (h_p), line of apsides, and angle of inclination (i). For this analysis, an example initial orbit with a periapsis altitude of 1000 kilometers and an apoapsis altitude of 20,000 kilometers was chosen. This corresponds to the nominal 1971 orbit. Results of this analysis are given below.



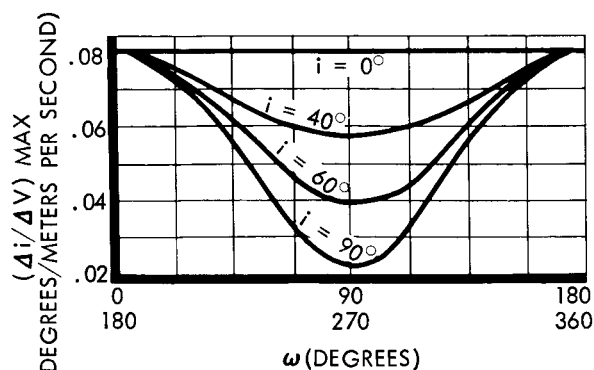
Periapsis Altitude Change

Significant variation in periapsis altitude is available with only modest ΔV when thrust is applied at apoapsis.



Line of Apsides Change

For the representative 1971 orbit, apsidal rotation is a relatively expensive maneuver.



Maximum Inclination Sensitivities

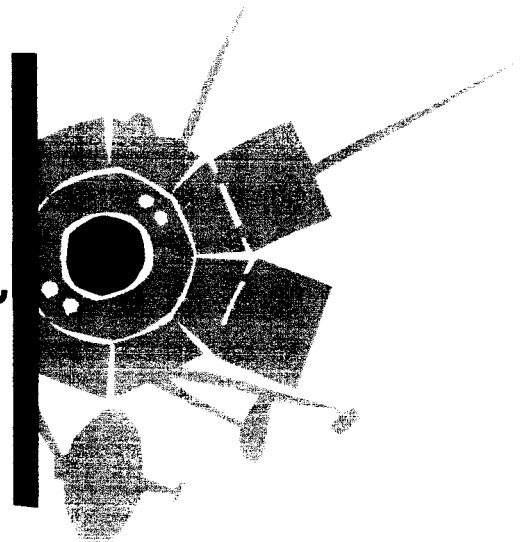
Significant variation of orbit inclination can be achieved with moderate ΔV from orbits with initial inclinations anticipated for the 1971 mission.

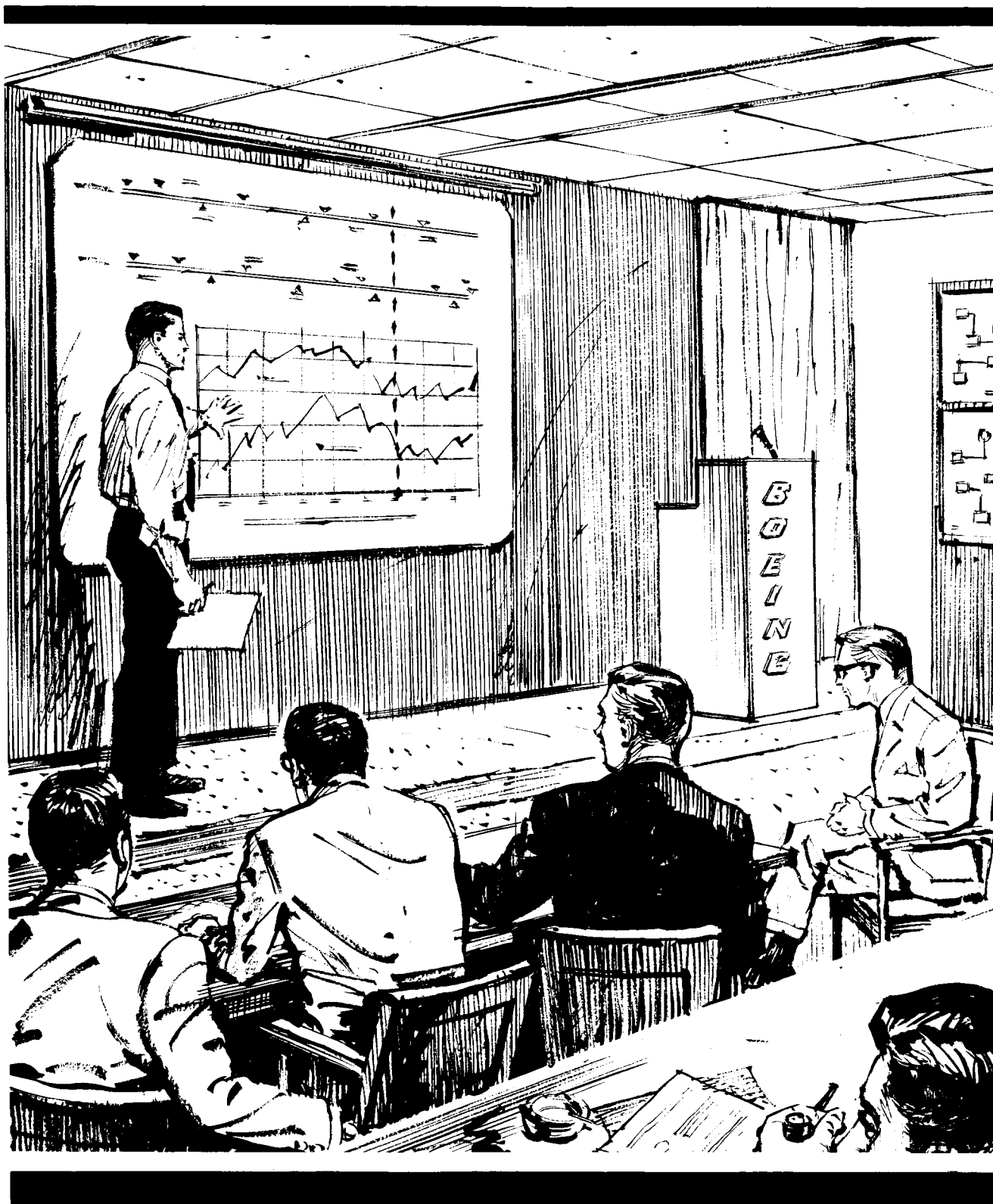
Provision for even a modest maneuver capability would provide considerable latitude for orbit adjustment. For example, an additional ΔV capability of only 200 meters per second would permit:

