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PHASE 1A STUDY REPORT

VOYAGER SPACECRAFT

VOLUME 1
SUMMARY

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TRW SYSTEMS GROUP

Redondo Beach, California

PREFACE

In preparing this report we have adhered to the outline provided by JPL in the contract statement of work. Although the subject matter is presented in the order defined by that outline, because of the relative size of the various sections it has not been convenient to retain a one-to-one relationship between the five major topics of that outline and the individual volumes of this report. After this summary volume, which is independent of the JPL format, the relationship between the volumes of the report and the JPL outline is as follows:

	JPL Format	Report
(A)	Presentation of the preferred design for the flight spacecraft and hardware subsystems	Volumes 2 and 3
	I. Mission objectives and design criteria	Volume 2, Section I
	II. Design characteristics and restraints	Volume 2, Section II
	III. Systems level functional description of spacecraft and its relationships and interfaces, following "100 series" functional specifications	Volume 2, Section III
	IV. Functional descriptions for the individual hardware subsystems	Volume 2, Section IV
	V. Schedule and related Voyager implementation plan for the recommended design	Volume 3
(B)	Presentation of alternate designs considered	Volumes 4 and 5, including appendix volumes
	I. Alternate mission objectives and design criteria	Volume 4, Section I
	II. Various design characteristics and restraints	Volume 4, Section II

	III. Alternate system philosophies and system mechanizations	Volume 4, Section III
	IV. Various subsystem mechanizations considered	Volume 5, including appendix volumes
	V. Effects and implications on the schedule and implementation plan	Volume 5
(C)	Design for the operational support equipment	Volume 6
(D)	Design for the 1969 test spacecraft	Volume 7, Sections I-VI
(E)	Design for the operational support equipment for the 1969 test flight	Volume 7, Section VII

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I. INTRODUCTION

The results of the Phase IA Voyager spacecraft study performed for JPL by TRW Systems Group (formerly TRW Space Technology Laboratories) and its two major subcontractors, Douglas Aircraft Company and the Radio Corporation of America, are presented in this technical report.

Here in the summary volume we have brought together the major points of the report to permit a relatively quick review of our results. We do not attempt here to justify these results or show completely the prior steps that have led to these conclusions; this is done in the other volumes. But we have attempted to make sufficient reference to the detailed discussion in these other volumes to allow the reader of the summary to turn readily to the relevant supporting material. After a discussion of our approach in arriving at an optimized spacecraft design, this summary volume reviews the individual tradeoffs and analyses completed during the process of that optimization, including the operational support equipment (OSE) as well as the spacecraft. Next we summarize our conclusions concerning the benefits of a 1969 test flight and the design for that flight. Finally we include the major points developed in the course of drawing up our implementation plan, a single plan which utilizes the 1969 test flight as a means of providing additional confidence in achieving a successful 1971 mission.

Within the framework of the JPL specifications and guidelines this study has led to the spacecraft design illustrated in Figure 1. This design, which is summarized in Section IV of this volume, is presented in specific detail in Volume 2. The detailed tradeoff studies leading to this design are discussed in Volumes 4 and 5. We believe it meets in good measure the intent of the JPL Preliminary Voyager 1971 Mission Specification. Design conservatism, simplicity of approach, and careful application of alternate operating modes and redundancy are key characteristics of the design approach. Specific spacecraft features include:

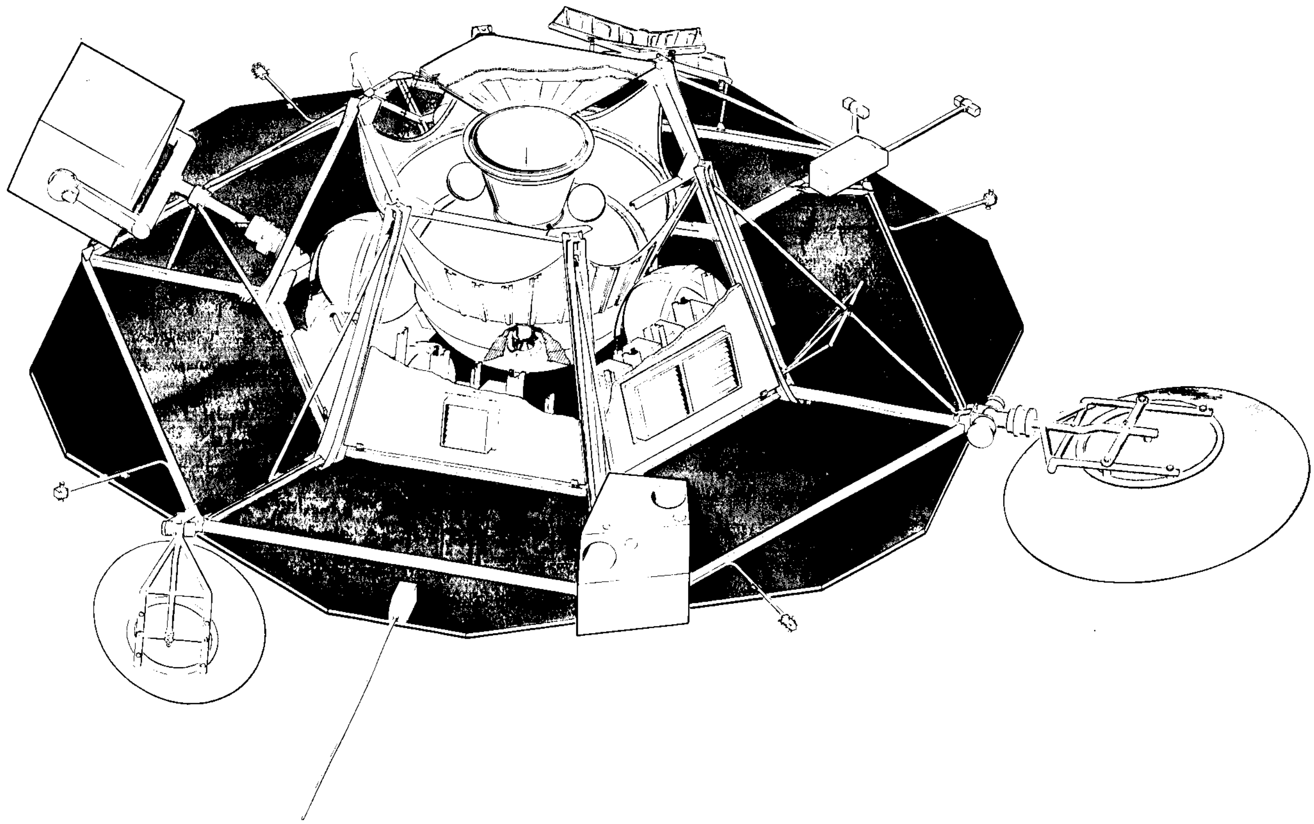


Figure 1. 1971 Voyager Spacecraft

- Straightforward, three-part structure with six-point attachment to the capsule and to the Centaur inter-stage
- Hinged equipment mounting panels which also serve as shear structure members
- Fixed solar array panels
- A double-gimballed 6-foot antenna, a functionally redundant single-gimballed 3-foot antenna, and a low-gain, broad-coverage antenna
- Fixed VHF antenna for receiving capsule telemetry
- Balanced double-gimballed planet-oriented package (POP)
- Fixed science payload package

- Temperature control by Mariner-type louvers on the equipment-mounting panels
- Removable solid-propellant retropropulsion engine with liquid injection thrust vector control
- Removable blow-down monopropellant midcourse propulsion system
- Cold gas attitude control system with nozzle heating available
- Cold gas blow-down propulsion for spacecraft evasion to facilitate capsule boost sequence
- Standardized equipment packaging and mounting.

Sufficient redundancy and alternate modes of operation are included to achieve a cumulative mission reliability of 0.817 for successful operation after 1 month in orbit about Mars and 0.708 after 6 months in orbit. The spacecraft weight is estimated as 320 pounds below the 2000 pounds allotted. Of this 133 pounds is assigned as contingency, and 187 pounds is available as margin to improve mission performance.

II. THE DESIGN STUDY

The initial period of this study was devoted to a careful, quantitative evaluation of the design constraints imposed by the mission objectives, other project elements, and the limitations of technology, especially in view of the general requirement that only proven systems and techniques should be employed. The desire for design conservatism, as expressed at the contractor's meeting on 21 May, played an important part in this evaluation.

A large number of initial system and subsystem concepts were then formulated which to varying degrees appear capable of fulfilling these design constraints. Through a series of rapid design iterations and comparative evaluations, the majority of these competing ideas were rejected as not adequately meeting the design constraints. Three distinct classes of spacecraft configurations emerged:

1. Configuration A, a sun-Canopus oriented spacecraft with a two-gimbal communications antenna, a monopropellant midcourse propulsion subsystem, and a solid-propellant Mars orbit injection rocket.

2. Configuration B, a sun-Canopus oriented spacecraft with a two-gimbal communications antenna and a liquid bipropellant engine for both midcourse and Mars orbit injection.

3. Configuration C, an earth-Canopus oriented spacecraft with a body-fixed communications antenna and the monopropellant and solid propulsion arrangement of Configuration A.

Table 1 compares the key characteristics of these three basic configurations, and Figures 2 through 4 illustrate their major features.

As can be seen the three configurations differ principally in the areas of communications and retropropulsion. In reviewing the key mission objectives of adequate space and weight for scientific experiments and adequate communications capacity to return the science data

Table 1. Key Differences of Basic Configurations

	A	B	C
Attitude Control	Sun-Canopus oriented	Sun-Canopus oriented	Earth-Canopus oriented
Power	Solar array, body-fixed	Solar array, body-fixed	Solar array, deployed paddles with fixed earth orientation; RTG's studies as an option
Communication Antenna	6-foot double-gimballed dish	6-foot double-gimballed dish	Body-fixed 16-foot dish
Propulsion	Monopropellant for midcourse, solid for orbit injection	Bipropellant for both midcourse and orbit injection	Monopropellant for midcourse, solid for orbit injection. Bipropellant briefly studied as an alternative

to earth, it became clear that communications capacity was a limiting design parameter. It was calculated that 20 watts of transmitted power and a spacecraft antenna six feet in diameter will permit a data rate of about 4000 bits/sec to earth from Mars orbit until approximately one month after 1971 encounter. Configurations A and B are based on this implementation for the nominal mode of operation. To augment data rate further, a 40-watt transmitter was considered. However, this was felt to involve additional development risk since the design would have to wait on qualification of a 40-watt S-band power amplifier. Attention was also directed toward the possibility of increasing the size and gain of the spacecraft antenna. The antenna for Configurations A and B (an elliptical 5.5 by 6.5 feet) was the largest rigid, articulated, antenna which could be fitted within the prescribed spacecraft envelope and which could be deployed without intricate mechanization. Review of unfurlable antennas indicated inadequate test experience to establish their reliability and hence dictated against their use in a conservative design approach.

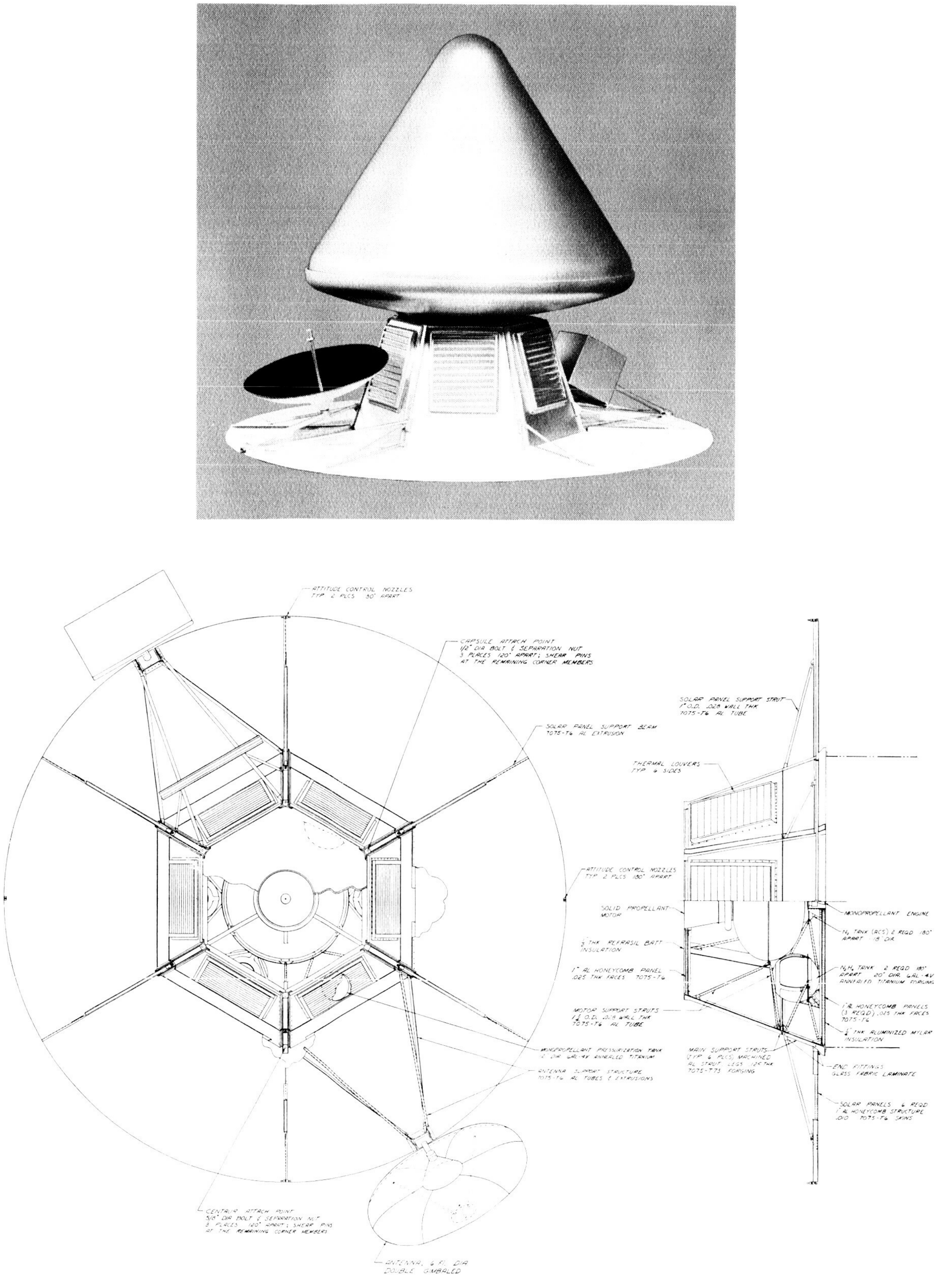
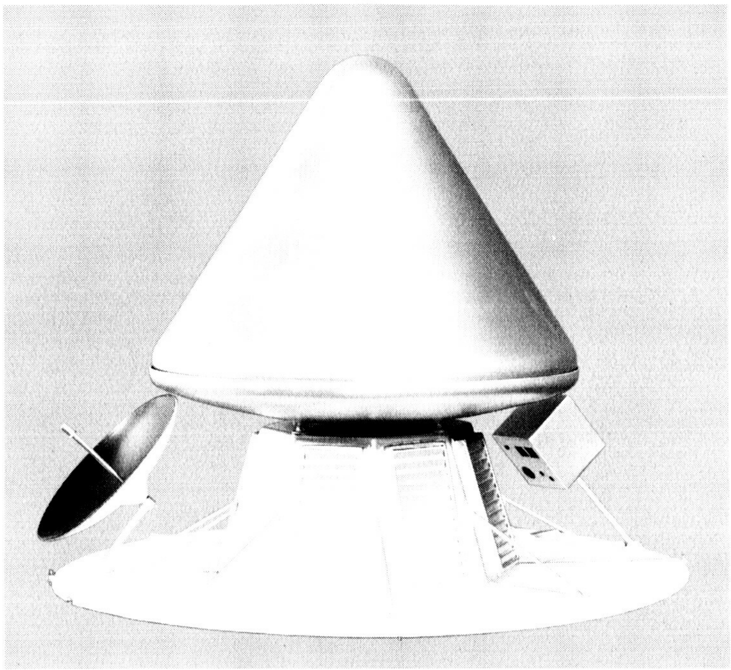


Figure 2. Configuration A



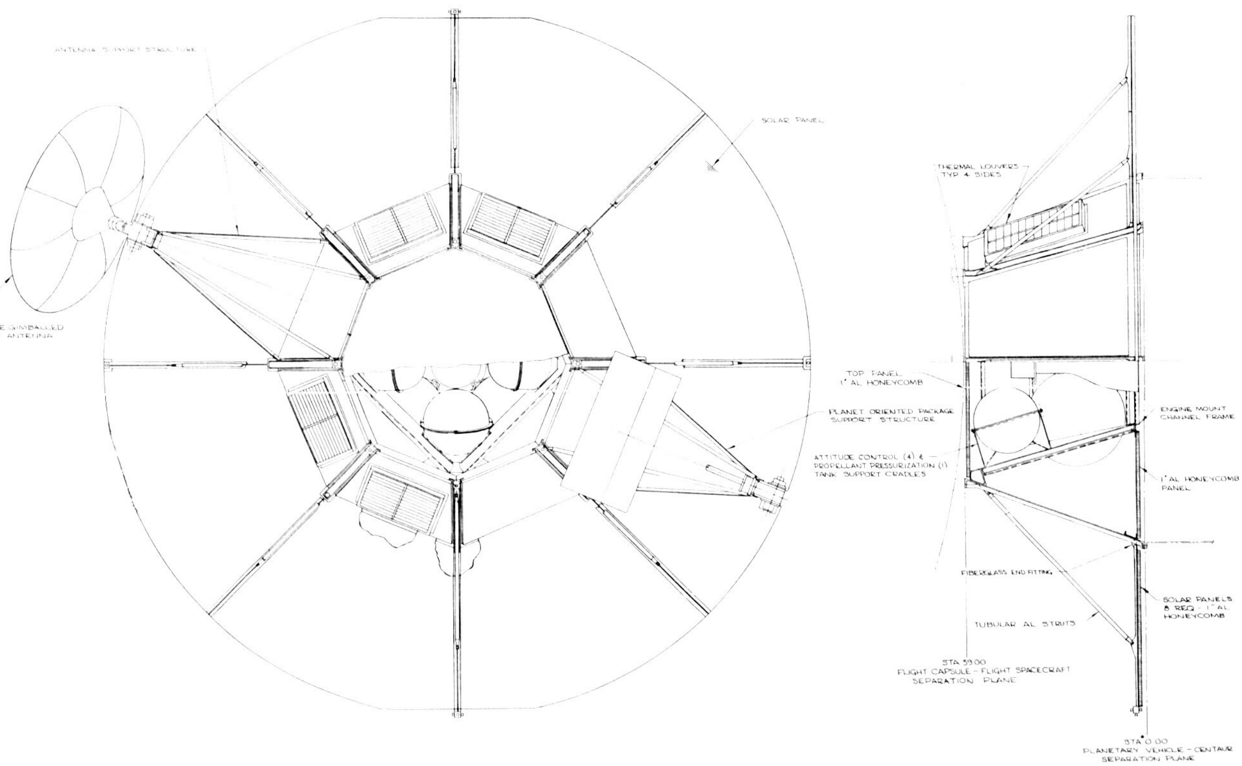
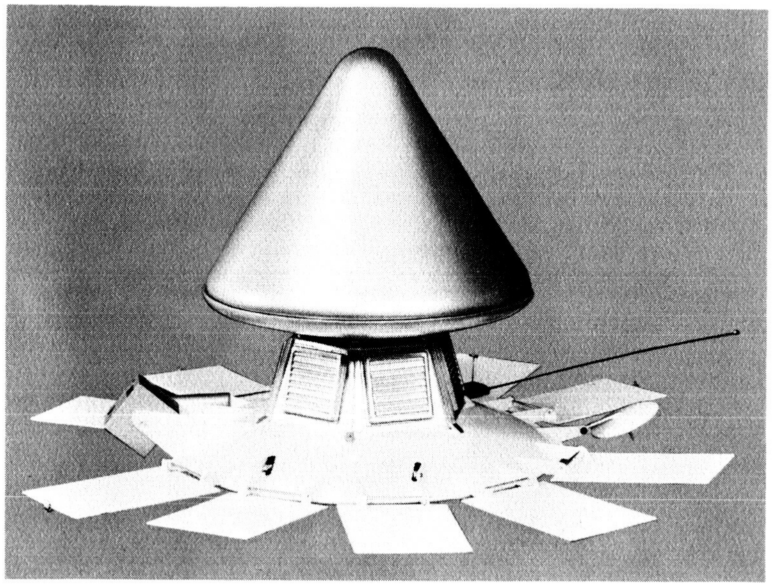


Figure 3. Configuration B



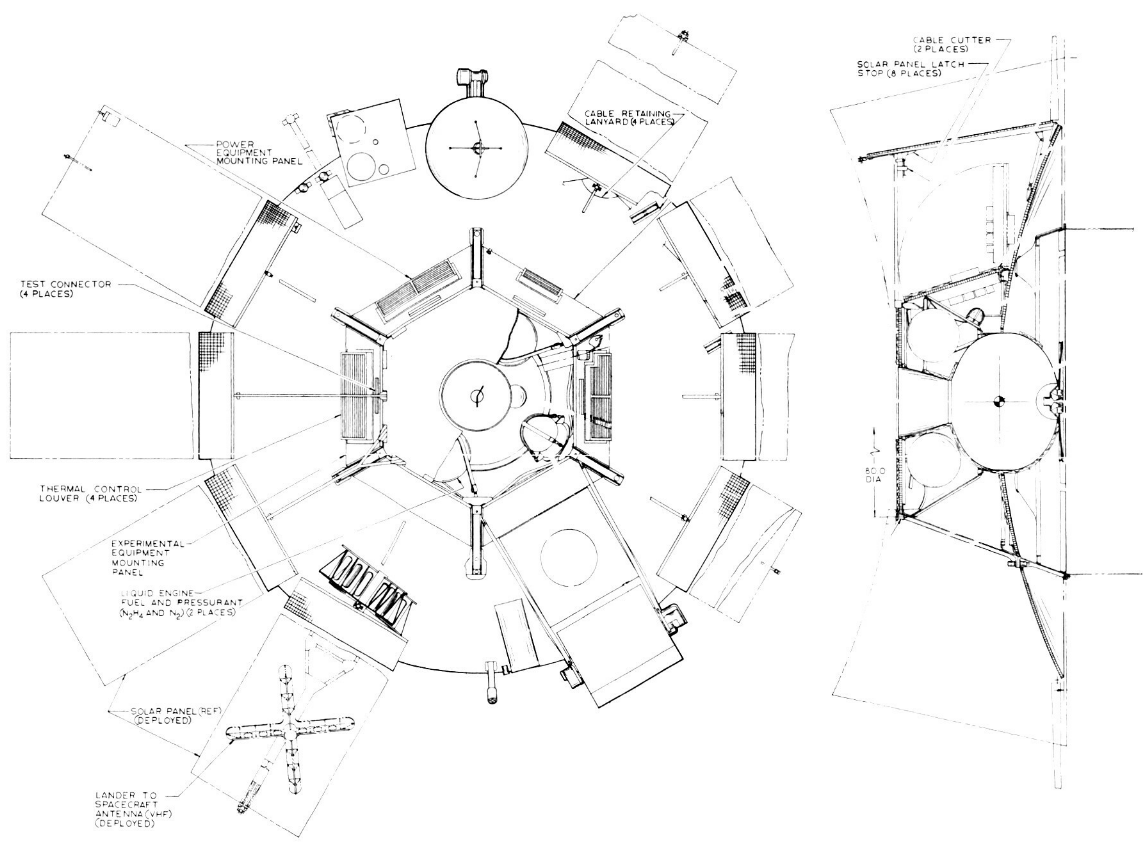


Figure 4. Configuration C

The search for a feasible larger antenna led to the body-fixed antenna of Configuration C shown in Figure 4. It was apparent that utilizing a body-fixed antenna necessitated a departure from the JPL specification that the attitude reference frame be based on sun-canopus orientation; for Configuration C an earth-Canopus reference was a more logical choice. Despite the departure, however, the potential gain in communication capability appeared to warrant continued study of the concept on the grounds that, should the benefits of the design prove finally to outweigh its negative aspects, it could prove advantageous to deviate from the sun-Canopus guideline As the analysis in Volume 4 shows, this configuration leads at most to a 10 per cent reduction in power supply margin if the end-of-life design date is taken as May 1972 (four to six months in Mars orbit), and the penalty essentially disappears if end-of-life is taken as August 1972 (seven to nine months in Mars orbit).

Although three basic configurations evolved during the first few weeks of the study, considerable flexibility in viewing of alternatives still existed in each of the subsystem areas, as illustrated in Table 2. For the most part, the specific design alternatives considered within each of the subsystems were relatively independent of the choice of configuration. Obvious exceptions occurred in the propulsion area and to a lesser extent the attitude control and thermal areas. This lack of first-order interaction allowed the structural and configuration designs to proceed relatively unhampered by the indecisions still existing within the electronic subsystem areas.

At this early stage of the study emphasis within the subsystem areas was placed upon arriving at the best subsystem mechanization which would meet the specified performance goals, and which utilized a conservative equipment design. The subsystem design was to be as reliable as possible, but was not to utilize any equipment redundancy or alternate operating modes. Wherever possible, consideration was first given to equipment designs flown on the Ranger and Mariner spacecraft. With the resulting nonredundant subsystem designs as the basis, the the weight, power requirements, and other details were completed for

Table 2. Subsystem Alternatives Considered for Baseline Configurations

	Configuration A (solid retro, gimballed antenna)	Configuration B (liquid retro, gimballed antenna)	Configuration C (solid retro, fixed antenna)
Communications			
High-Gain Antenna			
Rigid dish, double-gimbal	X	X	
Rigid dish, single-gimbal	X	X	
Body-fixed dish			X
Medium-Gain Antenna			
Rigid dish, double-gimbal			X
Rigid dish, single-gimbal			X
Low-Gain Antenna			
Crossed dipole	X	X	X
Deployed on boom	X	X	X
Mounted on fixed boom	X	X	X
Transmitter Output Stage			
Solid-state	X	X	X
TWT	X	X	X
Klystron	X	X	X
Receiver			
Narrowband, phase-lock	X	X	X
With preamplifier	X	X	X
Command Modulation			
Noncoherent FSK	X	X	X
Coherent PSK	X	X	X
Telemetry Modulation			
Mariner scheme	X	X	X
Pioneer scheme	X	X	X
Data Handling			
Sequential data	X	X	X
Interleaved data	X	X	X
Buffer storage	X	X	X
Capsule Link			
Coherent receiver	X	X	X
Noncoherent receiver	X	X	X
Fixed antenna	X	X	X
POP mounted antenna	X	X	X
VHF	X	X	X
Telemetry-matched modulation	X	X	X
Structure			
Basic Type			
Sandwich plus semimonocoque	X	X	X
Sandwich plus truss	X	X	X
Separation Joints			
Shaped charges	X	X	X
Explosive bolts	X	X	X
Explosive nuts	X	X	X
Primacord	X	X	X
Deployment Mechanisms			
Torsion springs	X	X	X
Tension springs	X	X	X
Metal reel (STEM)	X	X	X
Thermal Control			
Compartments			
Insulated, passive	X	X	X
Insulated, louvers	X	X	X
Batteries			
Heaters	X	X	X
Louvers	X	X	X
Heaters and louvers	X	X	X
Planet-Oriented Package			
Insulated, heaters	X	X	X
Insulated, louvers, heaters	X	X	X
Retropropellants			
Inside thermal envelope		X	
Insulated, heaters	X	X	X
Midcourse Propellants			
Inside thermal envelope	X		X
Insulated, heaters	X		X

	Configuration A (solid retro, gimballed antenna)	Configuration B (liquid retro, gimballed antenna)	Configuration C (solid retro, fixed antenna)
Power			
Source			
Solar cells	X	X	X
RTG			X
Regulation			
Shunt	X	X	X
Boost	X	X	X
Series	X	X	X
Form			
AC	X	X	X
DC	X	X	X
AC and DC	X	X	X
Battery			
AgCd	X	X	X
AgZn	X	X	X
Stabilization and Control			
Gyros			
Gas bearings	X	X	X
Ball bearnings	X	X	X
Heated	X	X	X
Unheated	X	X	X
Analog output	X	X	X
Pulsed output	X	X	X
Sun Sensor			
Solar aspect			X
Wide angle	X	X	
Narrow angle	X	X	
Canopus Sensor	X	X	X
Earth Sensor			X
Solar Pressure Vanes	X	X	X
Drives			
Linear with bellows seal	X	X	X
Wabble gear	X	X	X
Harmonic	X	X	X
Limited travel DC motor	X		
Differential bevel gear	X	X	X
Propulsion			
Thrust Vector Control			
Swivelling	X	X	X
Lateral swing only		X	
Liquid injection	X		X
Gas injection	X		X
Entire engine rotation or translation	X	X	X
Jet vanes	X		X
Feed			
All bellows		X	
Bellows and bladder		X	
Positive expulsion		X	
Monopropellant			
Additives to N_2H_4	X		X
N_2H_4 with catalyst	X		X
N_2 pressurant	X		X
He pressurant	X		X
Ablative Chamber		X	
Radiative Chamber		X	
Regenerative Chamber		X	
Ablative Nozzle		X	
Radiative Nozzle		X	
Central Sequencer and Command			
Decentralized Mariner concept	X	X	X
Centralized Mariner concept (single assy)	X	X	X
Centralized Mariner with backup CDU	X	X	X
Increased capability for stored commands	X	X	X

Configurations A, B and C mentioned above. These austere, nonredundant configurations were called "baseline" configurations by the study team. Each baseline configuration could fulfill the performance requirements of the Voyager mission, but all three were significantly lacking in their reliability capability (all three baseline configurations exhibited reliabilities about 1/10th those required by the Voyager mission specification).

