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PHASE IA TASK B

FINAL TECHNICAL REPORT

VOYAGER SPACECRAFT

Volume 1

PREFERRED DESIGN: SYSTEM CONSIDERATIONS

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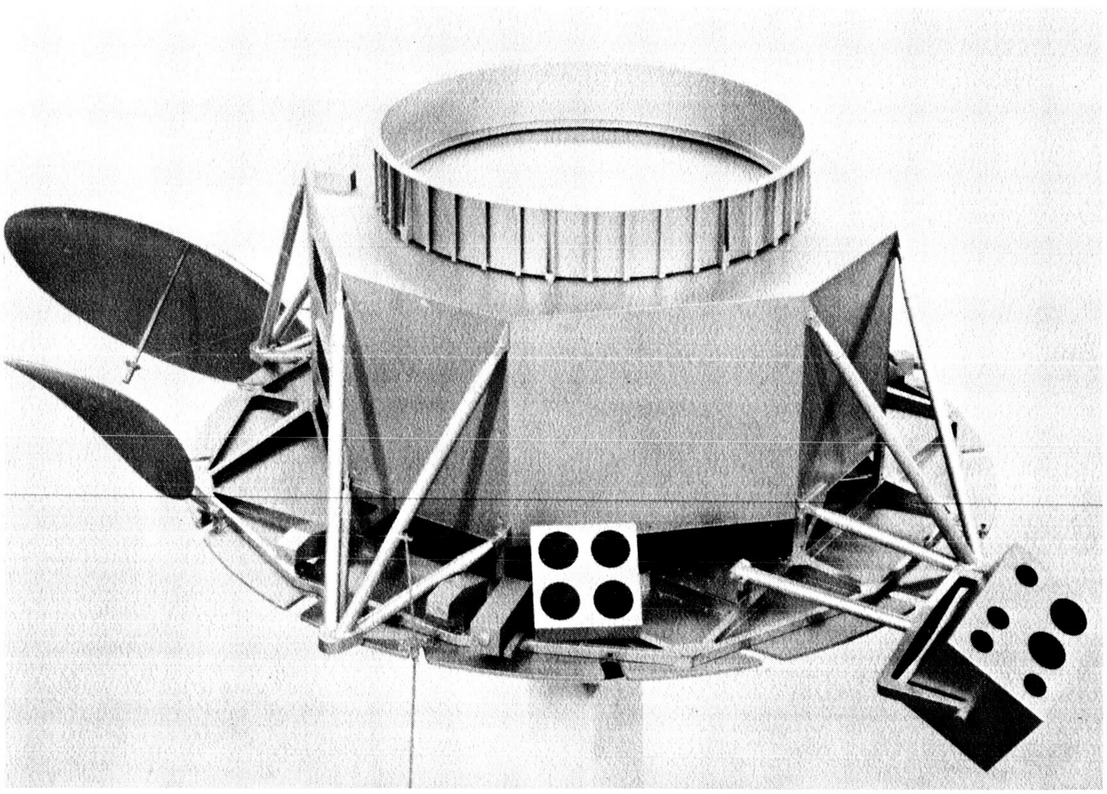

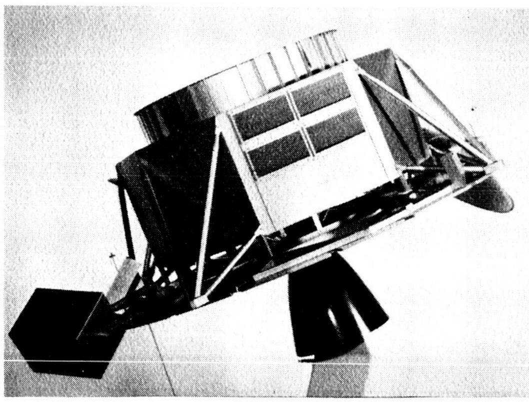

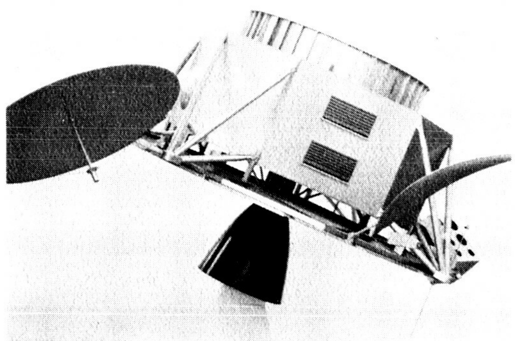

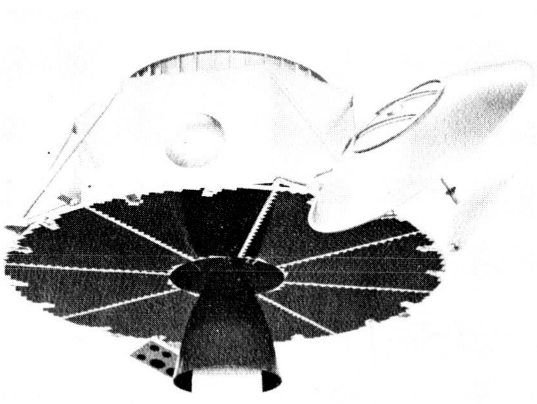

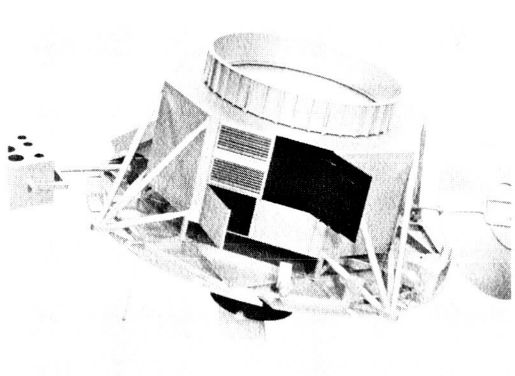

I. INTRODUCTION

Sections II, III, and IV of this volume correspond to I, II, and III of the Final Technical Report as defined in the Task B Work Statement issued by JPL, and present the TRW preferred design for the Voyager flight spacecraft. The functional descriptions of the individual hardware subsystems are given in Volume 2; schedule data and the related Voyager implementation plan for the spacecraft are given in Volume 3, as well as a discussion of support equipment implementation considerations; the recommended design for the operational support equipment is presented in Volume 4 along with the associated schedules and implementation plan; and Volume 5 presents the tradeoff analyses that were carried out at the system level to arrive at the selected design.

The presentation herein follows closely the approach indicated in the work statement, which corresponds to the Mariner C Spacecraft Functional Specifications Book. Section II contains the applicable mission objectives and design criteria for the selected design, Section III gives the applicable spacecraft design characteristics and restraints, and Section IV presents a system-level functional description of the flight spacecraft. In addition, several appendixes are included to present supporting system analyses developed during the study.

The mission objectives and design criteria given in Section II correspond to the top-level direction and guidelines for the system. These include over-all project direction regarding system responsibilities, design philosophy, priority for evaluating competing characteristics, and system design criteria covering safety, contamination, assembly and handling, and maintenance. In addition, Section II presents guidelines for how the operational mission is to be carried out. These guidelines take the form of mission restraints and include the gross mission plans and weight allocations, the mission profile, tracking and communication operations, trajectory characteristics, and spaceflight operational constraints.

Such over-all requirements as covered in Section II do not in general lend themselves to direct interpretation for spacecraft design purposes. Some of these requirements have a direct effect on design, but most of them take the form of constraints upon the general class of possible spacecraft design solutions. As such they are indirect requirements that must be translated by the system engineering process into specific spacecraft design characteristics and restraints. The results from such a process are presented in Section III. Much of this material has been established by work prior to the present study, both by TRW and JPL. This previous work has been updated and combined with the Task B results.

The data presented in Section III serves as a requirements baseline for the selected approach at the spacecraft design level. This data covers spacecraft performance for accuracy and reliability, configuration characteristics and constraints, intersystem interface requirements, science integration considerations, and general design standards and requirements for spacecraft equipment. This section also establishes the functional responsibilities for the spacecraft subsystems, and defines the performance requirements imposed upon the subsystems by system engineering considerations for the spacecraft.

The mode of presentation used in Sections II and III is a definitive one such as is generally associated with the documentation of requirements data and the function of establishing a requirements baseline. Accordingly, there is essentially no discussion or presentation of associated engineering data in those sections. The function of providing such background information is supplied by Section IV, which gives a functional description of the spacecraft. Such a description is analogous to an engineering data book presentation at the spacecraft level. It serves to convey an understanding of the design and the operational characteristics of the system, and also is a reference source for spacecraft engineering data.

In keeping with the above, Section IV includes a discussion of the mission sequence and the related spacecraft operations. Also discussed is the system operation against a framework of system functions and interrelations. The configuration is described and the underlying system design rationale is presented. Reliability design objectives are given and the assumptions and basis for reliability assessments are defined. System data is included for weights and mass properties, power, and engineering measurements; and component design parameters are summarized. A detailed sequence of mission events is also provided.

In keeping with the Task B Statement of Work, the following propulsion options were studied:

- Solid propellant motor for orbit insertion (sized for 1971-73 missions) plus liquid propellant midcourse system - (both monopropellant and bipropellant)
- Solid propellant motor for orbit insertion (sized for 1975-77 missions) plus liquid propellant midcourse
- LEM descent propulsion system
- Transtage propulsion system with modifications to adapt to Voyager.

As a result of the tradeoff analyses presented in Volume 5, the LEM descent stage has been selected as the basis for the Task B spacecraft design. This design is illustrated by the frontispiece and the accompanying photographs which show the quarter-scale spacecraft model developed during the study.

The advantages of the LEM descent stage for this application stem from the high level of development it represents and the simple, attractive Voyager spacecraft design that can be realized. The extensive testing and operational use of the LEM descent stage leads to significantly reduced program costs and development risk when it is utilized for Voyager.

The configuration approach in the study has been to limit modifications to the LEM descent stage to only those necessary to meet mission requirements. A modular approach has been achieved so that all nonpropulsive equipment can be assembled and tested independently of the LEM module. The LEM descent stage configuration naturally lends itself to installation into the Saturn V, and in that respect it has been designed for conditions quite similar to those of the study. The two-level LEM descent thrust capability has made possible a single propulsion system. Additional propulsion simplification is realized by the use of a blowdown mode for maneuvers prior to Mars orbit insertion. This provides positive isolation of the tanks from each other and from the pressurization components so as to prevent leakage, and allows a lower tank operating pressure for the long interplanetary flight. Other than the use of the LEM stage, the present design is similar to that of Task A. A fixed array is again used along with the same main compartment thermal concept. Also, the electronic subsystems are essentially the same as for the Task A design.

II. MISSION OBJECTIVES AND DESIGN CRITERIA

Section II presents over-all ground rules, guidelines, and the general system design philosophy for the Voyager program as applicable to the spacecraft system. These are, of course, established above the spacecraft contractor level, but are repeated here for completeness. The basic sources for such data have been References 1 and 2.

In addition to such over-all program aspects, this section also presents mission restraints and data on how the Voyager operational mission is to be carried out. This data deals with spacecraft functions and operations in mission terms. These considerations translate into specific spacecraft design characteristics and restraints, which are presented in Section III.

1. VOYAGER PROGRAM OBJECTIVE

The Voyager program will continue and extend the unmanned scientific exploration of the planets and solar system begun by Mariner, Ranger, and Pioneer. The primary objective of the Voyager program is to carry out scientific investigations of the solar system by instrumented, unmanned spacecraft which will fly by, orbit, and land on the planets. Emphasis will be placed on acquisition of scientific information relevant to the origin, evolution, and nature of life, and the application of this information to an understanding of terrestrial life.

2. MISSION OBJECTIVES

The primary objective of the Voyager missions to Mars, beginning in 1971, is to obtain information relevant to the existence and nature of extraterrestrial life; the atmospheric, surface, and body characteristics of the planet; and the planetary environment by performing unmanned experiments on the surface of and in orbit about the planet.

A secondary objective is to further knowledge of the interplanetary medium between the orbits of Earth and Mars by obtaining scientific and engineering measurements while the planetary vehicle is in transit.

In order to achieve a high level of success in achieving these objectives, it will be necessary in the 1971 mission to develop and to gain experience in the use of the basic capability 1) to place a science payload in orbit about Mars, 2) to conduct observations of Martian phenomena with this payload over extended periods of time, and 3) to transmit the results of these observations to earth. In addition, the 1971 mission will 1) develop and provide experience in the use of the basic capability to enter the Martian atmosphere and land on the Martian surface, and 2) conduct observations relating to critical Mars landing design parameters in orbit and in the atmosphere as required.

3. PROJECT ELEMENTS

3.1 Systems

The Voyager program is implemented by the Voyager project, which is divided into five systems. Each system includes the operational hardware, software and spare end items; associated operational support equipment; associated developmental test models and developmental test facilities; the management, engineering, and other personnel assigned to design, develop, fabricate, and test the system hardware and software; and the personnel assigned to support the prelaunch, launch, and flight operations phases of the mission. The Voyager systems are as follows:

3.1.1 Launch Vehicle System

The launch vehicle system includes the launch vehicle; all space vehicle umbilical lines; launch complex facilities assigned to Voyager; all other Kennedy Space Center (KSC) and Air Force Eastern Test Range (AFETR) facilities which support launch vehicle system prelaunch and launch operations; and the MSFC, KSC, and contractor personnel required to accomplish the functions of the launch vehicle system.

3.1.2 Spacecraft System

The spacecraft system includes the flight spacecraft; the planetary vehicle adapter; the spacecraft system test complex; all special test facilities required for spacecraft and planetary vehicle testing; the facilities at KSC utilized to assemble and prepare the flight spacecraft and planetary vehicle for launch; the flight spacecraft launch checkout equipment in the launch complex; and the JPL and contractor personnel required to accomplish the functions of the spacecraft system.

3.1.3 Capsule System

The capsule system includes the flight capsule; the relay link receiving and storage equipment mounted on the spacecraft bus; all special facilities required for capsule fabrication, sterilization, and testing; the facilities at KSC utilized to prepare the flight capsule for launch; the flight capsule launch checkout equipment in the launch complex; and the JPL and contractor personnel required to accomplish the functions of the capsule system

3.1.4 Mission Operations System

The mission operations system (MOS) includes the mission-dependent equipment for handling mission data and commands at DSN sites, KSC, and AFETR; the computer programs for processing mission data and commands; the operations teams in the SFOF which conduct the mission operations from injection to the end of Mars orbital and Mars entry operations; and all other DSN, AFETR, and KSC facilities assigned to support the MOS in the conduct of the spaceflight operations.

3.1.5 Tracking and Data System

This system includes the Deep Space Net and all other NASA and Department of Defense tracking and data acquisition stations, ships, and aircraft assigned to support the mission; all NASCOM and other circuits assigned to handle mission data and commands; and all other NASA and DOD personnel, physical facilities and general purpose equipment assigned to handle mission data and commands and support mission operations.

3.2 Flight Hardware

Flight hardware is that hardware which is designed to leave the ground in the conduct of launch operations. This corresponds to the Voyager space vehicle and is comprised of two identical planetary vehicles, two planetary vehicle adapters, and the launch vehicles as shown in Figure 1.

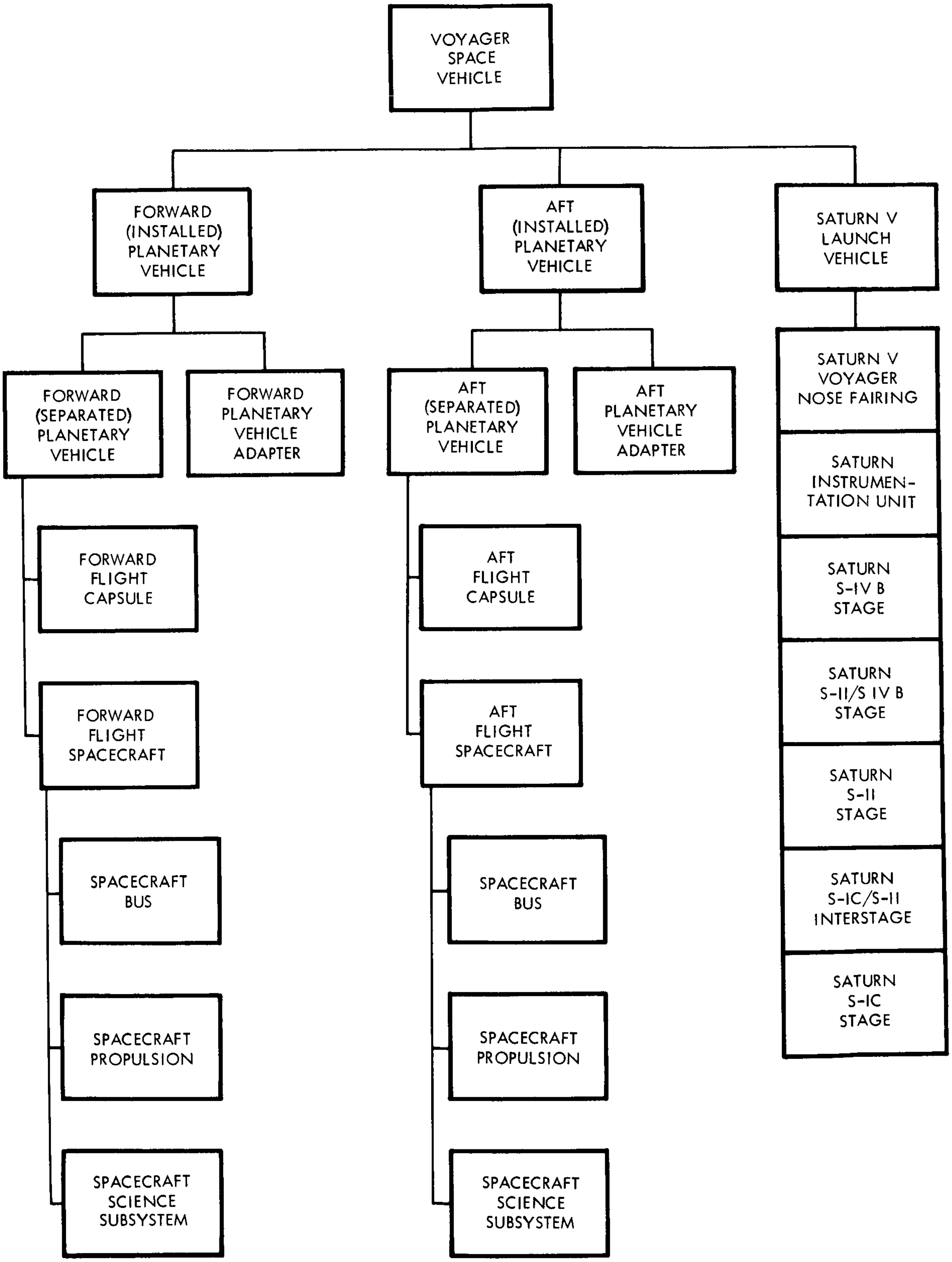

Figure 1. Over-all Breakdown for Voyager Flight Hardware

3.2.1 Launch Vehicle

The launch vehicle consists of the Saturn S-IC stage, S-IC/S-II interstage, S-II stage, S-II/S-IVB interstage, S-IVB stage, instrument unit, and the nose fairing.

3.2.2 Planetary Vehicle

A planetary vehicle is comprised of one flight capsule and one flight spacecraft.

3.2.3 Flight Capsule

A flight capsule is comprised of the entry, descent and impact equipment, capsule instrument payload, sterilization canister, capsule adapter, and all other ancillary equipment. The flight capsule includes the inflight separation joint for the separable capsule vehicle. The sterilization canister provides a bacteriological barrier between the separable portion of the flight capsule and its surroundings while on earth and during a portion of the cruise to the planet.

3.2.4 Flight Spacecraft

A flight spacecraft is comprised of a spacecraft bus, a spacecraft science subsystem, and spacecraft propulsion.

3.2.5 Planetary Vehicle Adapter

A planetary vehicle adapter is comprised of all structure, cabling, and hardware located between a planetary vehicle inflight separation joint and the associated points of attachment to the nose fairing.

4. MISSION RESTRAINTS

4.1 Schedule

The 1971 Mars opportunity places an absolute constraint on the mission schedule; consequently, all design, development, fabrication, testing, and deliveries are to conform to the established mission schedule milestones.

4.2 Number of Launches

One space vehicle is to be launched for each Mars opportunity.

4.3 Launch Site

Complex 39 at Kennedy Space Center will be utilized to launch the Voyager space vehicle during the 1971 Mars opportunity.

4.4 Launch Vehicle

The launch vehicle for Voyager is the Saturn V.

4.5 Spacecraft Provisioning

Two identical planetary vehicles will be launched on a single launch vehicle. To provide for a spare spacecraft, a total of three fully qualified flight spacecraft are required for each launch opportunity, along with all necessary support.

4.6 Tracking and Communications

The spacecraft is to be designed to be compatible with existing and planned DSN capability as defined in Reference 3.

4.7 Gross Mission Plans and Weight Allocations

Tentative gross mission plans and individual planetary vehicle weight allocations for each opportunity are given in Table 1. It is anticipated that these plans will be adjusted as the design of the 1971 mission and its systems progresses and as the program evolves from each Mars opportunity to the next. The mission for each opportunity will also vary according to the energy requirement for the particular opportunity as it

Table 1. Tentative Missions and Weight Allocations

		Weight Allocations (lb)				
Time Period		Spacecraft Bus and Payload	Spacecraft Propulsion	Flight Capsule	Orbiter Lifetime	Lander Lifetime
1971	Orbiter and capsule	2500	15,000	3,000	6 months	Entry to impact
1973	Orbiter and lander	2500	15,000	3,000	6 months	2 days
1975	Orbiter and lander	3500	15,000	10,000	6 months	2 months
1977	Orbiter and lander	3500	15,000	10,000	6 months	6 months

relates to the launch vehicle capability at that time, and will be defined to take advantage of the advancing state of the art within the constraints of the design criteria given in 5.

4.8 Planetary Quarantine

The probability that Mars is contaminated prior to calendar year 2021 as a result of any single launch is to be less than 10^{-4}.