- Alteration of orbit periapsis altitude by lowering it as much as feasible within contamination constraints or raising it as much as 3500 kilometers;
- Rotation of the line of apsides as much as 15 degrees;
- Change in orbit inclination of as much as 11.5 degrees for an orbit initially inclined 40 degrees with an argument of periapsis of 90 degrees (for the worst case).

Orbital maneuvering capability could be provided by extending the orbit trim capability beyond the required value as shown in the preceding section or by using the excess liquid propellant available if low VHP trajectories are employed.

VOYAGER





FACILITIES

For the past several years, The Boeing Company has budgeted an average of \$24 million annually for new capital improvements. At the present time, over \$43 million of company capital money is being invested in the new, 430-acre Boeing Space Center complex located at Kent, Washington. An additional \$12.6 million of specialized, unique Voyager facilities are planned for provisioning at the Boeing Space Center to support Voyager Spacecraft System requirements. The current laboratory complex, plus the additional laboratory and office facilities being constructed at this facility, will make this one of the most advanced space research centers in the industry.

A space environment and flight simulation laboratory is one of the key, existing facilities at the center that is committed to the Voyager Spacecraft Program. This \$8-million facility contains a large vacuum chamber capable of containing the complete, assembled spacecraft. Eleven small to medium-size vacuum test chambers are also located in this facility. The large chamber has an interior work zone 29 feet in diameter and 42 feet high. Vacuum capability of the large chamber is 1×10^{-9} torr; space temperatures can be simulated through a computer-controlled radiant heat source that will be supplemented by a new \$3.6-million solar simulator planned for the Voyager Spacecraft Program. The \$1.6-million first phase of the solar simulator is now under construction.

The flight simulation laboratory contains an unmanned satellite attitude control simulator incorporating an air-bearing-supported table that simulates the angular motion freedom of a spacecraft. Bottled-gas reaction jets enable the on-board attitude-control subsystems to be tested through complete signal and response control cycles. The air bearing permits the platform 360 degrees of freedom about one axis and 90 degrees about the other two axes. This laboratory also incorporates a unique instrumentation system to accurately simulate inflight characteristics and response to ground station control. A computer-controlled flight simulator supports technical studies of docking, landing, and maneuvering.

A \$3.3-million microelectronics laboratory is also located at the Boeing Space Center. This

environmentally controlled, ultraclean laboratory supports electronic technology experiments and studies. The facility is divided into three distinct work areas supporting:

- Semiconductor wafer preparation;
- Thin-film and semiconductor device preparation;
- Circuit and advanced packaging development research relating to circuitry and component packaging.