After satisfying ourselves that the baseline configurations indeed were as reliable as could be expected for equipment which did not utilize alternate modes of operation or redundancy, the next step was to carry the three baseline configurations forward in the design process to the point where each could meet the mission reliability requirements. This step is illustrated in Figure 5, where it is also noted that Configuration C with the 16-foot body-mounted antenna has the inherent capability to provide significantly improved communication performance over that of Configurations A and B. Considerable weight margin was available for this application of equipment redundancy; preliminary analysis indicated that after allowing for a suitable contingency, the three baseline configurations had the following weights available for improving reliability: Configuration A - 362 pounds, Configuration B - 369 pounds, and Configuration C - 355 pounds.

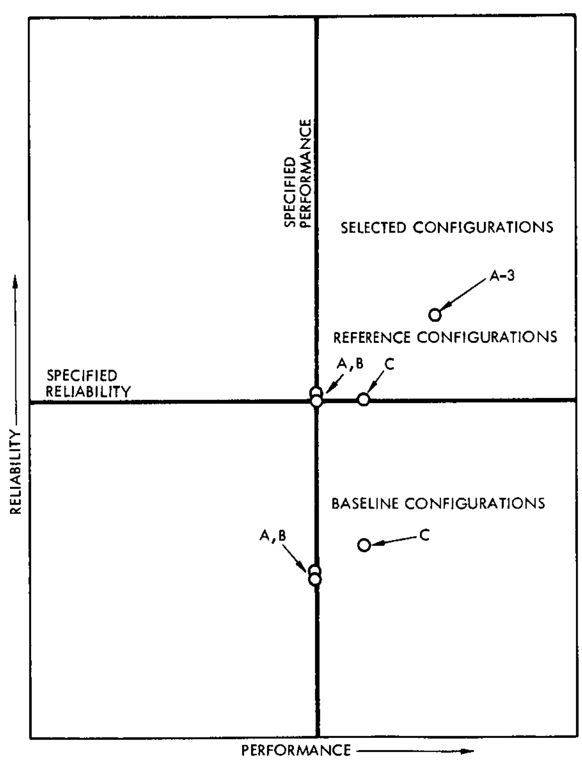


Figure 5. Spacecraft Configuration Evolution

It should be noted that baseline Configuration C has a capability for alternate mode operation not inherent in Configurations A and B.

To fulfill the requirement for telemetering position data to earth while the spacecraft is oriented for propulsive midcourse and orbit-injection maneuvers, a 3-foot single-gimballed antenna was added to baseline Configuration C, since the body-mounted 16-foot dish would be oriented away from the earth during these maneuvers. This second antenna is thus available while in Mars orbit as an alternate operating mode for increased spacecraft reliability. Figure 5 reflects this fact by giving baseline Configuration C a slightly greater reliability.

The judicious application of redundancy and alternate modes of operation in order to best improve mission success thus became the next goal of the study. What was desired was the maximum increase in mission success (over-all spacecraft reliability) for the minimum increase in spacecraft weight. Each subsystem engineering group investigated five to ten sets of new approaches. As each design was considered, weight and power were estimated and a reliability model was constructed. Then each new subsystem was assessed for reliability improvement per pound of added weight, and those new design features that produced the highest yield were selected. Through this process the three baseline configurations were brought up to or slightly beyond the specified mission reliability goals. These three spacecraft designs which now fulfilled or exceeded both mission performance and mission reliability goals were termed "reference" configurations. Configurations A and B had utilized 169 pounds of weight in this process; each still had over 190 pounds available for additional reliability and/or performance growth. Configuration C, which already exceeded the performance of A and B, had utilized 158 pounds of weight and also had over 190 pounds available for growth.

The time had now come for selection among the three reference configurations so that the study team could concentrate on the best ways of improving that one configuration. A preference for Configuration A over Configuration B was first established; the solid engine for retropropulsion was selected instead of the liquid bipropellant engine for reasons of performance, reliability, and development costs. The dimensional constraint on the spacecraft necessitated a relatively complex and inefficient tankage and feed system for the liquid engine; this required using weight to the point that the gain in specific impulse was more than offset in the final performance analysis by the increased weight. Although a second engine is required for Configuration A to permit midcourse corrections before the solid engine is fired at Mars, the capability of a single, multiple-firing liquid engine to meet both of these requirements still did not offset the greater simplicity and ease of development possible with the concept of a solid engine in conjunction with a monopropellant midcourse engine. A monopropellant midcourse propulsion module is also optimum for 1969, 1975, and 1977 flyby opportunities and if not used for 1971 would require a separate development for these opportunities. Moreover the hexagonal sides permitted by the solid engine permitted a more efficient structure than the octagonal sides required by the liquid engine, from the points of view of thermal control and structural simplicity. Midcourse propulsion accuracy was also significantly better when using the monopropellant engine as compared to the bipropellants, probably reducing the number of midcourse corrections required from three to two. Finally, development costs for the liquid bipropellant engine exceeded by several million dollars the development costs associated with the monopropellant-solid combination.

The choice between Configurations A and C was more difficult. On the one hand, Configuration A was clearly the more conservative approach and met all performance requirements quite adequately. On the other, Configuration C with substantially superior communication capability (a factor of 7 improvement) permitted attractive adaptability to the needs of missions beyond 1971. The comparison between Configurations A and C then mainly centered on conservatism in design versus communications performance. To bring the communications performance of the two configurations more in line, some emphasis was placed on improving Configuration A's capability in that area. It was found that additional solar cells and batteries could be added to drive a 40-watt TWT in Configuration A without exceeding the weight available for performance improvement. The factor of 7 difference in communications capability would then be reduced to 3.5.

Further analysis indicated that Configuration C would be data storage limited whereas Configuration A tends to be communications bandwidth limited. The 2×10^8-bit storage capacity of the selected tape recorders requires the full 14 hours of the specified Martian orbit for readout at a rate of 4000 bits/sec, the capability of the Configuration A spacecraft with a 20-watt TWT. The higher data rates of Configuration C could be utilized to provide more free time for the DSIF; however, to transmit more information would require either additional tape recorders or significant advances in data storage state-of-the-art. Configuration C also requires closer attitude control because of its narrower antenna beamwidth.

Because Configuration A is the more conservative design and since it can adequately fulfill all mission requirements with adequate weight margin available for spacecraft improvements (including but not limited to higher communication data rates), it was selected over Configuration C.

The study team was able then to concentrate on how best to utilize the remaining weight reserve of 193 pounds. It was rapidly determined that without some new invention, additional reliability improvements were marginal in that they would greatly complicate the spacecraft design and add disproportionate weight while achieving only minor improvement in mission reliability. Emphasis was therefore placed on improving performance, simplifying spacecraft design, minimizing the complexity of interfaces, and easing assembly and test problems. With these goals in mind, weight was allocated: to propulsion for substituting a simpler blow-down monopropellant midcourse propulsion system in place of the more complex constant-pressure system; to structure for increasing the flexibility for experiment attachment around the periphery of the solar array and for minimizing equipment attachment to the solar array to allow all six solar panels to be identical; and to the planet oriented package so that it could be relocated and hence simplified in its design and deployment. The resulting configuration, called A-3 in the study, is illustrated in Figure 6.

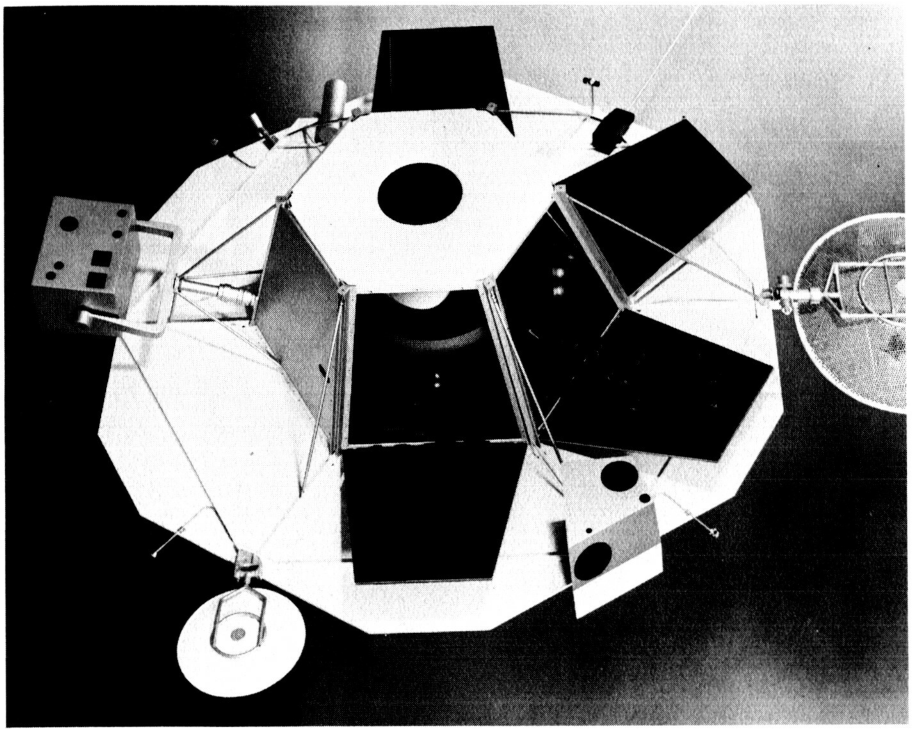


Figure 6. Model of Configuration A-3, with Equipment Panels Opened

Even with all of the selected features integrated into the design of the selected spacecraft, weight margin still remains. This weight is thus available for added safety margin in the design weights, for improving spacecraft performance by adding additional tape recorders or higher power transmitters, or for added scientific instruments.

III. DESIGN CRITERIA

In reaching the decisions just discussed with respect to spacecraft configurations, and particularly in formulating and selecting an approach to individual subsystem designs, it was necessary to adopt a set of ground rules or criteria. First, a review of the mission specification indicated that a strict interpretation of it as a criteria document would not allow the design flexibility indicated as being desired by JPL at the contractors' briefing. Secondly, many subsystems had a multitude of requirements to fulfill, some of which proved to be conflicting. Hence some judgement was required as to the relative importance to be placed upon the multiple requirements.

In arriving at a design which best fulfills our interpretation of what is desired, some 13 specific criteria have been applied. The criteria are discussed here approximately in descending order of importance. It is recognized, however, that they are not strictly comparable, since to some extent they overlap and to some extent they affect independent aspects of the spacecraft design.

1. QUARANTINE

The requirement that the probability of contaminating Mars by a Voyager flight be less than 10^{-4} has been accepted as an overriding ground rule for the Voyager mission. Although the severest implications of this constraint apply to the flight capsule rather than to the flight spacecraft, those requirements interpreted as being on the spacecraft are accepted as absolute. The main areas of applicability to the spacecraft are:

a) A provision in the prelaunch sequence for the surface sterilization of the external surfaces of the flight spacecraft and the capsule canister and the interior of the nose fairing of the launch vehicle

b) The requirement for control of the interplanetary trajectory, with provision for biasing successive aiming points and providing sufficient time after trajectory corrections for redetermination of the orbit and the institution of additional corrective measures, if required

c) Assurance in the design of the spacecraft-capsule interface that the sterilization integrity of that interface will not be jeopardized.

2. PROBABILITY OF SUCCESSFUL OPERATION

The reliability of the spacecraft design has also been accepted as an overriding criteria, in the sense that the predicted probability of mission success desired by JPL has to be demonstrable. In addition, however, and even in those areas where calculated reliability was already so close to 1.00 that further refinements did not affect the theoretical results we have tended to apply the principles that simplicity should have priority, that proven equipment and methods should be preferred to new designs, and that assumptions at the conservative end of the realizable spectrum should be made in estimating margins, stresses, tolerances, and the like.

3. FAILURE MODE CAPABILITY

The failure mode criterion is closely related to the preceding one. It pertains to the ability to achieve the mission objective even though some of the equipment or functions of the spacecraft system are disabled. In particular, great importance is attached to the ability to achieve functional reliability through parallel but nonidentical functional paths (alternate modes of operation). The advantage of this approach over that of pure equipment redundancy is twofold. First, the implementation of a backup mode for a certain function often need not be as complex as the primary mode or consume as much of the available weight, power, and space. Secondly, with identical equipment redundancy, a defect of materials or design which leads to the failure of one mechanization of a function has a possibility of disabling the second. With nonidentical approaches, complete loss of the function due to such a defect is less likely.

4. VERSATILITY AND ADAPTABILITY OF DESIGN TO VARIATIONS IN TRAJECTORY AND MARS ORBIT

The value of a flight spacecraft design which can be used in different modes of trajectory and orbit about Mars within a single launch opportunity, and which is adaptable to the changing requirements associated with the successive missions of the Voyager program, is recognized as quite high. Variations in trajectory geometry and characteristics with launch date must be accommodated. Accommodations to variations in arrival date need not be extreme to make a meaningful mission, but because all of the factors which will enhance the scientific missions have not been specified, the superior spacecraft design is the one which will be able to accommodate the families of trajectories which are chosen. For much of the launch opportunity, asympotic approach velocity and approach orientation relative to the sun are correlated with arrival date and are relatively independent of launch date. An important part of this criterion is adaptability of the design to Voyager missions for launch opportunities after 1971. This criterion places a value on ability to accommodate trajectories and possible Mars orbits for the 1973 opportunity and beyond. The ability to adapt to a flight capsule of increased mass, size, and moment of inertia, for 1975 and later, and the ability to delete the retropropulsion system from the design with a minimum effect on the configuration also are a part of this criterion.

5. ABILITY TO ACCOMMODATE A VARIABLE SCIENCE PAYLOAD

Flexibility with respect to the science payload is important because:

a) The complement of the science payload which will be carried on the 1971 spacecraft has not yet been detailed

b) An ability to accommodate changes in the science payload rather than a design appropriate only to one complement of scientific instruments is of value for the 1971 mission

c) It is likely that the scientific objectives of the Voyager spacecraft will be revised for successive launch opportunities.

The ability to accommodate a variable science payload includes availability of space, weight margin, electrical power, provision for required commands, data storage capability, and communications capability.

6. SCIENCE COVERAGE OF MARS

In view of the fundamental objectives of the Voyager mission, a major criterion with respect to the design of the spacecraft is its ability to facilitate scientific measurements. It must therefore provide for the pointing of sensors as needed, not mask the validity of their data by any mechanism such as induced magnetic fields, and be adaptable to various types and quantities of scientific instruments. The articulation of the POP, the ability of the spacecraft to maintain the required stable attitude, and the magnetic cleanliness of the design are affected in a major way by this criterion.

7. ADAPTABILITY TO CAPSULE REQUIREMENTS

The ability of the spacecraft to accommodate various capsule needs is the counterpart of the ability to accommodate a variable science payload. The capsule criterion is particularly desirable because the capsule design is not firm. Among the variations which the spacecraft should be capable of accommodating are different capsule sizes, shapes, and weights; different landing sites; different geometry and sequences for the separation of the capsule vehicle from the spacecraft; and variations in the command and telemetry requirements of the capsule, both while it is attached to the spacecraft and after it is separated.

8. SPACECRAFT PERFORMANCE MARGINS

Performance capabilities significantly beyond the minimum requirements need to be achieved for improved spacecraft reliability and flexibility to handle design changes (or additional science equipment). Spacecraft operation is enhanced when the demands of the components do not strain the resources of the spacecraft. A thermal control system, for example, which maintains temperatures substantially below the design limits for electrical equipment will improve system reliability and allow for design flexibility and the possible addition of new equipment. Similarly, an electrical power supply which more than meets the demands of the subsystems in the amount of power available, regulation, ripple, etc., will foster a more reliable mission and allow growth in subsystem power utilization without requiring power supply redesign.

The previous eight criteria deal largely with what it is that the spacecraft design is supposed to accomplish. The following criteria deal with how the design is to meet these goals. Although these criteria are listed below the others, the fact that they address a different aspect of the spacecraft design makes any ranking inferences not entirely applicable.

9. USE OF PROVEN DESIGNS

Even large analytical, developmental, and ground testing programs cannot replace the confidence generated by successful flight-proven performance. Therefore, the use of components which have been proven on successful spacecraft is preferred. Only when a distinct and substantiated improvement is apparent are new designs (in the sense that they have not been flown) to be considered.

10. DEVELOPMENTAL SIMPLICITY

When development is needed, that is, for those areas outside the preceding criterion, simplicity and minimum risk are to prevail. The acceptance of the value of this criterion has been formalized in the JPL Preliminary Voyager 1971 Mission Specification by the establishment of a July 1966 development freeze date.

11. SIMPLE INTERFACES WITH OTHER MISSION ELEMENTS

It is desirable that the flight spacecraft be designed in such a way as to provide the simplest interfaces with other mission elements and to impose the least constraints on other elements of the Voyager program. Simple interfaces allow for greater independence in the design of the various mission elements while also providing higher confidence that these elements will function properly when brought together.

12. MODULARITY

It is desired that the layout and design of the flight spacecraft be conducted so as to provide modularity, accessibility, ease of testing, and a minimum requirement for unusual handling and testing facilities. Modularity contributes to the versatility in handling different complements of subsystem components and science payloads. In addition, on a larger scale, modularity permits interchanging major subsystems (for example the propulsion subsystem) to meet the different requirements of successive launch opportunities. A further benefit of modularity is the reduction in the different types of handling equipment, testing equipment, and spares required.

13. COMPLIANCE AND COMPATIBILITY WITH THE INTENT OF THE PRELIMINARY VOYAGER 1971 MISSION SPECIFICATION

Essentially all of the aspects of the mission specification have been recognized in the preceding 12 criteria. The inclusion of this 13th criterion is in recognition of the fact that all of the details of the JPL specification constitute applicable criteria. Deviation from these criteria was allowed only if adequate justification could be proved, and this justification had to be with respect to one or more of the preceding criteria.

Such justification was demonstrated to our satisfaction in a few areas. Variances from the specifications appear in the propulsion subsystem for all three configurations and in the power and attitude control subsystems for Configuration C.

In Configuration A and C, both of which employ a solid retropropulsion motor, the 0.90 limiting value of μ, the mass ratio parameter, has been exceeded by 0.01. Compliance with this restraint is subject to interpretation since associated components of the propulsion system could reasonably have been included in the initial mass which would have decreased the value computed for μ. However, evaluating μ in what we regard to be the intent of the restraint gives the figure of 0.91.

A design restraint applicable to the bipropellant liquid propulsion system of Configuration B is that the propellant expulsion be of the positive displacement type. In considering alternate implementations of the bipropellant liquid engine, the most attractive possibility appeared to be one in which the propellant is expelled by positive displacement through all interplanetary trajectory correction maneuvers and through the start of the orbit insertion maneuver. For the remainder of the orbit insertion maneuver, however, acceleration forces on the propellant are used for propellant expulsion. Because the amount of propellant consumed by all the midcourse corrections is a small fraction of the total propellant, this alternate was chosen for Configuration B to reduce the inert weight without compromising reliability.

The constraint to use solar cells as the primary power source was set aside to permit evaluation of a configuration (Configuration C) which, by nature of its communication and attitude control implementation, is less dependent on orientation with respect to the sun than other spacecraft designs. The use of RTG power sources would be compatible with an orientation independent of a solar reference.

Probably that constraint of the Preliminary Voyager 1971 Mission Specification which we found most limiting was the allowable spacecraft dynamic envelope. The allowable spacecraft height of approximately 5 feet significantly affected: the propulsion subsystem design including both motor design and thrust vector control, the maximum size of a rigid, articulated communication antenna which could be carried, and the locating of science and attitude control sensors in order to obtain adequate viewing angles. However, since this dynamic envelope interfaced with both the Centaur and the flight capsule and since an almost infinite number of possibilities presented themselves for arriving at a different dynamic envelope, the one given by the mission specification was kept invariant for this study.

IV. SELECTED SPACECRAFT DESIGN

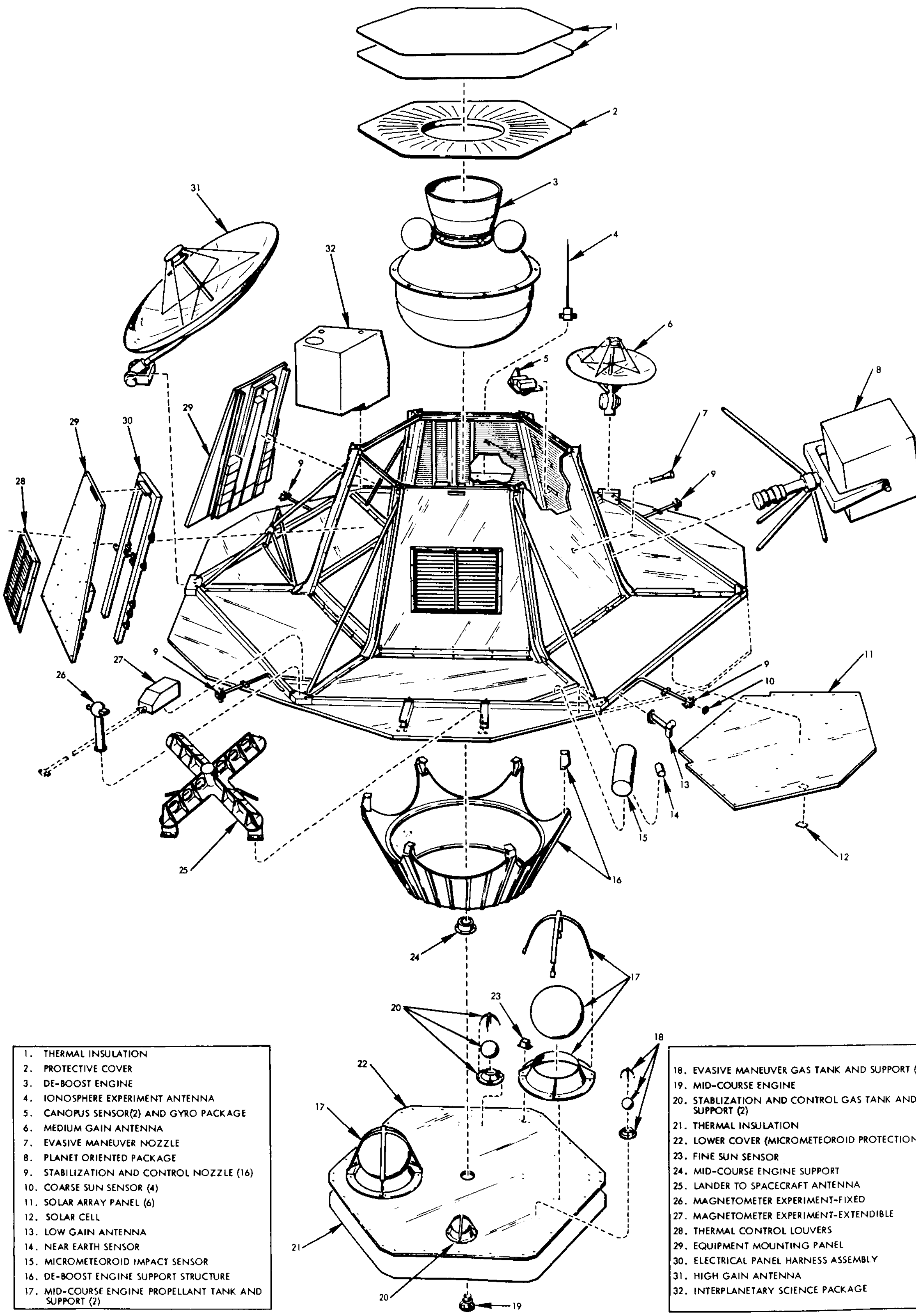
1. STRUCTURE AND CONFIGURATION

The basic structure of the proposed 1971 spacecraft, shown in Figure 7, consists of a hexagonal, truncated pyramid above the 19-foot hexagonal disk which forms the base for the solar array. Six aluminum π-section corner members run from the capsule attachment points to six hard points at the Centaur interstage and carry all vertical loads. The six side panels, 1-inch thick aluminum truss-grid core with aluminum skins, serve as mounting panels for the electronic subsystems, and provide meteoroid protection, heat sink, and the basic shear path of the structure. The panels are hinged along their lower edge to facilitate spacecraft assembly and maintenance. Four of the six side panels include extruded rails on their inner sides for equipment mounting and carry thermal control louvers appropriately located on their exteriors.

A semimonocoque truncated cone of aluminum is centrally located as support for the solid-propellant retropropulsion engine. To reduce heat conduction, a fiberglass attach angle is provided at the lower end of this cone for attachment of the solid motor. Six of the stringers provide uniform engine support into the six corner frame members of the main bus structure. Tanks for liquid injection thrust vector control are mounted at the base of the solid engine nozzle.

A 10-foot hexagonal aluminum truss-grid panel below the solid engine carries tankage for both attitude control and midcourse propulsion. Each pressurization tank is held in place by a cradle support structure and two tension straps. All tanks are made from titanium-forged hemispheres welded together.