4.9 Launch Period and Window

The launch period for the Voyager 1971 mission will not be less than 45 days. The minimum daily launch window will not be less than 2 hours. However, the system is to be designed for a capability to accommodate a launch window as short as 1 hour.

4.10 Launch on Time

The capability will be provided for accomplishing one launch from one launch pad in a 20-day period with a probability of 0.99. A daily launch window of 2 hours is to be used in calculating this probability.

4.11 Mission Profile

The operational activities for a Mars opportunity begin upon mission acceptance review and shipment of flight hardware to Kennedy Space Center. Mission operations terminate at the end of Mars orbital operations. The nominal 1971 mission events and sequences are defined below. Non-nominal considerations give rise to specific design requirements as presented in Section III, and provide alternate modes of operation as described in Section IV.

a) Prelaunch: Space vehicle final assembly, sterilization certification, system tests, and other activities resulting in the commitment to launch

b) Launch and Injection: Space vehicle countdown, launch, parking orbit insertion, interplanetary transit trajectory injection, and separation of two planetary vehicles from the launch vehicle

c) Celestial Reference Acquisition: Acquisition by a planetary vehicle of celestial attitude references and execution of all sequences leading to cruise status

d) Interplanetary Cruise: Events and sequences during transit flight until Mars orbit insertion, with the exclusion of interplanetary corrections

e) Interplanetary Trajectory Corrections: Events and sequences required to correct the transit trajectory of a planetary vehicle and return it to cruise status

f) Flight Capsule Canister Separation: Events and sequences required to separate a flight capsule biological barrier

g) Mars Orbit Insertion: Events and sequences required to insert a planetary vehicle into orbit about Mars and to reacquire planetary vehicle attitude references

h) Planetary Vehicle Orbital Operations: Orbital operations for a planetary vehicle from the time external references are reacquired after orbital insertion until spacecraft-capsule separation, with the exclusion of orbit trim

i) Planetary Vehicle Mars Orbit Trim: Events and sequences required to adjust the orbital parameters of a planetary vehicle satellite orbit and return to planetary vehicle orbital operations status

j) Spacecraft-Capsule Separation: Events and sequences required for a planetary vehicle to separate the capsule from the flight spacecraft, including return of the flight spacecraft to orbital operations status

k) Capsule De-Orbit Maneuver: Events and sequences required to place a capsule on a selected Mars impact trajectory

l) Capsule Orbital Descent: Events and sequences between a capsule de-orbit maneuver and capsule entry

m) Capsule Entry: Events and sequences from the time a capsule reaches an altitude of 800,000 ft until the entry configuration reaches terminal deceleration

n) Capsule Terminal Descent: Events and sequences from the time a capsule entry configuration reaches terminal deceleration until the terminal descent configuration impacts the surface of Mars

o) Flight Spacecraft Orbital Operations: Operations for a flight spacecraft from the time external references are reacquired after capsule separation until the last time the communications signal is received or the orbiter mission is declared to have ended, with the exclusion of orbit trim

p) Flight Spacecraft Mars Orbit Trim: Events and sequences required to adjust the orbital parameters of a flight spacecraft satellite orbit and return to flight spacecraft orbital operations status

4.12 Prelaunch Operations

Final assembly checkout and other prescribed activities will be performed at AFETR to ready the space vehicle for launch. Spacecraft prelaunch assembly and checkout will be conducted at a spacecraft assembly facility. An explosive safe facility will be used for propellant and gas loading, final spacecraft alignment, installation of other hazardous components, assembly of the flight capsules and flight spacecraft into planetary vehicles, encapsulation of the planetary vehicles, and final sterilization activities. The planetary vehicles will be mounted to the launch vehicle at the pad in an encapsulated condition. After encapsulation the planetary vehicles will be maintained in a sealed condition with access limited to radio telemetry, radio command, and umbilical links.

4.12.1 Maintenance and Repair

No modification to any spacecraft hardware will be planned to be accomplished after the hardware has been shipped to AFETR. Repairs to be effected at AFETR will be limited to those failures which are discovered at AFETR. Such repair will be accomplished only by replacement of equipment at the provisioned spares level. All such spares are to have had previous test history in a fully assembled spacecraft system. Failed equipment will be returned to its designated maintenance center for repair.

4.12.2 Cleanliness

The spacecraft will be assembled, tested, and handled at all times in accordance with 5.7. The cleanliness of the planetary vehicle at the time of shroud encapsulation in preparation for launch will be such that there will be no loose particles, either internal or external to the spacecraft equipment, greater than 4 mils in diameter.

4.12.3 Combined System Tests

Each system will demonstrate flight readiness and compatibility with all other interfacing systems. These tests will be designed to ensure operability of all systems when combined for the mission.

4.12.4 Commitment to Launch

The spacecraft will incorporate the necessary instrumentation and will perform the necessary monitoring during the countdown to detect nonstandard performance. The effects of nonstandard performance will be evaluated to determine the ability to meet the mission requirements. A commitment to launch or a request to hold will be made in accordance with established mission launch and hold criteria.

4.13 Spaceflight Operations

The mission operations system will control spaceflight operations from planetary vehicle injection through completion of the mission.

4.13.1 Tracking and Communication Coverage

Continuous tracking (after DSN acquisition) and communications coverage, within DSN view capability, will be provided for two planetary vehicles from launch until 30 days after the first interplanetary trajectory correction maneuver has been made for each vehicle. Twenty-four hours per day coverage will be provided for two planetary vehicles during subsequent interplanetary cruise on a time-shared basis. Continuous coverage will be provided for each planetary vehicle from 5 days prior to Mars encounter until 8 days after encounter. Time-shared coverage for tracking and command, and continuous coverage for data for the two planetary vehicles will be provided from 8 days after encounter through termination of flight spacecraft orbital operations.

4.13.2 Automatic Sequencing Capability

The system design will provide the capability for carrying out the flight mission automatically without ground command intervention on a nominal flight if trajectory corrections, trajectory biasing, instrument calibration, or updating of time dependent and trajectory dependent sequences are not required.

4.13.3 Ground Command Capability

Ground command capability will be provided to back up onboard command functions, to establish the interplanetary trajectory correction maneuver parameters, and to alter or adjust spacecraft-capsule separation, orbit insertion, and orbital sequence parameters if such adjustments are required.

4.13.4 Interplanetary Trajectory Corrections

The first interplanetary trajectory correction maneuver is to be performed at any selected time between 2 and 10 days after injection. This maneuver will accomplish the nominal Mars arrival separation in accordance with 4.14.4. One or more subsequent maneuvers may be required at a later time to improve the aiming point and flight time accuracy.

The interplanetary trajectory on which the final trajectory correction maneuver places the planetary vehicle will be a Mars fly-by trajectory with a Mars periapsis consistent with the location of the desired planetary orbit insertion maneuver. The choice of trajectory is constrained by the planetary quarantine constraint.

4.13.5 Flight Capsule Canister Separation

The capsule canister lid will be separated prior to Mars orbit insertion. The lid is not separated in Mars orbit because of its low weight-to-drag ratio, which might cause it to violate the planetary quarantine requirement of 4.8. The separation will occur as late as possible to minimize the possibility of capsule contamination.

4.13.6 Mars Orbit Insertion

The planetary vehicle will be placed into a suitable Mars orbit by a deboost maneuver utilizing spacecraft propulsion. To limit the probability of contaminating Mars, the system will have the capability for verifying that the desired planetary vehicle orientation has been achieved prior to deboost commitment. The orbit insertion maneuver is to occur in view of the DSIF station at Goldstone.

4.13.7 Mars Orbit Trim

The capability will be provided for accomplishing orbital trim maneuvers both before and after capsule-spacecraft separation.

4.13.8 Capsule Separation and Flight

The capsule de-orbit maneuver will nominally be performed 3 to 10 days after orbit insertion. The flight spacecraft will orient the planetary vehicle in an attitude specified prior to launch or as updated by ground command, and, after MOS attitude verification, will initiate the capsule separation sequence. The capsule will maintain the inertial attitude established by the spacecraft, and after achieving a suitable separation distance, will perform its de-orbit maneuver.

In the event that the maneuver is inhibited at the nominal separation time, the capability will exist for repeating the maneuver sequence on a subsequent orbital revolution.

The nominal capsule descent will be performed from an orbit which is trimmed to approach a preselected value. Consideration will be given to failure modes in which only one or no orbit trim maneuvers are performed.

The time of the maneuver is to be selected such that the orbiter is not occulted from earth at any time between capsule separation and capsule impact. The separation and de-orbit maneuvers are to be timed to occur during a view period of the Goldstone DSIF station.

The flight spacecraft will provide a relay link function during capsule flight as defined in 7 of Section III.

4.13.9 Spacecraft Orbital Operations

The flight spacecraft is to have the capability to acquire and play back scientific data without ground command for at least 2 months and preferably for 6 months after orbit insertion. The spacecraft will also have the capability for accepting ground commands to override this automatic mode of operation so as to incorporate timing changes within the basic program or to utilize direct commands from the MOS.

4.13.10 Post-Landing Operations

Post-landing operations for the 1973 capsule mission are to provide for the following additional considerations:

- Immediately after landing, direct link communications to earth are to be possible from an antenna oriented to the local vertical

- The duration of the first direct link view period is to be not less than 4 hours

- The first and second view periods of earth must occur during a view period of the Goldstone DSIF station. This constraint has priority over the requirement for the separation and de-orbit maneuvers to occur during a view period of the Goldstone DSIF station as given in 4.13.8.

4.14 Trajectories

The Voyager mission trajectories are to be designed to achieve the program objectives. They are to be consistent with the mission restraints presented earlier, and with the following specific trajectory constraints and characteristics.

4.14.1 Ascent Mode

The parking orbit ascent mode will be utilized for the Mars 1971 mission. The capability to coast in parking orbit for times between 2 and 90 minutes will be provided.

4.14.2 Launch Azimuth

Launch azimuths for the Voyager 1971 mission will lie between 60 and 115 degrees east of North. Subsequent Voyager missions may require launch azimuths from 35 to 120 degrees east of North.

4.14.3 Transfer Trajectories

Type-I transfer trajectories will be utilized for the 1971 mission. A maximum C_3 of 25 km^2/sec^2 is to be assumed. The hyperbolic excess velocity at Mars is not to exceed 4.5 km/sec.

The absolute value for declination of the launch asymptote (DLA) is not to be less then 5 degrees, and the inclination of the heliocentric transfer plane to the ecliptic plane is not to be less than 0.1 degree.

4.14.4 Mars Arrival

The separation between planetary vehicle arrival dates at Mars will not be less than 10 days.

4.14.5 Satellite Orbit Selection

The geometry selected for a satellite orbit about Mars will be favorable for obtaining the desired orbiter science data and for providing the required initial conditions for the capsule de-orbit maneuver. In particular:

- Occultation of the sun by Mars will not last for more than a prescribed per cent of each orbit period.

- No orbit will be selected which will cause loss of the spacecraft roll reference for a period of time which may impair flight spacecraft control in the nominal mode of operations.

- The orbit will be selected to meet the requirements of the capsule-spacecraft relay link and the capsule-Mars surface geometry.

4.14.6 Capsule Descent Trajectory

Constraints on the capsule descent trajectory are as follows:

1) The fixed velocity increment required to perform the de-orbit maneuver must not exceed 550 m/sec.

2) The orbital descent time (the time from capsule motor ignition until the capsule reaches an 800,000-ft entry altitude) is to be at least 50 minutes and is not to exceed 12 hours.

3) The descent trajectory is to be such that in the event an angle of attack orientation maneuver cannot be performed and the capsule remains inertially oriented in the de-orbit maneuver attitude, the capsule angle of attack at entry will be less than 60 degrees.

4) The actual entry angle (at entry altitude) is to be between the vacuum skip limit and 20 degrees.

5) The maximum atmospheric descent time (from the 800,000-ft entry altitude until impact) is to be less than 900 seconds.

4.14.7 Landing Site Constraints

The impact point is to be between 15 and 30 degrees from the terminator. The terminator for which the forward looking (roll axis) camera would not be looking into the sun is preferred as the reference terminator, because of the descent TV experiment.

4.15 Accuracies

The flight spacecraft will be capable of performing interplanetary trajectory corrections, orbit insertion, and orbit trim adjustment for the planetary vehicle and orbit trim adjustment for the flight spacecraft to accuracies specified in 2.1 of Section III.

5. DESIGN CRITERIA

5.1 Design Philosophy

The Voyager spacecraft is to be based on a simple and conservative design approach. Wherever applicable, the spacecraft design will take advantage of equipment designs, the operational techniques developed, and experience gained in the Ranger, Mariner-II, Mariner-IV, and other programs. Emphasis will be placed on simplifying interfaces between the systems in order to simplify design, flight operations, ground test and checkout operations, and to increase mission probability of success.

5.2 Mission Adaptability

The Voyager missions to Mars from 1971 through several subsequent opportunities will be conducted as an integrated program in which each individual mission forms a part of a logical sequence from both the scientific and engineering points of view. With this aim in mind, the spacecraft for the 1971 opportunity will be designed to be adaptable to subsequent missions; to accommodate a variety of spacecraft science payloads, capsule payloads, mission profiles, and trajectories; and to accept improvements in technology with minimum redesign.

5.3 Mission Success

The spacecraft will utilize design, manufacturing, test, operational techniques, and procedures designed to maximize mission success and partial success in the event of noncatastrophic failure. These efforts are to include, but not be limited to, the following:

- Comprehensive failure-mode and failure-effect analyses and design for partial mission success in the event of failure
- Establishment and demonstration of design margin adequacy
- Application of functional and parallel redundancy techniques where constraints can be met and increase in reliability can be domonstrated
- Systematic identification and elimination of unreliable items wherever possible.

5.4 Lifetime

Since a Mars-mission spacecraft must survive and function during Earth-to-Mars transit and subsequent orbital operations, it is anticipated that a major requisite for a successful mission will be the achievement of a "long-life" capability. Therefore, strong emphasis is placed on a thorough system approach to long-life needs, followed in the hardware stage by a comprehensive test program to expose potential failures in a working spacecraft over a long period.

The useful life of all spacecraft system equipment is to be sufficient to include all operating time from initial turn-on through subsystem checkout and acceptance tests, system checkout and acceptance tests, prelaunch tests, and flight operation to the end of its anticipated service either in normal or abort modes of operation.

5.5 Competing Characteristics

In the event of technical conflicts affecting the following mission characteristics, the relative priorities, in decreasing order of importance, are to be as follows:

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

5.6 Safety

All design features, operations, and procedures are to take into account achievement of a high degree of safety. Consideration will be given to safety techniques to be employed while testing and checking out flight hardware. The design of hoisting, handling, and testing fixtures will give special attention to minimizing hazard to both personnel and equipment. Reference 4 is to apply.

5.6.1 Toxic Fluids

The propulsion subsystem is to be designed for essentially zero leakage. All fuel handling is to be accomplished at the explosive safe facility just prior to transport to the launch pad. Propellant transfer is to be accomplished without significant spillage or escape of fumes and personnel will require special clothing and breathing apparatus.

5.6.2 Pressure Systems

All pressure systems are designed in accordance with Reference 4. Special attention is given to the method of mounting pressure vessels and components to avoid undue restraint that could induce high-stress concentration during pressurization and temperature variations. Designs with minimum welding (integral ports and integral mounting bosses) are preferred. A hazard factor of 1.76 will be used in the design of pressure vessels hazardous to personnel. Such pressure vessels will be fabricated of tough, ductile materials. A hazard factor of 1.0 will be used in the design of all other components, including pressure vessels nonhazardous to personnel.

5.6.3 Pyrotechnics

All electroexplosive devices are category B types. They and their associated wiring and firing circuitry will conform to AFETR P80-2. Electromechanical devices are incorporated to maintain spacecraft energized pyrotechnics in a safe condition until planetary vehicle separation. Safing of the planetary vehicle separation devices is accomplished within the launch vehicle, which supplies the firing signal for separation.

The AFETR P80-2 requirements for shielding of circuits associated with category A devices are to apply to spacecraft circuits associated with category B devices.

5.6.4 Spacecraft Sequencing and Monitoring

The OSE is to incorporate features to protect against damage to itself or the spacecraft by failure of the OSE, spacecraft or the test facilities. The OSE provides programming safeguards in the automatic mode of testing to prevent damage from improper sequencing and also temperature alarm monitoring while the spacecraft, or subsystems thereof, are under thermal environmental test.

5.6.5 Electrical Power

The OSE provides spacecraft main power isolation from other facility loads and their effects. The spacecraft is conditioned to a safe mode in the event of test facility power failure. Current limiting is provided in the OSE on all high-energy interfaces with the spacecraft. Safety interlock provisions are to be incorporated to control application of power to the spacecraft.

5.6.6 RF Radiation

Personnel will be protected from excessive RF radiation from the high- and medium-gain antennas by installation of a coupler/cover over the antenna feed arms until the planetary vehicle is encapsulated in the nose fairing. Subsequent to the encapsulation, it is expected that the nose fairing will attenuate the rf radiation to below hazard levels.

5.7 Nonbiological Contamination

The spacecraft design and construction is to be compatible with the control of nonbiological contamination within allowable limits, and the elimination of all contaminants which could have a detrimental effect on vehicle operation. This requires utilization of suitable processes during manufacture of subassemblies and assemblies, proper handling of equipment at all levels of assembly, use of appropriate facilities for accomplishing manufacturing and assembly, and use of suitable procedures to control all program activities and operations that could adversely affect equipment cleanliness. Approaches to contamination control by prevention are preferred over those that depend on curative measures.

5.7.1 Manufacturing Processes

Manufacturing processes will be specified to minimize the generation and/or retention of nonbiological contaminants. Tools, fluids, epoxies, etc., will be employed that have been designed for a low level of particle production. Suitable cleaning processes will be utilized at appropriate levels of assembly to remove particles that have been generated or collected

5.7.2 In-Process Equipment Handling

All equipment received or manufactured will be subject to handling in a manner contributory to the maintenance of spacecraft cleanliness. Transportation of equipment at any level will be accomplished either in sealed containers or under controlled environments. During inactive periods, the equipment will be sealed in a clean room and will remain sealed until required for further assembly at a subsequent clean area.

5.7.3 Facilities

Fabrication, Assembly, and test of the spacecraft will be accomplished in facilities meeting the requirements of Class 100,000 or better (Federal Standard No. 209).

5.7.4 Procedures

Procedures will be established to achieve and maintain cleanliness requirements. These will cover personnel access, attire and handling methods, as well as verification of the adequacy of cleaning materials, use of cleaning materials and equipment, cleanliness interfaces between OSE and flight equipment, and handling and shipping techniques.

5.8 Biological Contamination

The spacecraft system is to include a program for achieving the required control of biological contamination. This will include limiting the number of extraneous particles in accordance with 5.7 and destroying microorganisms by chemical or heat sterilization. The biological load buildup during manufacturing, assembly, test and shipment is to be controlled and limited to acceptable levels. Implementation considerations for control of biological contamination are included in Volume 3 A supporting technical discussion is presented in Appendix E of this volume.

The following decontamination processes will be employed:

a) Use of self-sterilizing surface coatings

b) Filter sterilization of gases

c) Internal sterilization of subsystems which may produce spacecraft efflux

d) Surface sterilization of the spacecraft with ethylene oxide gas.

5.9 Handling and Assembly

Assembly, handling, and shipping equipment (AHSE) will be designed to allow all associated operations to be performed in a manner that is both safe and comfortable for personnel. Holding and handling fixtures are to be provided as required and work stands are to provide ready access to all segments of the spacecraft at all levels of assembly. Movable and mobile equipment and shipping containers are to prevent the spacecraft and subsystem equipment from being subjected to excessive acceleration shock levels.

5.10 Transportability

All elements of the spacecraft system are to be transportable by truck. Consideration will be given to transportation by air. The size, weight, interconnection, and functional features of the OSE and cabling will be suitable for expeditious take-down, transport, and assembly in support of any testing required at remote test facilities. In general, the time necessary to transport and set up the OSE is to be equal to, or less than, the time necessary to transport and prepare a spacecraft for testing.

III. SPACECRAFT DESIGN CHARACTERISTICS AND RESTRAINTS

Section III presents characteristics and requirements data applicable to the preferred design for the Voyager flight spacecraft, as developed in the Task B study. It includes controlling data as specified by JPL, as well as the derived characteristics and requirements established by the spacecraft contractor. This corresponds mainly to the over-all spacecraft level, but it also covers the performance requirements imposed upon the subsystems by system engineering considerations for the spacecraft.

1. GENERAL

The Voyager 1971 flight spacecraft is designed in keeping with the mission restraints of 4 and the design criteria of 5 in Section II. It is designed to perform all flight spacecraft functions required to accomplish the mission profile given in 4.11, Section II.

1.1 Description

The flight spacecraft is capable of automatic operation without ground command intervention for a nominal flight, and will accommodate ground commands for trajectory-dependent sequencing and to override or modify automatic operations. It incorporates a two-way communications capability for transmitting data to earth, for receiving commands from earth, and for angle, doppler, and range tracking. The capability is provided for achieving propulsive velocity increments, as required, and to maintain temperatures for proper operation.

The flight spacecraft is fully attitude stabilized, using the Sun and Canopus as reference objects, except during maneuvers. It operates on battery power from launch until solar acquisition is completed, and during maneuvers or when it is in the shadow of Mars. Power is derived otherwise from photovoltaic solar cells arranged on panels.

The equipment that performs these functions is assembled into a unifying structure which allows operation as a complete hardware system. The structure provides load and thermal paths, suitably rigid support points for equipment, support points for interface items, and protection against environmental factors as required.

1.2 Functions

The functions to be performed by the flight spacecraft include the following:

a) Achieve separation of the planetary vehicle from the launch vehicle

b) Transport the flight capsule and provide power, telemetry data handling, command handling, and other support

c) Provide the capability for executing two or more interplanetary trajectory correction maneuvers based upon ground command transmissions

d) Insert the planetary vehicle into a specified Mars orbit

e) Orient the flight capsule and initiate the capsule-spacecraft separation sequence

f) Provide the capability for executing orbital trim maneuvers based on ground command transmissions, both prior to and subsequent to capsule-spacecraft separation

g) Provide the capability for updating trajectory and time-dependent control sequences by ground command

h) Measure and transmit engineering performance data to earth

i) Execute the science mission as specified and transmit science data to earth

j) Relay flight capsule telemetry data to earth after capsule-spacecraft separation.