A \$4.5-million space technology materials and processes laboratory supports the development and testing of spacecraft materials and processes. Typical tests determine long-term characteristics of bearings and lubricant materials in extreme vacuum with severe temperature variations. Temperatures in the test chambers are varied from -270 to 1300°F and pressures are reduced to 1×10^{-10} torr to obtain friction and wear-rate data on materials. Other special facilities applicable to Voyager include laboratories in which to conduct thermal coating tests, high-temperature and ablation material tests, solar absorptance measurements, and formulation of special finishes.

Additional facilities at the Boeing Space Center that are either under construction or are approved for construction total \$26.8 million. These facilities consist of an expansion to the space environment laboratory, new laboratories for space technology research and engineering development, support shops, and general-purpose office buildings.

A 45,000-square-foot, \$2.3-million addition to the space environment laboratory is being constructed to house \$3.5 million worth of vibration and structural test equipment that is being relocated from other facilities. These facilities will have direct application to Voyager development test requirements.

Some \$4.6 million is programmed to construct a 120,000-square-foot space technology laboratory. This building will house \$10 million of laboratory equipment that is being transferred from buildings in the Seattle area. The engineering development shop will be a 60,000-square-foot building

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costing \$3.3 million and containing \$1.2 million worth of developmental shop equipment.

Four office buildings costing \$13.2 million are under construction and will accommodate over 5000 Space Division and other Aerospace Group personnel.

All additional facilities described above will be completed and available for use before the start of Voyager Phase II.

Planned future facility additions at the Boeing Space Center include a 100,000-square-foot Voyager assembly and test building incorporating 20,000 square feet of Class 100,000 (Federal Standard 209) laminar-down-flow clean area. This facility will be uniquely designed and provisioned for Voyager Phase II assembly and test activity, and will cost \$7 million including equipment installations.

A 250,000-square-foot general-purpose fabrication building is also planned. This building will contain new and relocated fabrication equipment to support program activity.

In addition to these facilities at the Space Center, extensive capability also exists at other Aerospace Group and Space Division locations in the

Seattle area. Four million square feet are assigned for related aerospace effort. This area contains a variety of modern laboratories, computer facilities, production areas, and office and support facilities that will directly support the Voyager Spacecraft System program. Development, qualification, and acceptance testing of spacecraft systems requiring significant quantities of fuel, propellant, ordnance devices, or other hazardous materials can be accommodated at the Tulalip test site. This complex is located on a 2100-acre tract about 45 miles north of Seattle.

Boeing principal subcontractors have also made major capital investments in facilities applicable to Voyager Spacecraft System requirements.

Autonetics Division of North American Aviation, located at Anaheim, California, has facilities covering the entire electronic spectrum in a single geographic area. These facilities represent an investment of \$65.4 million and are unequaled in diverse capabilities and available resources. Buildings have been designed and constructed within the past 3 years, with layout and equipment reflecting more than 15 years of experience on major electronic programs. Equipment is the most advanced available; highly specialized equipment not available on the com-



Autonetics Division of North American Aviation

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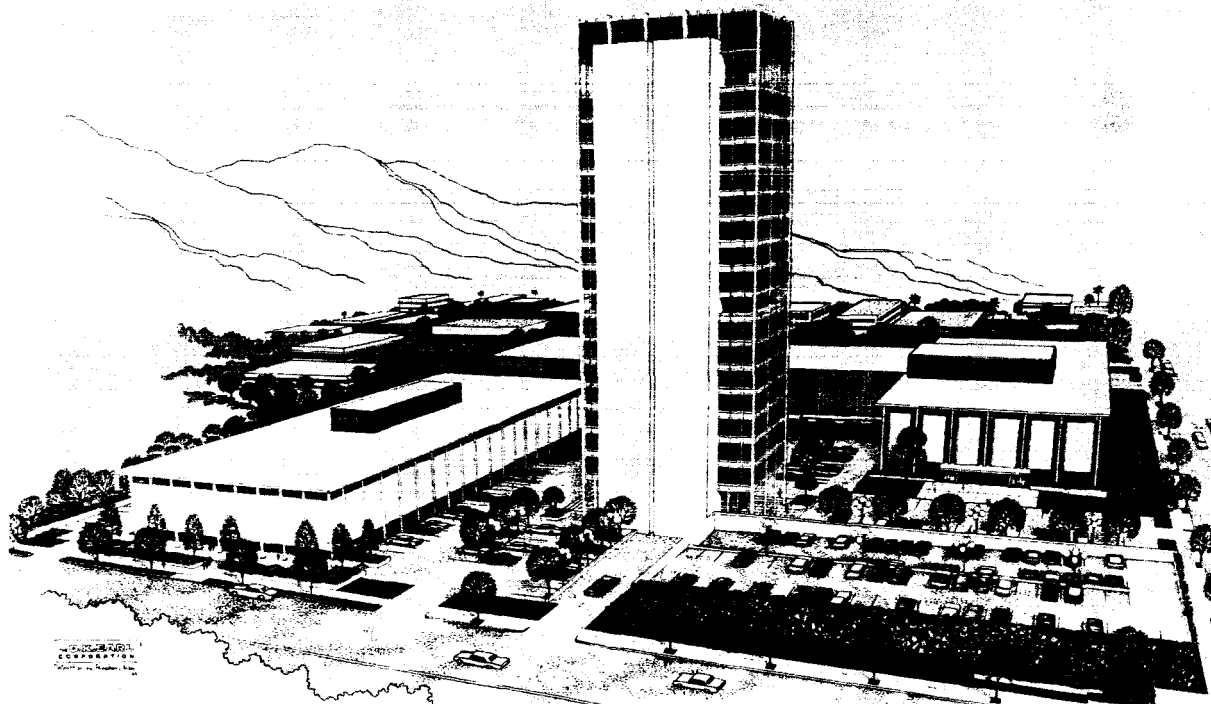
mercial market has been developed by Autonetics. The modern complex of buildings, all of which are owned or privately leased, covers more than 3 million square feet. Virtually every technological and scientific discipline can be found in the specialized areas, from research laboratories to production and test capability for present and future programs. Of the Autonetics plant complex, approximately 300,000 square feet are devoted to laboratories, 1 million square feet to equipment fabrication, with the remainder for offices and general-purpose support areas.