The solar array utilizes six identical, 1-inch thick, aluminum honeycomb panels attached by fiberglass angles to the bus structure. These panels provide vibration stiffness and act as heat sinks to assist in thermal control of the solar array. Six horizontal radial beams run



1. THERMAL INSULATION
2. PROTECTIVE COVER
3. DE-BOOST ENGINE
4. IONOSPHERE EXPERIMENT ANTENNA
5. CANOPUS SENSOR(2) AND GYRO PACKAGE
6. MEDIUM GAIN ANTENNA
7. EVASIVE MANEUVER NOZZLE
8. PLANET ORIENTED PACKAGE
9. STABILIZATION AND CONTROL NOZZLE (16)
10. COARSE SUN SENSOR (4)
11. SOLAR ARRAY PANEL (6)
12. SOLAR CELL
13. LOW GAIN ANTENNA
14. NEAR EARTH SENSOR
15. MICROMETEOROID IMPACT SENSOR
16. DE-BOOST ENGINE SUPPORT STRUCTURE
17. MID-COURSE ENGINE PROPELLANT TANK AND SUPPORT (2)
18. EVASIVE MANEUVER GAS TANK AND SUPPORT (1)
19. MID-COURSE ENGINE
20. STABLIZATION AND CONTROL GAS TANK AND SUPPORT (2)
21. THERMAL INSULATION
22. LOWER COVER (MICROMETEOROID PROTECTION)
23. FINE SUN SENSOR
24. MID-COURSE ENGINE SUPPORT
25. LANDER TO SPACECRAFT ANTENNA
26. MAGNETOMETER EXPERIMENT-FIXED
27. MAGNETOMETER EXPERIMENT-EXTENDIBLE
28. THERMAL CONTROL LOUVERS
29. EQUIPMENT MOUNTING PANEL
30. ELECTRICAL PANEL HARNESS ASSEMBLY
31. HIGH GAIN ANTENNA
32. INTERPLANETARY SCIENCE PACKAGE

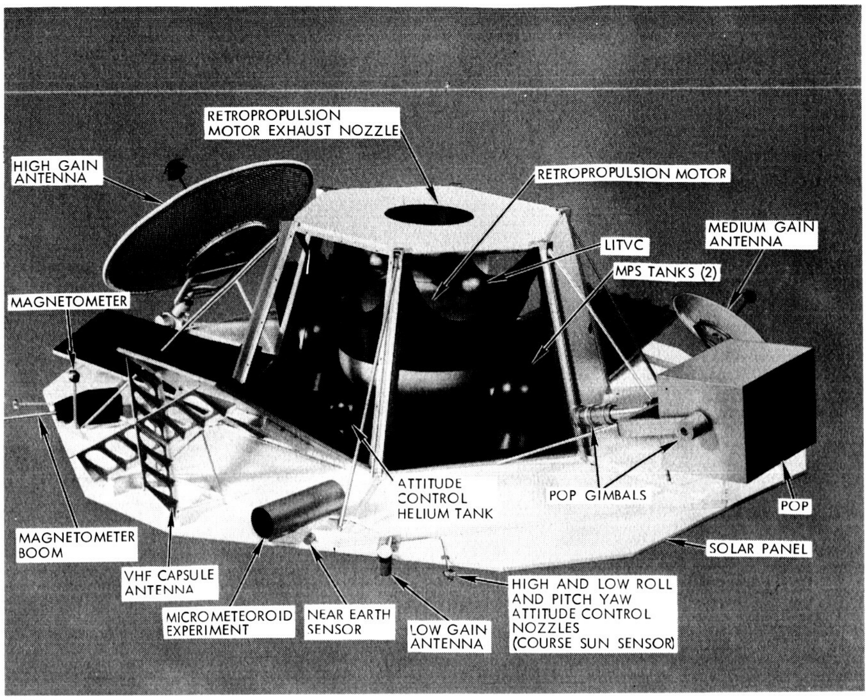
Figure 7. Exploded View of Selected Design

from the corners of the bus to support the edges of the solar panels and the equipment mounted around the periphery of the solar array. Vertical loads are carried by the tubular struts running from the ends of the horizontal support members to the corners of the bus.

The 6-foot antenna, stowed on a triangular brace during boost, is free to rotate on its two gimbals after it is released; the 3-foot antenna moves in one plane only. The X-shaped VHF antenna, a turnstile with reflector, is mounted behind the solar array between the base from which the magnetometer boom unrolls and the micrometeoroid detector. The POP, holding those instruments which face Mars after the spacecraft is in orbit, is articulated by a rotating shaft and fork-mount permitting motion in two orthogonal planes. Photographs of the model of the selected configuration in Figure 8 illustrate assembly of the elements shown in the exploded diagram of Figure 7. In Figure 1 can be seen the positioning of the monopropellant nozzle inside a cavity at the rear of the solid engine and the thermal coupling between the heat generating electrical equipment and the thermal control louvers.

A weight breakdown of the spacecraft by subsystem is given in Table 3. In arriving at these weight calculations, a contingency of 6 per cent has been applied to the nominal weights. This contingency reflects the over-all level of confidence of the weight estimates; it allows for uncertainties in weight estimation techniques, slight modifications of the design, and balance weights; it also includes an allowance for normal weight growth during development.

Since in Configuration A the spacecraft separation plane is located at the field joint, there is little weight associated with separation above the combined field separation joint. Thus, of the allocated 250 pounds for the spacecraft adapter and support above the field joint, only 12 pounds is estimated for cabling, the mechanical disconnect system, and other separation provisions. The remainder of the allocated weight is available as additional margin.



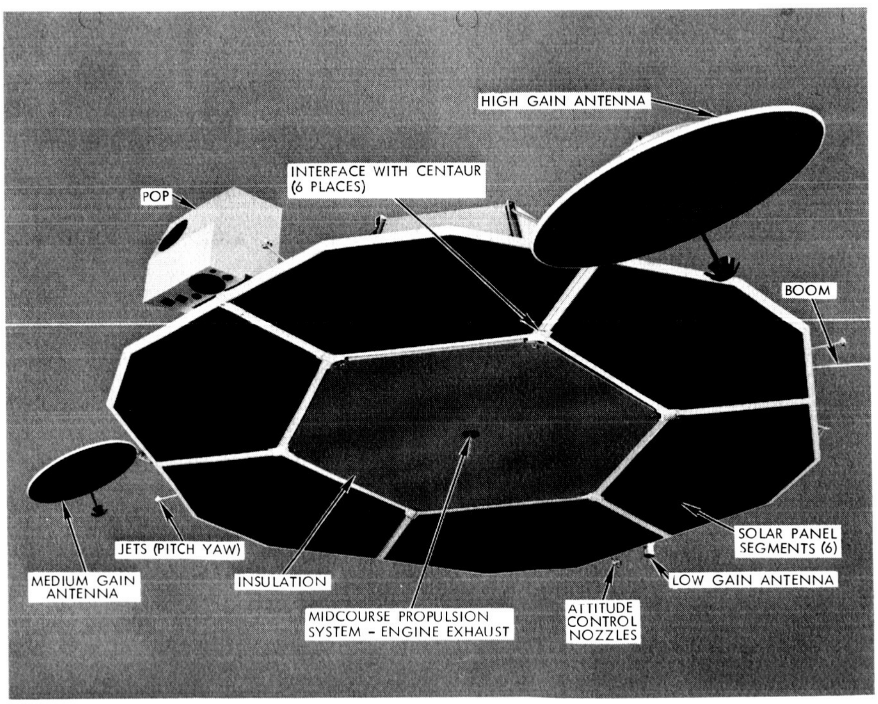
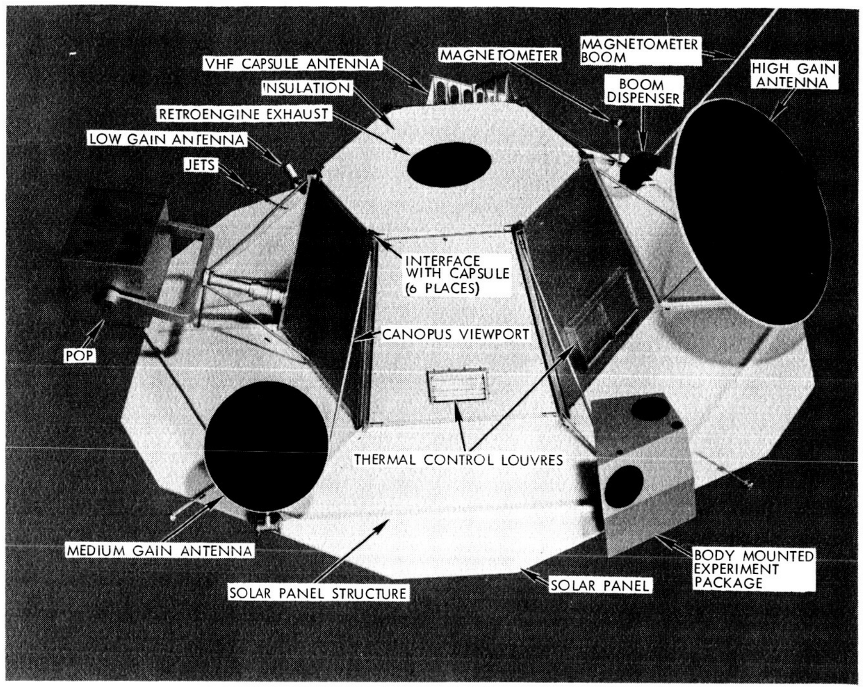


Figure 8. Model of Selected Design



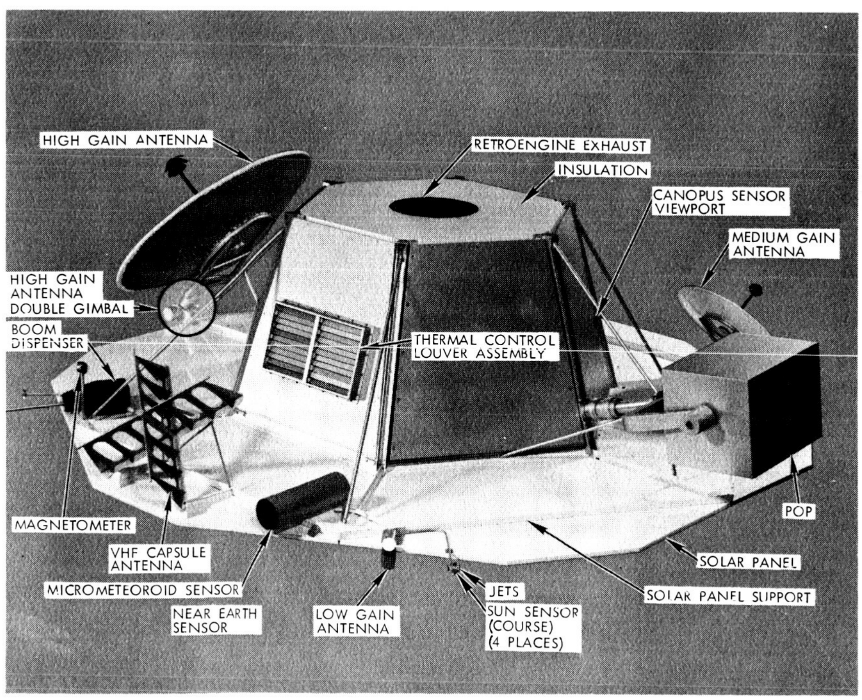


Figure 8. Model of Selected Design (Continued)

Table 3. Weight Analysis of Selected Spacecraft Design

	Propulsion	Capsule	Bus	Total
Spacecraft Bus				
Mechanical and pyrotechnics			37	37
Spacecraft structure			489	489
Thermal control			50	50
Telecommunications			160	160
Electrical power			314	314
Electrical distribution			142	142
Central sequencing and command			27	27
Stabilization and control			100	100
Science support			114	114
Margin			187	187
Contingency			113	113
Spacecraft Propulsion System				
Retropropulsion				
Inert weight	315			315
Module structure	21			21
Midcourse propulsion				
Inert weight	75			75
Midcourse propellant used	215			215
Evasive maneuver propulsion	2			2
Contingency	29			29
Spacecraft Science Payload			267	267
Spacecraft Weight in Orbit				2657
Propulsion				
Retropropellant for deboost	2733			2733
Inerts expended	70			70
Spacecraft Weight After Capsule Separation				5460
Flight capsule				
Remaining capsule components (ejected after capsule separation)		150		150
Capsule vehicle		1950		1950
Jettisoned canister		200		200
Spacecraft Weight Before Capsule Separation				
Propulsion				
Median midcourse propellant used	40			40
Separated Planetary Vehicle				7800
TOTAL	3500	2300	2000	7800
Adapter Allocated Weight Above Field Joint				
Adapter weight above field joint remaining with Centaur				12
Adapter allocated weight not used				238
TOTAL PLANETARY VEHICLE WEIGHT				8050

2. SCIENCE PAYLOAD

After studying the possibilities for interplanetary experiments of the next generation, it was concluded that these experiments will very likely consist of instrumentation operating on the same principles as those carried on Mariner and other spacecraft but with additional capacity. Thus, a set of instruments similar to those carried on Mariner was selected but outputs of higher data rates are postulated. A typical set is listed in Table 4.

Table 4. A Possible Set of Scientific Instruments

Interplanetary Experiments	
	Directional micrometeoroid detectors (4)
	Solar plasma detectors (2)
	Cosmic ray detectors (4)
	Solar flare detectors (3)
	Helium magnetometer (1)
Body-Mounted Planetary Experiments	
	Flux-gate magnetometer
	Radio noise
	RF occultation
Planet-Oriented Experiments	
	Pulsed digital scan mapping camera
	High-resolution camera
	UV spectrophotometer (0.11 to 0.34 μ)
	IR spectroradiometer (0.7 to 20 μ)
	IR multichannel scanning radiometer
	UV photometer flash detector (0.25 to 0.45 μ)

At the available telemetry rate of 4096 bits/sec, a partial map of Mars at a resolution of 1 km can be obtained during an orbital period of 6 months. Complete coverage over 120 degrees of latitude can be obtained in three colors. The television equipment postulated, which incorporates the capability for nested pictures with a resolution of better than 1 km, uses a pulsed digital scan of 1020 by 1024 points, obtaining 6.3×10^6 bits/picture, and a storage vidicon to permit reading into the bulk data storage equipment at a rate of 163,000 bits/sec.

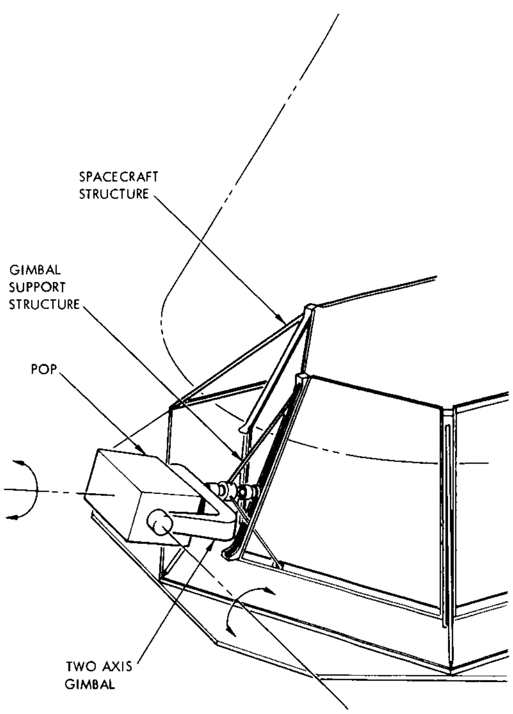


Figure 9. Selected POP Configuration

The POP, holding all those experiments which benefit from articulation with respect to the spacecraft body, provides 8 cubic feet for experiments and a 6-square foot Mars viewing area. POP configurations having no gimbals, one gimbal, and two gimbals were evaluated in terms of operational utility, vehicle interface, and implementation complexity. In addition, two-gimbal subsystems received fairly detailed design evaluation covering equipment layouts, gimbal and drive design, and design difficulty. The tradeoff factors associated with overall POP design and considerations related to mechanization techniques are presented in Volume 5, Section II-2. Figure 9 shows the selected POP configuration. It is supported by a fork on the end of a rotating shaft; the fork provides ±130 degrees of rotation and the shaft ±180 degrees.

3. TELECOMMUNICATIONS

3.1 Communications Subsystem

In the study of the communications subsystem for the selected configuration, the alternates considered covered S-band transmitter outputs in the range of 10 to 80 watts with telemetry rates between 128 and 8000 bits/sec, single- and double-gimballed antennas of various sizes, and the options available for bulk data storage in the spacecraft. Critical choices were made with respect to the low-gain antenna coverage and the type of telemetry link between the capsule and the spacecraft.

The selected configuration (Figure 10) has three S-band antennas: low, medium, and high gain. The medium-gain antenna was added to the baseline subsystem since the weight-reliability studies demonstrated that the redundancy backup provided for the high-gain antenna was an efficient means for enhancing total spacecraft reliability.

The low-gain antenna provides approximately hemispherical coverage (-8 db minimum for 90 degree cone angle) for short range, and at least 2 db over a 45-degree cone angle as required for encounter range. (See Volume 2, VS-4-310, Section 5. 3. 2) The high-gain antenna is a double-gimballed 5.5 x 6.5 foot elliptical paraboloid providing 30-db gain, the slight ellipticity necessitated by fairing constraints. For this configuration, an unfurlable antenna is required to obtain higher gain. Until the reliability of unfurling mechanizations is established, it did not appear desirable to incorporate them in a conservative design. The broader beamwidth of the 3-foot medium-gain antenna (10 degrees), enables the use of a single-gimballed drive without excessive pointing loss after the first 30 days of flight.

Optimization studies showed that the reliability gained by incorporating a separate S-band receiver with each antenna was well worth the cost in weight, providing that automatic receiver output selection was provided. Moreover, having three receivers each permanently connected to a separate antenna eliminated the need for receiver/antenna RF switching. The transmitter power amplifiers considered included klystrons, amplitrons, triodes, and travelling wave tubes. On the basis of known lifetime, reliability and availability, 20-watt TWT's from the Apollo program were selected. Redundant power amplifiers, cross-strapped to redundant modulator-exciters through a single four-port hybrid, were selected as the most reliable configuration.

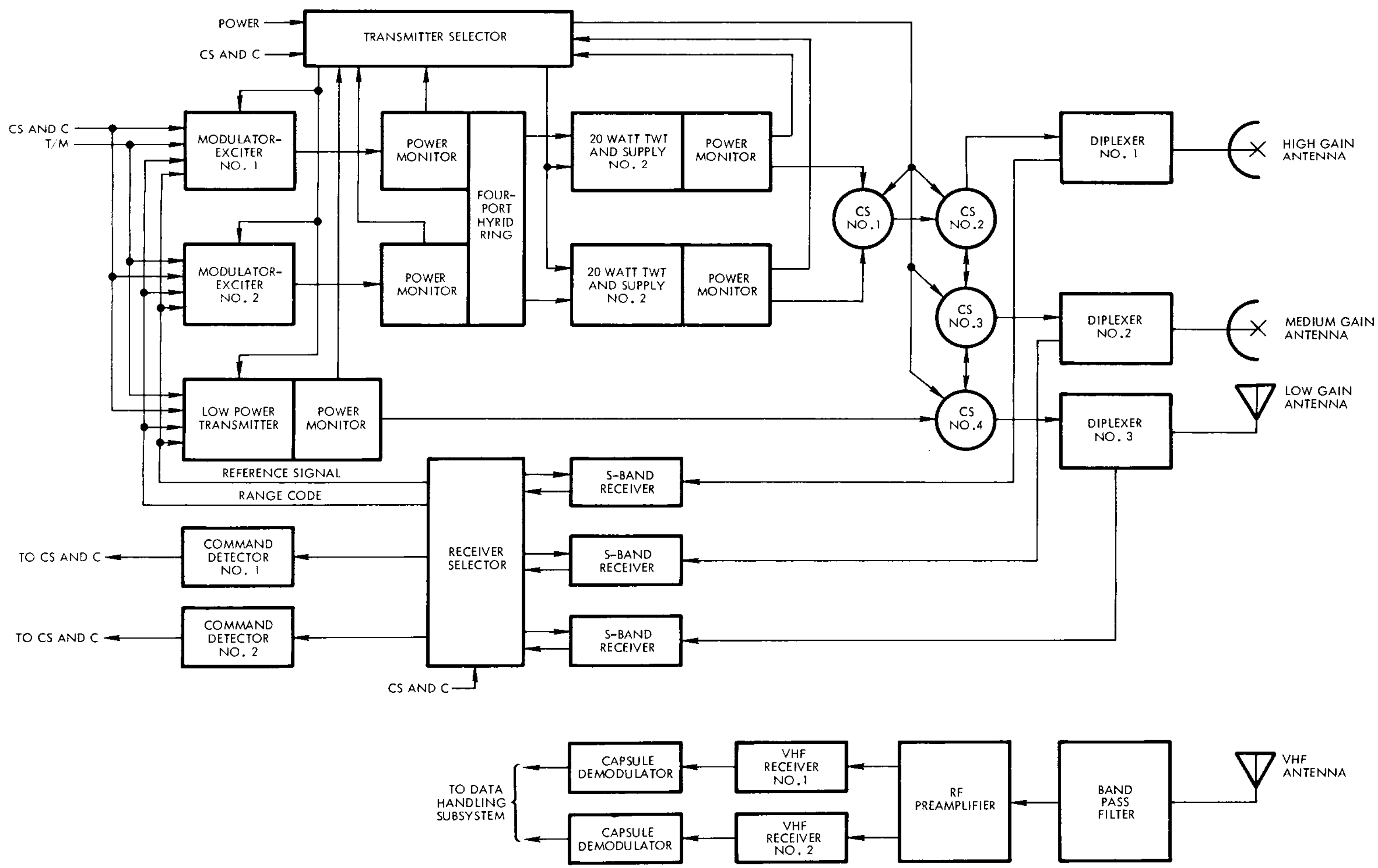


Figure 10. Communication Subsystem Block Diagram

Normal and backup modes of telemetry transmission are listed in Table 5. The maximum ranges and bit rates assume the 210-foot antenna and low noise maser preamplifier at the DSIF ground stations. During cruise, to a range of 1.3×10^8 km, operation at 1024 bits/sec is possible using the 85-foot diplexed antenna. Figure 11 shows telemetry system performance as a function of time from launch.

Table 5. Transmission Modes

Mode	Power (watts)	Antenna
Normal		
I (launch)	1	low-gain
II (after sun-Canopus lock)	1	6 ft
III (cruise, maneuver, encounter, orbit)	20	6 ft
Backup		
IV	1	3 ft
V	20	3 ft
VI	20	low-gain

Using the 85 foot dish and a 100-kw transmitter at the ground station and the low-gain antenna on the spacecraft, command of the spacecraft is possible to a range of 2.5×10^8 km. Beyond this range, the medium- or high-gain spacecraft antennas are required. Figure 12 shows the command link performance as a function of time from launch. Ranging to 2.5×10^8 km requires the 100 kw transmitter and 210-foot dish at the DSIF. Diplex operation with the 10-kw transmitter and the 85-foot antenna will permit ranging to about 6×10^7 km.

For the capsule-spacecraft link, frequency and modulation studies evaluated coherent PSK and noncoherent FSK and FSK/AM systems. Noncoherent FSK was selected since it is the system most tolerant of

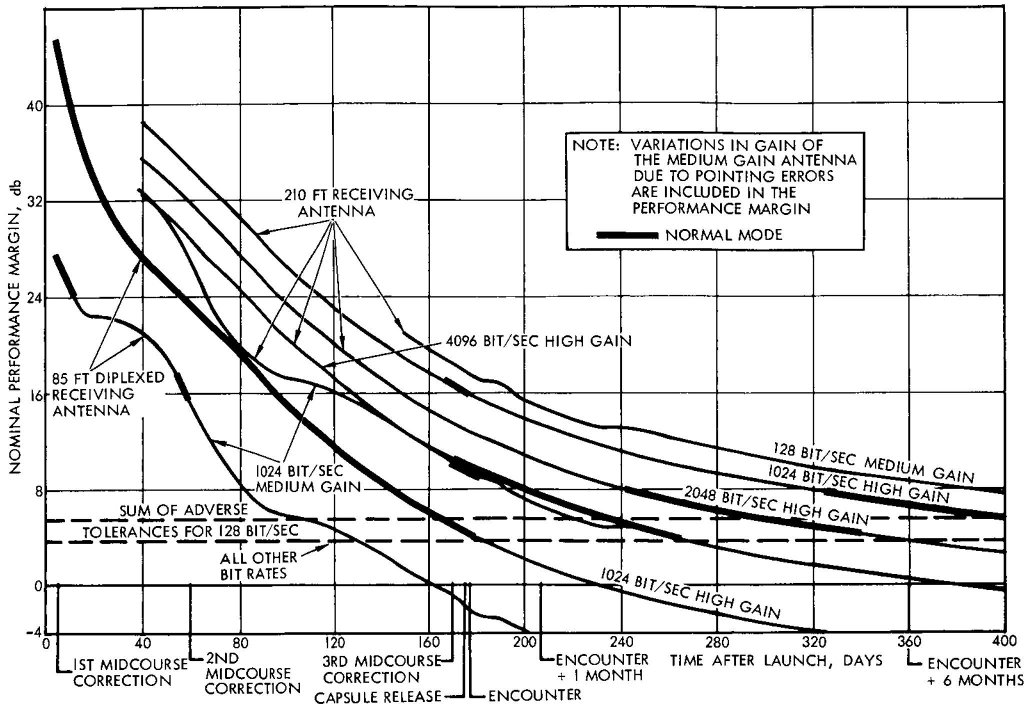


Figure 11. Typical Voyager Performance Margin versus Time, Telemetry Link (Spacecraft-to-Earth)

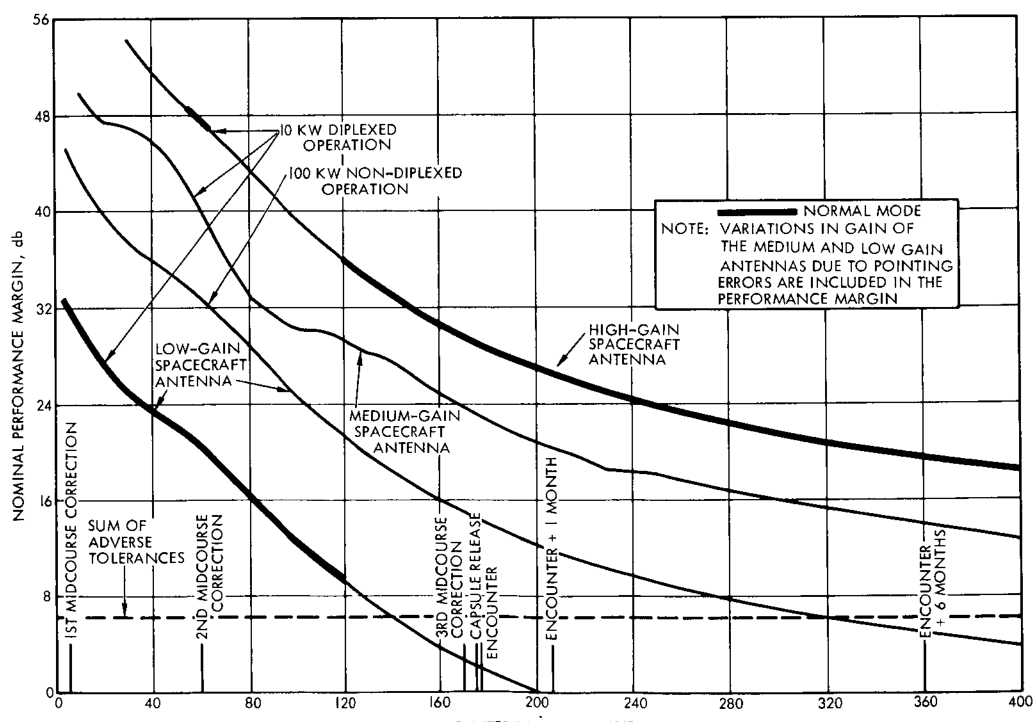


Figure 12. Typical Voyager Performance Margin versus Time, Command Link (Earth-to-Spacecraft)

the multipath propagation effects; in addition, the post blackout acquisition problem associated with a coherent system is eliminated. Redundant, parallel-connected VHF receivers and demodulators with separate outputs to the telemetry system were chosen for high reliability. A fixed VHF turnstile antenna providing a 110-degree beamwidth is connected to a low-noise preamplifier feeding the redundant receivers and demodulators.

3.2 Data Handling

The communications system has a maximum transmission rate of 4096 bits/sec, with commanded or programmed alternates of 2048 and 1024 bits/sec. In addition, studies showed that system reliability could be usefully improved by incorporating a capability for an emergency telemetry rate of 128 bits/sec.

Data storage in the spacecraft is required by the Mars video pictures at a rate which precludes real-time transmission. Based on resolution and other factors discussed in Section 2 of Volume 5, it appears desirable to provide storage for 24 pictures together with other high-rate scan data for a total storage requirement of 2×10^8 bits. Readout directly from the tape recorder is possible at the higher telemetry rates, but for the backup mode of 128 bits/sec, a two-step process is required: reading into the core buffer until it is filled (about 115,000 bits) and then stopping this transfer until the buffer is emptied.

The selected data-handling subsystem (Figure 13) performs various multiplexing functions as well as encoding, conditioning, and storing data. The hardware elements of the subsystem consist of two PCM encoders, two bulk storage tape recorders, a buffer core memory, and a signal conditioner. Redundant units have not been provided for the buffer core memory and the signal conditioner since these units are not in line with the flow of the major portions of the data in normal operating modes. The redundant PCM encoder can be switched by ground command. The subsystem operates in seven data-gathering modes as shown in Figure 14.