1.3 Functional Areas

The flight spacecraft functional areas under the cognizance of the spacecraft contractor are as follows:

1) Spacecraft bus

   a) Power subsystem

   b) Guidance and control subsystem

   c) S-band radio subsystem

d) Telemetry subsystem

e) Data storage subsystem

f) Command subsystem

g) Computing and sequencing subsystem

h) Structural and mechanical subsystem

i) Pyrotechnics subsystem

j) Temperature control subsystem

k) Cabling subsystem

2) Spacecraft propulsion

3) Planetary vehicle adapter

1.4 Flight Spacecraft Interchangeability

The three flight spacecraft supplied for the 1971 mission as per 4.5 of Section II will be interchangeable. Mechanical and electrical interchangeability will exist between all assemblies, subassemblies, and replaceable parts that are intended to be identical, regardless of manufacturer or supplier.

1.5 Mars Contamination

The probability that the flight spacecraft will impact the planet Mars before the year 2021 as a result of any single launch is to be less than 2×10^{-5}. The probability that any nonsterile ejecta from the flight spacecraft will impact the planet Mars before the year 2021 as a result of any single launch is to be less than 3×10^{-5}.

1.6 Emergency Capsule Separation

The flight spacecraft will provide for an emergency separation of the flight capsule. This may be commanded from the ground to take place at any selected time after injection. It involves orienting the planetary vehicle in a prescribed direction and activating the emergency capsule separation device so as to impart a suitable separation velocity to the flight capsule.

1.7 Absence of Capsule

In the event that the flight capsule is not launched with the flight spacecraft or that an emergency capsule-spacecraft separation as in 1.6 is required, the flight spacecraft will be capable of performing the mission as planned.

2. SYSTEM PERFORMANCE

2.1 Accuracy

Accuracy requirements on the flight spacecraft are specified in mission terms by IV.B.2.c.(3) of Reference 1. These have been related to spacecraft design characteristics as given below. Detailed G and C error budgets are given in 2 of Section II, Volume 2. Presentation of supporting analysis is given in Appendix C.

2.1.1 Interplanetary Trajectory Correction

The accuracy for achieving a vector velocity increment for an interplanetary trajectory correction is as follows:

- Velocity magnitude error: 3σ nonproportional error, 0.04 m/sec; 3σ proportional error, 0.1 per cent
- Velocity direction error: 3σ error between the direction of the actual velocity increment vector and a specified direction in each of two references axes, 0.8 degree
- Initiation time error: 3σ error from time selected by MOS, 15 sec.

For the reference trajectory given in IV.B.2.c.(3)(a) of Reference 1, the above spacecraft characteristics result in a 3σ semi-major axis for the dispersion ellipse in the $\overline{R}$, $\overline{T}$ plane at Mars of 114 km* (100 km required) and a 3σ error in time of arrival of 34 sec (60 sec required).

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*This dispersion ellipse can be improved by considerations discussed in 2 of Section II, Volume 2. In line with the competing characteristics given in 5.5 of Section II, the emphasis in the present study has been placed on a simple, reliable design in preference to improved performance. The estimated accuracy given is considered to represent adequate performance in keeping with the main purpose of Task B. Additional study relative to this is required in Phase IB.

2.1.2 Mars Orbit Insertion

The accuracy for achieving a vector velocity increment for Mars orbit insertion is as follows:

- Velocity magnitude error: 3σ nonproportional error, 1 meter/sec; 3σ proportional error, 0.1 per cent
- Velocity direction error: 3σ error between the direction of the actual velocity increment vector and a specified direction, in each of two reference axes, initial 1.15 degrees, final 0.86 degree
- Initiation time error: 3σ error from time selected by MOS, 15 sec.

2.1.3 Orbit Trim

The accuracy for achieving a vector velocity increment for orbit trim is as follows:

- Velocity magnitude error: 3σ nonproportional error, 0.04 meter/sec; 3σ proportional error, 0.1 per cent
- Velocity direction error: 3σ error between the direction of the actual velocity increment vector and a specified direction, in each of two axes, (with capsule) 0.68 degrees, (without capsule) 1.13 degrees
- Initiation time error: 3σ error from time selected by MOS, 4 sec.

2.1.4 Spacecraft-Capsule Separation

For spacecraft-capsule separation, the normal to the spacecraft-capsule interface reference plane will be oriented in a specified direction relative to the Sun-Canopus system to a 3σ accuracy of 0.75 degree, in each of two references axes. The time that the separation initiation signal is furnished to the flight capsule by the flight spacecraft will be at the time selected by the MOS to a 3σ accuracy of 4 sec.

2.2 Reliability

The flight spacecraft will be designed in accordance with the design criteria for mission success in 5.3 of Section II. Primary consideration is given to defining a basic system with functions, operations, equipment, and devices with maximum simplicity and proven reliability levels, while at the same time meeting performance requirements. Redundancy is added to this basic system to increase the probability of mission sucess.

2.2.1 Redundancy

Redundancy is to be employed only when comparable design reliability cannot be achieved by simplification or alternate means of accomplishing the intended function, and when the reliability improvement offered by the redundancy can be shown to be real in spite of the increased complexity.

The following design approaches to redundancy are specified in order of preferences:

a) Cooperative multichannel subsystem designs

b) Alternate path or functional redundancy

c) Ordinary block of element redundancy.

2.2.2 Functional Dependency

The independence of the functional elements of the flight spacecraft is to be maintained to the maximum extent practical. Where dependency must exist, a strong effort will be made to achieve a degraded performance from the dependent element despite a failure in the interface or in the element upon which it depends.

2.2.3 Catastrophic Failure

No single failure mode of an electronic or electrical part or component is to cause a catastrophic effect on the mission.

2.2.4 Reliability Assessments

Reliability assessments for the flight spacecraft broken down by subsystem are given in Table 2. Probabilities by individual mission phase are given along with over-all mission values. A discussion of these assessments and supporting data are given in 4 of Section IV.

2.3 Weight Allocations

The individual planetary vehicle weight allocations for the 1971 Voyager mission are given in Table 3. Additional weights data is given in 5.1 of Section IV.

Table 2. Reliability Assessment for the Flight Spacecraft

Subsystem	Spacecraft Reliability by Individual Mission Phase* 1A	1B	2A	2B	3A	3B	4	Mission Reliability
1.0 Structure and mechanical	.99961	.97889	.99999	.99996	.99999	.99795	.99385	.97044
2.0 Command	.99998	.99582	--	.99969	--	.99997	.99895	.99442
3.0 Propulsion	.99989	.99148	.99901	.99905	--	--	--	.98945
4.0 Cabling	.99999	--	--	--	--	--	--	.99999
5.0 Computing and sequencing	.99865	.95451	--	.99730	--	.99975	.99097	.94183
6.0 S-band radio	.99222	.96688	--	.99541	--	.99976	.99170	.94680
7.0 Guidance and control	.99700	.94400	.99994	.99999	.99995	.99999	.99400	.93539
8.0 Telemetry	.99995	.99039	--	.99931	--	.99992	.99764	.98724
9.0 Temperature control	.99998	.99939	--	.99998	--	--	.99990	.99929
10.0 Power	.99997	.99631	--	.99973	--	.99996	.99908	.99505
11.0 Data storage	.98529	.80713	--	.99163	--	.99914	.97295	.76661
12.0 Pyrotechnics	.97733	--	--	--	--	--	--	.97733
Mission success probability (one planetary vehicle)	.95071	.67020	.99894	.98212	.99994	.99644	.94040	.58574
Mission success probability (at least one of two planetary vehicles)								.82839

*Definition of phases is given in 4 of Section IV.

Table 3. Planetary Vehicle Weight Allocations for 1971 Voyager Mission

Flight Hardware		Weight Allocation (lb)
Flight spacecraft		17,500
Spacecraft bus	2,100	
Spacecraft propulsion	15,000	
Spacecraft science payload	400	
Flight capsule (a maximum of 250 lb may remain with spacecraft)		3,000
Planetary vehicle		20,500
Planetary vehicle adapter		1,500
Total		22,000

3. CONFIGURATION

The flight spacecraft is configured to combine with the flight capsule to form the planetary vehicle. Two planetary vehicles are to be installed in tandem within the launch vehicle nose fairing. The flight capsule is mounted forward of the flight spacecraft when the planetary vehicle is mounted to the launch vehicle. The flight spacecraft is configured so that the flight capsule will be completely shaded when the planetary vehicle is in the sun-stabilized attitude.

3.1 Allowable Envelope

The flight spacecraft configuration is to be in accordance with the planetary vehicle arrangement and dynamic envelope as shown in Figure 2.

3.2 Use of Lunar Excursion Module Descent Stage

On the basis of tradeoff studies presented in Volume 5, the LEM descent stage is to be used as a propulsion module for the flight spacecraft.

A standard LEM descent stage at an appropriate level of assembly is to be diverted with as little effort as possible for use in Voyager. Modifications are restricted to only those necessary to meet mission requirements, so as to achieve minimum program cost. No major changes in propellant tank size, mounting, or basic structure are allowed, and any changes which invalidate LEM descnet stage qualifications are to be avoided as much as possible. Emphasis is to be placed on maintaining compatibility with developed LEM descent stage support equipment.

The following modifications to the LEM descent stage are required:

1)* For thrust vector control, lower the engine to provide adequate moment arm, and change actuators to obtain proper response characteristics

2)* Add nonrefillable positive displacement start system

*The proposed modifications make it possible to use only one propulsion system. This is different from LEM, which utilizes a separate ullage and attitude control rocket system on the ascent stage. The elimination of this ascent stage propulsion system leads to modifications 1) and 2).

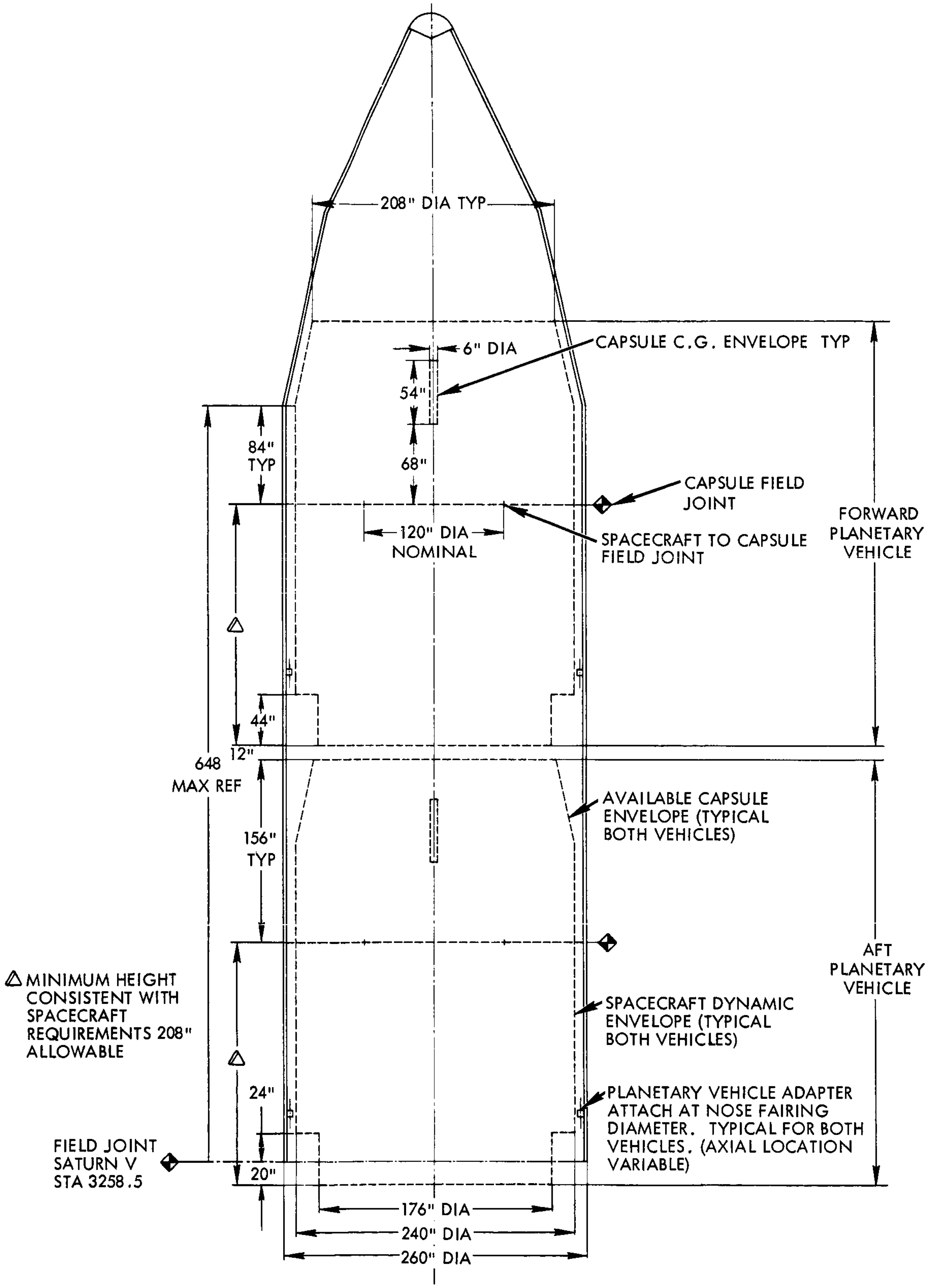


Figure 2. Planetary Vehicle Arrangement and Dynamic Envelope

3) Replace the radiation cooled engine nozzle extension with an ablative skirt to minimize radiant energy interchange with the fixed solar array

4) Replace engine continuous throttling capability with a simpler two-level (low and high) thrust capability

5) Modify external micrometeorite shield as required

6) Replace ascent engine blast deflector with more conventional skin as appropriate

7) Replace cryogenic helium pressurization tanks with a single available (qualified) pressure vessel for storing helium at ambient temperature

8) Modify feed system and engine for long-term space storage requirements

9) Delete all life support and electronic equipment, etc., inappropriate to the Voyager mission

10) Delete thermal insulation and replace with a tailored temperature control subsystem

11) Remove landing gear structure and replace with truss-type outriggers for attachment to shroud adapter.

3.3 Modularity and Configuration Breakdown

The flight spacecraft configuration is to provide for a buildup from a number of modules or major elements. These modules are to be utilized so as to simplify spacecraft assembly and test. They are to allow for pretesting and parallel assembly operations to a large extent. Emphasis is to be placed on simplifying interfaces between modules.

The spacecraft major modules are listed below:

a) LEM descent propulsion module

b) Main compartment equipment mounting panels attached to two sides of the LEM module so that equipment is mounted within LEM module

c) Aft equipment module, including reaction control assembly, cabling, and the solar array with replaceable solar panels.

The remaining flight spacecraft elements are as follows:

a) New outriggers

b) Flight capsule interstage

c) Insulation and louvers

d) Antennas

e) Science appendages.

3.4 Solar Array

A fixed solar array will be utilized. For the design condition that sizes the array the ratio of solar absorptivity to infrared emissivity (α_s/ϵ) is not to be taken as less than unity.

3.5 Externally Mounted Equipment

The spacecraft is to accommodate the installation of the following externally-mounted equipment as shown in Figure 3.

a) Double gimballed high-gain antenna

b) Single gimballed medium-gain antenna

c) Deployable boom-mounted low-gain antenna

d) Fixed relay link antenna

e) Double gimballed planetary scan platform

f) Reaction control jets

g) Spacecraft sensors

h) Fixed science packages

i) Deployable science instruments

3.6 Temperature Control

The configuration is to be amenable to continuous temperature control. The main module will utilize the insulated compartment concept, with thermal louvers mounted externally to the equipment mounting panels. Provisions for mounting insulation and for controlling heat leaks will be made.

3.7 Sensor Fields of View

Provisions will be made to accommodate fields of view as required for the following sensors, as shown in Figure 3.

a) Canopus sensors

b) Sun sensors

c) Earth detector

d) Limb and terminator crossing detectors.

3.8 Attachment and In-Flight Separation Joints

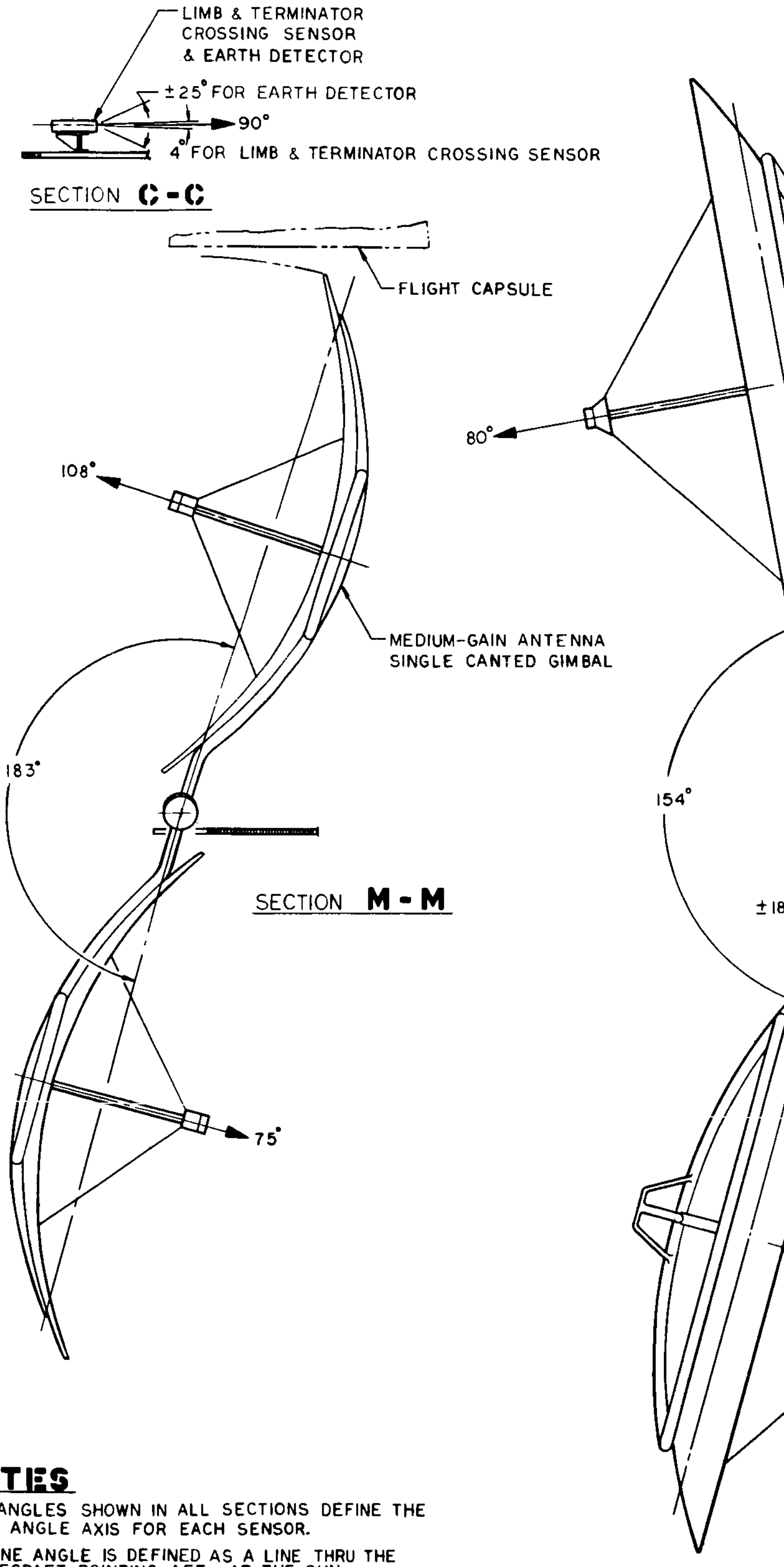
Provisions are to be made for attaching the flight capsule to the flight spacecraft and for attaching the planetary vehicle adapter to the launch vehicle nose fairing. These attachments are to be field joints. The designs are to be compatible with the launch vehicle nose fairing and the flight capsule.

Separation from the launch vehicle is to be accomplished by an inflight separation joint between the planetary vehicle and the planetary vehicle adapter. An emergency inflight separation joint is also to be provided to separate the flight capsule from the flight spacecraft. These inflight separation joints are to be separate from the above field joints.