Electro-Optical Systems, Inc., located at Pasadena, California, is completing a facilities expansion program that provides a consolidated capability for Voyager management, engineering, fabrication, assembly, and test. Most of the facilities for Voyager will be centered in the research and development building, flight sciences building, and the environmental engineering building, representing a \$9.4 million capital investment.

The research and development building houses corporate offices and most of the technical functions. This facility includes engineering laboratories, fabrication rooms, hardware performance test facilities, and environmental test facilities used in performing such space-hardware-oriented projects as Ranger and Mariner power sub-systems.

The 110,000-square-foot flight sciences building houses operations in plasma physics, gas dynamics, plasma devices, physical metallurgy, metals fabrication, surface physics, mechanical and electrical technology, fluid physics, and program management and systems engineering. The building will have 16,000 square feet of load-bearing roof for outside environmental testing of antennas and spacecraft solar panels; 23,000 square feet of the third floor will be used for clean-room laboratories. The 60,000-square-foot environmental engineering building will provide 20,000 square feet (35 feet high) of unobstructed underhook assembly bay area for aerospace environmental testing. The remainder of the building will house laboratories containing such equipment as vibration tables, a shock test stand, centrifuge, and vacuum chambers. The new \$1.2-million space-simulator test facility will be cylindrical in shape, 21 feet in diameter, 25 feet in height, and capable of pressures as low as 1×10^{-6} torr.

Philco Western Development Laboratories are located on a 40-acre tract in Palo Alto, California. The current occupied space of 636,000 square feet is being expanded to 650,000 square feet by an addition to the space vehicle assembly area. These leased buildings represent an investment of approximately \$19 million and contain an additional \$6.1 million of Philco capital-assets equipment.



Electro-Optical Systems, Inc.

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Most of Philco's Voyager activities will be consolidated in an integrated facility for management, engineering, fabrication, and test. Laboratories for detailed circuitry and design work are equipped with an experimental shop, welding machines, equipment for handling printed circuit cards, encapsulation facilities, and telecommunication test equipment adequate for all Voyager requirements. Development test facilities are equipped with special and general-purpose electronic test equipment for designing and developing solid-state circuits, analog-to-digital converters, differential amplifiers, magnetic and diode-transistor storage, and logic timing systems. Radio-frequency circuits can be designed and measured from d.c. through X-band. Additional clean rooms are planned for fabricating Spacecraft System communication equipment.

These rooms will be provided by expansion of existing laminar-flow facilities. Equipment in these rooms will include in-line environmental test machines for vibration, thermal vacuum, and magnetic balance testing. A remote, magnetic mapping facility is also available for Voyager. The ambient field is controlled by a 20-foot-diameter, three-axis Helmholtz coils system, which will provide the necessary magnetically controlled field.

Boeing and subcontractor team facilities described above, augmented by the unique government-owned facilities for launch and mission operations, will provide overall facilities capability to successfully accomplish the Voyager Program.



Philco Western Development Laboratories