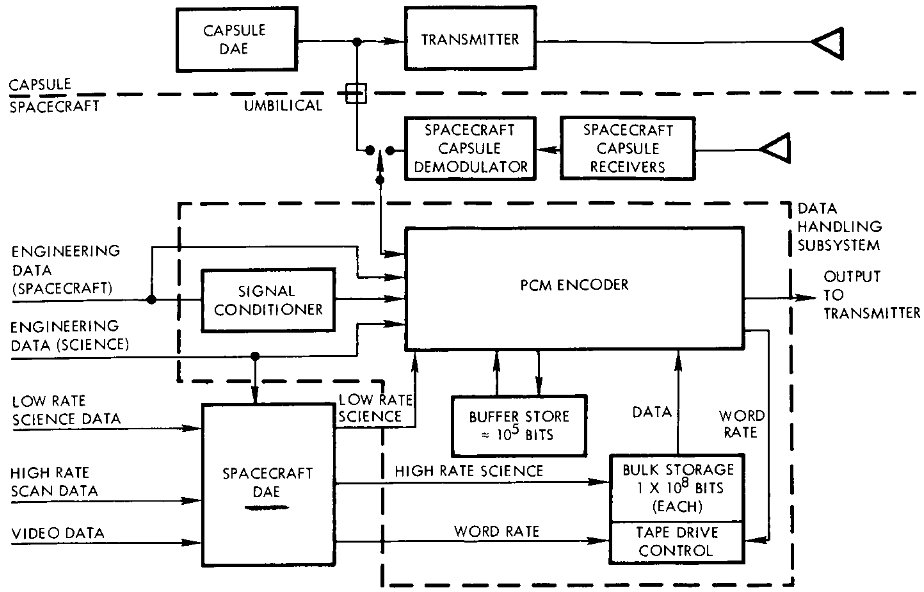


Figure 13. Data Handling Subsystem Simplified Block Diagram

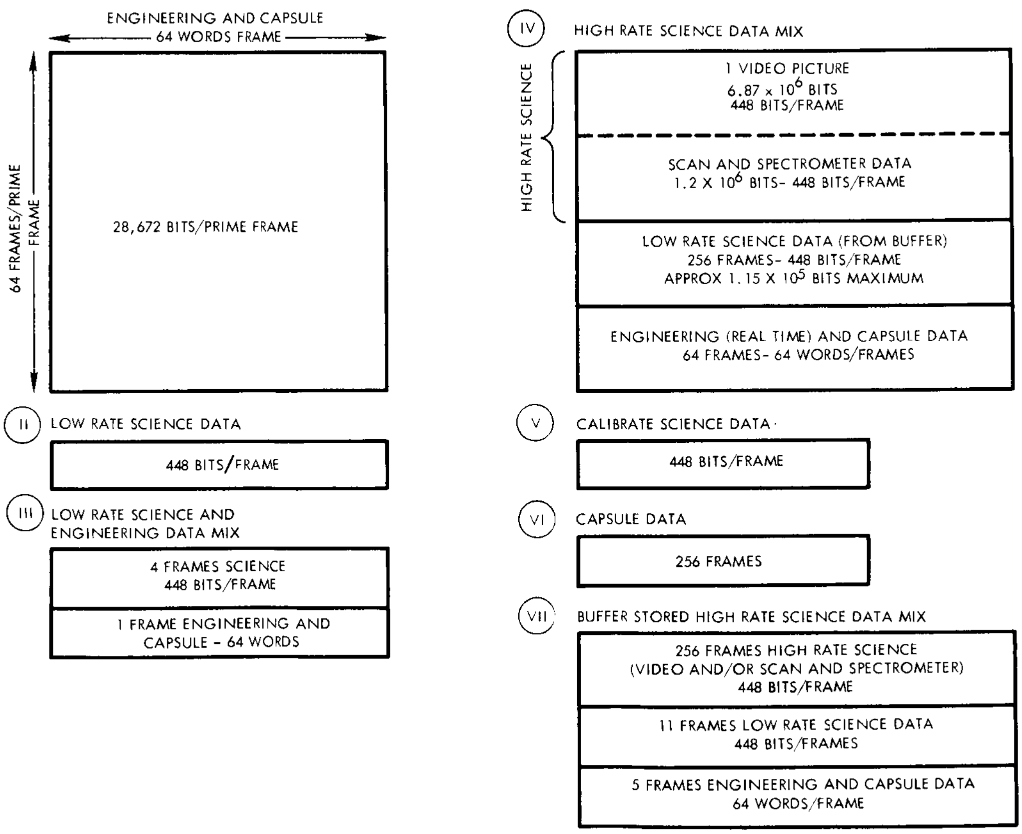


Figure 14. Data Gathering Modes

4. POWER AND CONTROL

4.1 Power Subsystem

It is evident from the spacecraft design criteria that solar cells constitute the preferred primary power source. With the capsule located at one end of the spacecraft, the other end is available for mounting an extensive solar array with unobstructed view toward the sun. For the selected configuration this face provides 190 square feet of useful area for mounting body-fixed solar cell modules; there is no need to deploy solar panels.

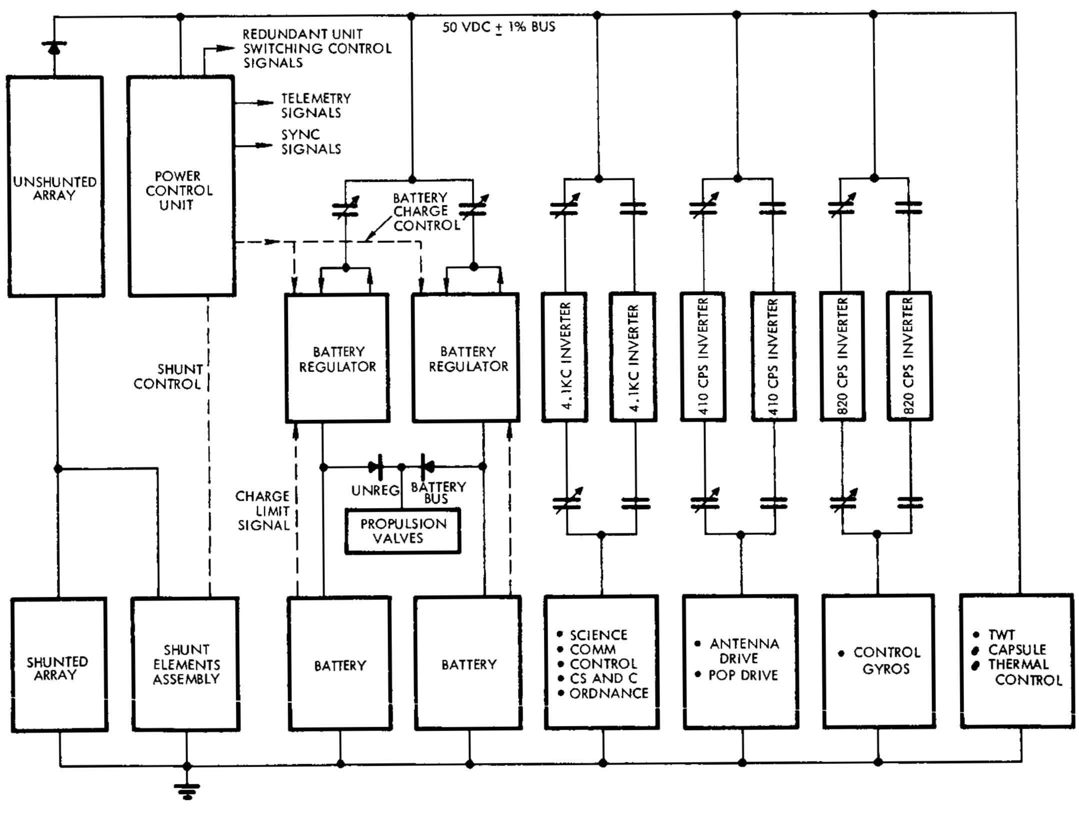


Figure 15. Power Subsystem Block Diagram

The power subsystem (Figure 15) consists of the solar array, solar array shunt voltage limiter, secondary batteries, battery regulators, power conditioning equipment, and a power control unit. The solar cells are installed on six identical panels, each panel having two parallel-connected sections. Each parallel section consists of 116 series by 28 parallel connected 2 x 2 cm, n-on-p silicon solar cells provided with 6-mil fused silica cover slides, making a total of 38,976 solar cells for the array. Tap points permit partial shunt regulation of the array output to 50 VDC ± 1 percent. The electrical output of the solar array is approximately 10 amperes at 50 volts at 1 AU and 8.4 amperes at 50 volts (420 watts) at 1.67 AU, corresponding to encounter plus 6 months and conservatively assuming a Mars radiation flux equivalent to that at earth. At encounter plus 6 months approximately 5 per cent power margin exists.

The two 30-cell, 25 amp-hr silver-cadmium batteries, each with a charge-discharge regulator, operate in parallel under normal conditions. Should a battery or regulator malfunction, the associated battery and regulator are disconnected by the power switching and logic circuitry. A single battery can support essential spacecraft loads through eclipse and maneuver phases. The batteries are charged from the 50 VDC bus through simple dissipative current limiters. Whenever the solar array is incapable of supporting the system load, as during maneuvers and eclipses, the batteries discharge through boost regulators to maintain the regulated 50 VDC bus.

The two main outputs from the power subsystem are the regulated 50 VDC bus and a 50 VAC ± 2 per cent, 4.1 kc, single-phase, square-wave bus. In addition, 410-cps single-phase and 820-cps two-phase inverters supply AC power to drive motors and control gyros. Sequential inverter redundancy is provided by sensing AC bus under-voltage and switching to standby inverters in the event of inverter failure.

4.2 Stabilization and Control

A series of studies directed toward defining the simplest and most reliable mechanization for stabilization and attitude control considered performance requirements, development status of components, and the possibilities for incorporating alternate modes of operation. These studies culminated in the adoption of a subsystem very similar to that of Mariner C. In the over-all system definition of the stabilization and control subsystem (SCS), the establishment of easily acquired attitude references, which remain useful throughout the mission, was a prime objective. Consideration was given to earth-Canopus and sun-Mars as well as sun-Canopus references; the sun-Canopus reference was selected because of its over-all simplicity, a simplicity which reflects into reduced requirements on the spacecraft sequencer as well as the SCS.

A functional diagram of the SCS is given in Figure 16. It provides automatic acquisition of the inertial reference and three-axis stabilization throughout the mission, and controls reorientation of the spacecraft upon command for velocity adjustment or capsule separation. Mode control to enable use of the various sensors and torque sources is done by commands from the spacecraft sequencers and by logic based on sensor outputs. The modes and associated equipment elements are summarized in Table 6. Table 7 lists the sensors used for attitude reference signals. The near-earth sensor is used early in flight to verify that the star tracker has in fact locked on Canopus.

Table 6. Voyager Stabilization and Control Subsystem Modes

Control Mode	Sensors						Actuators		
	Coarse Sun Sensor	Fine Sun Sensor	Star Sensor	Gyro Rate Mode	Gyro Attitude Mode	Earth Sensor	Gas Jets	Jet Vanes	LITVC
Initial Acquisition	x	x	x			x	x		
Cruise		x	x				x		
Reorientation					x		x		
Midcourse Correction					x		x	x	
Inertial Control					x		x		
Reacquisition		x	x	x			x		
Retropropulsion					x		x		x

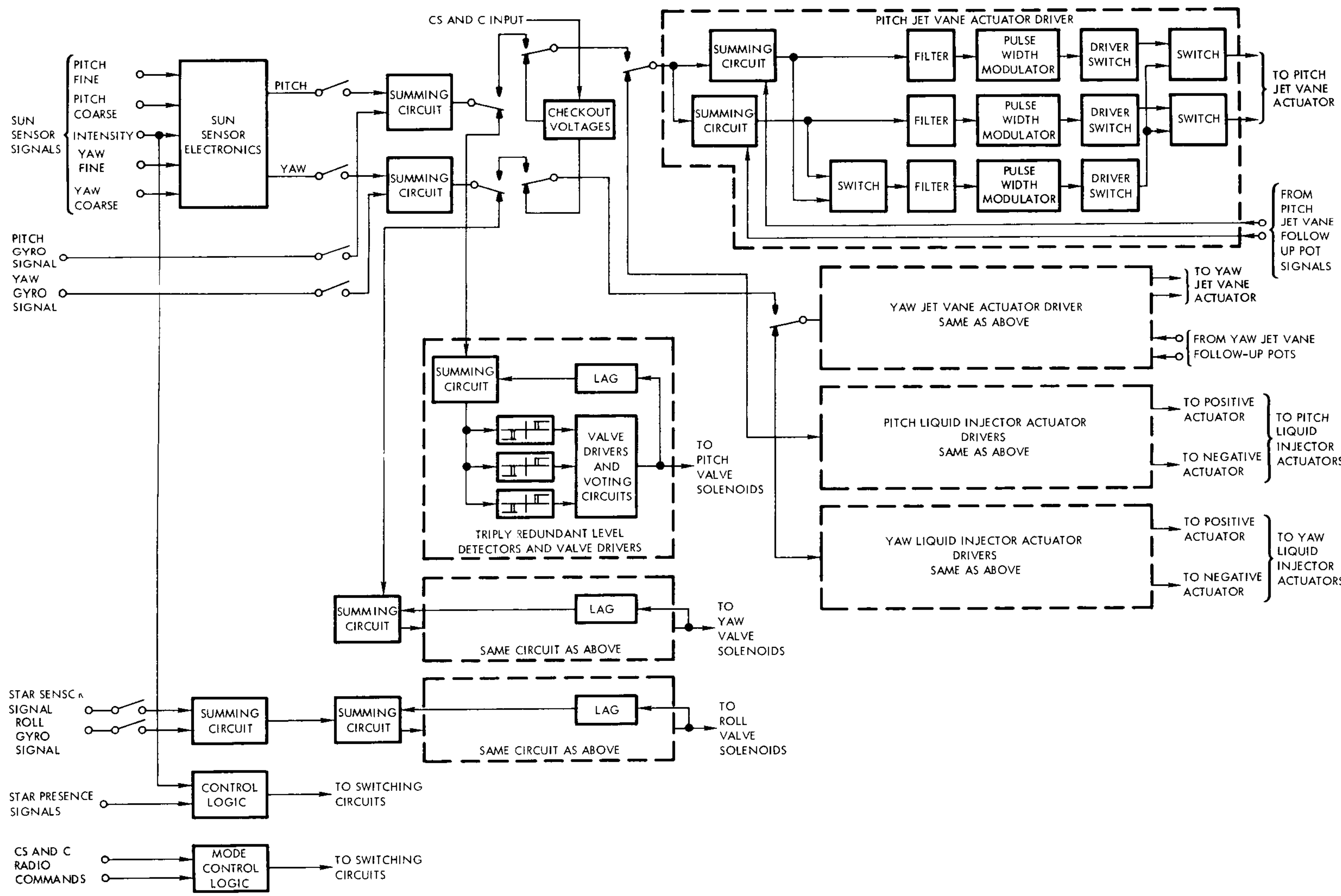


Figure 16. Stabilization and Control Subsystem Block Diagram

Table 7. Attitude Reference Sensors for Stabilization and Control Subsystem

Gyros	One-degree-of-freedom, integrating, temperature controlled with heaters (3)
Coarse Pitch Sun Sensor	Solar cells, back-to-back with lens (2)
Coarse Yaw Sun Sensor	Solar cells, back-to-back with lens (2)
Fine Sun Sensor	Shaded silicon photovoltaic quad cell
Canopus Sensor	Image dissector tube (Mariner-C design)
Near-Earth Detector	Cadmium sulfide cell, with lens

Control torques for cruise and in-orbit operation are obtained by expelling heated nitrogen gas through nozzles that produce couples about each of the principal control axes. When the midcourse monopropellant rocket is firing, pitch and yaw axis control torques are obtained by deflecting jet vanes in the rocket nozzle. During the solid retropropulsion firing, pitch and yaw axis torques are obtained by liquid injection. Roll control during operation of both engines is provided by the pneumatics system, using special nozzles of higher thrust during retromotor operation.

The reaction control system selected for the Voyager spacecraft stores gaseous nitrogen, and incorporates the capability for electrical resistance heating of the gas immediately upstream of the nozzles. This system, illustrated schematically in Figure 17 consists of two redundant storage tanks and feed systems, 12 normal thrust nozzles, four high thrust roll nozzles, 16 on-off solenoid valves, two pressure regulators, relief valves and charging valves, and four pressure transducers. The purpose of using heated nitrogen is to provide an increase in potential life by approximately a factor of two when compared with a cold nitrogen system of the same weight. The nitrogen is heated

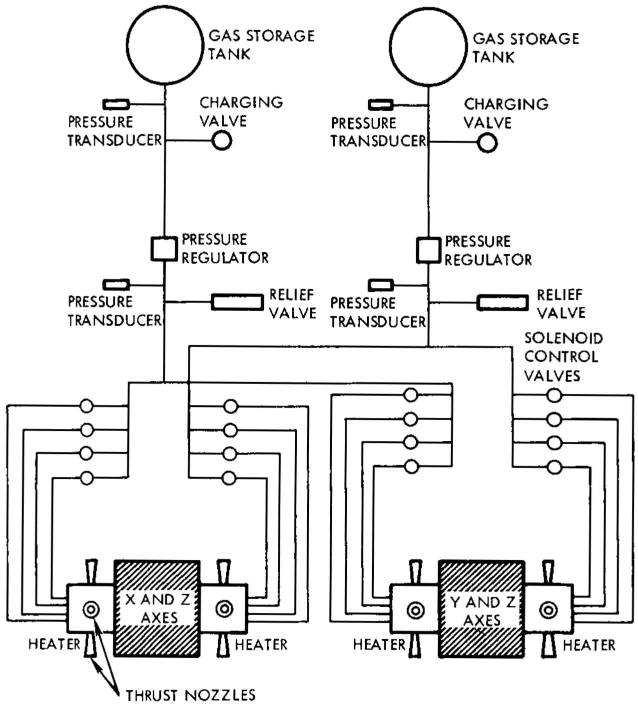


Figure 17. Heated Gaseous Nitrogen Reaction Control System Schematic

only during periods in which an excess of electrical power is available. The Voyager system is sized to provide at least twice the required impulse, not considering the effect of the heaters; the heaters therefore have no reliability implications for the required mission life.

5. CENTRAL SEQUENCING AND COMMAND SYSTEM

The nature and complexity of a central sequencing and command system (CS & C) for the Voyager mission depends on the number and kinds of functions it must perform. As a minimum requirement it must perform timing and sequencing operations, but it might also provide computational support in navigation, data compression, experiment sequencing, and subsystem failure analysis. Thus the system at one extreme may be a relatively simple sequencing device with command decoding capability or on the other a rather complex computer.

In our studies two areas of over-all systems tradeoffs strongly affected the choice among the alternatives: choices pertaining to the distribution of on-board functions between the CS & C and the other subsystems, and choices relating to ground versus on-board distribution of control. The selection of on-board commands had to consider the resulting spacecraft mechanization, and the selection of ground commands had to supply adequate real-time ground control and provide any required backup to on-board control. The final selection favored the simpler configuration which provides on-board sequencing for the major maneuvers with the sequencing initiated by ground commands.

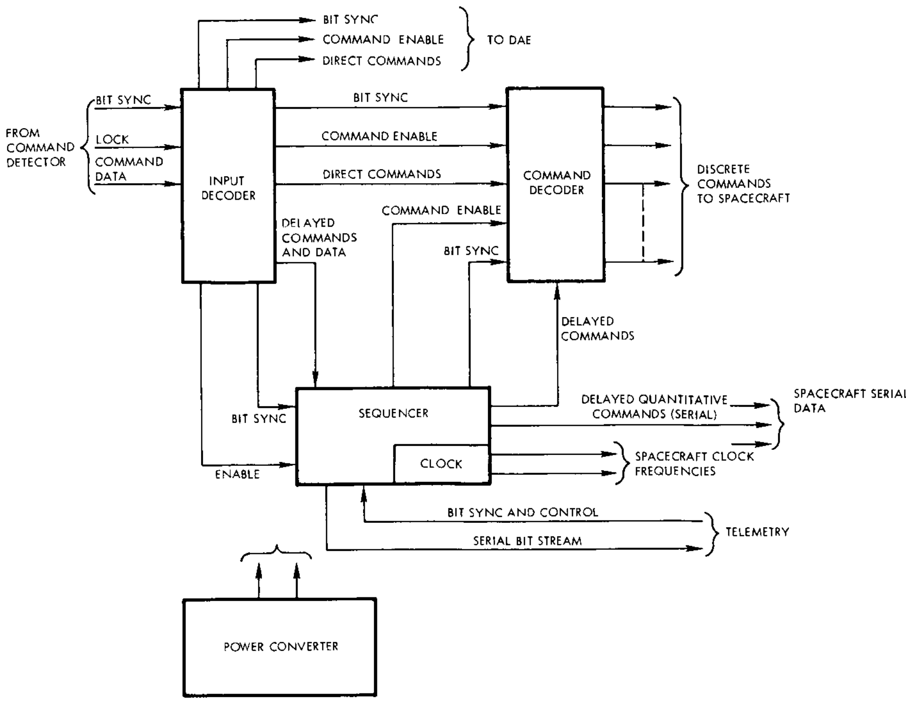
The selected subsystem consists of an input decoder, a command decoder, a sequencer, and a power converter; each of these units is supported by another identical unit so that the subsystem is fully redundant. A block diagram is shown in Figure 18.

In operation, the CS & C subsystem accepts messages from the command detector and routes discrete signals to other spacecraft subsystems in response to direct commands; it also stores command data as required. Commands are stored in a random access core memory which holds 256 18-bit words. The memory is function-oriented in that each word location is identified with a specific function. Each stored command discrete contains a time of command execution, a mode identification tag, and a verification bit. Associated data is located in an adjacent cell as required.

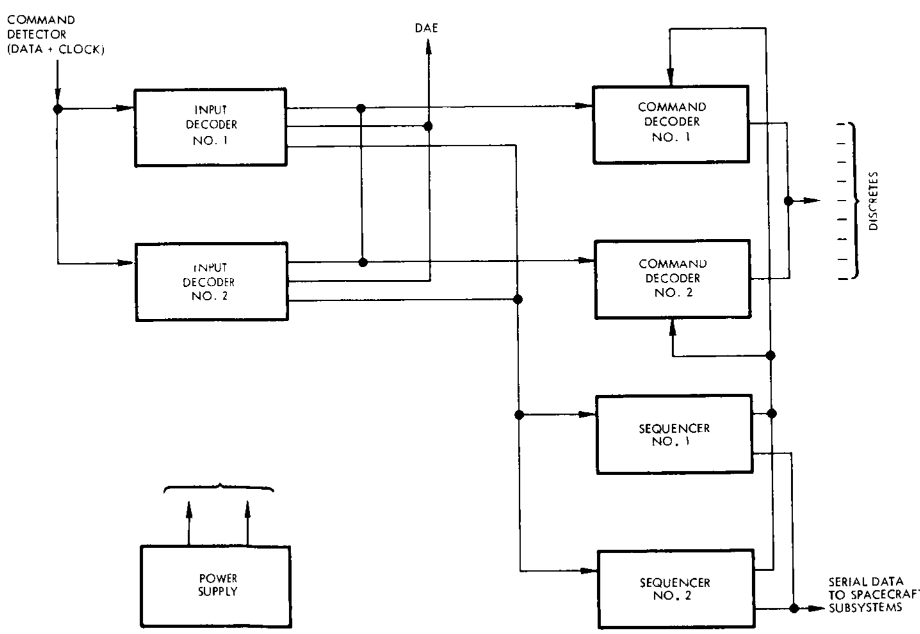
In addition to controlling the operation of the spacecraft, the CS & C performs a synchronizing function by distributing clock frequencies to the data handling and power supply subsystems. These frequencies are also used in the CS & C to control the issuance of serial data and discrete commands. A 26-bit clock provides a continuous 1-second lapsed time record for 400 days.

6. ELECTRONICS PACKAGING

Except for sensors and science equipment which require mounting at particular places on the spacecraft structure, all electronic assem-



Basic Block Diagram



Subsystem Redundancy

Figure 18. Central Sequencer and Command Subsystem

assemblies are mounted on temperature-controlled panels (Figure 19). These panels form the exterior surface of the spacecraft, act as heat sinks, and provide micrometeorite protection for the electronics. Louvers on the facing sides of the panels provide thermal control. At present, four of the six bus faces, each only partially filled, accommodate all the electronics.