3.9 Spacecraft Geometry

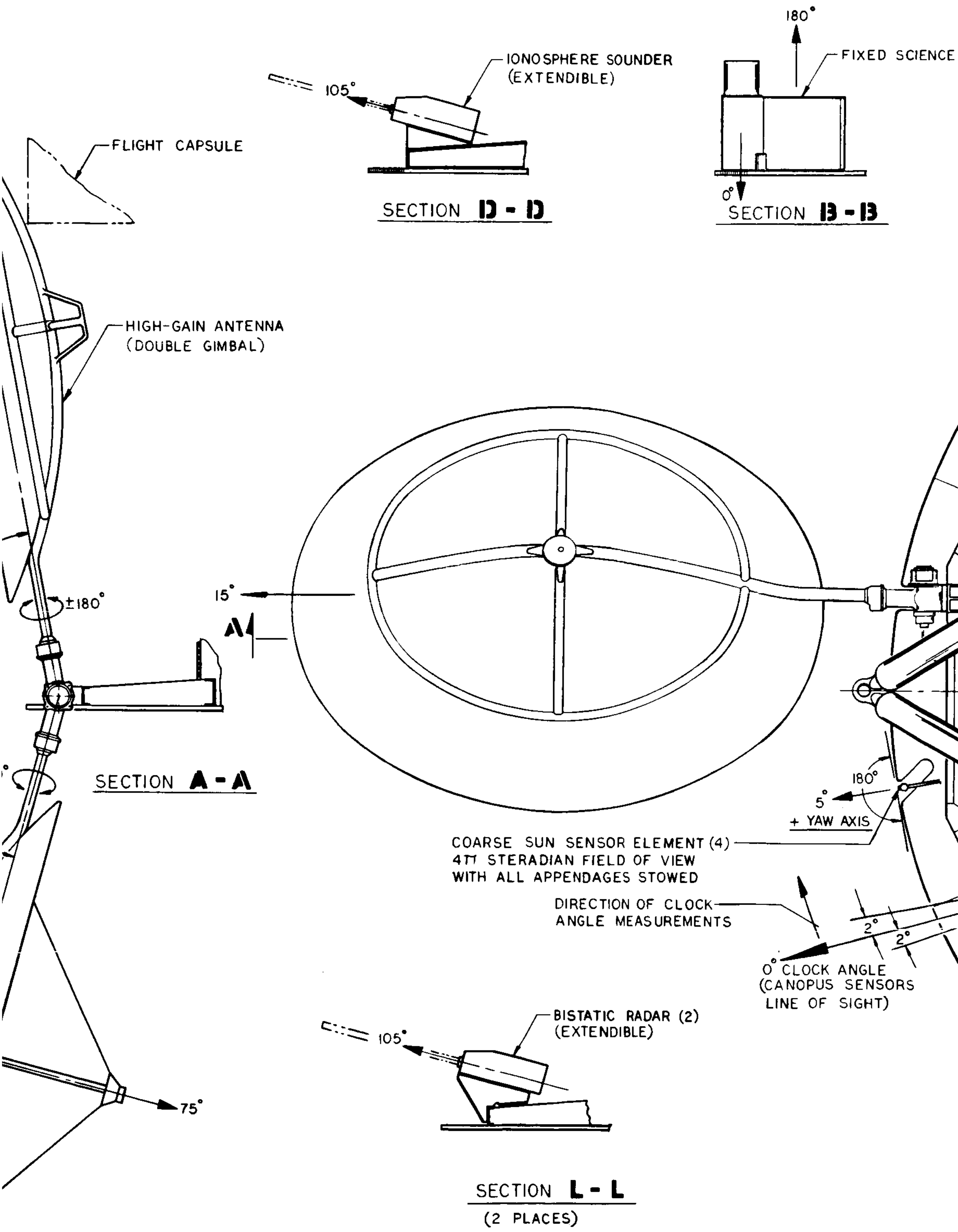
3.9.1 Station Coordinate

A spacecraft station coordinate is defined in the axial direction so as to be measured positively in the launch vehicle forward direction, with the spacecraft in the installed position. The spacecraft forward and aft directions correspond to increasing station and decreasing station, respectively. Spacecraft station designations will apply for the planetary vehicle.



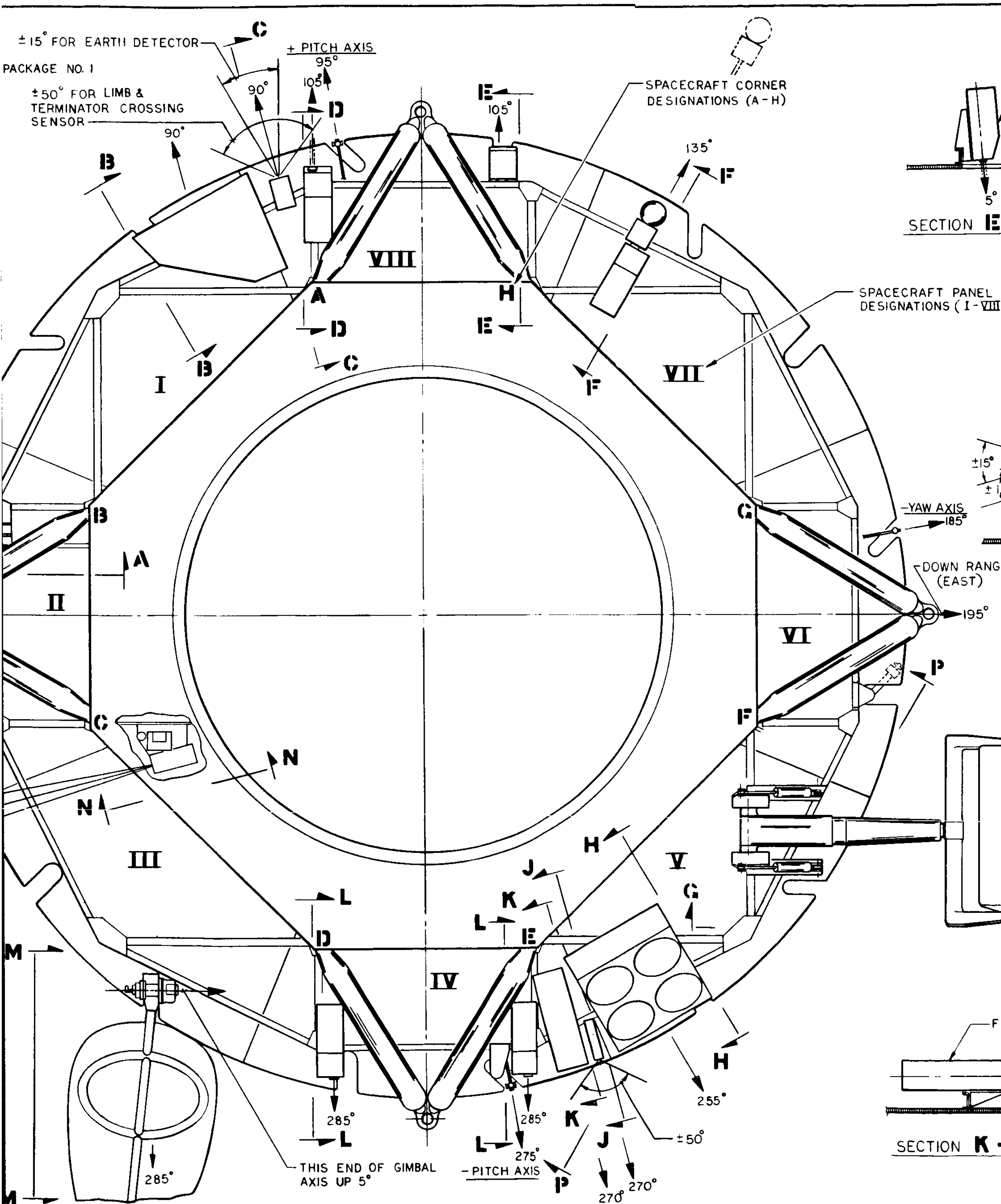
NOTES

1. ALL ANGLES SHOWN IN ALL SECTIONS DEFINE THE CONE ANGLE AXIS FOR EACH SENSOR.
2. 0° CONE ANGLE IS DEFINED AS A LINE THRU THE SPACECPAFT POINTING AFT — AT THE SUN.
3. SECTIONS SHOWN HAVE BEEN ROTATED FOR CLARITY.



NOTES

1. ALL ANGLES IN MAIN VIEW DEFINE THE CLOCK ANGLE FOR EACH SENSOR AS MEASURED CLOCKWISE FROM THE LINE OF SIGHT OF THE CANOPUS SENSOR, LOOKING AFT.
2. 0° CLOCK ANGLE IS DEFINED AS THE LINE OF SIGHT OF THE CANOPUS SENSOR.



± 15° FOR EARTH DETECTOR
PACKAGE NO. 1
± 50° FOR LIMB & TERMINATOR CROSSING SENSOR
+ PITCH AXIS
95°
105°
90°
SPACECRAFT CORNER DESIGNATIONS (A - H)
135°
SECTION E
SPACECRAFT PANEL DESIGNATIONS (I - VIII)
I
II
III
IV
V
VI
VII
VIII
-YAW AXIS
185°
DOWN RANGE (EAST)
195°
285°
275°
- PITCH AXIS
255°
± 50°
270°
THIS END OF GIMBAL AXIS UP 5°
SECTION K

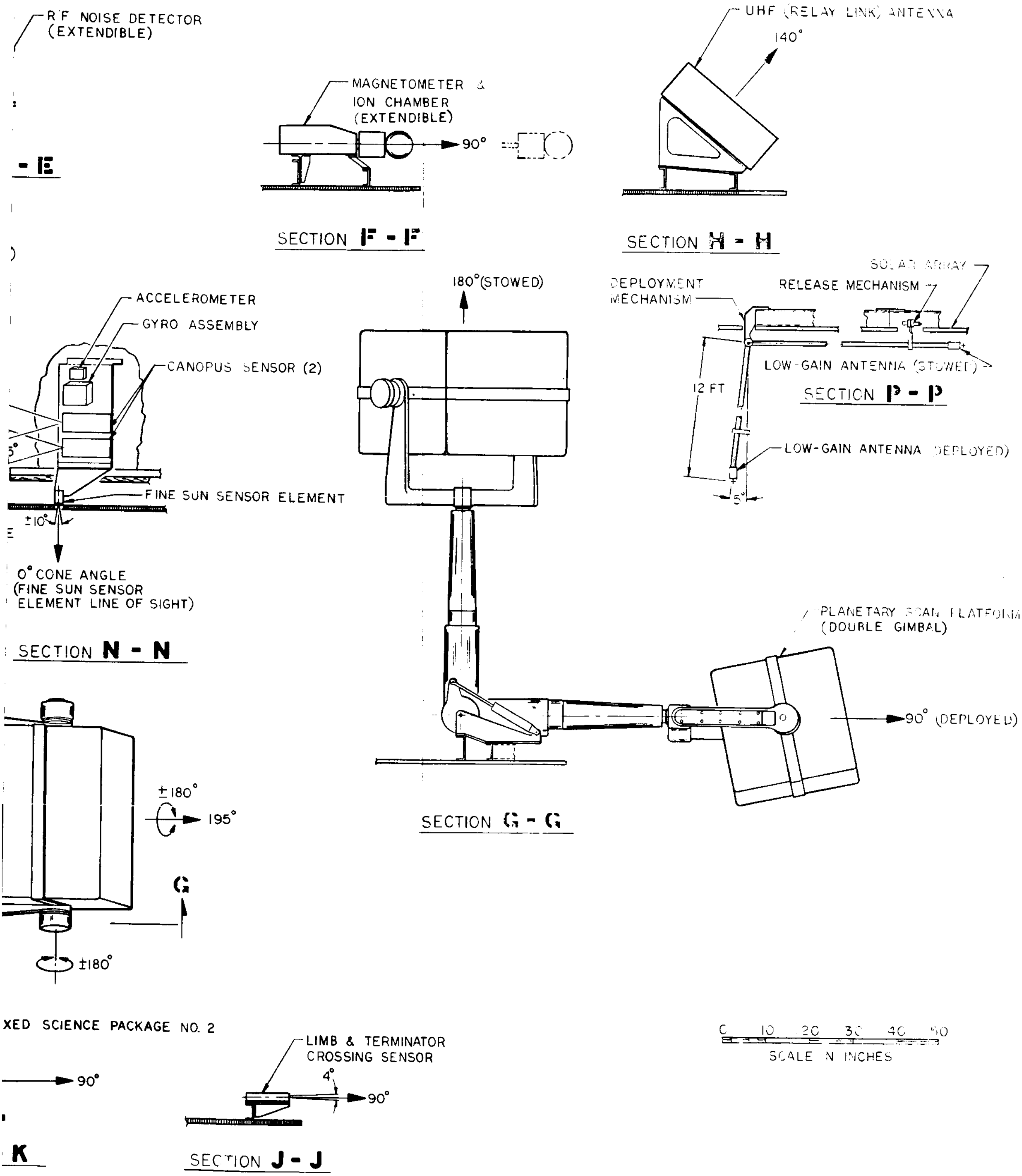


Figure 3. 1971 Voyager Spacecraft—Sensor and Appendages Geometry

4

3.9.2 Clock and Cone Angles

A line of direction (independent of location) relative to the spacecraft is defined by spacecraft clock angle and cone angle as follows:

1) Spacecraft cone angle is measured as the angle a line of direction makes with the spacecraft centerline. Zero cone angle corresponds to the aft direction (towards the sun in the cruise orientation).

2) The plane through the spacecraft centerline and parallel to the nominal line of sight of the Canopus sensor is called the zero clock angle plane. For any line of direction, consider its projection in a plane normal to the spacecraft centerline. The angle this projection makes with the zero clock angle plane measured positive in the clockwise direction looking aft is designated as the corresponding spacecraft clock angle.

3.9.3 Spacecraft Orientation Axes

Right-hand orientation reference axes $\vec{U}$, $\vec{V}$, $\vec{Z}$, in the spacecraft are defined as follows (see Figure 4).

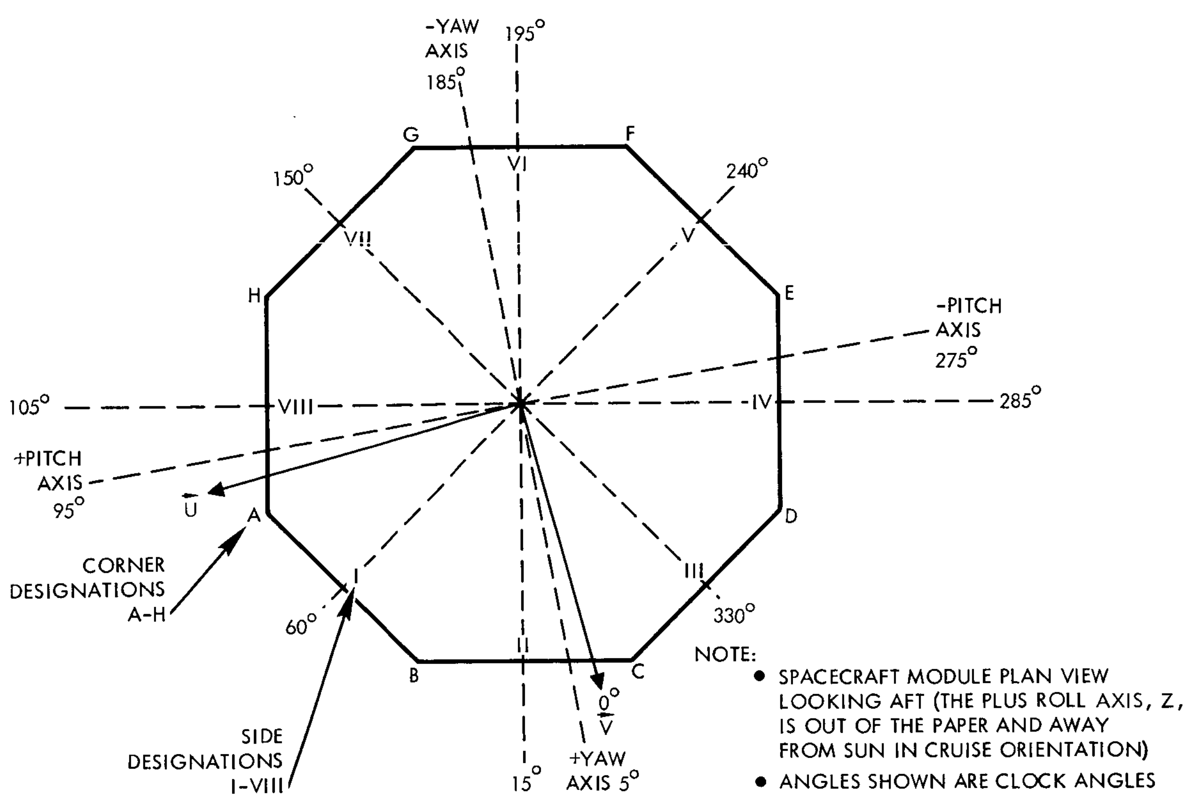


Figure 4. Spacecraft Geometry

a) The vector $\vec{Z}$ is along the spacecraft center-line and pointed in the forward direction.

b) The vector $\vec{V}$ is perpendicular to $\vec{Z}$ and pointed outward in the radial direction so as to have zero clock angle.

c) The vector $\vec{U}$ is equal to $\vec{V} \times \vec{Z}$ (with clock angle of 90 degrees).

Right-hand roll, pitch, and yaw control axes are defined for the spacecraft (planetary vehicle) as shown in Figure 4.

3.9.4 Spacecraft Corner and Side Designations

The corners and sides of the spacecraft module are designated as shown in Figure 4.

4. SUBSYSTEM REQUIREMENTS

This section defines the functional responsibilites of each subsystem and the salient requirements imposed upon the subsystem by system engineering considerations. Detailed performance, design, and interface requirements are given for each subsystem in Volume 2, including descriptive data. These subsystems, listed in paragraph 1. 3, will be designed in compliance with applicable design characteristics and restraints presented elsewhere in Section III.

4.1 Power

The flight spacecraft power subsystem will distribute power as required. When sun-oriented, the spacecraft derives power from photo-voltaic solar cells arranged on panels. When not sun-oriented, such as during maneuvers or during planetary orbit when the sun is eclipsed by Mars, power is derived from batteries. The batteries will also provide power for reacquisition in the event of a non-catastrophic loss of attitude control. Provisions will be made for charging the batteries as required. The subsystem design is described in 1 of Section II, Volume 2. The power requirement for the flight capsule is given in 7 and for individual equipment items in 5.5 of Section IV. The solar array configuration considerations are in 3.

4.1.1 Mission Considerations

a. Sun-Spacecraft Distance

The distance between the sun and the spacecraft may vary from 1.0 AU to 1.67 AU during the mission. For the 1971 mission, the distance at capsule separation may be as great as 1.47 AU.

b. Battery Operation

The batteries must be capable of supplying all required loads (this excludes capsule load) during the following periods:

- Launch to acquisition - maximum duration of 230 minutes
- Maneuvers - maximum duration of 120 minutes for each trajectory correction, orbit insertion, and orbit trim maneuver
- Eclipses - maximum duration of 138 minutes in orbit. Eclipse season begins 3 months after arrival.

c. Radiation

The intensity of trapped radiation at Mars is assumed to be 10^{-5} times that of earth, with the same particle energy distribution.

4.1.2 Power Requirements

Estimated average and peak load power requirements are listed as a function of mission phase in 5.2 of Section IV. The power requirement of 90 watts for reaction control gas heaters is regarded as a nonessential load and is not included in the total figures. The control subsystem gas heaters are not essential to the success of the mission and are used to increase lifetime only when the solar array capability exceeds the essential load requirements by a suitable margin.

a. Solar Array Performance Requirements

The required solar array outputs during the various mission phases in which the solar array is illuminated by and oriented to the sun are summarized below:

Mission Phase	Solar Array Output Required (watts)
Cruise	577
Planetary vehicle orbital operation	719
Flight spacecraft orbital operation	539

b. Battery Performance Requirements

The battery energy requirements listed in Table 4 consist of average spacecraft loads plus short duration loads for propulsion valves and engine gimbal actuators.

Table 4. Battery Energy Requirements

Mission Phase	Battery Output Energy Required (w-hrs)
Prelaunch, launch, injection and acquisition	944
Trajectory correction	1, 103
Orbit insertion	1, 283
Planetary vehicle orbital trim	1, 084
Capsule separation	1, 112
Flight spacecraft orbital trim	1, 082
Eclipses in Mars orbit (150 cycles)	1, 270

4.1.3 Transient Loads

The power subsystem is to be capable of accommodating transient loads of up to 150 per cent of the normal steady state for any load switched in flight.

4.1.4 Inverter Synchronization

Synchronization will be provided for the spacecraft by a precision oscillator located in the power subsystem. The frequency accuracy for inverter synchronization will be ±0.01 per cent. The inverters will have free-running capability.

4.2 Guidance and Control

The guidance and control subsystem controls the orientation of the Voyager spacecraft at all times after separation from the launch vehicle, including acquisition of celestial references and use of inertial devices for short term reference when required. It provides instrumentation to control velocity adjustments, implements antenna pointing control, and generates signals associated with terminator and limb crossings during Mars orbital operations. The subsystem design is as given in 2 of Section II, Volume 2. It will be in accordance with configuration considerations given in 3, the accuracy characteristics given in 2.1, and the mission sequence given in 1 of Section IV.

4.2.1 Celestial References and Cruise Orientation

The G and C subsystem will utilize a Sun-Canopus celestial reference system, and will be capable of automatic acquisition of these references from any attitude. In the nominal spacecraft cruise orientation, the aft roll axis points toward the sun and the spacecraft zero clock angle plane will contain the star Canopus.

4.2.2 Roll Maneuvers

The G and C subsystem will allow the spacecraft to roll about the sun-stabilized roll axis at a constant rate to permit magnetometer and star track calibration. It will provide a backup during such roll maneuvers so as to preclude spacecraft spin-up and depletion of the reaction gas supply. The G and C subsystem will also allow an incremental roll turn of ±2 degrees upon command.

4.2.3 Attitude Hold

For the period after separation from the launch vehicle until the sun is sensed (when separation occurs during earth eclipse), the G and C subsystem will maintain the attitude of the planetary vehicle to within 10 degrees of the initial separation attitude, in the presence of an initial angular rate of 1.5 degrees/sec.

In the event that the Sun or Canopus is occulted, the G and C subsystem will automatically maintain the spacecraft's attitude at time of the occultation to an accuracy of 2 degrees (3σ) in each axis for a period of 3 hours.

4.2.4 Antenna Pointing

The high-gain antenna and medium-gain antenna will be positioned by the G and C subsystem relative to the spacecraft so that they point towards the earth as required. The high-gain antenna has freedom about two axes, hinge and shaft, while the medium-gain antenna is only controllable about its hinge axis.

The antenna axes will be pointed to an accuracy of 1 degree (3σ) relative to the desired direction in the celestial reference system.

4.2.5 Thrust Vector Control

The G and C subsystem will provide thrust vector control during engine firings to the accuracy given in 2.1. Actuators will be utilized to rotate the thrust direction relative to the spacecraft by means of the engine gimbals.

4.2.6 Velocity Increment Control

The G and C subsystem provides a pulse rebalance accelerometer output signal to the computer and sequencer (C and S) subsystem so as to allow the engine to be shut down when the desired velocity increment has been achieved. The proportional error is to be no more than 0.1 per cent.