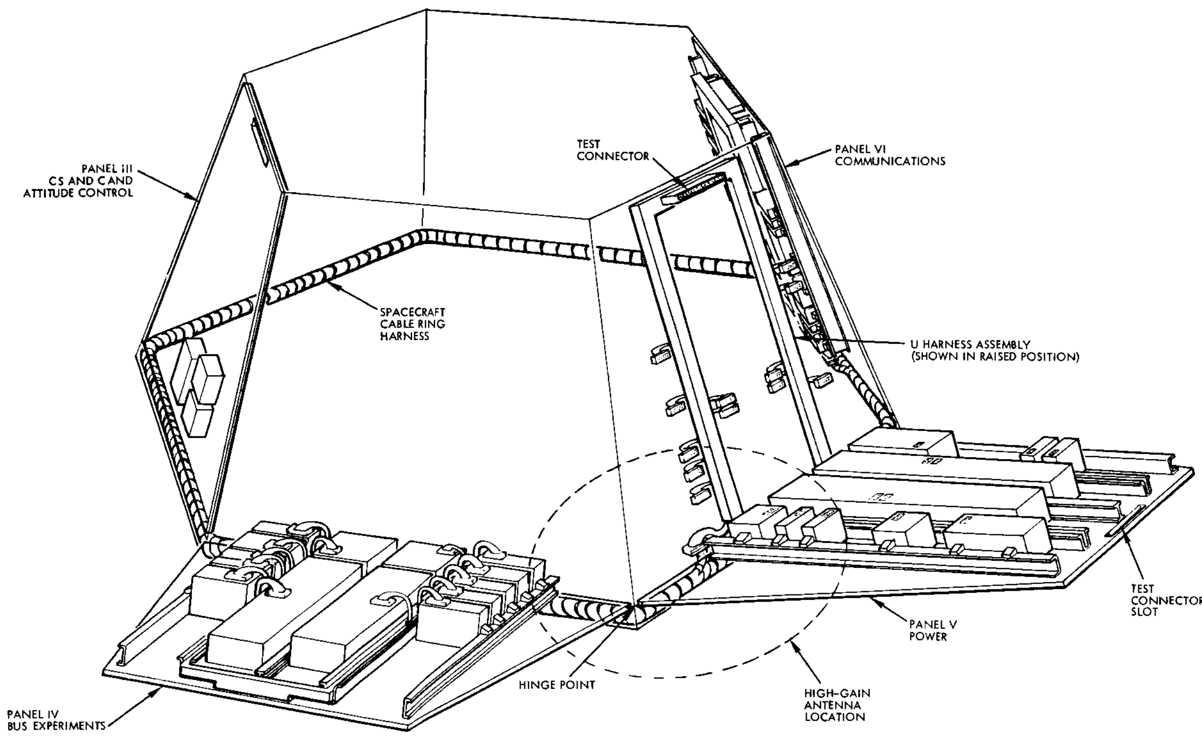


Figure 19. Electrical Equipment Subsystem Panel Installation Accessibility Concept

A standardized packaging concept is used for virtually all of the electronics, except those devices, such as sensors, whose shape and mounting requirements are unique. The level of standardization which was selected encompassed both standard external shapes and standard methods of internal construction. The proposed method of internal construction varies with the circuit type to be packaged as illustrated in Figure 20. For chassis construction, two approaches have been selected, as shown in Figure 21. In general a single integral chassis

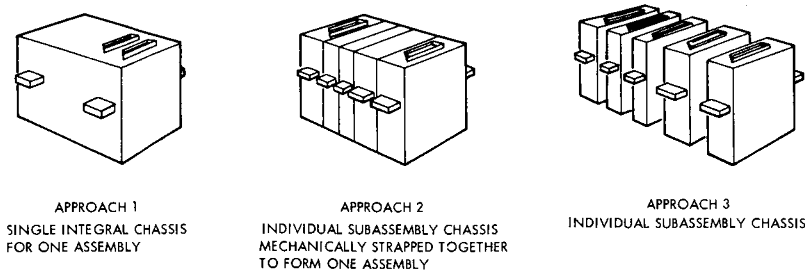


Figure 20. Electronics Assembly Packaging Techniques

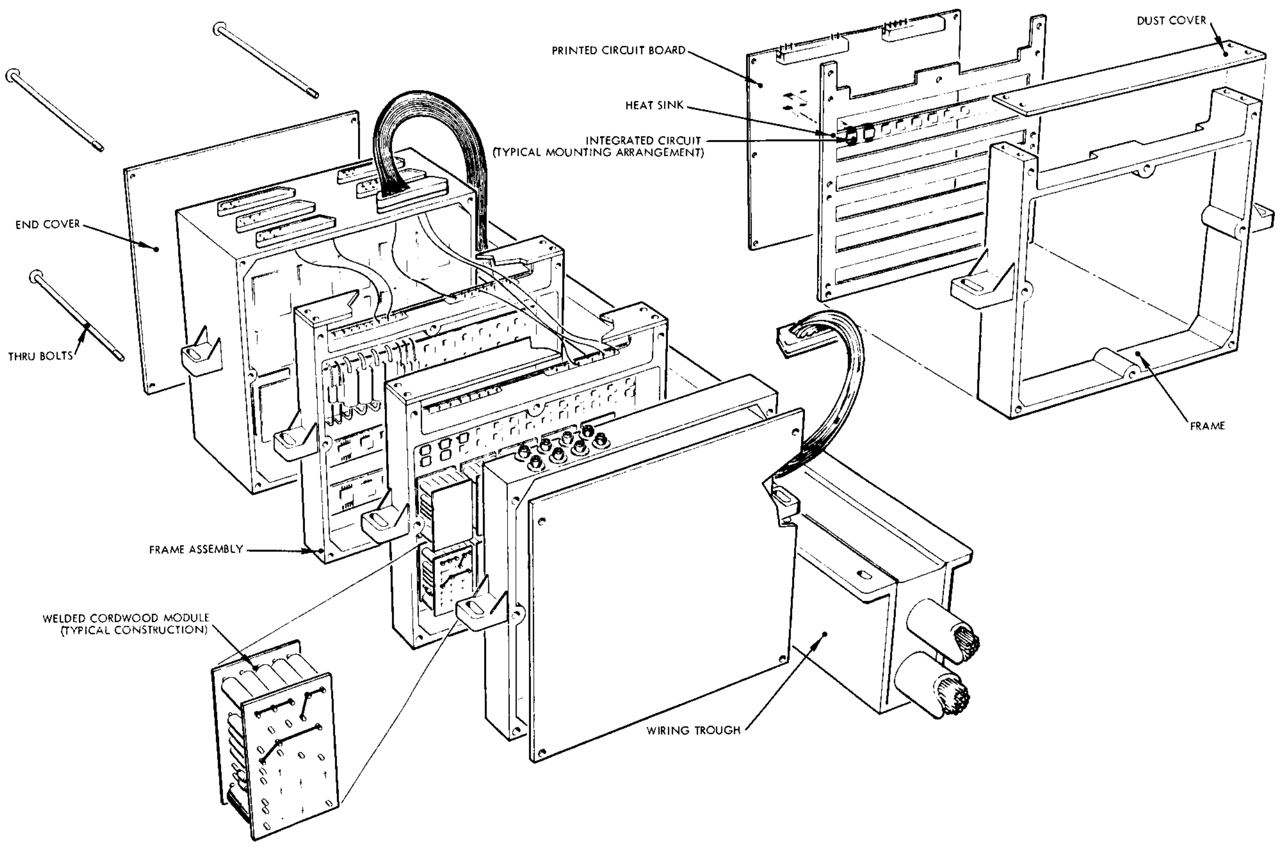


Figure 21. Perspective of Generalized Packaging Concept

construction is preferred for digital circuitry, battery packs, and electromechanical devices such as the tape recorder. Individual subassemblies, mechanically strapped together, are preferred for analog and RF circuits.

The selected packaging concept allows considerable flexibility in the choice of packaging techniques ranging from the conventional printed circuit boards and cordwood modules for discrete components to multilayer circuit boards for integrated circuits. The selected concept is also readily compatible with compartmentalized construction such as for RF circuitry.

7. THERMAL CONTROL

The thermal control subsystem design studies have centered principally on the Mariner concept of an insulated compartment with internally dissipated power radiated to space through individually actuated louvers. Initial studies showed that a completely passive system is inadequate. The electronic components are mounted to the internal surfaces of the honeycomb side panels from which heat is conducted to the external surface and radiated as controlled by louvers to space. The detailed tradeoff and alternative system mechanization considerations have dealt primarily with various techniques of louver blade construction and actuation and the positioning of insulation boundaries. Bimetal actuating springs for the louvers have been selected.

The features of the spacecraft design which contribute to its temperature control are indicated in Figure 22. All nonradiating areas are insulated to constrain heat flow into and out of the spacecraft. Spacecraft surfaces having a large view of the solid motor exhaust plume are protected by high-temperature insulation.

Within the spacecraft bus, heat-producing components are located so as to provide for fairly uniform thermal distribution. The centrally located tankage is shielded from the sun. The tank support structure and its surface characteristics promote thermal coupling with the spacecraft structure. To limit conduction from operating propulsion units, low-conductance plastic structural attachment fittings are used.

8. PROPULSION

As has been discussed in Section II, the selection of the type of propulsion subsystem represents one of the main decision points in

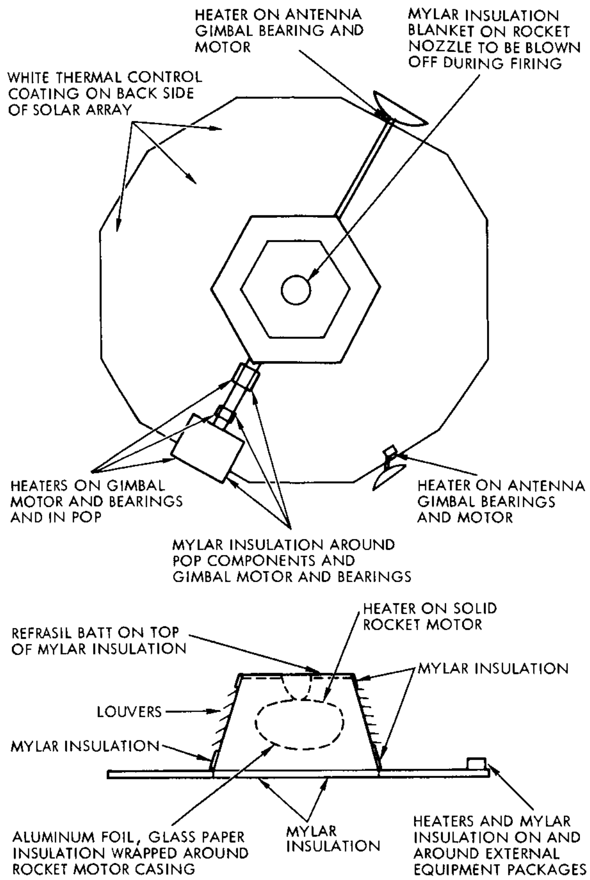


Figure 22. Thermal Control Features of the Selected Voyager Configuration

the study, not so much because the alternate types differ significantly in performance and reliability, but because the choice has a strong influence on the over-all spacecraft configuration.

The basic alternatives considered for spacecraft propulsion were: 1) a combination system in which a monopropellant hydrazine subsystem is used to provide impulse for midcourse velocity corrections and orbit trim maneuvers and a solid propellant motor is used to provide impulse for the retromaneuver, and 2) a storable liquid bipropellant system in which impulse for both the midcourse correction and the retromaneuver is provided by a single engine. More elaborate approaches such as multiple monopropellant engines or a side-firing engine were reviewed and eliminated early in the over-all configuration studies. In view of the fact that both choices are essentially equivalent insofar as ultimate performance is concerned, the major factors which decided the choice in favor of the first alternative above were:

a) The relatively compact nature of the combination system resulted in significantly greater flexibility in the vehicle design.

b) The status of propulsion technology is well established for all elements of the combination system. No problem areas requiring extensive development testing for reliability verification are anticipated, a conclusion that cannot be applied to the single-engine bipropellant system. Although the feasibility of the bipropellant system is sufficiently well established to qualify for consideration under the general guidelines, several components including the main engine will require relative lengthy development programs to verify the design and ensure that the reliability potential has been achieved.

c) The bipropellant engine, as configured, does not have orbit trim capability because of the limited positive displacement approach. If trim capability is provided, the engine is no longer comparable in weight with the solid system.

d) The bipropellant engine could achieve the required mid-course maneuver accuracy but could not achieve the desired accuracy goal. This goal is achievable with the combination of solid and monopropellant engines.

8.1 Retropropulsion

The retropropulsion subsystem is a solid-propellant motor (Figure 23) equipped with liquid injection thrust vector control. Nominal motor performance is summarized in Table 8. The motor consists of a solid propellant grain in a fiberglass pressure case, filled-rubber internal insulation, an ablative exhaust nozzle, a refractory throat insert, a nozzle seal, an igniter with safe and arm unit, and the liquid injection thrust vector control. The liquid injection system consists of four electrically-controlled modulating injector valves, an injectant tank and pressurization system, a supply of Freon, and associated electronics. Welded fittings are used throughout, and the valve ports and gas generator outlet are sealed with metallic burst diaphragms.

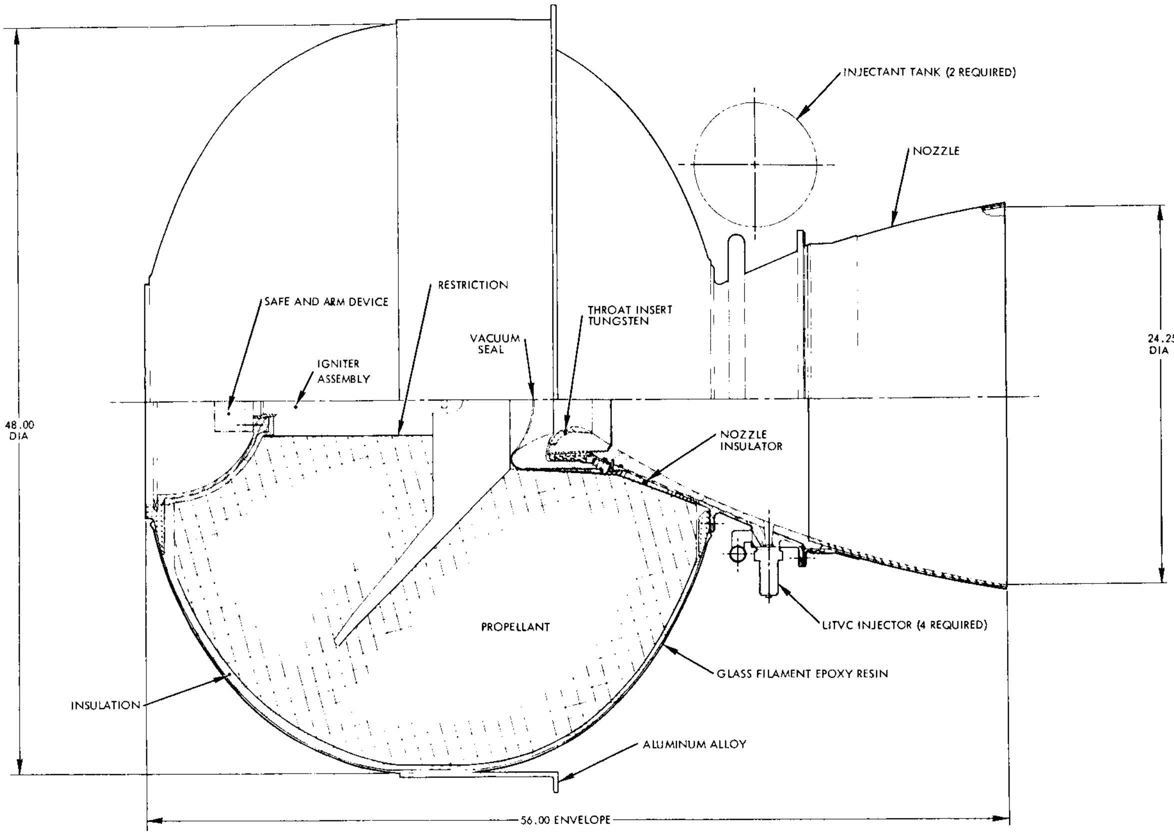


Figure 23. Solid Propellant Engine

Table 8. Performance Parameters of Selected Retromotor Design

Performance:	
Standard specific impulse (sec)	249
Effective vacuum specific impulse (sec)	293
Mass fraction (propellant / total)	0.87
Mass fraction (expended /total)	0.91
Maximum thrust (lb)	15,000
Average thrust (lb)	8500
Maximum chamber pressure (psia)	700
Expansion ratio	50
Burn time (sec)	90-100
Propellant Properties:	
Density (lb/in^3)	0.064
Burning rate (in /sec)	0.21-0.25

8.2 Midcourse Propulsion

The midcourse propulsion system (Figure 24) consists of two combination gas storage and propellant tanks, propellant flow control valves, and a monopropellant rocket thrust chamber assembly. The propellant is anhydrous hydrazine. The thrust chamber contains a catalyst which initiates spontaneous decomposition of the hydrazine.

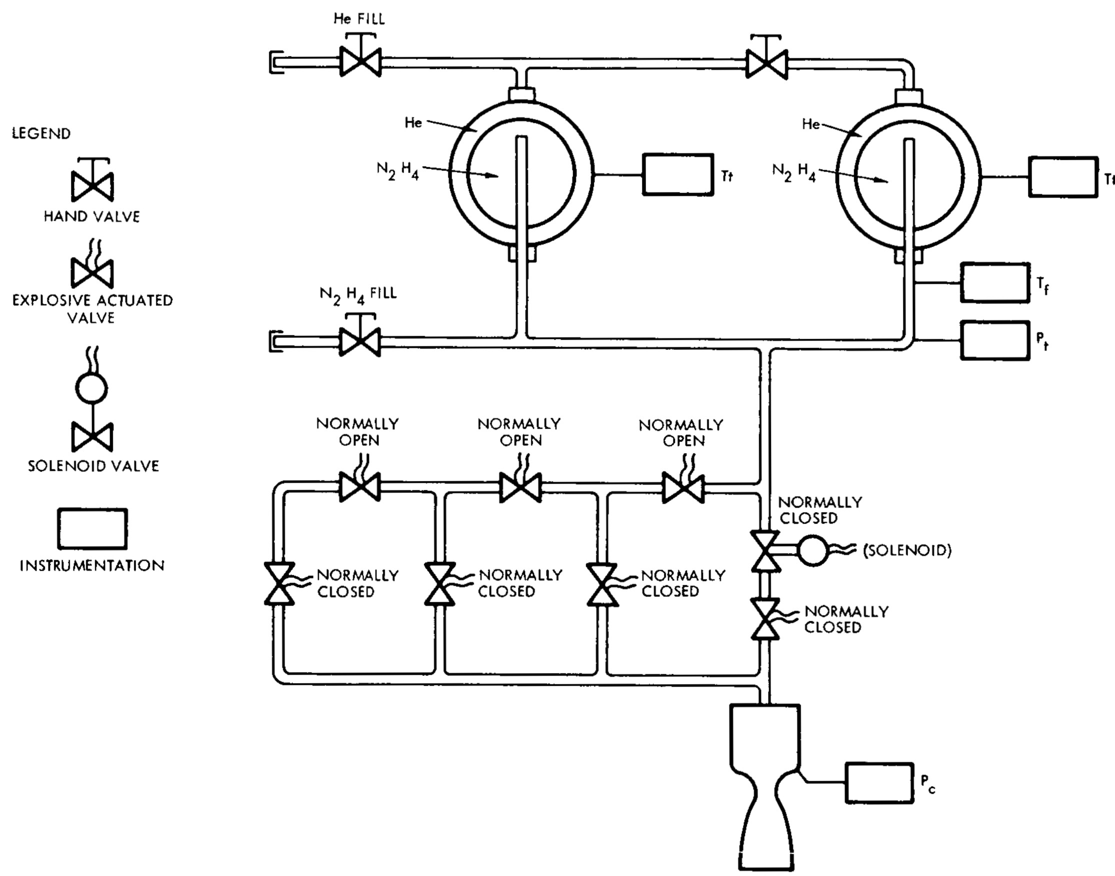


Figure 24. Liquid Propellant Rocket Engine and Associated Feed System

Multiple start capability is achieved by the ganged explosive valves shown in Figure 24. Redundancy and additional start capability are provided by a solenoid-operated backup valve which takes over after the explosive valves have been used. Other characteristics of the engine are as follows:

Specific impulse	230 sec
Initial thrust, lb	50 lb
Minimum velocity increment	0.1 m/sec
Pressurant gas	He

Thrust vector control is by four jet vanes, each producing approximately 2 pounds lift when fully deflected. The nominal thrust vector is parallel with the spacecraft roll axis and passes through the center of gravity of the planetary vehicle. Thrust vector of the engine is adjustable within $\pm$ 0.2 degree of the geometric engine centerline.

8.3 Evasive Maneuver Propulsion System

A cold gas, blow-down propulsion system is carried for the specific purpose of assuring that the spacecraft and capsule will not collide after the capsule separates. It consists of a nitrogen tank initially at 2000 psia, an explosively-actuated valve, and a nozzle. In operation, a firing command from the CS &C actuates the squib valve. Starting at an initial thrust level of 0.1 pound, the system imparts a 0.2 ft/sec velocity increment to the spacecraft.

9. DEPLOYMENT AND SEPARATION

At specified points in the Voyager mission the following separation and deployment events are programmed:

- Booster separation
- High-gain antenna release
- Medium-gain antenna release
- Magnetometer boom deployment
- Capsule cannister separation
- Capsule separation
- Jettison capsule adapter and cannister.

Several mechanisms have been studied for booster separation; stress analysis has indicated that an arrangement incorporating three bolts and separation nuts and three shear pins (Figure 25) is the preferable approach. Pending further information on the capsule design, it appears that a similar arrangement is desirable for the capsule interface, in which three hard points take all shear and three hard points take all tension. Preliminary analyses indicate that three 1-2/inch studs will adequately hold the capsule in position during launch and flight. These same three studs are released to jettison the portion of the capsule cover remaining after capsule separation. This mechanization was specifically designed so as not to intrude into the capsule attachment point adapters as shown in Figure 4 of the JPL Preliminary Voyager 1971 Mission Specification.

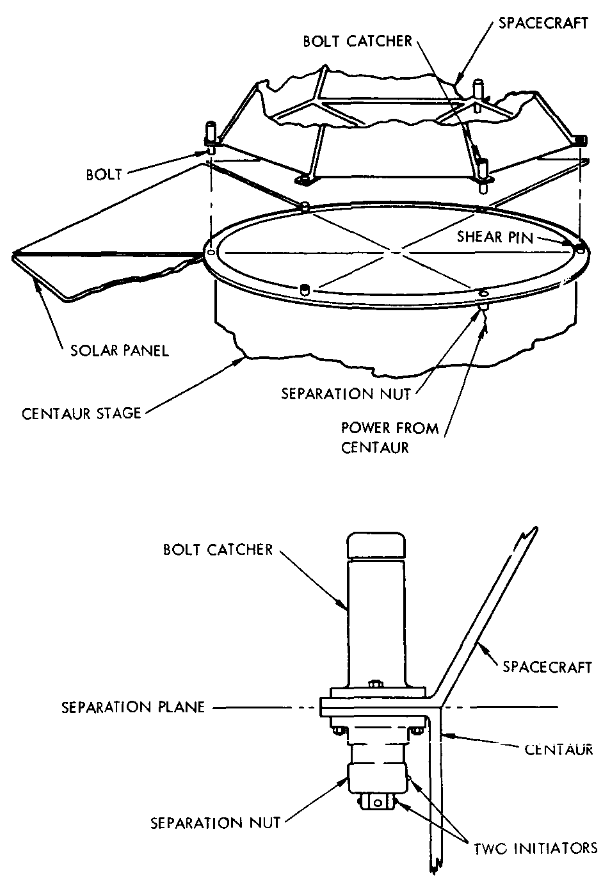


Figure 25. Separation Joint Assembly

V. OPERATIONAL SUPPORT EQUIPMENT

In the design of all operational support equipment (OSE) for the Voyager spacecraft, five principles have been followed:

1) Contamination of the capsule will be prevented

2) Insofar as possible OSE will be applicable to both the 1969 test flight and the 1971 mission

3) The OSE will readily adapt to payload changes

4) Straightforward techniques, previously used wherever possible, will be followed

5) Insofar as possible launch complex OSE will duplicate factory OSE.

In designing the electrical OSE it was necessary further to determine the level of automation (and hence of OSE sophistication) to support the Voyager spacecraft. On the one hand, the desire to launch in a relatively narrow launch window argues for a high degree of automation because of the rapidity with which faults can thereby be detected and isolated. On the other hand, spacecraft programs with relatively few launches tend more toward the use of manual checkout equipment, because the total cost is lower despite the greater number of checkout engineers required. The policy which we have adopted is one of a relatively large degree of automation. Preprogrammed consoles incorporating self-check capabilities, mission simulation, and automatic fault isolation have generally been adopted. The deciding factor was the conclusion that an automated approach is more conducive to prelaunch confidence in the reliability of the spacecraft. Morever, automated checkout equipment could also provide more detailed and consistent record keeping on the spacecraft during assembly and test and thus be of greater utility in the analysis and rectification of any difficulties encountered during spacecraft flight.

Following JPL's nomenclature, the OSE has been categorized into four major groups:

1) System Test Complex

2) Launch Complex Equipment

3) Mission-Dependent Equipment

4) Assembly, Handling, and Shipping Equipment

1. SYSTEM TEST COMPLEX

The system test complex, illustrated in Figure 26, covers all support equipment for powering, monitoring, and recording during tests of the spacecraft and its subsystems. The complex includes unit test sets, system test sets, and an automatic data handling system.

During unit and subsystem acceptance tests, unit type approval qualification tests, and panel qualification tests, the unit test sets simulate the output loading and the inputs which the unit experiences during spacecraft operation, including varying the input parameters beyond normal tolerance requirements.

The system test set is the central point for conducting and controlling the integrated systems test. In conjunction with the automatic data handling system, it contains the command stimulus generators and the data acquisition, processing, measurement, display, and related equipment for exercising and evaluating the operation of the Voyager spacecraft. A simplified block diagram is shown in Figure 27. Functional requirements are divided into the following categories:

- Commands to spacecraft
- Data acquisition from spacecraft
- Data processing and display
- Stimulation
- Simulation
- Ground power
- Critical spacecraft monitoring
- Self-test and fault isolation.