4.2.7 Fine and Coarse Attitude Control

The G and C subsystem will provide a coarse attitude control mode so that each spacecraft axis will oscillate in a limit cycle within 0.5 degree of its nominal reference orientation. The G and C will also provide a fine attitude control mode such that the corresponding accuracy is 0.25 degree.

4.3 S-Band Radio

The S-band radio subsystem implements a two-way radio link between the spacecraft and the DSN stations to provide telemetry to earth; command capability to the flight spacecraft; and doppler, ranging, and angular tracking for orbit determination. The subsystem design will be as given in 3 of Section II, Volume 2. The RF interface with the launch vehicle is given in 6 and the interface with the TDS is given in 8.

4.3.1 Communication Capability

As a minimum, communications will be maintained from prelaunch to encounter plus 2 months. The capability to maintain communications to encounter plus 6 months is a design goal.

Uplink communications with each planetary vehicle can be accommodated from injection through end-of-mission by a low-gain nonsteerable antenna. Downlink telemetry is required from each planetary vehicle within the view capability of the DSN, from prelaunch to interplanetary orbit injection.

From interplanetary orbit injection to injection plus 30 days, the capability for simultaneous doppler tracking, telemetry and command of both vehicles is required on a continuous basis, within view of the DSN. Simultaneous turnaround ranging capability is required for both vehicles (except during command operation).

From injection plus 30 days to Mars encounter minus 5 days, the capability for simultaneous doppler tracking, turnaround ranging (except during command operation), telemetry, and command is required, with DSN operation on a time-shared basis for the two vehicles.

From Mars encounter minus 5 days to Mars encounter plus 8 days, the capability for simultaneous doppler tracking, turnaround ranging (except during command operation), telemetry, and command is required for the planetary vehicles. From Mars encounter to termination of orbital operations, continuous telemetry from both planetary vehicles is required. Doppler tracking, turnaround ranging, and command of the vehicles will be on a time-shared basis during this period.

4.3.2 Communications During Maneuvers

The capability for uplink command and downlink telemetry will exist during roll maneuvers following sun acquisition. Continuous communication while the spacecraft is changing orientation for a maneuver is not required. (The data storage subsystem is required to store engineering data during maneuvers for subsequent playback.)

4.3.3 Communication Distance

The S-band radio subsystem will be compatible with communication distances as follows:

At encounter:	80×10^6 km minimum
	180×10^6 km maximum
Encounter + 2 months:	150×10^6 km minimum
	265×10^6 km maximum
Encounter + 6 months:	310×10^6 km minimum
	390×10^6 km maximum

4.3.4 Data Rate

The telemetry data rate of each planetary vehicle will not exceed 15,000 bits per second. The peak total telemetry rate from both vehicles will not exceed 15,000 bits per second.

4.3.5 Bit Error Rate

The signal characteristics and radiated power of the downlink transmission will be such that the telemetry bit error rate for reception by DSN stations does not exceed 5×10^{-3}.

4.3.6 Radiated Power

The nominal radiated power will be 50 watts.

4.4 Telemetry Subsystem

The telemetry subsystem accepts data outputs from the flight capsule, other flight spacecraft subsystems and transducers and processes this data for input to the radio transmitter as a serial pulse-code-modulated digital signal. The subsystem design will be as given in 5 of Section II, Volume 2. The interface with the flight capsule and the relay link subsystem will be in accordance with 7.

4.4.1 Data Modes

The following four data modes are to be provided:

- Mode 1 is engineering data only, and includes very low rate capsule engineering data while the capsule is part of the planetary vehicle. This mode is used during launch, maneuvers, and when required for failure analysis
- Mode 2 used during cruise is a combination of engineering data and real-time science data, such as field and particle measurements
- Mode 3 is the playback of non-real-time data such as stored science instrument data, stored capsule data, and data recorded during maneuvers
- Mode 4 is selected engineering data plus C and S memory readout. This mode is used prior to maneuvers for verification of maneuver commands and for checkout of guidance and control equipment prior to use.

4.4.2 Data Rates

The following six data rates will be provided:

Use	bits/sec
Orbital operations	15,000
	7,500
	3,750
	1,875
Launch, cruise maneuver	234.4
Emergency	7.3

4.4.3 Synchronization

Synchronization will be accomplished by a pseudonoise (PN) code of 63 bits cycling at the telemetry word rate and providing bit and word synchronization. Data and the PN code are used to modulate two sub-carriers, the outputs of which are linearly mixed to produce a signal for input to the radio transmitter.

The engineering data format includes a 21-bit code at the beginning of each frame, enabling rapid frame synchronization in the decommutation computer. A word of the data frame contains a frame count, enabling identification of subframe and sub-subframe channels, and a data mode word indicating the telemetry mode and data rate.

4.4.4 Data Sampling

Four data sampling rates are to be provided, with the following channels available for data, after fixed word requirements are met:

- 27 channels sampled at the frame rate
- 70 channels sampled at 1/20 of the frame rate
- 126 channels sampled at 1/200 of the frame rate
- 209 channels sampled at 1/400 of the frame rate.

The total number of channels devoted to engineering measurements, considering each channel to have seven bits, is 432. The manner in which the commutators are implemented will be flexible enough to permit adjustments between sampling requirements and commutator capacities.

4.5 Data Storage

The data storage subsystem will record digital data from the flight spacecraft science subsystem and engineering and cruise science data from the flight spacecraft telemetry subsystem during maneuver phases. The capability of recording the high rate capsule data from the relay link receiver is within the capsule system and not part of the flight spacecraft data storage subsystem. The subsystem design will be as given in 6 of Section II, Volume 2. The interface with the science subsystem is given in 9.

4.5.1 Tape Recorder Performance

The following tape recorders will have the characteristics as specified:

a) Television tape recorders: 2 recorders, each with a capacity of 10^8 bits; data will be accepted at 90 k-bits/sec

b) Spectrometer tape recorder: 1 recorder with a capacity of 10^7 bits; data will be accepted at 2 k-bits/sec

c) IR scanner tape recorder: 1 recorder with a capacity of 10^7 bits; data will be accepted at 5 k-bits/sec

d) Fields/particles tape recorder: 1 recorder with a capacity of 10^6 bits; data will be accepted at 700 bits/sec

e) Spacecraft Tape Recorder: 1 recorder with a capacity of 1.5×10^6 bits; data will be accepted and played back at 234.4 bits/sec.

The science tape recorders (a to d above) will be capable of being played back synchronously at data rates of 15, 7.5, 3.75, and 1.875 k-bits/sec.

4.5.2 Recording and Playback

The tape recorders will be capable of recording data, clock, and a data gap signal. The science subsystem will supply these signals for science data and format the data as required. Data will be in serial digital form. Recording will be initiated and halted by pulse-type commands. Gaps in the recorded data will be left to allow real-time data to be interleaved during playback. Signals will be recorded to indicate to telemetry the beginning and end of such gaps.

The recorders will embody a servo speed control system which is capable of controlling the playback tape speed and synchronizing the recorded clock and data with the telemetry data rate supplied by the telemetry subsystem. The tape recorders will supply the telemetry with a data gap signal, indicating the absence of data on the recorder output. Playback will be initiated and halted by pulse-type commands. To permit the reproduction of stored data more than once, the recorders will incorporate a tape advance mode, initiated and halted by pulse-type commands.

4.5.3 Control and Formatting

All required input buffering and data formatting for the science data will be performed by the DAE. The science subsystem will furnish signals to control the start and stop operations of the science data tape recorders in the record mode. The C and S subsystem will furnish signals to control the start and stop operation of the tape recorders in the playback mode. A signal from each science tape recorder will be furnished to the science subsystem, indicating that the tape is fully loaded.

4.6 Command

The command subsystem demodulates command signals from the output of the S-band radio subsystem, decodes the resulting digital commands, and provides outputs to the C and S subsystem, the science subsystem, the flight capsule, and to other spacecraft components as required by the addresses in the command data. The subsystem design will be as given in 7 of Section II, Volume 2, and will implement the commands listed there.

4.6.1 Classes of Commands

The command subsystem will be capable of processing commands as given below, with associated data:

a) Direct discrete commands: approximately 170

b) Quantitative commands (serial outputs)

- 7-bit serial words to each of 2 science decoders
- 20-bit serial words to each of 6 registers in the C and S
- 12-bit serial words to 3 antenna pointing registers in the G and C
- 32-bit serial words to 2 C and S command input registers, 2 science sequencer command input registers, and 6 flight capsule registers.

4.6.2 Performance

The probability of no response action in the command subsystem attributed to false out-of-lock indications and/or bit errors will be less than 10^{-4} for each command transmission. The bit error probability for the command link at threshold will be less than 10^{-5}, and probability of a word error will be less than 10^{-8}.

Synchronization acquisition time will be minimized, and the command subsystem will process command words at the rate of 1 bit/second ± 10 per cent, after detector lock has been acquired.

4.6.3 Hard Line Command Input

The command subsystem will accept and process commands received by a hardline as well as by the RF link.

4.6.4 Command Frequency Spectrum

The command frequency spectrum will be far enough removed from the RF carrier so that spectrum components do not interfere with the phase tracking of the carrier and that the frequency components are not distorted by the carrier tracking loop.

4.7 Computing and Sequencing

The computing and sequencing (C and S) subsystem provides timed outputs to other flight spacecraft subsystems and the flight capsule to achieve suitable switching and sequencing. The design is as given in 8 of Section II, Volume 2, and the C and S is to be capable of implementing the commands listed there. The interfaces with the capsule system and the science subsystem will be in accordance with 7 and 9 respectively. The C and S will be compatible with the mission sequence given in 1 of Section IV.

4.7.1 Spacecraft Sequencing

The C and S will provide the capability for automatically controlling a nominal spacecraft mission sequence, including all events and operations not associated with trajectory dispersions. However, it will provide a capability allowing preselected commands to require enabling by the MOS prior to their implementation. Provision will be made such that all mission dependent stored commands can be revised by ground command.

Long term predictable events (such as Canopus sensor update and antenna pointing) will be timed from a mission clock initiated prior to launch, and covering a period of 15 months. A capability to reference some sequences from mission events such as planetary vehicle separation, Mars orbit insertion, or terminator/limb crossings is also to be provided.

4.7.2 Test Operations

During ground testing, the C and S will be capable of being automatically or manually controlled by the OSE through a real time or speedup mission, while verifying all input and output interfaces. The capability will be provided for individual initiation of all inputs and outputs in a nonsequential mode. The design will allow all modes to be commanded by either onboard logic, ground command via RF link, or via a direct access circuit.

4.7.3 Thrust Termination

The C and S will provide for integration of the output from a pulse rebalanced accelerometer so as to generate an engine cutoff command.

4.7.4 Telemetry Readout

Outputs to telemetry are to include data mode, data rate, recorder playback commands, and C and S engineering data. C and S memory words will be issued one at a time to a register which telemetry will sample at a rate determined by the data mode. When the sampling is completed, the next word in memory will be loaded into the register. Both a low rate and a high rate telemetry data mode will be provided. The high rate is to be capable of being commanded separately as a pre-maneuver mode (telemetry mode 4).

4.7.5 Mission Clock Calibration

The C and S will have the capability to calibrate the mission clock against astronomical time by readout of the mission clock register to telemetry, so that the time of readout can be determined by the MOS.

4.7.6 C and S Outputs

The C and S will issue quantitative data as a serial bit stream synchronized to the spacecraft master oscillator. Discrete commands will only be low-level outputs.

4.8 Structural and Mechanical

The structure serves to physically integrate the many equipment elements comprising the flight spacecraft. It provides sufficient strength, rigidity, and other physical characteristics necessary to maintain alignment between components, acceptable static and dynamic load environments, and to support components, assemblies, and the flight capsule during preflight, boost, and spaceflight operations. Other design objectives are to provide meteoroid protection, ease of maintenance, accessibility, and flexibility to changes in the mission and subsystem requirements. The mechanical elements provide deployment and release of spacecraft equipment as required.

The structural and mechanical subsystem design is as given in 9 of Section II, Volume 2. It will be in accord with the structural design criteria given there, the configuration requirements of 3, the environmental requirements of 5.5, the launch vehicle interface of 6, and the flight capsule interface of 7.

4.9 Pyrotechnics

The pyrotechnic subsystem includes the electro-explosive devices and their associated firing circuitry to provide various actuation and disconnection functions for the flight spacecraft. The subsystem design is as given in 10 of Section II, Volume 2 and will be in accordance with the safety criteria in 5.6 of Section II and the electrical design requirements of 5.3.

4.9.1 Functions

The pyrotechnic subsystem is utilized as follows:

- Launch vehicle - planetary vehicle separation (commanded from LV)
- Spacecraft-capsule emergency separation and disconnect

- Antenna release (3)
- PSP release and uncaging
- Propulsion valve openings (N.C. valves)
- Propulsion valve closings (N.O. valves)
- Jettisoning of science covers (if required)

4.9.2 Firing Considerations

The use of electro-explosive devices will be inhibited until conditions of a safe and arm device are satisfied. The safing circuitry is actuated by double-dual redundancy at planetary vehicle-launch vehicle separation. Arming is caused by redundant circuits in groups at some time before actuation of the respective events is required.

Firing circuitry is concentrated in a pyrotechnic control assembly. Initiating current is independent of the main power system, requiring only raw AC power input for charging the firing discharge circuits. All firing circuitry is to be redundant. All fragments and gas produced by electro-explosive devices will be retained by the element.

4.10 Temperature Control

The temperature control subsystem ensures satisfactory thermal conditions for spacecraft components. The design for this subsystem is given in 11 of Section II, Volume 2. The thermal operating conditions to be provided for spacecraft equipment are given in 5.5 of Section IV.

4.10.1 Design Conditions

Temperature control design depends on configuration and layout data and power data given in 3 and 5.2, respectively, of Section IV. Design conditions with additional data are given below:

a. Prelaunch

When encapsulated in the fairing, the spacecraft will be exposed to a 100^{o}F gaseous environment (12 per cent ethylene oxide, 88 per cent freon) for about 10 hours. This will be followed by a nitrogen purge.

Cooling of the spacecraft, when encapsulated in the fairing, will be accomplished by a flow of air (45 to 65°F) within the fairing. Flow rate, humidity and particulate matter content are to be specified.

b. Launch and Injection

Thermal effects from the launch vehicle shroud and other environmental considerations are given in 6.7. A coast period of 2 to 90 minutes in a 100 nmi parking orbit must be accommodated, during which earth albedo, infrared emission, and eclipse conditions are to be expected.

c. Injection to End of Mission

The planetary vehicles will be attitude stabilized, using the Sun and Canopus as reference objects, except for periods associated with earth eclipse; initial stabilization; maneuvers for midcourse corrections; retropropulsion firing; Mars orbit trim and capsule orientation; and Mars eclipses. A Mars eclipse of 2.3 hours represents the longest such period anticipated. Solar thermal radiation, corpuscular radiation, micrometeoroid distribution, and Mars albedo and infrared emission will be as given in Reference 6.

4.10.2 Flight Capsule

The flight capsule will be thermally coupled to the flight spacecraft essentially by conduction alone, through the capsule interstage structure. This coupling will be as small as possible. Radiation coupling to the external equipment, the louvered radiative areas and to the insulated main compartment and solar array, is also be be small. As a goal, the design of the capsule system will minimize the heat loss from the spacecraft forward of the field joint.

4.10.3 Plume Heating

The radiative and convective characteristics of the spacecraft engine will be evaluated, with a firing duration of 400 seconds. The apparent emissivity of the plume is to be taken as 0.1. Impingement on the magnetometer and low-gain antenna will be represented by a heating of 20 Btu/hr sq ft.

4.11 Cabling

The cabling subsystem provides the means by which the various elements of the flight spacecraft are electrically connected with each other and with the launch vehicle and capsule as required. It consists of interconnecting cabling throughout the spacecraft, hardware for electrically connecting the spacecraft with interfacing systems, junction boxes for the distribution of electrical functions, umbilical cabling peculiar to the spacecraft, and system level test points including hard-line test connectors.

The subsystem design is to be as given in 12 of Section II, Volume 2. It is to be in accord with the electrical design criteria in 5.3, the launch vehicle electrical interface in 6.4, the capsule interface in 7.5, and the system functional block diagram in 2 of Section IV.

4.12 Spacecraft Propulsion

The spacecraft propulsion subsystem provides velocity increments for interplanetary trajectory corrections, Mars orbit insertion, and orbit trim both prior and after capsule separation. The subsystem design is to be based on the LEM Descent stage as given in Section III of Volume 2. It will be in accordance with configuration considerations in 3 and the weight allocation for the complete flight spacecraft given in 2.3.

4.12.1 Propulsive Capability

The nominal propulsive capability is given in Table 5.

4.12.2 Impulse Control

The propulsion subsystem will be capable of providing a minimum nominal impulse of 750 lb/sec at the low thrust level. The impulse predictability for propulsion shutdown will be 75 lb-ft/sec (3σ) at low thrust and 470 lb/sec (3σ) at high thrust.

Table 5. Propulsive Capability

Maneuver	Thrust Level (lbf)	Required Velocity Increment (km/sec)	I_{sp} (sec)	Mass* Fraction
Interplanetary trajectory correction	1040	0.2	285	0.070
Mars orbit	7750	2.0 (required)	305	0.488
		2.16 (design estimate)		0.512
Planetary vehicle orbit trim	1040	0.1	285	0.035

*Propellant mass consumed divided by initial mass.

4.12.3 Engine Firings

The propulsion subsystem will be capable of at least the following number of firings:

- 3 interplanetary trajectory corrections
- 1 Mars orbit insertion
- 1 planetary vehicle orbit trim
- 1 flight spacecraft orbit trim

The propulsion subsystem will provide within itself for any propellant settling or propellant feed capability arising from an initial free fall condition.

4.12.4 Space Storage

The propulsion subsystem will be capable of space storage as required by the mission profile. The subsystem will be free of sensible leakage. This corresponds to leakage below 10^{-6} standard cubic centimeters per year.

4.13 Planetary Vehicle Adapter

The planetary vehicle adapter comprises all spacecraft-supplied structure, cabling, and hardware located between a planetary vehicle inflight separation joint and the associated points of attachment to the launch vehicle nose fairing. The adapter design will be as given in Section V of Volume 2. It will be in accordance with the structural design criteria in 9 of Section II, Volume 2, the configuration requirements of 3, and the environmental requirements of 5.5. The interface with the launch vehicle will be as given in 6.

5. SYSTEM DESIGN STANDARDS AND REQUIREMENTS

5.1 Parts, Materials, and Processes

Parts, materials, and processes are to be standardized and controlled throughout design and fabrication. This includes selection, procurement, and utilization of parts and materials, and selection and utilization of processes.

All parts and materials will be selected on the basis of suitability for the intended application with emphasis on reliable performance during the prelaunch, launch, Earth-to-Mars transit, and Martian-orbit phases of the mission. In addition, all parts, materials, and processes will be selected on the basis of capability to perform in accordance with requirements during the complete test and operational lifetime, as established by test program evaluations and applicable specifications.

All materials used in the spacecraft will be selected from a list of materials compiled by TRW and approved by JPL. All manufacturing processes used in spacecraft manufacture will be selected from a list of process documents compiled by TRW and approved by JPL.

The spacecraft system contractor will establish a parts, materials, and processes standardization and control program for implementing JPL requirements.

Attempts to advance technology by the utilization of parts, materials, and processes which cannot demonstrate a history of reliability are prohibited unless such advances are demonstrated to be necessary to meet minimum system performance requirements.

All parts, materials, and processes will be selected on the basis of capability to perform in accordance with functional requirements under the test and operating environments of the assemblies in which they are used. Environmental requirements are given in 5.5.

The number of electronic piecepart types employed in the design of the spacecraft system equipment will be held to a minimum.

Parts, materials, and processes will comply with the magnetic requirements given in 5.4.

5.2 Electronics Equipment Packaging

Arrangement of equipment in the spacecraft is to meet the following guidelines:

a) Grouping of subsystems for short cable runs, minimum electrical interference, and minimum line losses

b) Placement of heat-dissipating equipment for low power density

c) Placement of equipment for proper spacecraft mass distribution

d) Grouping of equipment for ease of installation and checkout.

5.2.1 Equipment Mounting Panels

In general, internally installed electronics equipment assemblies are to be mounted to sides III and VII of the LEM module. Each of these sides is made up of four panels to accommodate mounting of equipment. Equipment panel layouts and related parameters are described in 3 of Section IV.

The individual equipment panels will be hinged along the sides to facilitate access to installed equipment. Cable loops are to be provided between the system harness and the individual panel assembly harnesses to permit rotation of the panels about the hinges without undue strain and flexing of the harness.

Positive mounting provisions for assemblies and harness are to be incorporated in the equipment mounting panels as shown in Figure 5. The assemblies are to be of rigid construction and are secured to the mounting rails so as to contribute to the over-all mechanical strength and stiffness of the panel. The natural frequency of a panel with all equipment installed is to be greater than 75 cps, with a design goal of 100 cps. The lowest primary vibration resonance within an assembly is to be above 400 cps.