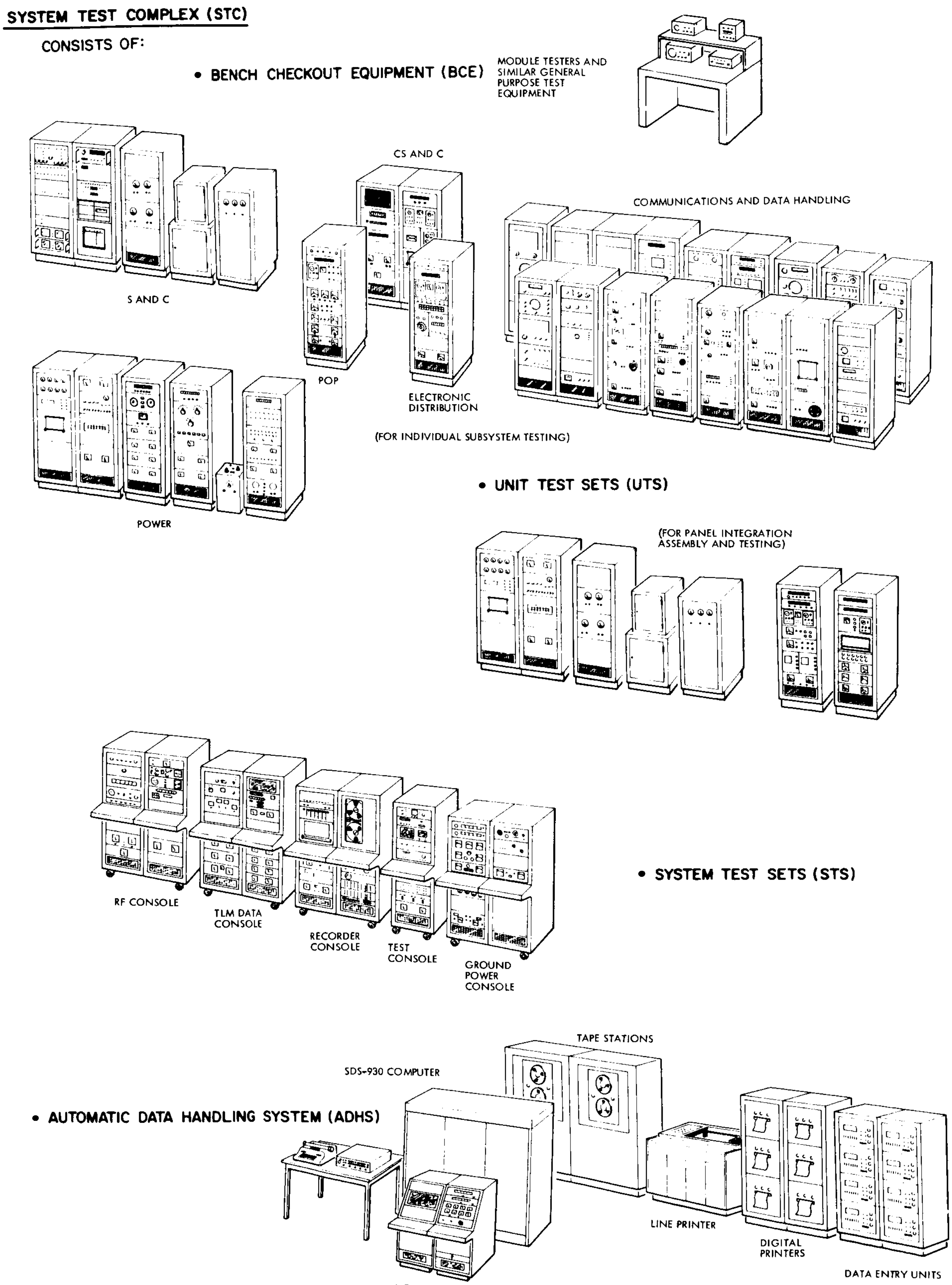
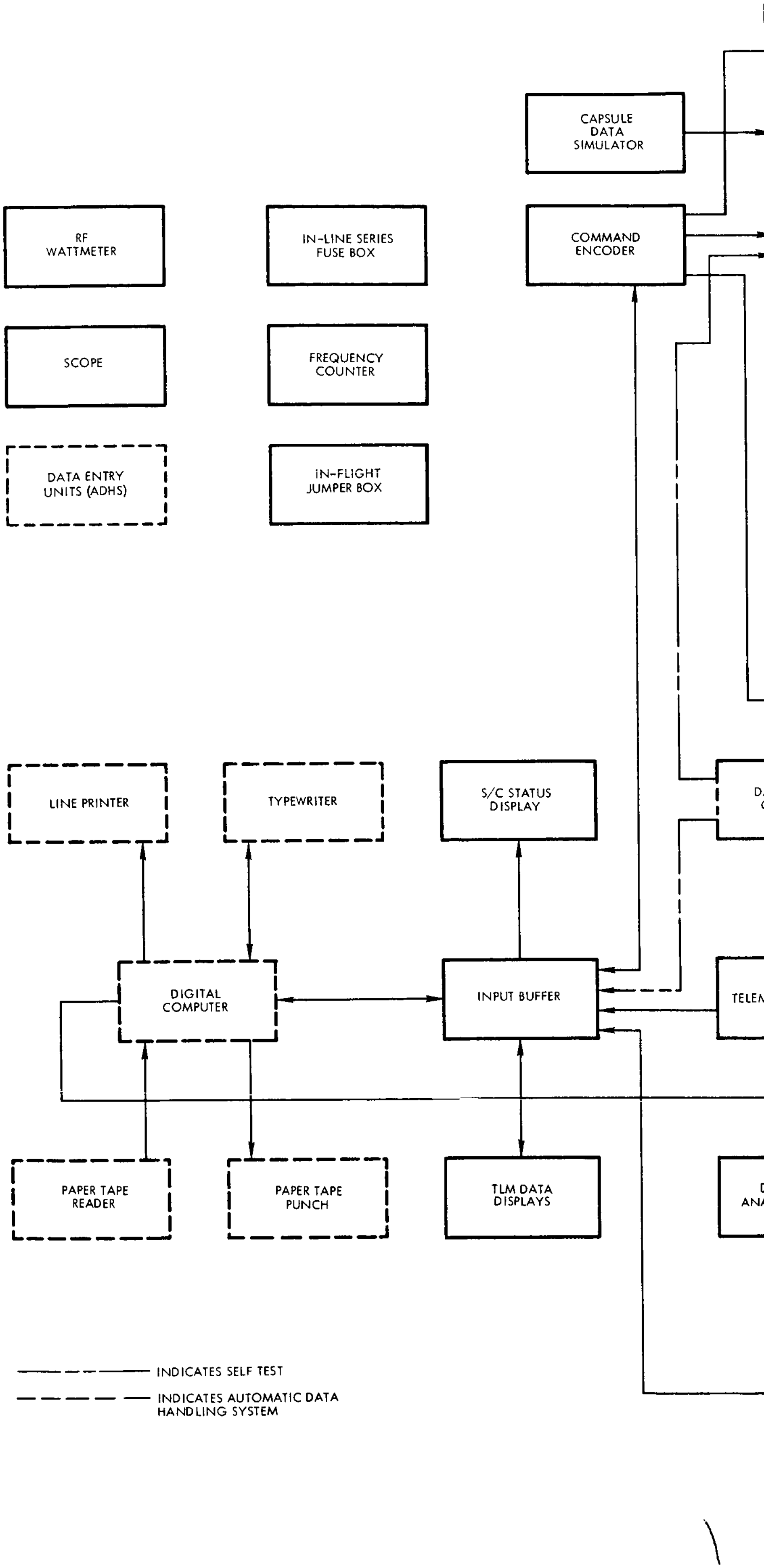
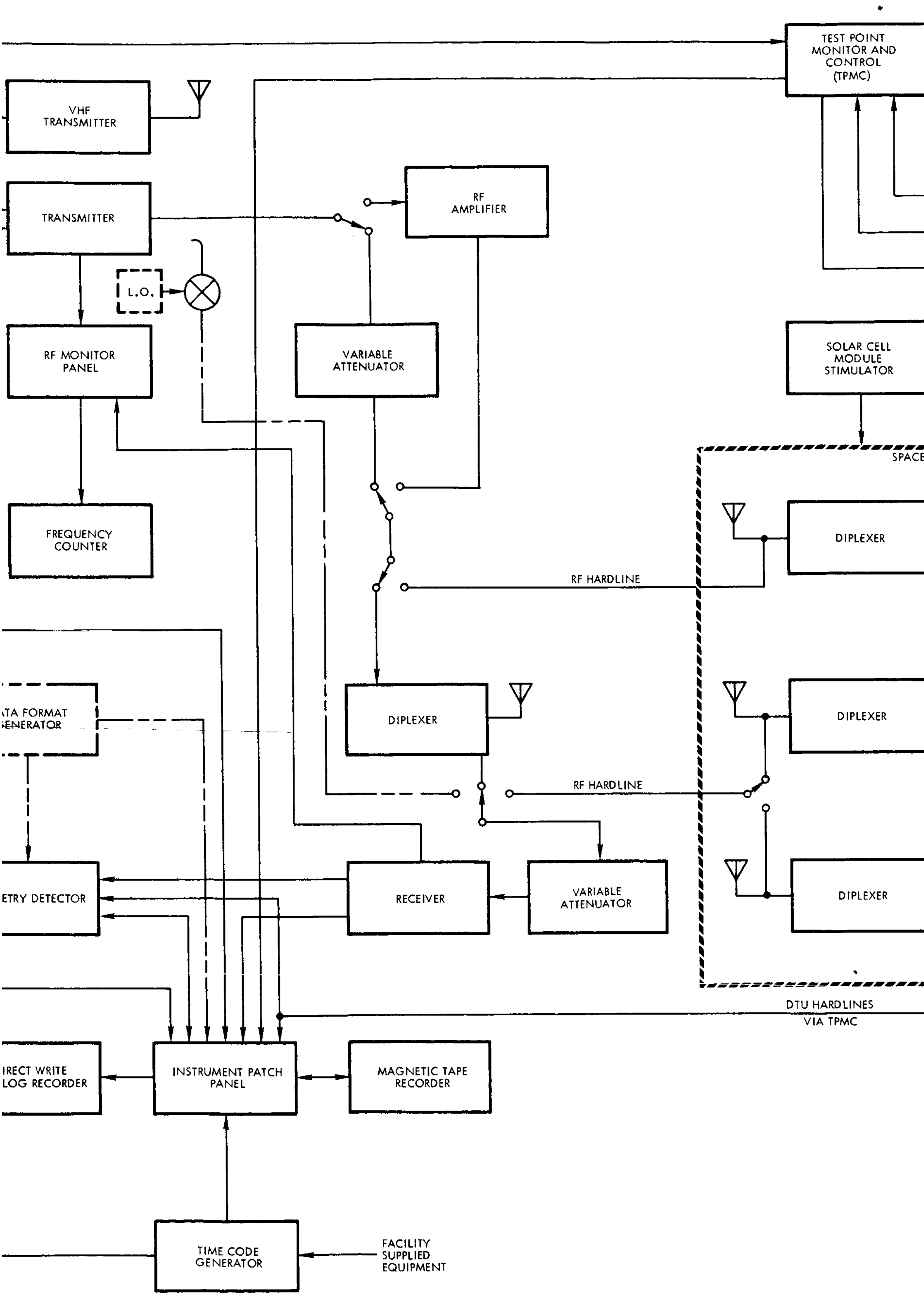


Figure 26. System Test Set Complex


CAPSULE DATA SIMULATOR
RF WATTMETER
IN-LINE SERIES FUSE BOX
COMMAND ENCODER
SCOPE
FREQUENCY COUNTER
DATA ENTRY UNITS (ADHS)
IN-FLIGHT JUMPER BOX
LINE PRINTER
TYPEWRITER
S/C STATUS DISPLAY
DIGITAL COMPUTER
INPUT BUFFER
PAPER TAPE READER
PAPER TAPE PUNCH
TLM DATA DISPLAYS
INDICATES SELF TEST
INDICATES AUTOMATIC DATA HANDLING SYSTEM



TEST POINT MONITOR AND CONTROL (TPMC)
VHF TRANSMITTER
TRANSMITTER
RF AMPLIFIER
L.O.
RF MONITOR PANEL
VARIABLE ATTENUATOR
SOLAR CELL MODULE STIMULATOR
SPACE
FREQUENCY COUNTER
DIPLEXER
RF HARDLINE
TA FORMAT
ENERATOR
DIPLEXER
DIPLEXER
RF HARDLINE
DIPLEXER
ETRY DETECTOR
RECEIVER
VARIABLE ATTENUATOR
DTU HARDLINES VIA TPMC
IRECT WRITE LOG RECORDER
INSTRUMENT PATCH PANEL
MAGNETIC TAPE RECORDER
TIME CODE GENERATOR
FACILITY SUPPLIED EQUIPMENT

2

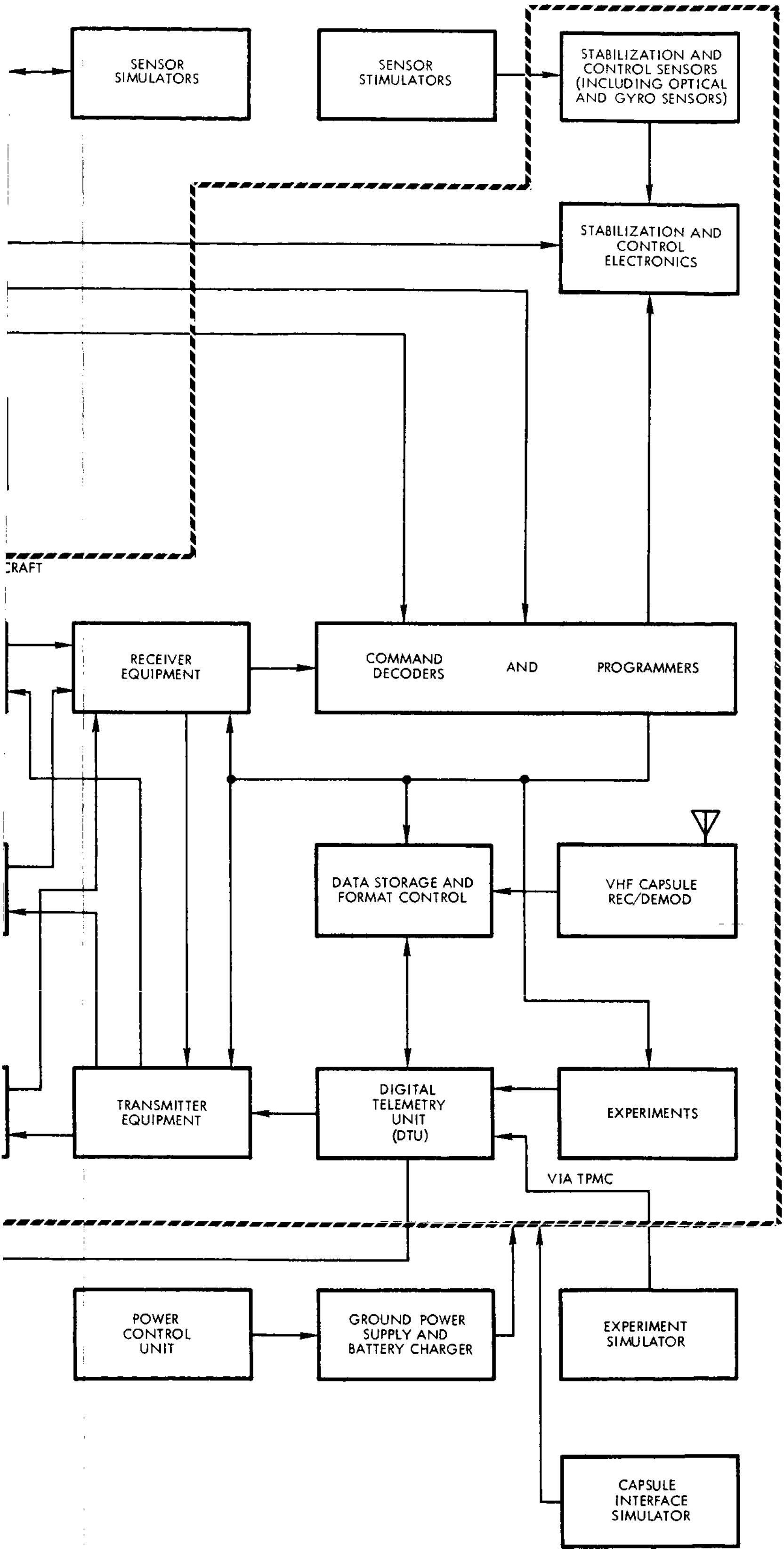


Figure 27. System Test Set, Block Diagram

The prime communication path between the system test set and the spacecraft is the RF link; subsystem performance evaluation is conducted largely by telemetry, although a few hardlines carry simulation and fault isolation signals (e.g., sun sensor simulation and command monitoring) which cannot be sent over the RF link. The set is self-tested by closed loop checking of the RF functions. Fault isolation to a replaceable unit in the set makes use of general purpose test equipment.

The automatic data handling system has four prime functions:

1) Real time processing of spacecraft data from both telemetry and hardline sources

2) Test sequencing

3) Displays of various kinds in formats meaningful to test personnel

4) Various off-line functions such as program generation and data reduction.

As shown in Figure 28 the system consists of an SDS-930 computer, manual input devices, and computer peripheral equipment. Growth capacity has been incorporated in the design of the equipment. For example, the system can handle double the present 4096-bit/sec telemetry rate.

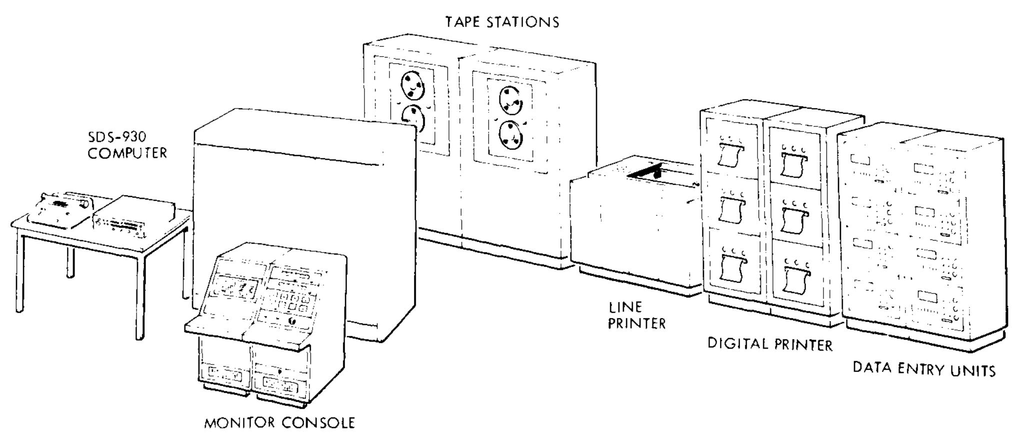


Figure 28. Automatic Data Handling System

2. LAUNCH COMPLEX EQUIPMENT

Support equipment for powering, monitoring, and recording during spacecraft preflight tests at the launch site includes all OSE in the spacecraft assembly facility, the explosive safe facility, the Centaur mating facility, the launch pad, and the blockhouse. Functionally, the launch complex equipment is identical with the test set and automatic data handling system which are part of the system test complex.

3. MISSION-DEPENDENT EQUIPMENT

The equipment and computer software needed to interface with the mission equipment at the DSIF are shown in Figure 29. This

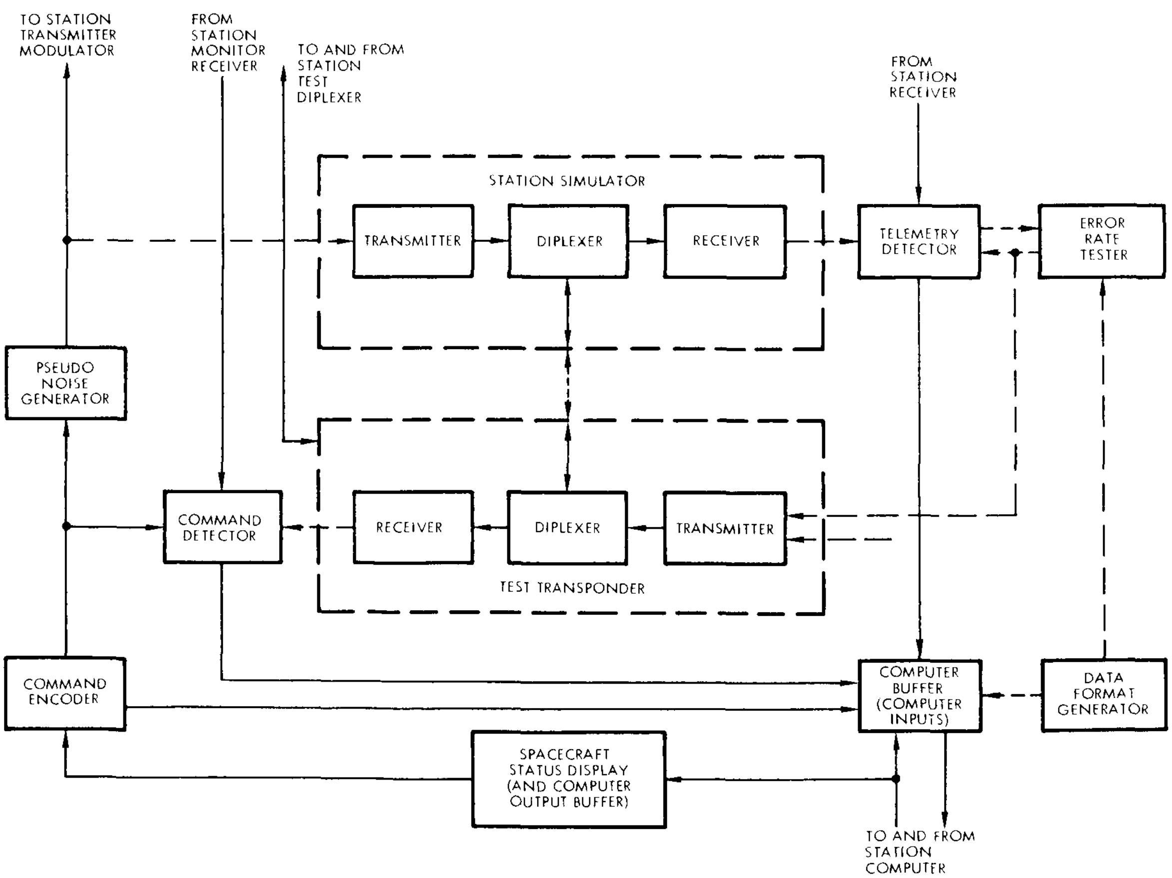


Figure 29. Mission Dependent Equipment, Block Diagram

equipment is designed to perform three sets of functions:

1) In-line functions essential to the DSIF link with the Voyager spacecraft, including command generation, telemetry detection, and computer buffering

2) In-line functions desirable for operations monitoring, such as command detection and verification and spacecraft status display

3) Functions related to compatibility testing, station readiness testing, fault isolation, maintenance, and calibration.

The command generation equipment includes redundant command encoders and pseudonoise generators. The command encoder provides for both computer-generated and manual commands to the spacecraft. Redundant telemetry detectors extract the telemetry bit stream and synchronization signals from a subcarrier of the spacecraft-to-ground link. The computer buffer transforms the telemetry data signal and its associated synchronization signals into a format acceptable to the station computer. The display panel (including malfunction alarms) indicates operational status of the Voyager spacecraft, transmits command verification and inhibit signals to the command encoder, and responds to system malfunctions sensed by the computer. A test transponder can simulate the RF portions of the spacecraft for compatibility and readiness testing, and station simulation equipment is included for testing when the actual station equipment is occupied with other missions. The data format generator simulates telemetry data for test of the combination of the telemetry detector, computer buffer, and computer.

The station computer is programmed to decommutate the Voyager telemetry data, to send telemetry data to teletype lines and to the mission dependent equipment, to make command checks, and to accept station time signals. A written copy of telemetry, command, and status data is provided by the computer typewriter.

4. ASSEMBLY, HANDLING, AND SHIPPING EQUIPMENT

Study of the assembling, handling, and shipping requirements for the spacecraft and planetary vehicle has shown that the equipment listed in Table 9 is needed. In addition to this system level equipment, standard mechanical support equipment items are also required for assembling, handling, and shipping the following subsystems: science payload, telecommunications, stabilization and control, structure, pyrotechnics, POP, and propulsion.

5. OSE FOR 1969 TEST FLIGHT

The test philosophy for the 1969 mission OSE is the same as for the 1971 in its effect on OSE design. Thus, unit test sets are heavily instrumented to support subsystem testing, and system test sets perform end-to-end testing through many subsystems in series. Moreover, the design characteristics of the OSE apply equally to the two missions except for provisions relating to the capsule and its OSE. Details of the 1971 OSE are given in Volume 6, of the 1969 OSE in Volume 7.

Table 9. Assembly, Handling and Shipping Equipment

	Use Location: Transportation	Subcontractor Plant	TRW Plant	Remote Test Sites	JPL	ETR
Transporter, spacecraft	x				x	
Assembly, handling, and tilt fixture, spacecraft			x		x	x
Transport recorder	x					
Weight and center of gravity fixture, spacecraft planetary vehicle			x			x
Shipping container, standardized modules	x	x	x	x	x	x
Work platforms, mobile		x	x			x
Adapter kit, Centaur shroud-transporter						x
Sling assembly, planetary vehicle and nose fairing						x
Purge unit, freon-ethylene oxide						x
Nose fairing mating and assembly fixture, planetary vehicle						x
Sling, capsule			x	x	x	x
Hoist sling, spacecraft		x	x	x	x	x
Tag lines						x
Launch stand access platforms						x
Universal mounting ring, spacecraft and planetary vehicle	x	x	x	x	x	x
Environmental cover, spacecraft	x				x	
Hoist sling, environmental cover		x	x	x	x	x
Platform auxiliary accessories			x		x	x

VI. 1969 TEST FLIGHT

1. 1969 TEST FLIGHT GOALS

Viewed as an engineering test for the Voyager spacecraft bus, the 1969 test flight would significantly enhance the probability of mission success in 1971. As such a test, the 1969 flight can serve three main purposes:

1) To test the 1971 spacecraft equipment in space under actual environmental stress conditions. Such a test would provide a realistic assessment of the spacecraft life under these environmental stress conditions and would allow for checking failure and redundancy modes of operation.

2) To develop the operational aspects of the Voyager program, with respect to the spacecraft bus, before the 1971 mission. These operational aspects include:

   - Crew training
   - Spacecraft assembly operations and test procedures
   - OSE, facilities checkout, computer programming and software checkout, and DSIF interface
   - Manufacturing experience in component procurement, parts screening, equipment fabrication
   - The development of an integrated team of Voyager personnel who are familiar with the idiosyncrasies of the spacecraft equipment and all of the many interfaces, including the working relationships with other Voyager Project teams.

3) To obtain environmental or other engineering data (such as Mars radiation levels, micrometeorite flux, atmospheric density, horizon characteristics, etc.) which are necessary for, or enhance the probability of, subsequent successful missions.

The principal limitation on the 1969 test flight arises from the reduced performance and restricted envelope associated with the Atlas-Centaur launch vehicle. Separated spacecraft weight for a Mars mission is of the order of 1400 to 1500 pounds, depending on the launch window, with additional margin available from the anticipated upgrading of the Atlas launch vehicle and from the later 1969 launch dates associated with extended transit times.

The major benefit, of course, to a 1969 test flight results from flying an identical spacecraft bus configuration on a Saturn-Centaur launch vehicle. Such an approach would be simpler from the spacecraft bus contractor's point of view, since identical drawings and checkout procedures could be used for the 1969 and subsequent launches. These factors, in turn, would alleviate some of the tightness in the 1969 launch schedule. In spite of these advantages, the fact that a Saturn-Centaur launch vehicle is not available does not argue against the utility of a 1969 flight. With the Atlas-Centaur, all three categories of objectives can also be attained.

There appear to be four alternatives for the 1969 test flight:

1) A Mars-orbiting mission

2) An earth-orbiting mission

3) An interplanetary mission

4) A Mars flyby.

The first alternative is precluded by the launch vehicle. The capability of the Atlas-Centaur is not great enough to launch a useful test spacecraft plus a retropropulsion motor to Mars, since the weight of the motor alone exceeds the payload capability to Mars.

An earth orbiter would be attractive if the retropropulsion motor could be tested in space after a delay equivalent to that experienced on a Mars trajectory. But here again the Atlas-Centaur is inadequate to launch a useful test spacecraft plus a full-scale retropropulsion engine. It is possible, however, to orbit a test spacecraft with a retropropulsion unit scaled down from that needed on the Mars mission, and this alternative is in fact a possibility for the 1969 engineering test flight.

The spacecraft could be dispatched on an interplanetary flight with no specific target. Like the earth orbiter, this approach lacks the aspect of the realistic rehearsal for a 1971 launch. The space environment that the spacecraft encountered would be reasonably representative of a Mars flyby except, of course, for the vicinity of Mars. Such an interplanetary flight would lack the realistic training simulation for the Voyager team in attempting to meet the narrow launch window of a biannual Mars opportunity; hence the engineering and operations team would not have rehearsed their test, launch, and flight procedures under the scheduled urgency of a limited launch opportunity.

Next to the orbiting mission, a Mars flyby most closely resembles the 1971 mission. Within the weight limits, the 1969 spacecraft bus on a Mars flyby could exercise authentic predecessors of all of the spacecraft subsystems (except the retropropulsion), including operational support equipment, software, and operating procedures and science interface.

A fifth possibility, of course, is not to launch in 1969 at all. The best argument for skipping a 1969 engineering test flight appears to be cost reduction. However, in terms of total program cost, over the entire series of Voyager missions, a well-executed 1969 engineering test would prove highly cost effective if it can provide significantly improved confidence of success in 1971 and subsequent years.

Of the alternatives, the 1969 flyby mission appears to do the best job of satisfying the three main objectives. It provides not only a test of the spacecraft equipment, but also exercises the operational aspects and can provide environmental information. The main elements that it does not check are the retropropulsion subsystem and the thermal and power problems associated with recurrent eclipses; however, it is possible that one Mars eclipse could be achieved during flyby, and programmed spacecraft reorientation during cruise can partially simulate recurrent eclipses.

The main disadvantages of the 1969 attempt are associated with the schedule constraints, particularly with the fact that time might not allow full implementation of MPC 200-2. Moreover, a modified parts-traceability program would be required and complete qualification of screened parts would probably not be achieved before the 1969 launch.

Based on the above reasoning, we propose the Mars flyby as the best choice for the 1969 engineering test flight. As discussed in the next section the design of the 1969 spacecraft, despite the limitations imposed by the smaller boost vehicle, is a close duplicate of the 1971 spacecraft. All elements are authentic predecessors and all elements can be tested, except for retropropulsion and the 1971 structure. The entire three panels of 1969 spacecraft electronic equipment, as opposed to the scientific equipment, are identical to the 1971 panels. Except for the quantity of propellant and number of tanks, the midcourse propulsion system is the same. Except for size, thermal control is the same. Power supply electronics are identical; the deployable solar panels use solar cell modules identical to those for 1971. The CS¢C subsystem is identical.

Moreover, the utility of the 1969 test flight is not limited to the configuration selected in this study. Studies of the applicability of the test flight were carried out with respect to all three reference configurations. As discussed in Volume 7, each of the 1969 test spacecraft corresponding to each reference configuration will adequately fulfill the three main objectives.

In our development plan the completion of the 1969 ground and flight test program is a major factor contributing towards improving the success of the 1971 mission. The ground test program begins to provide significant data on the performance of the spacecraft subsystems during the engineering model phase of the development of the 1971 spacecraft. Confidence is gained in terms of subsystem size, weight, power consumption, and interactions with other elements of the flight spacecraft.

Assembly and checkout of the 1969 test spacecraft will provide an opportunity to validate a large portion of the 1971 operational support equipment, assembly and checkout procedures, computer programs, and test facilities. 1969 launch and prelaunch operations will provide a means of rehearsing and validating much of the 1971 launch control equipment, checkout and on-stand operations, and terminal count procedures. As the 1969 flight progresses, data on the performance and survival of the subsystems will add to confidence in the success of the 1971 mission. Any failures that may be uncovered early in the 1969 flight will provide design data for application in the 1971 design. Problems occurring late in the 1969 flight will provide data that may be applicable for the 1973 spacecraft design and will bear on the launch decisions for the 1971 mission.

2. 1969 TEST SPACECRAFT DESIGN

To perform the maximum number of engineering tests bearing directly on the components of the selected design for the 1971 spacecraft, as many of these components as possible have been used in the 1969 configuration. Only minor differences exist in the communications, power, stabilization and control, and central sequencer and command subsystems. As has been mentioned, however, major differences exist in the propulsion and structure subsystems.

To permit a better approximation to the 1971 spacecraft, the Atlas-Centaur fairing has been lengthened by 42 inches. As shown in Figure 30, the 1969 spacecraft bus has four sides which are used as the equipment mounting panels; these are identical to their 1971 counterparts. Moreover, similarly to the 1971 model, four corner longerons and upper and lower frames attach the equipment mounting panels to the bus. The panels are hinged, as on 1971, for access to the bus interior. The provisions for equipment modularization and thermal control are identical. Upper and lower thermally insulated truss-core sandwich panels complete the meteoroid protection of the bus.