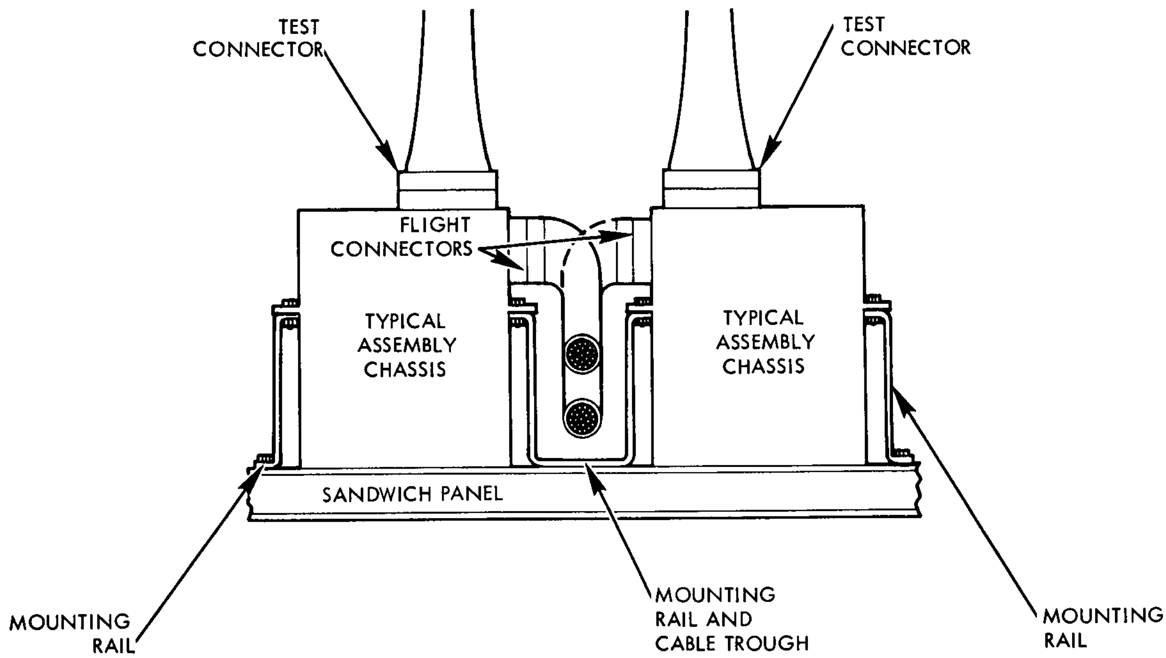


Figure 5. Panel Mounting of Equipment

5.2.2 Standard Package Form Factor and Installation

Panel-mounted electronics equipment is to be packaged so that external dimensions and mounting provisions correspond to the outline and mounting drawings as shown in Figure 6. The standard dimensions are 7 inches wide by 6 inches high, while length may vary up to a maximum of 30 inches. The width dimension is fixed by the spacing between the mounting rails, and can be varied only by eliminating a rail and/or

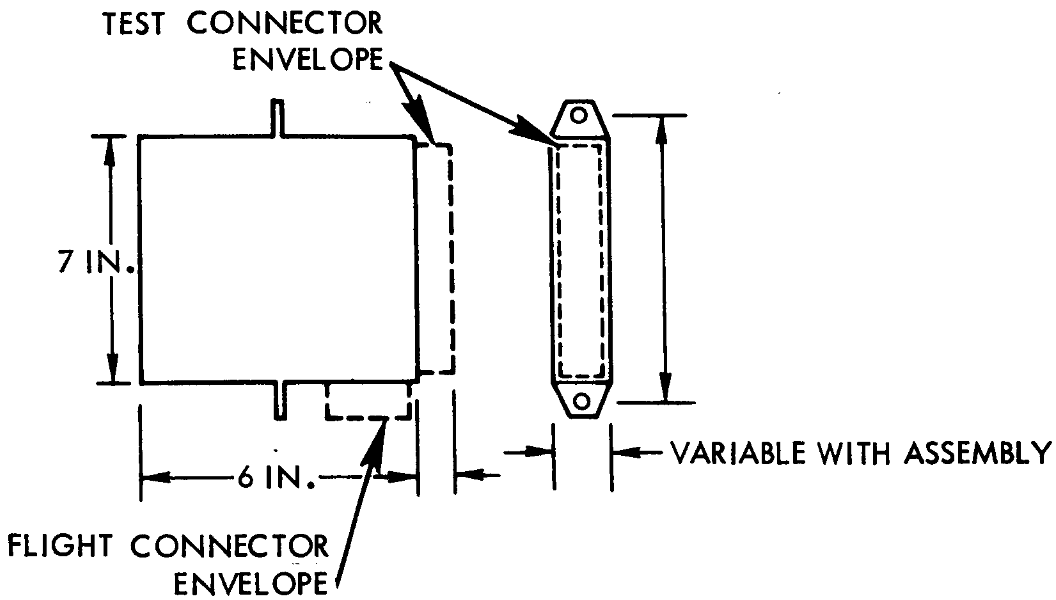


Figure 6. Standard Package Geometry

cable channel and spanning across two rows. The width in this case is 19 inches or 15.5 inches. The height dimension is restricted to a maximum of 6 inches near the hinge side of a panel, but at free end locations could be as high as 10 inches. Deviation from the standard configuration is permissible for odd-shaped equipment.

Equipment which is not panel-mounted, such as sensors, need not conform to the standard package size. Special equipment mounting and location requirements are accommodated. Subsystems will be mounted on single equipment panels whenever possible.

5.2.3 Thermal

Panel-mounted assemblies are to be attached to the rails with positive dimensional tolerance to assure intimate contact between the base of the assembly and the panel.

Special provisions will be required to ensure good conductance for equipment possessing a base area average power density greater than 0.3 watt per square inch. The base area of high heat dissipating assemblies will be sized so that the power density in no case exceeds 1.4 watts per square inch.

Temperature control within an assembly is generally passive, depending principally on conduction to the base and from there to the mounting panel for radiation to space. High heat dissipating components are to be mounted as close as practical to the base of the assembly. Mechanical joints in the conduction path are held to a practical minimum.

Active temperature control within as assembly is employed only on gyros and other devices where temperatures must be maintained within a very close tolerance.

Part temperatures are not to exceed derated temperatures as established from mission requirements, when the equipment is operating within the temperature extremes provided by the spacecraft thermal control system.

The design allows for maximum radiative heat transfer between assemblies and between surrounding spacecraft structure.

5.2.4 Weights

The equipment is designed to achieve minimum weight consistent with performance, reliability, and producibility requirements.

5.2.5 Electrical and Magnetic Considerations

Design criteria for electrical interface grounding and cabling is presented in 5.3. Criteria for test points and connector locations are given in 5.6.

Packages containing RF circuitry are to be designed RF tight to the extent required for compatible operation with other equipment. Seals are generally of a tongue and groove type labyrinth with gaskets added as required.

Shielded or twisted wires are used to satisfy spacecraft magnetic field requirements. The location and orientation of magnetically critical components are to be controlled in accordance with 5.4.

5.3 Electrical Design

The electrical design of the Voyager spacecraft system will be such as to insure compatibility and satisfactory interfaces with the other Voyager systems. Similarly, it will provide for compatibility and proper operation within the spacecraft system. The proposed basis to achieve this is summarized here and discussed in detail in Appendix G.

5.3.1 System Electrical Ground Reference

A common electrical ground reference will be established for any system configuration. This applies to interconnected arrangements involving the flight spacecraft, its electrical operational support equipment, and any other associated equipment. The mechanical structures of the flight spacecraft, the EOSE, etc., in conjunction with the following techniques, provide the basic low-impedance reference.

1) Electrical bonding, when possible, will be accomplished by metal-to-metal contact over the entire area of surfaces which are held in mechanical contact.

2) The electrical bonding technique employed, where metal-to-metal contact is not used, will provide a bonding impedance not to exceed 2.5 milliohms DC and 80 milliohms at 20 mc across any bond.

3) The use of bonding straps will be minimized. When required, bonding straps will be of solid metal, having a length-to-width ratio not be exceed 3 to 1.

4) All subsystem mechanical structures (both EOSE and spacecraft) will be electrically bonded to the system electrical reference.

5) All electrical connector shells will be electrically bonded to the associated equipment ground structure.

5.3.2 Interface Circuit Grounding

The system (EOSE and spacecraft) electrical circuit interface will conform to the following grounding requirements. The specific design details are contained in the cabling subsystem description in 12 of Section II, Volume 2.

1) Both primary and secondary system power circuits in the EOSE and spacecraft will be connected to the system electrical reference (either EOSE or spacecraft structure) at only one point per circuit.

2) The various system power circuits, AC (400 cps and 4 kc) and DC (internal equipment power), will be DC isolated from each other by a minimum of 10 megohms everywhere in the system, except at the point of connection to the system electrical reference.

3) Interface signal circuits which are unbalanced to ground, including pulse, change of state, and RF circuits, will be connected to the system electrical reference at both the source and load end.

4) Analog interface circuits will be connected to the system electrical reference at the load end of the circuit only.

5) Balanced interface circuits will be connected to the system electrical reference at the point of balance only.

6) Interface signal circuit ground (returns) will be DC isolated from each other and from power ground everywhere in the system except at the circuit (electrical reference) ground points.

7) Shields of all interface circuits will be individually connected to the system electrical reference at both the source and load end and at each shield discontinuity.

8) All EOSE-spacecraft interface circuits will conform to interface circuit grounding requirements.

5.3.3 Shielding and Cabling

Shielding and cabling considerations are given below:

1) The outer enclosure of each assembly will be designed to provide the maximum amount of RF shielding that is practical.

2) The EOSE racks will be designed to provide a high degree of RF shielding. Special provisions will be made to incorporate rack shielding requirements into the design of the EOSE.

3) The spacecraft basic structure, solar panels, forward and aft covers, and propulsion structure will be electrically bonded together to provide RF shielding at the spacecraft transmitter/receiver frequencies.

4) Shielded cables will be employed on all RF, digital, pulse clock, change of state, and command signal lines within the system.

5) Power cables will generally be routed on twisted pair, triad, or quad cabling. These cables will generally not be shielded.

6) S-band and UHF, RF and video signals (transmitting and receiving) will employ solid-jacketed or double-shielded cables.

7) Twisted pair shielded cables will be employed on all pyrotechnic cabling.

8) Analog signals will generally not be shielded. These signals will be routed on twisted cabling.

9) Individual signal types and power will be physically and electrically isolated within the system cable harnesses to minimize common circuit impedance, crosstalk, radiation, and pickup.

5.3.4 Circuit Isolation and EMI Suppression

Any individual equipment item or group will meet the established requirements for generated EMI, susceptibility, and power characteristics while performing normal system operations. General design considerations in this regard are as follows:

1) Power conductors will be decoupled directly at the entering or exiting interface of an equipment. RF decoupling components of the feedthrough variety will be employed.

2) Analog circuits will be decoupled via RF filtering components as required to eliminate power switching and digital signal coupled energy.

3) The squarewave AC power rise and/or fall time will be limited to a minimum value of 20 microseconds.

4) Command the telemetry pulse, clock, and change of state signal rise and/or fall times will be no shorter than 10 microseconds.

5) Suppression devices will be employed to minimize power bus transients during heavy load switching.

6) Suppression devices will be employed across all relay switching contacts and solenoid devices to reduce generated power bus transients.

5.3.5 Pyrotechnic Firing Circuits

The electrical design requirements for pyrotechnic firing circuits are presented in 10 of Section II, Volume 2.

5.4 Magnetics

All equipment will be designed using Voyager approved parts, materials, and processes as per 5.1. All Voyager parts and materials will be nonmagnetic, except that a deviation will be allowed for the use of a magnetic part or material in a specific application when such use can be shown to enhance the probability of mission success through increased reliability or reduced technical and schedule risk. Processes will be specified and controlled so as to avoid any deleterious effects on magnetic properties.

Each equipment item will meet a specified magnetic constraint (expressed as the maximum field referred to 1 foot). Current estimates of these constraints are given in the table of component design parameters in 5.5 of Section IV. These magnetic constraints will be allocated in accordance with the analysis procedures outlined in Appendix F.

When a deviation from the magnetic allocation is necessary for reasons given above, the vector magnetic moment of the equipment item will be specified in the corresponding specifications. Such a deviation will be incorporated in the current table of component design parameters, and the location and orientation of the end item will be shown on the appropriate magnetic properties control drawing.

A drawing (or set of drawings) will be maintained and controlled to show the current status of the location and orientation of all magnetically critical equipment items, the location and orientation of the magnetometer sensor, the allocated levels for all equipment items, and the predicted field at the magnetometer.

The flight spacecraft will be designed to meet the magnetic requirements of 9.1.4.

5.5 Environmental Requirements

All spacecraft equipment will be designed to survive and operate as appropriate in the environments established by the applicable specifications. Definition of specific environmental conditions for particular equipment items will be based on the Voyager Environmental Predictions Document (Reference 6) and on analysis and test data relative to spacecraft system characteristics. Environmental considerations for the Task B preliminary design are based mainly on Reference 6.

5.5.1 Launch Phase Loads

The spacecraft will be designed to withstand the environment associated with all launch and boost phase quasistatic and transient loads. The ability of the design to sustain these loads will be assessed. The design load factors are to be derived by the logical combination of the rigid body and low-frequency accelerations presented in Reference 6. The dynamic amplifications of the forcing functions are to be based on the following assumptions: a) at lift-off, the longitudinal excitation will decay to 10 per cent of the initial values in 20 cycles, which corresponds to approximately 1.8 per cent of critical viscous damping for the launch vehicle; b) the lift-off transient will occur in the first two longitudinal modes of the launch vehicle (f_1 = 4.0 cps and f_2 = 8.7 cps), with energy participation factors of 0.7 and 0.3, respectively; c) at engine cutoff, the appropriate transient will occur in the first longitudinal mode (f_1 = 7.1 cps) of the launch vehicle after the thrust has rapidly diminished to zero.

The stiffness characteristics of major spacecraft assemblies are to be selected to avoid deleterious coupling with the launch vehicle resonant frequencies and to minimize the dynamic amplification to the flight capsule. The following natural frequencies for major structural assemblies are to be exhibited:

- Spacecraft with respect to nose fairing (axial) $\geq$ 10 cps
- Spacecraft with respect to nose fairing (lateral) $\geq$ 5 cps

- Capsule with respect to spacecraft > 100 cps
- Aft equipment module with respect to spacecraft > 50 cps
- Appendage support with respect to spacecraft > 75 cps
- Equipment mounting doors with respect to spacecraft > 75 cps

5.5.2 Micrometeorite Protection

The particle flux, velocity, and density data of Reference 6 will be used in conjunction with rational analysis to determine the level and amount of micrometeorite protection required. The depth of penetration is to be considered uniform over all exposed compartments which contain pressurized units and sensitive electronic equipment. The exposed area is to be based on the requirement that the flight spacecraft must perform its intended mission with the flight capsule removed from its interface.

A double wall low-density core shield to disperse the particles encountered will be considered. This concept will be used for all exposed surfaces except for the two faces of the prismatic equipment compartments and the aft or sun-oriented face of the spacecraft. For these exceptional cases, resistance to penetration is to be inherent in the aluminum structural sandwich panels of the solar array and equipment mounting doors.

5.5.3 LEM Descent Engine Loads

The spacecraft is to be designed to have sufficient strength, rigidity, and other physical characteristics to survive the steady state and transient loads generated during the operation of the LEM descent engine. For the design of engine support structure a maximum axial thrust load of 9000 pounds will be used. This is derived by combining the medium thrust level of 7800 pounds with a 3σ thrust overshoot of 1200 pounds. This forcing function is characterized as a narrow-band random oscillation having a center frequency of 35 cps.

For equipment other than the thrust structure, an axial design limit load factor of 1.0 g is to be used. This is intended to account for the condition which would occur at retropropulsion cutoff with the capsule having been previously jettisoned, and with the allocated propellant for interplanetary trajectory corrections having been consumed.

The effect of the cyclic application of load will also be considered. Cyclic loads include those transmitted during start, shutdown, normal burning, and thrust vector control. The final assessment of the environment induced by the LEM descent engine and the integrity of the flight spacecraft structure will be based on suitable test data.

5.5.4 LEM Descent Engine Plume Heating

The radiative and convective characteristics of the LEM descent engine plume, as a heat source, will be determined in order to evaluate its effect on the solar array and other spacecraft equipment. The radiant energy interchange between the engine and spacecraft equipment, in conjunction with plume heating, is to be considered; however, combined effects are to be based on the premise that the radiatively cooled nozzle extension is replaced with an ablatively cooled skirt. Heating of the high, medium, and low gain antenna from plume radiation and impingement will also be evaluated.

5.5.5 Equipment Temperature Limits

The estimated operating and nonoperating temperature limits for spacecraft equipment are given in 5.5 of Section IV.

5.6 Support System Considerations

The following requirements are placed on flight spacecraft design by the ground test operation. These requirements will also be taken into account in the design of the OSE.

5.6.1 Telemetry Usage for Test

The spacecraft test philosophy is to be based on making maximum use of telemetry information for ground checkout. This is to minimize the number of hardlines to the spacecraft and to allow testing in a mode more closely approximating the flight configuration. This philosophy requires allocation of sufficient telemetry to isolate faults to the provisioned spares level. Analog telemetry functions are to be sampled at a sufficiently high rate so that all system parameters can be adequately evaluated. This may require a commutator speedup mode for ground test pruposes.

Provisions to verify receipt and execution of all commands as well as current command status is to be provided via telemetry. This is to include delayed commands sent to storage in the computer and sequencer as well as the direct commands sent via the command subsystem.

5.6.2 Command Allocation

Consistent with the minimum hardline test philosophy, a certain number of commands will be allocated for test purposes. These commands will initiate such functions as sequencer speedup and simulation of certain failure modes so that redundant elements can be tested. These commands will also provide a useful function in flight for failure analysis and operation.

5.6.3 Test Points

All assemblies will have adequate test points brought out to separate test connectors to allow the following functions to be performed.

a) Calibration of all flight telemetry transducers conditioned within the assembly

b) Confirmation of separate fault isolation performed via telemetry, to the provisioned spare level

c) Provide for simulation (signal injection) and for hardline monitoring required to perform subsystem level testing

d) Verification of subsystem interface, check of noise and transient conditions, and performance of EMI checks.

Test points required for system testing and subsystem interface checks will be brought to a subpanel test connector by means of a flight hardware test harness. Test points required for pad testing will be brought through the subsystem interface to an umbilical junction box. The combination of the aforementioned test harnessing constitutes the direct access test circuitry. All other test points will be available only at the individual boxes and will appear on separate test connectors. These connectors will be mounted such that they can be used without requiring demating of the direct access test harness for the assembly. Some system level tests, such as space simulation testing, may require special test cables between the EOSE and the separate test connectors.

5.6.4 Power Bus Control

The spacecraft system will provide for EOSE control over the main power bus via the umbilical. This includes the ability to remove power from all spacecraft components, and to switch the spacecraft internal power supply (batteries and solar array) off the power bus and replace it with an external power supply. The reverse action will be a fail-safe mode. Separate control of power supplied to the flight capsule is required.

5.6.5 Connector Locations

Electrical connectors will be located on the boxes in a position which allows maximum ease of access and minimum flexure of the harness when demating of connectors is required. Removal of an assembly from the spacecraft will not require demating connectors on any other unit.

5.6.6 Hardpoint Locations

A sufficient number and variety of mechanical hardpoints will be designed into the structure to allow handling of the spacecraft for all required operations. These hardpoints are to allow such operations to be performed without undue complexity or hazard, and human engineering study recommendations will be considered in the design.

5.6.7 Maintainability

Maintenance of the flight spacecraft during system level testing is to be restricted to the removal and replacement of equipment at the provisioned spares level. All electrical parameter trimming, mechanical adjustment or alignment performed at the subsystem or lower level is to be performed prior to subsystem flight acceptance testing and will not be changed subsequently. Spacecraft system level electrical trimming, mechanical adjustment or alignment will be such that these parameters are exactly and uniquely reestablished in the event of subsequent disassembly and assembly of the spacecraft. Adjustment settings required in the OSE are to remain sufficiently stable to permit a complete span of test scheduling without adjustment during the span. Adequate access to spacecraft equipment will be provided.

6. LAUNCH VEHICLE SYSTEM INTERFACE REQUIREMENTS

This section presents spacecraft-launch vehicle system interface requirements as extracted and interpreted from References 1 and 2. It also includes requirements data as assumed for purposes of the Task B preliminary design effort, to establish a specific basis for the spacecraft design. It is not intended to represent a definitive or exhaustive treatment of the interface, which will be developed in Phase IB.

All interfaces between the launch vehicle and the planetary vehicle will be considered as being between the launch vehicle system and the spacecraft system. Emphasis will be placed on reducing complexity in system interfaces in order to simplify design, flight operations, ground test, and checkout operations, and to maximize mission probability of success.

The interface will allow as much independence in the design, test, and operation of the spacecraft and launch vehicle systems as is consistent with over-all performance and reliability goals.

6.1 Launch Vehicle System

The launch vehicle system includes the Saturn V launch vehicle; all space vehicle umbilical lines; launch complex facilities assigned to Voyager; all other KSC and AFETR facilities which support launch vehicle system prelaunch and launch operations; and the MSFC, KSC, and contractor personnel required to accomplish the functions of the launch vehicle system.

The launch vehicle consists of the Saturn S-IC stage, S-IC-IIS interstage, S-II stage, S-II-S-IVB interstage, S-IVB stage, instrument unit, and the nose fairing.

Complex 39 at Kennedy Space Center will be utilized to launch the Voyager space vehicle during the 1971 Mars opportunity.

6.2 Space Vehicle Configuration and Geometry

The space vehicle is defined to be the combination of two planetary vehicles and the launch vehicle as shown in Figure 7.

The reference axes for the planetary vehicle, the launch vehicle, and the relationship between the two are presented in Figure 8.

The available dynamic envelope for the planetary vehicle as installed to the launch vehicle is given in 3.1.

6.3 Mechanical Interface

The structural, electrical, and mechanical design at the spacecraft interface with the launch vehicle will not require critical mechanical adjustments during matchmate. The interface design will minimize the assembly operations required on pad to mate the encapsulated planetary vehicles to the launch vehicle. Recognition will be given to the reverse operation for demating from the launch vehicle.

6.3.1 Attachment and Umbilicals

The mechanical attachments between the planetary vehicles and the launch vehicle are to be field joints at the locations shown in Figure 7. The attaching field joint is to be separate from the in-flight separation joint that releases the planetary vehicle. Details of the interface are shown in Figure 9.

No umbilical provisions for mechanical functions are required for the encapsulated planetary vehicles. In particular, the propulsion subsystem will not require liquid or gaseous umbilicals, and the capability for propellant loading or unloading while mated to the launch vehicle will not be provided.

6.3.2 Boost Powered Flight Environmental Instrumentation

One or more instrumentation equipment items will be mounted on the planetary vehicle adapter to measure the boost powered flight environment. This instrumentation is assumed to be part of the launch vehicle system and is to be mounted on the adapter at the time of final field installation of the planetary vehicle with adapter in the shroud section. Details of the physical interface have not been defined.