At the aft end of the spacecraft is the same double-gimballed high-gain antenna as used on the 1971 configuration. The same low-gain antenna is also carried. An additional low-gain antenna is installed on

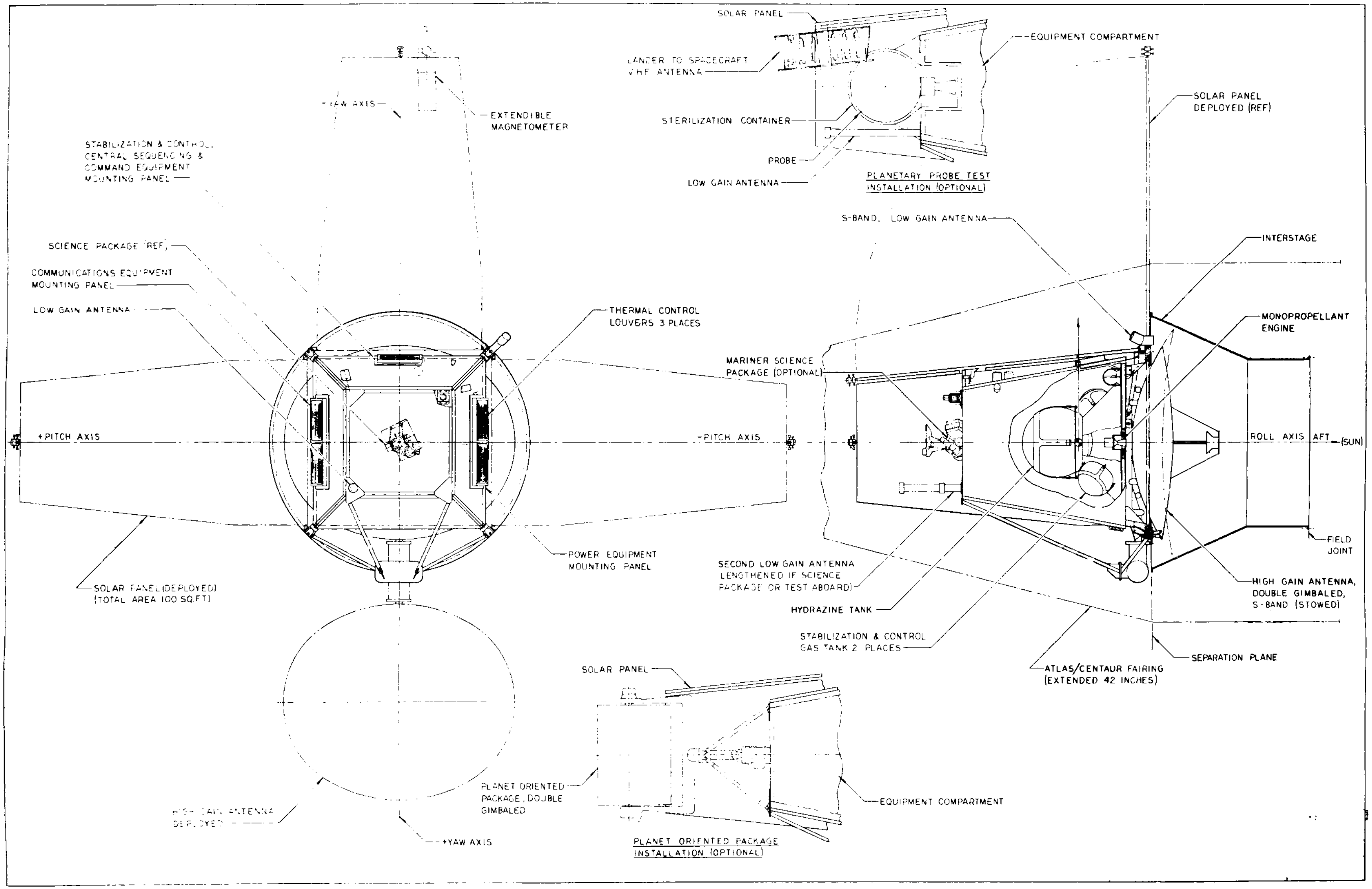


Figure 30. 1969 Test Spacecraft

the forward end of the spacecraft to allow complete testing of all operating modes of the communications subsystem (the 1971 spacecraft uses high-, medium-, and low-gain antennas connected to redundant communications equipment). Communications electronics are identical for the two spacecraft.

For midcourse propulsion, the 1969 test spacecraft uses the same monopropellant engine, one of the same pressurized propulsion tanks (off-loaded to 45 pounds of propellant) and identical valves and plumbing as on the 1971 configuration. The evasive maneuver propulsion package (used to translate the 1971 spacecraft out of the way of the capsule) is carried on the 1969 test vehicle. Although the solid retropropulsion engine is not carried on the 1969 test spacecraft, it is possible to carry and partially test its liquid injection equipment.

Except for the reduced size of the tankage and the relocation of some low-thrust nozzles, the stabilization and control system for 1969 is the same as on 1971. The sensors associated with the stabilization and control system are mounted in a way similar to the 1971 installation. As on the 1971 configuration, the aft cover and the propulsion and stabilization and control systems located in the bus are modularized for ease of assembly, installation, and test. Since the equipment mounting panels and the thermal-control louvers are those of the 1971 spacecraft, the thermal control equipment can also be tested in 1969.

Three deployable solar panels, adjacent to three of the equipment mounting panels, utilizing 1971 solar cell modules and circuitry, are sized for 100 square feet of solar cells. Except for the solar array, the remainder of the power supply matches the 1971 configuration. The fourth side of the spacecraft has been left clear to permit adequate look angles for the double-gimballed, high-gain antenna.

The Mariner science package has been shown in phantom on the forward end of the spacecraft in Figure 30 to indicate the 1969 test spacecraft's potential for data gathering as a Mars flyby. Partial views shown on the figure also indicate the possibility of using the spacecraft to deliver an atmospheric probe to Mars, or to test an off-loaded or complete POP with its Mars horizon sensor. In addition to these options, it is possible to fly a VHF antenna and associated propagation experiment or the thrust-vector control system. Which option is chosen depends upon which of these spacecraft design areas is determined to be the more critical.

3. OPERATIONAL SUPPORT EQUIPMENT FOR 1969

As noted above, the majority of the subsystems for the 1969 test spacecraft are identical to their 1971 counterparts. Exceptions exist in the areas of retropropulsion, structure, and interface with the science payload. Because of this similarity, the OSE design for each of these programs can be basically the same. The system-level electrical OSE to support system-level testing of the 1969 test spacecraft includes the system test set used for integrated system testing; the automatic data handling system to support the systems tests; the launch complex equipment; and the mission dependent equipment. Except for minor differences in panel details, all of this system-level electrical OSE is identical for both the 1969 and 1971 Voyager missions.
1969 and 1971 Voyager missions.

Changes in the structure and elimination of the solid-propellant retro-engine necessitate some changes in the mechanical OSE for 1969. However, as noted in Table 10, the assembly, handling, and shipping equipment required for 1969 is for the most part similar to its 1971 counterpart.

As mentioned in the discussion of 1971 OSE in Section V above, the OSE consists of both system level test equipment and subsystem test equipment. The electrical subsystem test equipment, called unit test sets, for the 1969 test flight will consist of:

Table 10. 1969/1971 Mechanical OSE Comparison Matrix (Assembly, Handling, and Shipping Equipment)

1971 Equipment	1969 Application of 1971 Equipment	1969 Functional Require-ments	1969 Adaptation			1969 All New Equip. Req'd.
			Use As is	Add Mod Kit	Modify Equip.	
Transporter, flight spacecraft	Yes	Same	Yes	Yes	-	-
Assembly, handling, and tilt fixture	Yes	Same	Yes	Yes	-	-
Transport recorder	Yes	Same	Yes	-	-	-
Fixture, weight, c. g. and MDI	Yes	Same	-	Yes	Yes	-
Shipping container group, standard modules	Yes	Same	Yes	-	-	-
Work platforms, mobile	Yes	Same	Yes	Yes	-	-
Adapter kit, Centaur/shroud transporter	No	None	-	-	-	-
Sling assembly, planetary vehicle and nose fairing	No	None	-	-	-	-
Purge unit, freon/ethelene oxide	No	None	-	-	-	-
Planetary vehicle/nose fairing mating and assembly fixture	No	None	-	-	-	-
Sling, flight capsule	No	None	-	-	-	-
Hoist beam and slings, flight spacecraft	No	Same	-	-	-	Yes
Tag lines	Yes	Same	Yes	-	-	-
Platform, launch stand access	No	Possibly	-	-	-	Possibly
Universal mounting ring, flight spacecraft and planetary vehicle	No	Same	-	-	-	Yes
Environmental cover, flight spacecraft	No	Same	-	-	-	Yes
Hoist sling, environmental cover	Yes	Same	Yes	-	-	-
Platform, auxiliary access	Yes	Same	Yes	-	-	-
Transporter adapter cradle, 1969 test spacecraft	-	1969 Only	-	-	-	Yes

- Four telecommunication subsystem unit test sets
- Five stabilization and control subsystem unit test sets
- A central sequencer and command unit test set
- Five power subsystem unit test sets
- An electrical assembly subsystem unit test set.

Except for minor differences in panel details, all of these subsystem unit test sets are identical for the 1969 and 1971 missions.

Because of the different structural arrangement and design for the 1969 test spacecraft and the need for deployable solar panels, about half of the subsystem mechanical support equipment is new equipment for 1969. The remainder is either not required or will be the same as its 1971 counterpart.

VII. IMPLEMENTATION

1. INTRODUCTION

The implementation plan for the Voyager spacecraft, discussed in Volume 3, encompasses Phase IB design engineering and the complete cycle of development and operations in Phase II. The policy used in generating the schedules and task descriptions has been that the 1969 flight test effort is an integral portion of the development cycle for the 1971 mission. To this end, the ground rule for the design of the 1969 spacecraft has been to retain a one-to-one identity with the elements of the 1971 spacecraft, within the constraints imposed by the difference in launch vehicles and the absence of scientific objectives.

2. SCHEDULES

2.1 Phase IB

The major efforts during Phase IB involve the system and subsystem engineering leading to a clear definition of system and subsystem design requirements and interfaces by the seventh week. The ensuing five weeks is used to prepare preliminary design concepts in accordance with the design requirements. Thus at the end of the twelfth week, a design review is programmed, to verify that all requirements are defined and that the design approach is satisfactory.

The effort following this review includes the detailed design of both the 1969 and 1971 spacecraft systems culminating in a second design review, scheduled for the 28th week. The material to be reviewed at this time includes:

- Detailed layout and schematics
- Material and parts
- Equipment and process specifications
- Development test results
- Reliability data

- Weight, volume and power requirement
- Technical work statements (including engineering model test plans)
- Management plans and controls
- Operations plan.

The completion of this second design review provides approval for the release of the system, subsystems, and OSE specifications and also the various management and operational plans. The Phase II detailed work package and cost plan is submitted for JPL approval within three weeks after this design review.

2.2 Phase II Schedule

The major milestone schedule for the combined 1969, 1971, 1973 effort is presented in Figure 31 and the highlights of the task flow for the assembly, test, and launch operations scheduled for Phase II are shown in Figure 32. Scheduling is based on pacing the in-line operations backwards from the launch date through the required time spans for type-approval fabrication and test to the drawing release date. Satisfactory phasing among the three programmed spacecraft is apparent in terms of the facilities and manpower loading.

2.2.1 1969 Test Flight Schedule

For the 1969 test flight the normal pacing of events can lead to considerable overlap of type approval tests and the fabrication of flight equipment unless the drawing release date is moved forward. The imposition of an earlier drawing release date, however, incurs the risk of more design changes. In the light of these conflicting requirements, a compromise was selected which favored the design and development cycle (e.g., later drawing release) at the expense of some concurrency of the subsystems type approval tests and the fabrication cycle of flight units. This concurrency can be kept within tolerable limits, it is felt, by the acceleration of design effort during Phase IB, resulting in an earlier drawing release cycle. Other possible critical areas, as discussed in Section II-4 of Volume 3, are listed below:

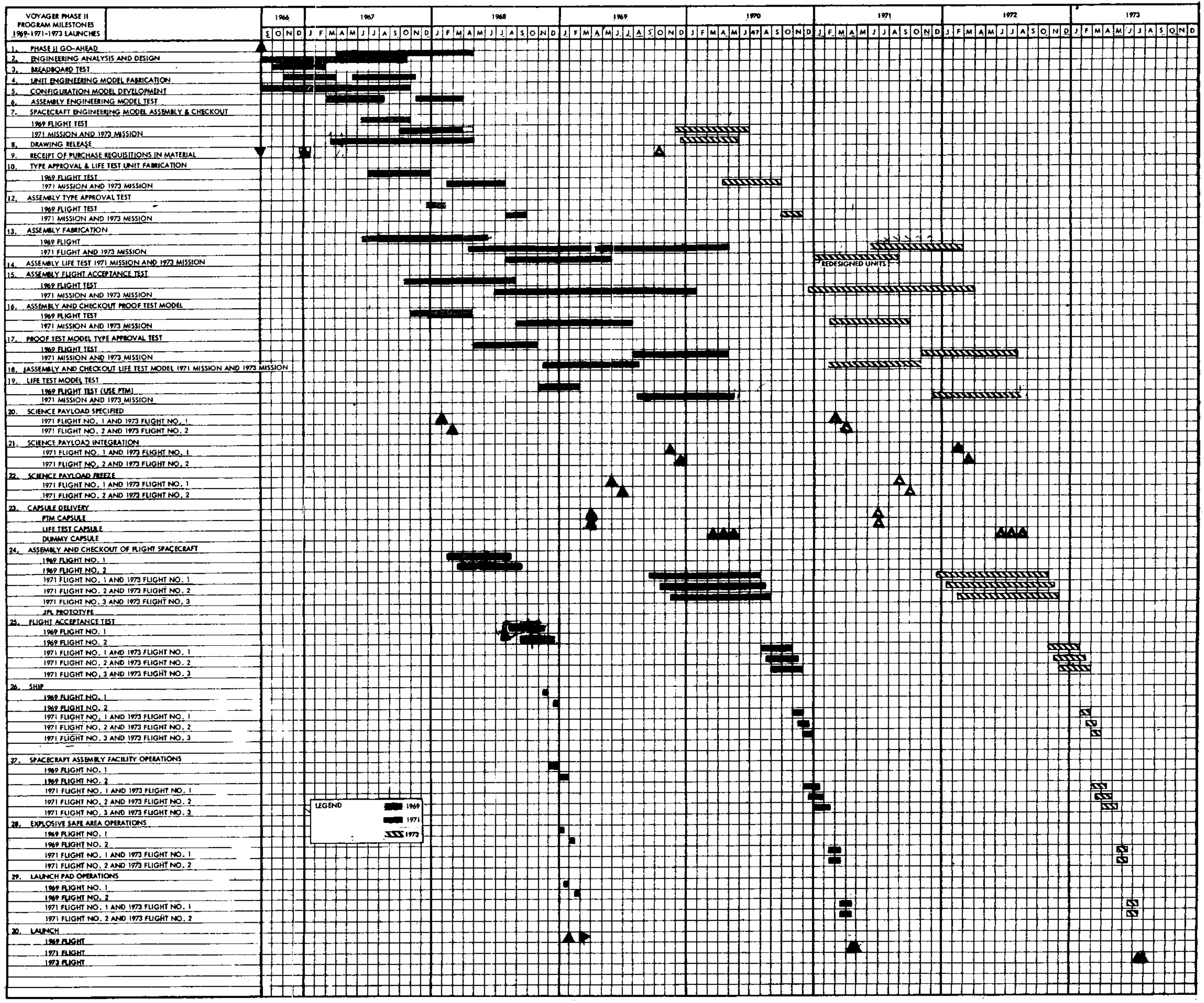


Figure 31. Voyager Phase II Program Milestones, 1969-1971-1973 Launches

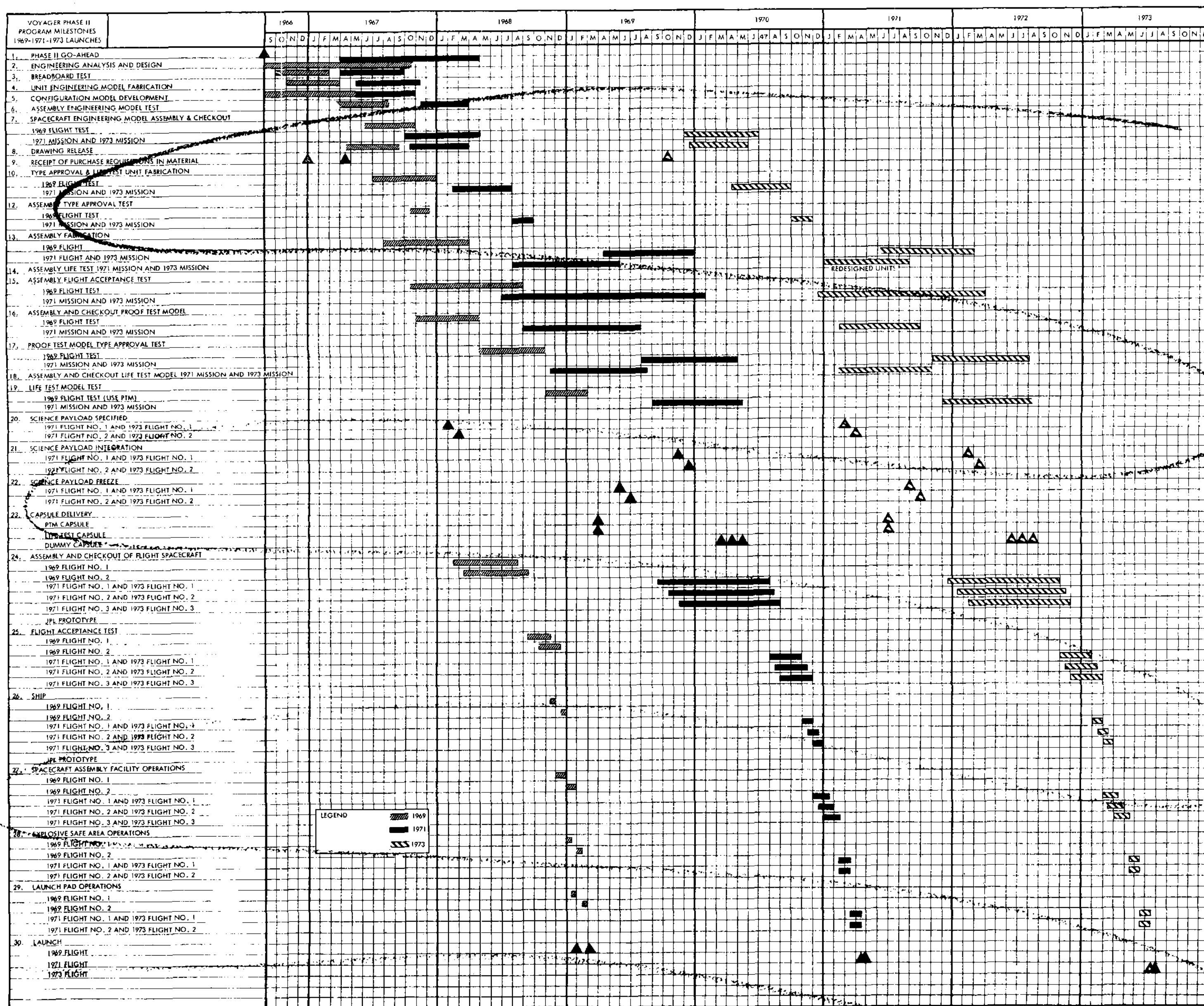
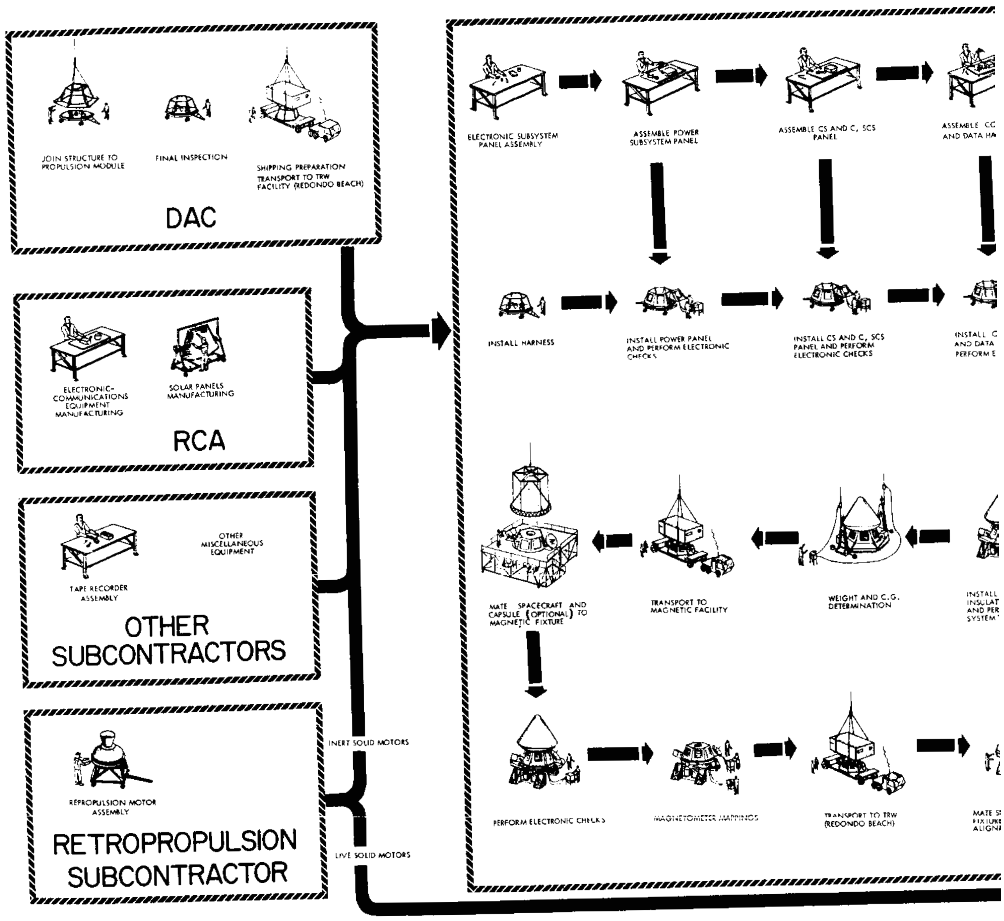
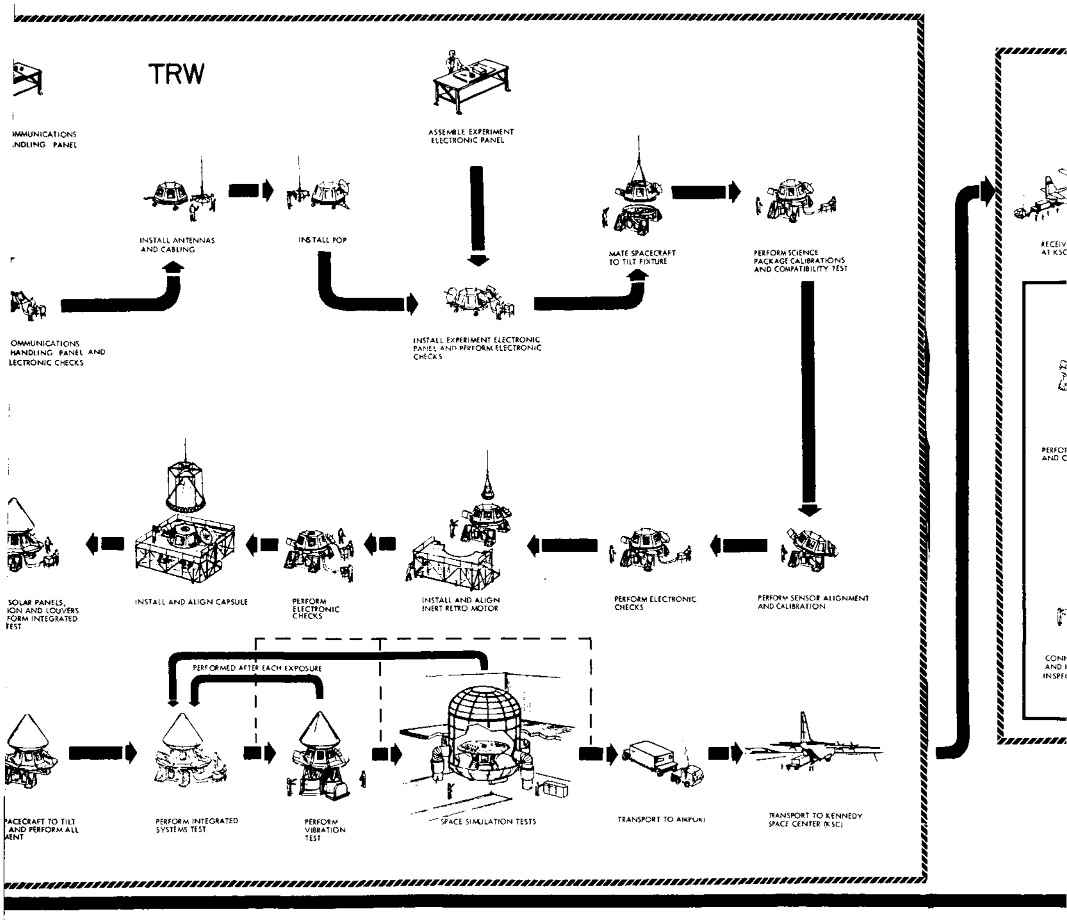


Figure 31. Voyager Phase II Program Milestones, 1969-1971-1973 Launches



JOIN STRUCTURE TO PROPULSION MODULE
FINAL INSPECTION
SHIPPING PREPARATION
TRANSPORT TO TRW FACILITY (REDONDO BEACH)
DAC
ELECTRONIC-COMMUNICATIONS EQUIPMENT MANUFACTURING
SOLAR PANELS MANUFACTURING
RCA
TAPE RECORDER ASSEMBLY
OTHER MISCELLANEOUS EQUIPMENT
OTHER SUBCONTRACTORS
REPROPULSION MOTOR ASSEMBLY
RETROPROPULSION SUBCONTRACTOR
INERT SOLID MOTORS
LIVE SOLID MOTORS
ELECTRONIC SUBSYSTEM PANEL ASSEMBLY
ASSEMBLE POWER SUBSYSTEM PANEL
ASSEMBLE CS AND C, SCS PANEL
INSTALL HARNESS
INSTALL POWER PANEL AND PERFORM ELECTRONIC CHECKS
INSTALL CS AND C, SCS PANEL AND PERFORM ELECTRONIC CHECKS
MATE SPACECRAFT AND CAPSULE (OPTIONAL) TO MAGNETIC FIXTURE
TRANSPORT TO MAGNETIC FACILITY
WEIGHT AND C.G. DETERMINATION
PERFORM ELECTRONIC CHECKS
MAGNETOMETER MAPPINGS
TRANSPORT TO TRW (REDONDO BEACH)



TRW
ASSEMBLE EXPERIMENT ELECTRONIC PANEL
INSTALL ANTENNAS AND CABLING
INSTALL POP
MATE SPACECRAFT TO TILT FIXTURE
PERFORM SCIENCE PACKAGE CALIBRATIONS AND COMPATIBILITY TEST
INSTALL EXPERIMENT ELECTRONIC PANEL AND PERFORM ELECTRONIC CHECKS
INSTALL AND ALIGN CAPSULE
PERFORM ELECTRONIC CHECKS
INSTALL AND ALIGN INERT RETRO MOTOR
PERFORM ELECTRONIC CHECKS
PERFORM SENSOR ALIGNMENT AND CALIBRATION
PERFORMED AFTER EACH EXPOSURE
PERFORM INTEGRATED SYSTEMS TEST
PERFORM VIBRATION TEST
SPACE SIMULATION TESTS
TRANSPORT TO KENNEDY SPACE CENTER (KSC)

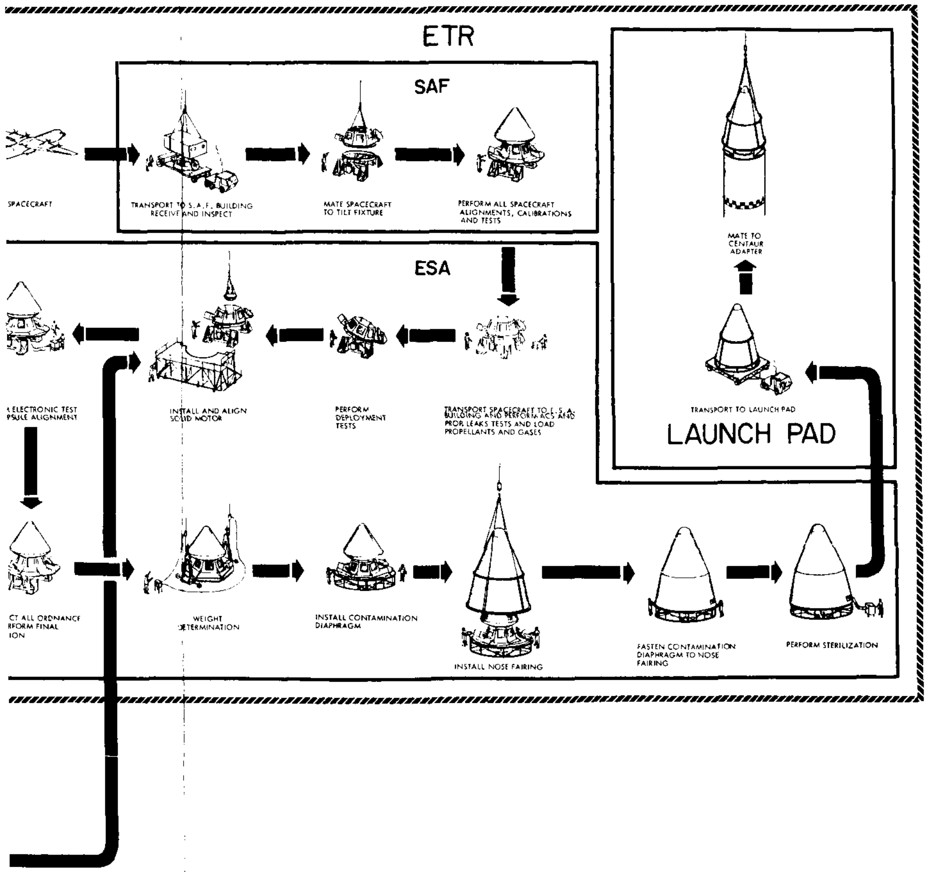


Figure 32. Assembly and Test Flow

3

2.2.2 1971 Mission Schedule

The time available from the start of Phase II to the start of manufacturing of the 1971 flight units clearly allows a degree of freedom not contained in the 1969 schedule. There are two basic choices of how to best use the available time. One choice would be to delay the 1971 drawing release date sufficiently to allow any 1969 ground test results to be included in the 1971 design. This approach then would require that a series of test models immediately precede the start of fabrication of the flight units. The other choice is to continue the design effort from the end of the 1969 design effort and release the final 1971 drawings as soon thereafter as possible.