6.3.3 Mass Properties

The weight of the planetary vehicle and planetary vehicle adapter will not exceed 22,000 pounds for the 1971 and 1973 missions or 30,000 pounds for the 1975 and 1977 missions.

The center of mass (c.m.) of the installed planetary vehicle and adapter is estimated to be within a 0.5-inch radius of the planetary vehicle centerline and is nominally at planetary vehicle station 168.

The estimated moments of inertia of the planetary vehicle and the planetary vehicle adapter combination are given below. These are calculated relative to the nominal center of mass of the combination given above and about axes X, Y, and Z, corresponding to space vehicle roll, pitch and yaw, respectively, as shown in Figure 8.

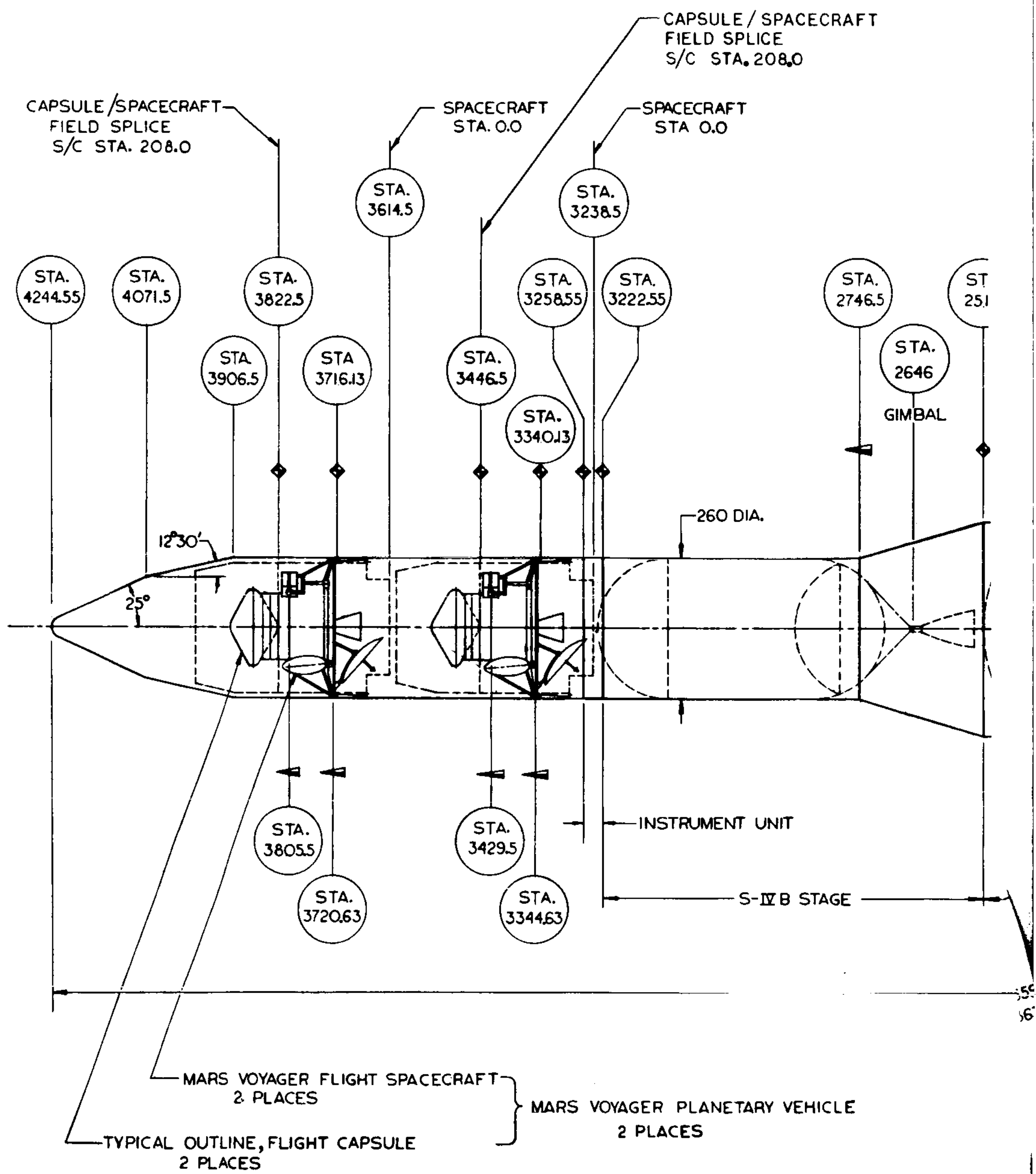
	slug ft^2
I_{roll}	= 21,840
I_{pitch}	= 28,550
I_{yaw}	= 28,420

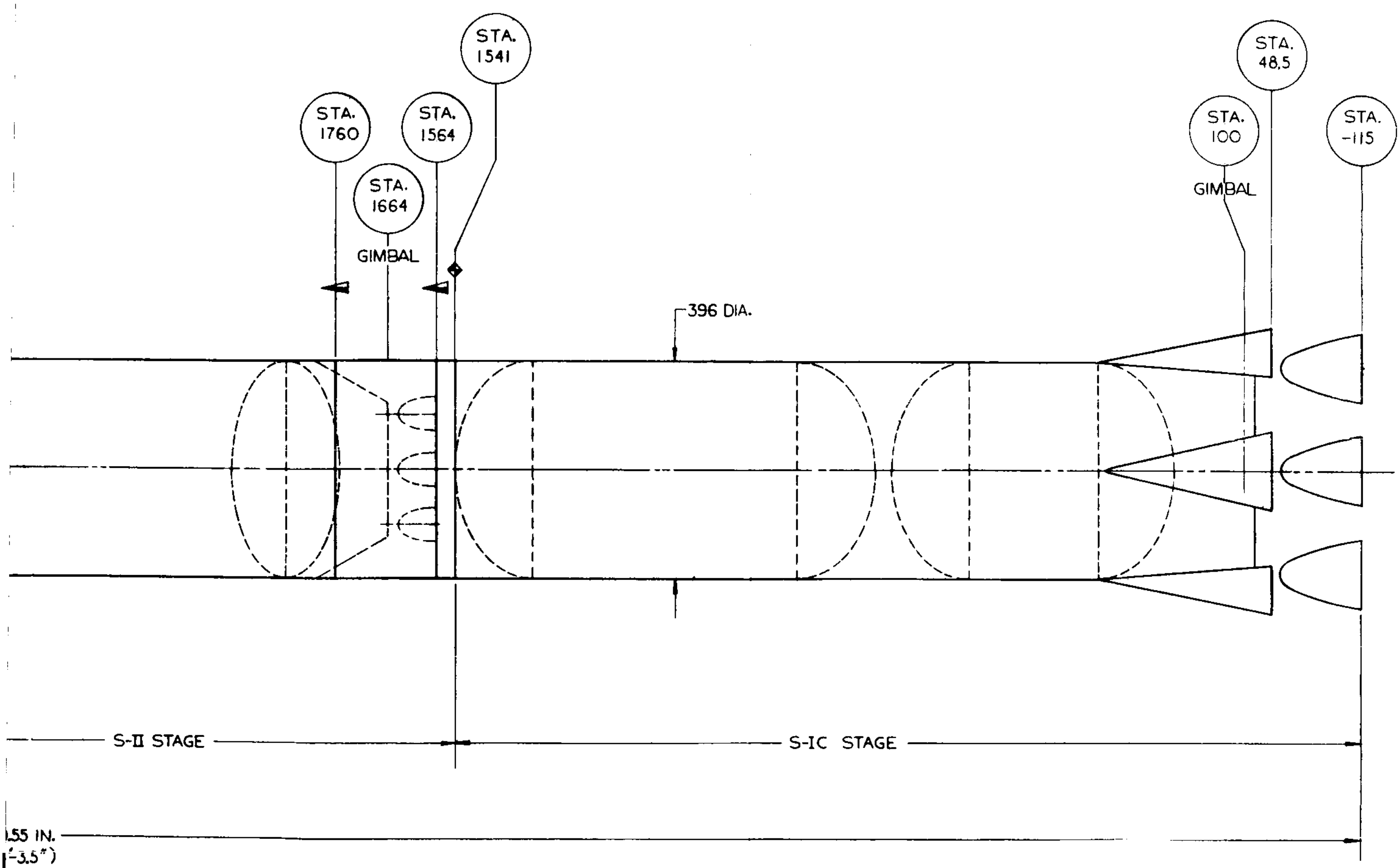
6.4 Electrical Interfaces

The flight spacecraft will provide all electrical circuits to the launch vehicle for the planetary vehicle. The electrical interfaces will meet the design requirements of 5.3. Each installed planetary vehicle will require the following electrical interface circuits as shown in Figure 9:

a) Umbilical circuits from planetary vehicle to launch complex equipment (LCE), disconnected prior to lift off

b) Flight connectors from the launch vehicle to separation devices to carry separation firing current


CAPSULE / SPACECRAFT FIELD SPLICE S/C STA. 208.0
CAPSULE /SPACECRAFT FIELD SPLICE S/C STA. 208.0
SPACECRAFT STA. 0.0
SPACECRAFT STA 0.0
STA. 3614.5
STA. 3238.5
STA. 4244.55
STA. 4071.5
STA. 3822.5
STA. 3258.55
STA. 3222.55
STA. 2746.5
STA 3906.5
STA 3716.13
STA. 3446.5
STA. 3340.13
STA. 2646
GIMBAL
260 DIA.
12°30'
25°
STA. 3805.5
STA. 3720.63
STA. 3429.5
STA. 3344.63
INSTRUMENT UNIT
S-IV B STAGE
MARS VOYAGER FLIGHT SPACECRAFT 2 PLACES
TYPICAL OUTLINE, FLIGHT CAPSULE 2 PLACES
MARS VOYAGER PLANETARY VEHICLE 2 PLACES



◆ FIELD SPLICE

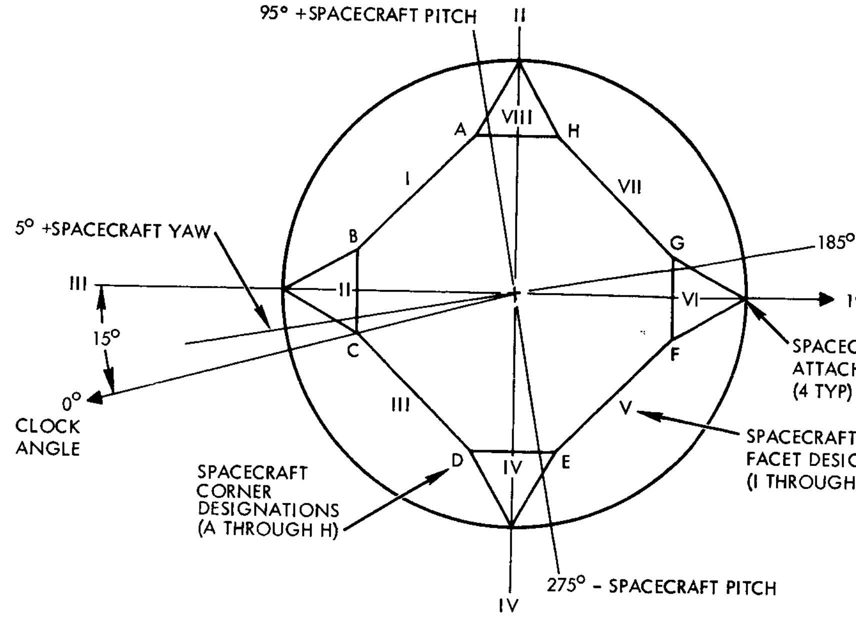
◄ SEPARATION

NOSE FAIRING JOINTS ARE NOT SHOWN

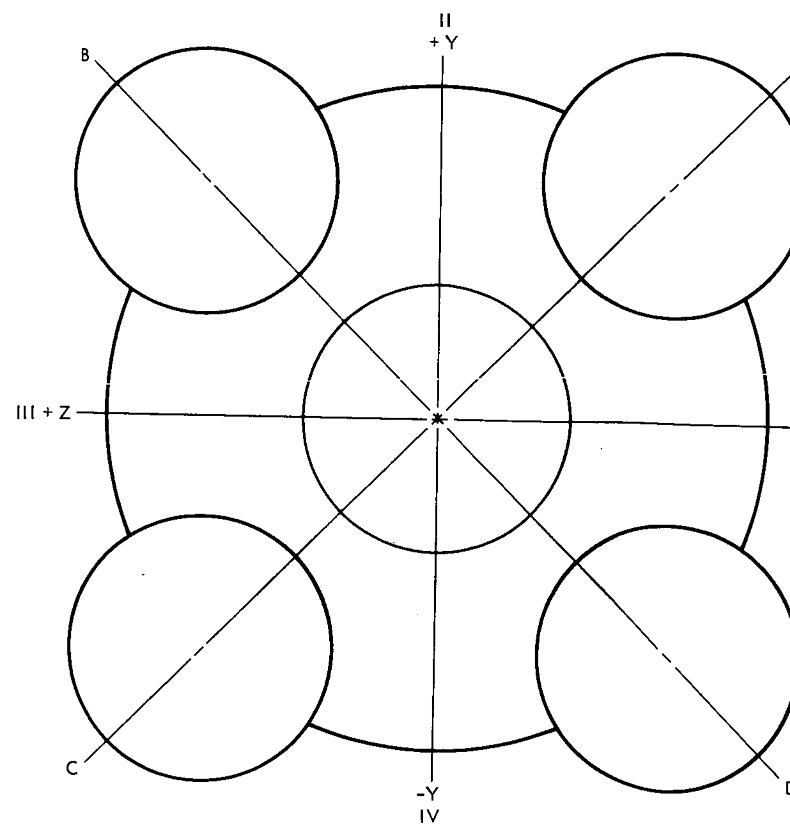
0 100 200 300 400 500

SCALE IN INCHES

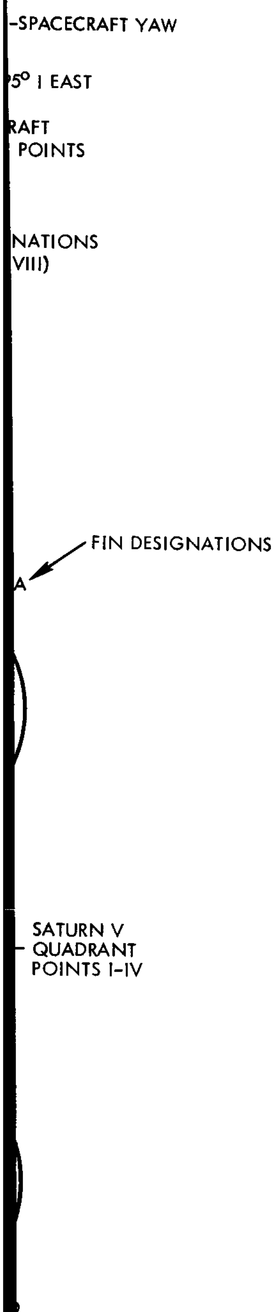
Figure 7. Space Vehicle Arrangement



SECTION A-A
SPACECRAFT COORDINATE SYSTEM
(LOOKING AFT)



SECTION B-B
SATURN V COORDINATE SYSTEM
(LOOKING AFT)



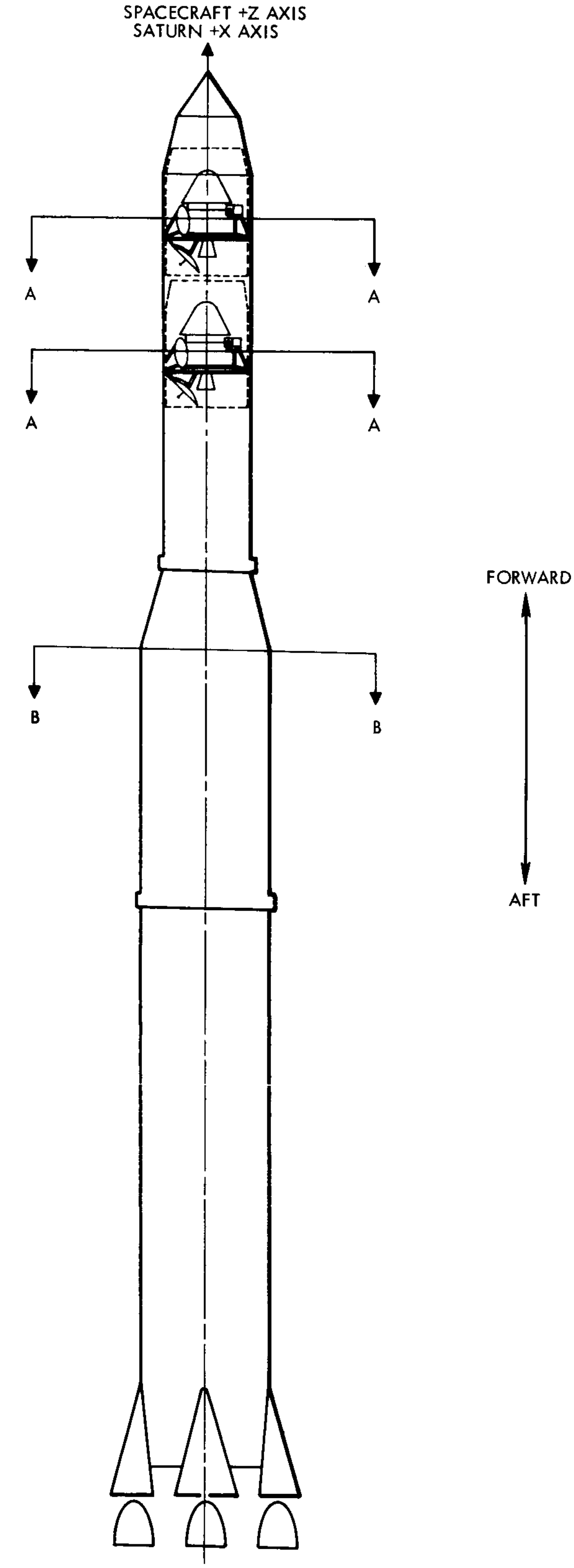
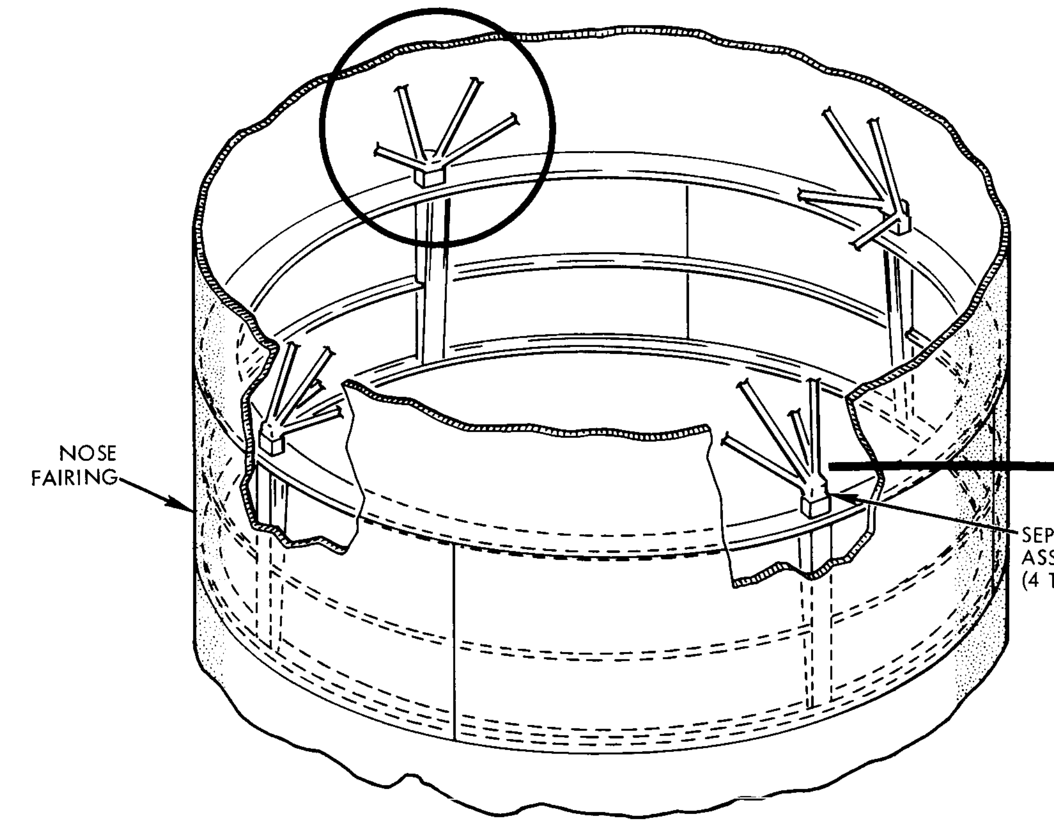
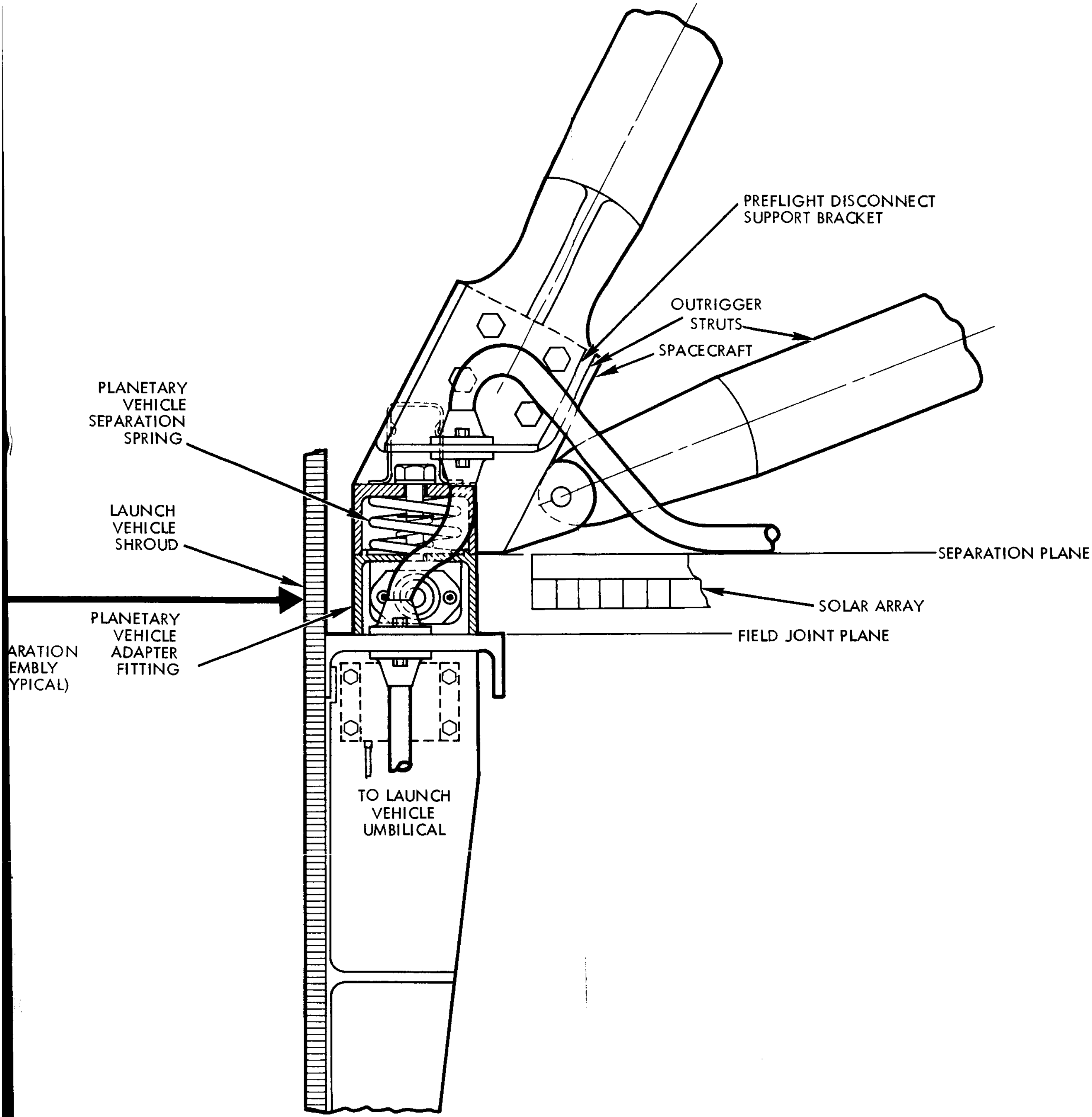