This second approach is preferred since it allows the 1971 type approval, life test, and proof test model units to be fabricated at an early date, allowing these units to accumulate a significant test history prior to fabrication of the flight units. This approach still allows for any design adjustments that may result from the 1969 test program.

The 1971 mission schedule has no critical schedule areas in the development cycle. The drawing release cycle occurs during late 1967 and early 1968, thus providing a development time of approximately 24 months from Phase IB start or 16 months from Phase II start. This time is more than adequate, particularly since much of the 1971 design is identical so that for 1969. The subsystem fabrication and type approval cycle in fact allows 7 months for design adjustment if needed before fabrication of the flight hardware begins. The start of flight fabrication is so placed as to allow for the inclusion of the 1969 test results up to and including the early portions of the test flight as well as the results of the 1971 subsystem life testing.

In the case of a failure in the 1969 test flight, there is still sufficient time to include changes in the 1971 spacecraft as late as 14 months after 1969 launch. A failure during type approval testing of the 1971 proof test model spacecraft is most likely to occur during vibration or space simulation testing; this portion of the tests is

a) Parts. The requirement to retain a one-to-one identity wherever possible between the 1969 test flight and the 1971 mission requires that identical parts be used for both. Procurement of magnetically acceptable high reliability parts dictates a long lead time effort. To circumvent any problem, TRW recommends that an approved parts list be negotiated early in Phase IB from which the designs must be selected and that deviations to this list be identified during Phase IB testing in order to initiate a special effort to qualify such parts. In addition it is recommended that an early release be negotiated for long lead time parts.

b) Structure. The need for an early structural vibration and static load test imposes a requirement to provide detailed structural layouts during Phase IB to enable early fabrication and test.

c) Midcourse Propulsion System. To meet the 1969 schedule requirements for a completely tested midcourse propulsion system, it is necessary to begin fabrication and test of the development and prototype midcourse engines in June of 1966.

d) Stabilization and Control. The long lead time procurement of the gyro reference assembly represents a possible critical area in the stabilization and control system. It is planned to initiate this procurement early in Phase IB to ensure delivery of this assembly for engineering model tests and subsequent spacecraft.

e) Communication and Data Handling. The critical equipment in the communications and data handling subsystem includes the development of a three-speed tape recorder and the prototype antenna gimbal drives. The fabrication and test of an engineering model tape recorder with breadboard electronics will be provided during Phase IB. Prototype models of the gimbal drives will also be fabricated and tested.

f) Power. The critical factor in the development of the power subsystem is the design of the solar array for the low temperature condition. This requires that Q-boards of solar panel segments be fabricated and tested over extremes of temperature during Phase IB.

completed by the end of December 1969, allowing approximately 6 months to include design refinements. The 1971 life test model is scheduled to enter life test in August of 1969 and could therefore proceed as long as 8 months before a detected failure would pose a 1971 launch schedule problem.

3. EFFECTS OF THE 1969 TEST FLIGHT ON THE 1971 MISSION

As has just been suggested the completion of the 1969 subsystem type approval testing provides for high confidence in the proper functioning under severe environment conditions and verifies the procedures and processes used in the manufacturing phase. Failures uncovered during this test phase are useful in correcting design deficiences in the 1971 hardware.

An additional and important test benefit is provided by the 1969 ground test program in terms of providing reliability data on parts, subsystems, and systems. Life testing of the 1969 proof test model spacecraft (see Section IV, Volume 3) will add to the confidence in the ability of the subsystem designs to survive the expected life requirements.

The conduct of the 1969 test flight effort also provides additional confidence in the success of the 1971 mission in the following areas:

a) <u>Crew Training.</u> The assembly, checkout, test and launch crews will receive experience in the conduct of their respective operations. The conduct of the engineering model and proof test model interface tests assist in training at the Deep Space Network, SFOF, and Mission Support centers.

b) <u>Procedure and Computer Program Checkout.</u> A large portion of 1969 test procedures and computer programs will be directly applicable to the 1971 mission. The 1969 test effort provides an opportunity for their real time validation.

c) <u>OSE Checkout.</u> A great deal of the OSE used in the 1969 effort is identical to that used for the 1971 mission, and an early opportunity is afforded to validate this equipment and to improve its design.

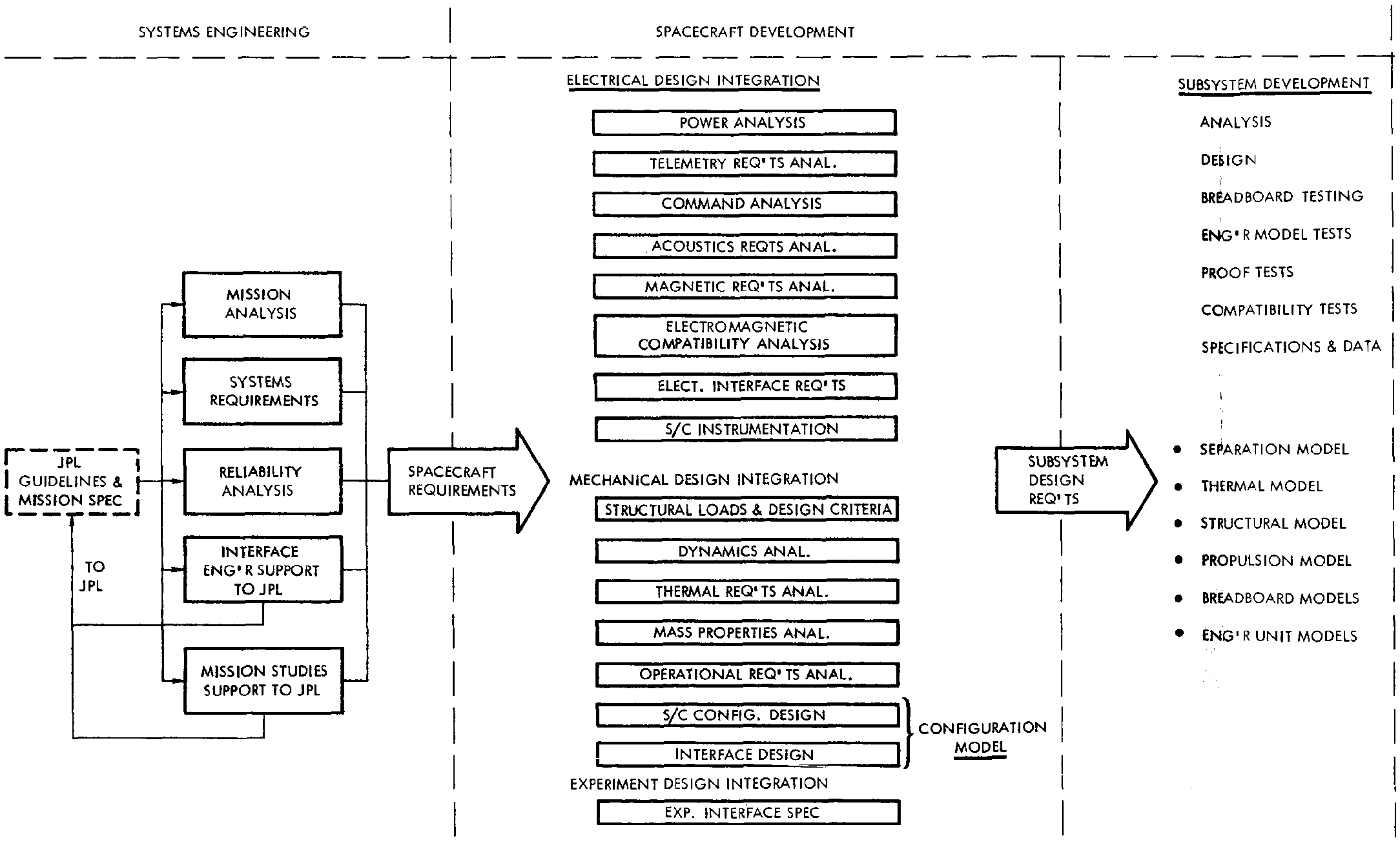
d) Test Facility Checkout. It is planned to use the same test facilities for the 1969 test flight spacecraft as for the 1971 mission spacecraft. The use of the 1969 equipment in these facilities will provide a high confidence in their design and operations.

e) Manufacturing Checkout. The identical designs of much of the equipment fabricated for both the 1969 and the 1971 programs provides a checkout of the manufacturing processes, assembly, lines, test equipment, and software controls. This will contribute to the confidence in fabricating high quality 1971 equipment and on-schedule performance. The qualification of the various vendors and subcontractors will be verified.

f) Schedule Confidence. The performance of the 1969 program provides high confidence through learning in performing to the 1971 schedule.

4. SPACECRAFT DEVELOPMENT

Development of the Voyager spacecraft will begin with system engineering tasks and extend to the conduct of launch and mission support operations as depicted in Figure 33 and as discussed in Section V of Volume 3. The flow of tasks begins with the definition of spacecraft requirements resulting from analyses and studies at the mission level by the systems engineering group. These requirements are then converted into subsystem design requirements by a series of design integration studies. Subsystem development, including breadboards and models, leads to the release of manufacturing drawings. Manufacturing then proceeds and subsystems are acceptance tested and assembled into the spacecraft. Each spacecraft undergoes flight approval testing prior to its shipment to the launch site.


SYSTEMS ENGINEERING
SPACECRAFT DEVELOPMENT
JPL GUIDELINES & MISSION SPEC
TO JPL
MISSION ANALYSIS
SYSTEMS REQUIREMENTS
RELIABILITY ANALYSIS
INTERFACE ENG'R SUPPORT TO JPL
MISSION STUDIES SUPPORT TO JPL
SPACECRAFT REQUIREMENTS
ELECTRICAL DESIGN INTEGRATION
POWER ANALYSIS
TELEMETRY REQ'TS ANAL.
COMMAND ANALYSIS
ACOUSTICS REQTS ANAL.
MAGNETIC REQ'TS ANAL.
ELECTROMAGNETIC COMPATIBILITY ANALYSIS
ELECT. INTERFACE REQ'TS
S/C INSTRUMENTATION
MECHANICAL DESIGN INTEGRATION
STRUCTURAL LOADS & DESIGN CRITERIA
DYNAMICS ANAL.
THERMAL REQ'TS ANAL.
MASS PROPERTIES ANAL.
OPERATIONAL REQ'TS ANAL.
S/C CONFIG. DESIGN
INTERFACE DESIGN
CONFIGURATION MODEL
EXPERIMENT DESIGN INTEGRATION
EXP. INTERFACE SPEC
SUBSYSTEM DESIGN REQ'TS
SUBSYSTEM DEVELOPMENT
ANALYSIS
DESIGN
BREADBOARD TESTING
ENG'R MODEL TESTS
PROOF TESTS
COMPATIBILITY TESTS
SPECIFICATIONS & DATA
SEPARATION MODEL
THERMAL MODEL
STRUCTURAL MODEL
PROPULSION MODEL
BREADBOARD MODELS
ENG'R UNIT MODELS

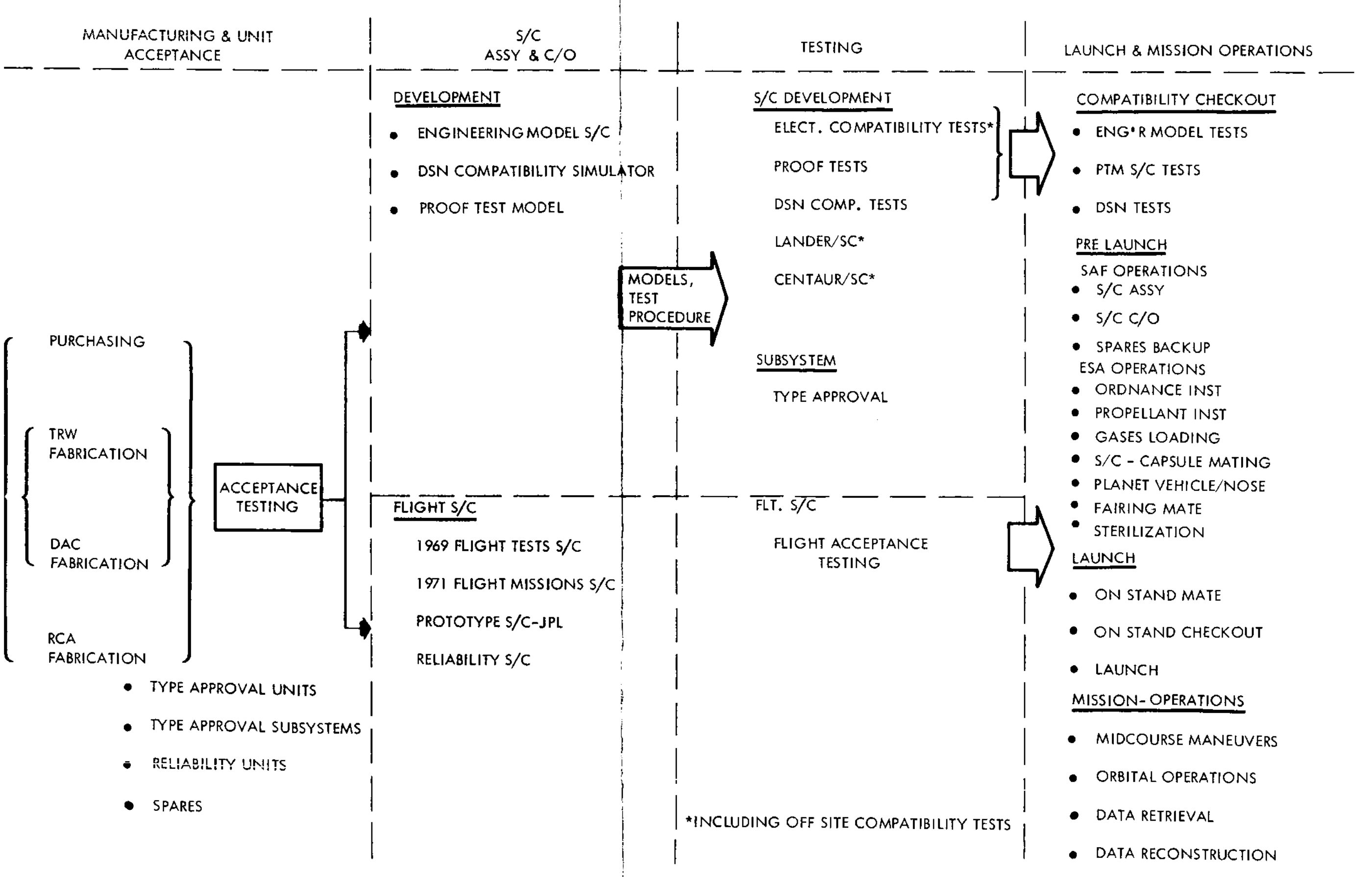


Figure 33. Voyager Spacecraft Implementation

AUG 12 1965

SIGNIFICANT ERRATA. TRW Systems, Phase 1A
Study Report, Voyager Spacecraft
August 11, 1965

N66-21047

Volume 1. Summary

Substitute new p. 79 attached.

Volume 2. 1971 Voyager Spacecraft

p. 18. Item h) "necessary landed operations" should read "necessary lander operations."

p. 143. Section 3.4.1.a. second line should read "threshold of 0.25 gamma"

p. 282. Lines 3 and 4. Delete "or incorrect spacecraft address"

p. 284. Figure 5. Change "128 Word DRO Core Memory" to "256 Word DRO Core Memory"

p. 32 . Denominator of second term on right hand side of equation should read

$$\left(\frac{1}{\epsilon_1} + \frac{1}{\epsilon_2} - 1\right)\left(N - 1\right)$$

p. 351. Figure 1, Section F-F. "separation nut" should read "bolt catcher"

Volume 3. Voyager Program Plan

Substitute new p. 12 attached.

p. 13. Figure 2-3. PTM Assemblies in item 7 move 1.5 months to right

p. 16. Figure 2-6. First milestone date should be September 1, 1969, instead of mid-January 1970, and all subsequent dates should be correspondingly adjusted 4.5 months earlier.

p. 20. Table 2-2. Third item in 1969 column should read "coincident with completion of proof test model assemblies. Fifth item in this column change "2 weeks" to "3.5 months." Fourth item in 1971 column, change "4 months" to "5 months."

p. 67. Figure 5-2. Under Intersystem Interface Specification add a block entitled "Spacecraft to OSE Interface Specification"

p. 120. Last line of paragraph c should read "shown in Table 5-2."

p. 126. Figure 5-13. Year should be 1966 instead of 1965.

p. 153. Figure 5-18. Ignore all numbers associated with lines in figure.

p. 167. Figure 5-[illegible]. In line 20 change "design revisions" to "design reviews"

p. 254. Second paragraph, third line, "The capability of the transmitter to select" should read "The capability of the transmitter selector to select."

p. 258. Section heading n should read <u>Experiment Data Handling</u>

p. 604. Section 3.2.1 beginning of second paragraph should read "The hydrazine fuel . . . "

Volume 4. <u>Alternate Designs: Systems Considerations</u>

p. 103. Figure 3-19. Caption should read "Radial Center of Mass. . . "

p. 151. Last paragraph, second line, "For the baseline, the reliability. . ." should read "The reliability . . . "

p. 158. 8th line, replace "0.06 pound/watt" by "0.6 pound/watt"

p. 215. Figure 3-50. Dot in ellipse at right should be 0.

p. 230. Section 5.3.2, second paragraph, 7th line, should read "Figure 3-52."

p. 239. Second line, "with a variable V" should read "with a variable ΔV"

p. 247. First line, "3250 km/sec" should read "3.250 km/sec"

p. 261. Figure 3-64. Interchange coordinates, clock angle and cone angle

p. 293. Figure 3-81. An arrow should connect "Low-gain spacecraft antenna" and the dashed line at 73×10^6 km

Volume 4. <u>Alternate Designs: Systems Considerations Appendix</u>

p. [illegible]6. Figure A-2. The shaded portion under the lower curve should extend to the right only as far as 325 lb.

p. 9. Table A-1, part (1). In last column heading change "W_3" to "W_1". In part (4) last column heading change "W_3" to "W_4"

p. 22. Second line below tabulation, replace "575 × 35" by "570 × 35"

p. 29. Tabulation at bottom of page, change "18" to "30" and "400" to "240"

p. 207. Numerator of equation for λ best at bottom of page should read "0.0201," and numerator of equation for λ worst should read "9.21"

p. 209. Table 5B, fifth line. Delete "$\times 10^{-}$." Also p. 213, Table 7A, seventh line, and p. 232, Table 3B, fifth line.

p. 217. Top portion of Table 9B should be labeled "primary mode" instead of "other modes"

p. 326. In equations following words "clearly" and "thus" insert "$>$" before second summation.

Volume 5. Alternate Designs: Subsystem Considerations

p. 3-15 Fifth line, "... is extended, spacecraft" should read "... is extended, two spacecraft"

p. 3-38 Last line, change "$= \frac{32}{4500} = M$" to "$\left(\frac{32}{4500}\right)(M)$"

p. 3-51 Two equations at bottom of page should read

$$D = 4\pi A/\lambda^2$$

$$A = \frac{D\lambda^2}{4\pi} = \frac{1000\,\lambda^2}{4\pi}$$

p. 3-67 Third line, last parenthesis "$\left(\frac{\pi}{2} + \phi\right) -$"

p. 3-82 6th line should read "50 degrees" instead of "50-140 degrees," and seventh line should read "140 degrees" instead of "50-140 degrees"

p. 3-111 Last line, change "50 Mc" to "1 Mc"

p. 3-137 Item g) for "... followed by 5 frames of real time" substitute "... followed by 11 frames of low rate science data and 5 frames of real time" pp. 3-150 and 3-151 are interchanged.

p. 3-156 Last line, should read "gates, a 7 bit"

p. 5-21 Second paragraph, third line, for "others since they are" substitute "others which are"

p. 5-33 Bjork equations should identify 0.18 as an exponent, and the exponent for (ρ_p/ρ_t) in the Hermann and Jones equation should be 2/3 in both cases.

p. 5-33 Figure 5-12 should be replaced with Figure C-7 of Appendix C.

p. 5-40 Three lines above Table 5-10 substitute "permanent set" for "experiment"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix I

p. B-11 Bottom of page, for "$r^{2/3}$" substitute "$(V/C)^{2/3}$ r"

p. C-4 The title of Figure C-2 should read "Figure C-2. Meteoroid Influx Rate Circular Orbit Mars", and the title of Figure C-3 should read "Figure C-3. Meteoroid Influx Rate Cruise"

p. C-5 At bottom of page, add the following: "*Within 50,000 km of Mars"

p. C-6 Line 13 should read: "... of low density ($\rho_p < 2.4$ gm/cm^3..."

p. C-6 Figure C-4. The ordinate "2" should read "100"

pp. C-17 C-21 The figures C-6 and C-7 on pages C-17 and C-21 should be reversed.

p. C-28 The title of Figure C-8 should read "Meteoroid Shield Test Specimen"

p. C-29 The title of Figure C-9 should read "Cutaway of Meteoroid Shield Test Specimen

p. C-34 In Section 1.8 the first sentence should be replaced by the following two sentences: "Preceding sections of this appendix contain derivations of the probability of penetrations of the spacecraft outer skin by meteoroids. It is clear that to design an outer skin of sufficient thickness to reduce the probability of no penetrations to a low level, such as 0.05 to 0.01, would be prohibitive in terms of the weight required."

p. C-35	In the first equation, the expression "(t in m^2)" in two places should read "(t in cm)" and "A" in two places should read "(A in m^2)"
p. C-38	In Table C-2, all values in inches should be in centimeters. A zero should be inserted immediately following the decimal point, for example: (0.020-inch) = 0.05080, (0.020-inch) = 0.06096, (0.020-inch) = 0.04064, etc.
p. C-40	In Section 1.8.7 Computation of R_i's, the sixth line should read "... than 10^6 are neglected"
p. C-45	In listing under "Values of t Used for Extreme Environment Analysis," under Inch, the first number should read 0.020 instead of 0.202
p. C-52	In 1.10 NOMENCLATURE, "K_2" should be defined as "$K^{-2/3}(4 \pm 2)$" and "B" should be $\frac{1000\,\rho_t V^2}{9.806\,H_t}$

pp. C-150 and C-151 should be reversed.

p. C-208	Along the ordinate in the graph, "Stress $\times 10^{-3}$" should read "Stress $\times 10^{-2}$"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix II

p. F-23	Lines 7 and 10 change all subscript τ to T
p. F-24	Line 14, change "ME_1" to "mE_1"
p. F-29	Figure F-9 title should be "Reflection Phase Angle ϕ (deg)" and Figure F-10 title should be "Reflection Magnitude R"
p. F-30	Last line, change "0.27" to "0.175"
p. F-31	Lines 14 and 15, change "14,700 ft/sec to 460 ft/sec" to 14,700 ft/sec minus 460 ft/sec" and "14,700 ft/sec to 10,000 ft/sec" to "14,700 ft/sec minus 10,000 ft/sec"
p. F-32	Last line in item 4), change "27 per cent" to "17.5 per cent"
p. F-35	Table F-4, under Assumed Parameter for item 2 insert "$\pm 2 \times 10^{-5}$", for item 3 insert "$\pm 3 \times 10^{-5}$", and for item 4 insert "$\pm 2 \times 10^{-5}$"

p. F-53	Item d. Noise Figure, change "4 db" to "3.5 db"; Gain, change "20 db" to "10 db", last line change "10 db" to "4 db"
p. F-58	Figure F-21. Change 102 kc to 112 kc.
p. F-59	Line 22, change to "M_1 = 21.5 deg or 0.375 radians (rms, peak)"
p. F-60	Line 2, change to $$"M_2 = \sqrt{(1.1)^2 - (0.375)^2}"$$
p. F-60	Line 3, change to "M_2 = 1.03 radians (rms) or 1.46 radians (peak)"
p. G-6	Paragraph 1.4, second line, change "from $E_M = 10' E_o$ to $10^4 E_o$ ..." to read "from $E_M = 10^{-1} E_o$ to $10^4 E_o$ ..."

Volume 6. <u>Operational Support Equipment</u>

p. 25	Figure 6. Caption should be "Typical Grounding Scheme"
p. 39	Section 1.3.3, change opening of first sentence to read "Launch pad equipment consists of the ground power and RF consoles and the test flight program power and control equipment ..."
p. G-31	Figure 1. Lines enclosing Data Format Generator should be solid.
p. G-102	Last line substitute "4500" for "45"
p. G-113	In Section 4.4.2, change "25 per cent" to "250 per cent"
p. G-184	Section 4.5, substitute "6.5 feet" for "six feet"
p. G-311	Fifth line, change "30 per cent" to "20 per cent"
p. G-398	Section 4.2 should begin with "The hoist beam is ..."
p. G-419	Second line "4 optical alignment targets" instead of 8. Same correction top of p. G-421.
p. G-423	Section 4.9.2, substitute "20 per cent" for "50 per cent"

Volume 7. 1969 Flight Test Spacecraft and OSE

p. 90 First line should read "Launch pad equipment consists of the ground power and RF consoles and . . . "

p. 107 Last line, change Volume 5 to Volume 6.