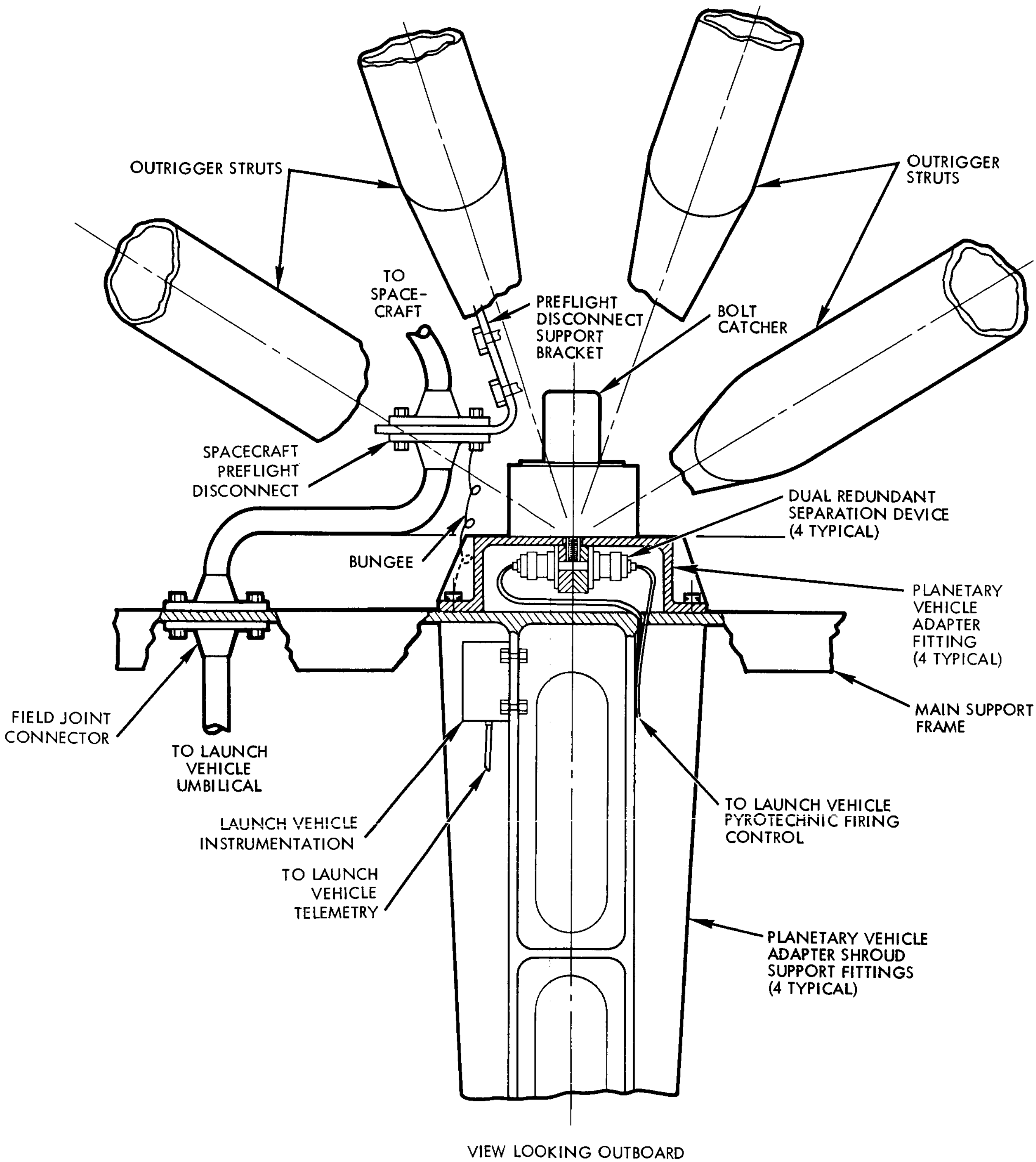
Figure 8. Vehicle Reference Axes



LAUNCH VEHICLE SEPARATION ASSEMBLY AS SHOWN IN VIEWS TO THE RIGH



VIEW TANGENTIAL TO SHROUD



3

Figure 9. Launch Vehicle-Planetary Vehicle Interface

c) Flight connectors carrying signals from adapter-mounted instrumentation to launch vehicle telemetry subsystem.

6.4.1 Umbilical

The circuits from the planetary vehicle to the LCE will pass through a planetary vehicle preflight disconnect, a planetary vehicle adapter field joint disconnect, and the nose fairing cabling and umbilical. The planetary vehicle preflight disconnect will be pyrotechnically separated at some short interval prior to launch. The planetary vehicle adapter field joint disconnect will be utilized to provide a readily accessible interface connector during installation and encapsulation of the planetary vehicle in the shroud section. This field joint interface connector will remain connected and attached to the planetary vehicle adapter subsequent to encapsulation and is separate from the remotely-actuated disconnect.

6.4.2 Separation Circuitry Connectors

The separation circuitry connectors and associated cabling are supplied as part of the nose fairing. Separate connectors are supplied to connect directly to each of the planetary vehicle separation pyrotechnic devices. This is a field joint that is made at encapsulation and no in-flight disconnect is required for the separation circuit, as the spacecraft half of the connection is mounted on the planetary vehicle adapter and remains with the launch vehicle at separation.

6.4.3 Instrumentation Connectors

Connectors will be required to connect boost powered flight environment instrumentation to the launch vehicle telemetry subsystem. This instrumentation is installed on the planetary vehicle adapter so that in-flight disconnection is not required. The required connectors and cabling are supplied as part of the nose fairing and are connected as field joints during encapsulation. If additional instrumentation forward of the separation joint is required, the design can of course be adjusted to accommodate this.

6.4.4 Pyrotechnic Circuitry

Circuitry for pyrotechnic devices will conform to 5.6.3 of Section II. Access to firing circuit connectors will be provided such that continuity and voltage status can be verified by tests conducted during the final field installation.

6.4.5 Boost Powered Flight Environmental Instrumentation

The instrumentation described in 6.3.2 will require no electrical support from the planetary vehicle. The measurements are to be telemetered by the launch vehicle. Electrical cabling and connectors are to be in keeping with 6.4.3.

6.4.6 RF Interface

The RF interfaces are the launch vehicle RF windows for radiating radio signals from both installed planetary vehicles.

In meeting the requirements for checkout and monitoring of spacecraft RF system parameters while on pad, transmission of the spacecraft RF signal will not require a mechanical connection between the planetary vehicle and the nose fairing.

To allow launch pad checkout, monitoring, and command, RF windows or equivalent will be required in the nose fairing to accommodate transmissions between ground stations and all antennas for both planetary vehicles in the installed configuration. This includes the high-gain antenna, medium-gain antenna, low-gain antenna, and relay link antenna. In addition, provisions are required for powered flight transmission from the low-gain antenna, which is located approximately in the launch vehicle pitch plane and on the down range side (see Figure 3).

During coast flight the launch vehicle pitch plane is nominally in the local vertical, with the roll axis horizontal and in the direction of flight. This orientation is maintained approximately during powered injection flight and through separation of the planetary vehicles.

6.4.7 Separation Signal

The launch vehicle will send separation signals to the planetary vehicles at the appropriate times for their separation. These signals will be suitable to fire the planetary vehicle separation pyrotechnic devices. Safe and arm requirements after planetary vehicle installation will be met by the launch vehicle system.

6.5 Environment

All elements of the spacecraft system will be designed to withstand the appropriate environments as defined in 5.5. Environmental considerations with design implications on the launch vehicle are given below.

6.5.1 Prelaunch Temperatures

Planetary vehicle subsystems will be designed to tolerate local temperatures during prelaunch operations corresponding to a steady state bulk gas temperature within the nose fairing between 45 and 65°F. In the encapsulated condition, suitably clean gas will be ducted into the planetary vehicle shroud sections to assure a positive internal pressure (to preclude inward leaks) and to provide the means for temperature conditioning as required.

6.5.2 Cooling

The spacecraft cooling requirements on-pad will not require the use of cooling gases at differing temperatures ducted preferentially to specific subsystems. The design will be compatible with natural convective distribution of the gas within the shroud section. To the maximum extent possible, the spacecraft will depend on radiation as the primary mode of heat dissipation.

6.5.3 Structural Flight Loads

Structural flight loads as a result of the flight environment are defined in Reference 6.

6.5.4 Shroud Radiation

During ascent the spacecraft will tolerate a totally enveloping IR heat flux equivalent to 0.5 solar constant eminating from the shroud innersurface.

6.5.5 Nose Fairing Separation Altitude

The forward launch vehicle fairing will be jettisoned at approximately 350,000 feet altitude.

6.5.6 Depressurization

No spacecraft component will be adversely affected by the sudden release of gas entrapped in the nose fairing cavity. The amount of unvented gas will be assumed to be equivalent to a 0.001 psi differential pressure across the shroud wall.

6.5.7 Sterilization

The planetary vehicle while within the nose fairing section will be subjected to surface sterilization in accordance with JPL specification GMO-50198-ETS.

6.6 Safety

The requirements of 5.6 in Section II will apply.

6.7 Contamination Control

The requirements of 5.7 and 5.8 in Section II will apply.

The planetary vehicle will be totally enclosed in a section of the nose fairing and sealed at both ends in preparation for launch. This encapsulation will be accomplished in a clean room area remote from the launch pad. The cleanliness of the planetary vehicle and the shroud section interior at the time of encapsulation will be such that there will be no loose particles, either internal or external to the equipment, greater than 4 mils in diameter. Surface sterilization will be accomplished in keeping with 6.5.7.

In the encapsulated condition suitably clean gas will be ducted into the nose fairing cavity as per 6.5.1. No physical access to the encapsulated planetary vehicle is allowed.

6.8 Separation

The planetary vehicle in-flight separation joint is within the spacecraft system.

The allowable inertial angular rate imparted to the planetary vehicle at separation, attributable to the launch vehicle, will be less than 1 deg/sec.

The separation sequence will include the following events and operations:

a) The launch vehicle will send a separation signal (see 6.4.7) to the forward planetary vehicle separation device.

b) The forward planetary vehicle separation device will release the forward planetary vehicle and impart a differential velocity between it and the remaining space vehicle.

c) After a suitable time delay, the launch vehicle will release and fold back the mid portion of the nose fairing and send a separation signal to the aft planetary vehicle separation device.

d) Step (b) will be repeated for the aft planetary vehicle.

e) After a suitable time delay the launch vehicle will retrothrust as required.

7. CAPSULE SYSTEM INTERFACE REQUIREMENTS

This section presents the flight spacecraft-capsule system interface requirements as extracted and interpreted from References 1, 2, and 5. The data presented is preliminary, pending further definition of the capsule system. It is developed as a concrete and representative basis for establishing the spacecraft design for purposes of the Task B study, and is therefore not intended to be a definitive and exhaustive treatment. The capsule system is defined in 3.1.3 of Section II. The flight capsule is defined in 3.2.3 of Section II.

The spacecraft contractor will be responsible for operations to integrate the capsule with the spacecraft to form the planetary vehicle, and for all interfaces between the planetary vehicle and the launch vehicle. The flight capsule will be compatible with all requirements on the planetary vehicle given in 6.

The flight spacecraft will provide the necessary support to the flight capsule during transit and until separation of the capsule vehicle. Such services include power, timing and sequencing, telemetry, and command. The spacecraft is designed to serve as a communications relay for the capsule during entry and during Mars surface operations.

The system for normal inflight separation of the capsule is forward of the spacecraft-capsule field joint and is within the capsule system. Each contractor is responsible for the structural and electrical design of his side of the field joint. The capsule emergency separation joint is within the spacecraft system.

The relay link receiving and recording electronic equipment installed in the flight spacecraft is to be supplied to the spacecraft contractor as GFE. The relay link receiving antenna is to be supplied by the spacecraft contractor to meet requirements as specified.

The selection of the descent trajectory within established constraints is considered to be a spacecraft system responsibility.

7.1 Mission Description

The Voyager mission profile is given in 4.11 of Section II. The spacecraft-capsule interface requirements relate to the prelaunch phase through flight spacecraft relay of capsule data. The associated phases will include operations and activities as presented in 1 of Section IV.

The flight spacecraft is capable of accomplishing its mission in the absence of the capsule, as described in 1.7.

7.2 Planetary Vehicle Configuration and Geometry

The planetary vehicle is defined to be the flight spacecraft-capsule combination arranged and installed as in Figure 7. The available dynamic envelope is defined in Figure 2.

The nominal 1971 capsule configuration is given as follows:

- Sphere cone (60 degrees)
- Diameter: 12 feet
- Total capsule weight: 2000 pounds
- Over-all length: 90 inches

7.3 Mechanical Interface

The structural, electrical, and mechanical design of the flight capsule interface with the flight spacecraft will not require critical mechanical adjustments during matchmate. The interface design will minimize the assembly operations required to mate the flight capsule to the flight spacecraft. Recognition will be given to simplifying the reverse operation in which the flight capsule is demated from the flight spacecraft.

The mechanical interface between the flight capsule and the flight spacecraft will be a field joint as depicted in Figure 10. The field joint is separate from the capsule emergency separation joint.

The spacecraft structure and the planetary vehicle adapter will be capable of supporting the structural and inertial loads of the flight capsule for all acceleration, vibration, shock, acoustic, and nonlinear load characteristics associated with launch vehicle and planetary vehicle powered flight. The mechanical attachment of the flight capsule to the flight spacecraft will be in keeping with 3.8.

The design of the mechanical interface will preclude deleterious dynamic coupling between the flight spacecraft and flight capsule.

The misalignment between a perpendicular to the plane of the flight capsule field joint and the flight spacecraft geometry roll axis will not exceed 2 mr.

Relay link equipment, with the exception of the relay link antenna, will comply with the modular packaging and mounting requirements in 5.2. The relay link antenna will be mounted at a perimeter location on the aft equipment module structure of the flight spacecraft as shown in Figure 3.

7.4 Flight Capsule Mass Properties

The weight of the flight capsule will not exceed 3,000 pounds for the 1971 and 1973 missions and 10,000 pounds for the 1975 and 1977 missions. The GFE relay link equipment estimated weights are given in 5.5 of Section IV.

The center of mass of the flight capsule for the 1971 Voyager mission will be within a 3-inch radius of the planetary venicle centerline, and within ± 27 inches of planetary vehicle station 303.

The weights and moments of inertia of the flight capsule for design estimation purposes are as follows:

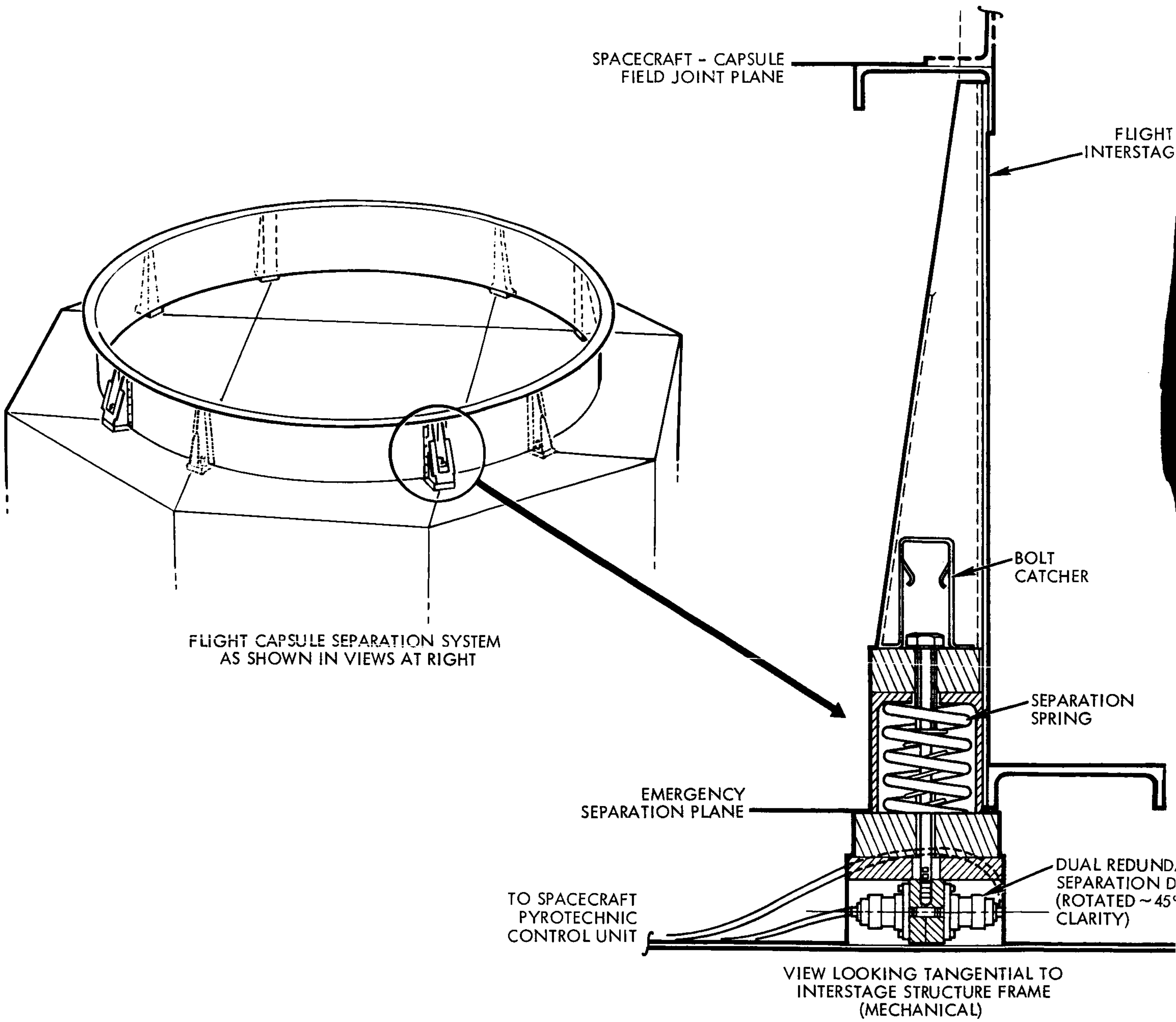
	1971	1975
• Weight:	2,000 pounds	8,000 pounds
• I_x:	1,500 slug ft^2	8,000 slug ft^2
• I_y:	1,500 slug ft^2	8,000 slug ft^2
• I_z:	1,500 slug ft^2	8,000 slug ft^2.

7.5 Electrical Interface

The electrical interface circuits will meet the design requirements of 5.3. Short circuit protection will be used on the spacecraft side of the interface. Fuses will not be used. Conducted energy between the flight capsule and the flight spacecraft systems must not interfere with the operation of either system.

7.5.1 Connectors

An electrical interface field joint connector will be provided to accommodate power, signal, command, and hardline telemetry functions between the flight capsule and the flight spacecraft. Circuits from the flight capsule to the capsule LCE will be accommodated through the flight spacecraft umbilical if required. The inflight disconnect associated with the emergency capsule separation is within the spacecraft system and is not an interface connector.


SPACECRAFT - CAPSULE
FIELD JOINT PLANE
FLIGHT
INTERSTAG
BOLT
CATCHER
FLIGHT CAPSULE SEPARATION SYSTEM
AS SHOWN IN VIEWS AT RIGHT
SEPARATION
SPRING
EMERGENCY
SEPARATION PLANE
DUAL REDUND
SEPARATION D
(ROTATED ~45
CLARITY)
TO SPACECRAFT
PYROTECHNIC
CONTROL UNIT
VIEW LOOKING TANGENTIAL TO
INTERSTAGE STRUCTURE FRAME
(MECHANICAL)

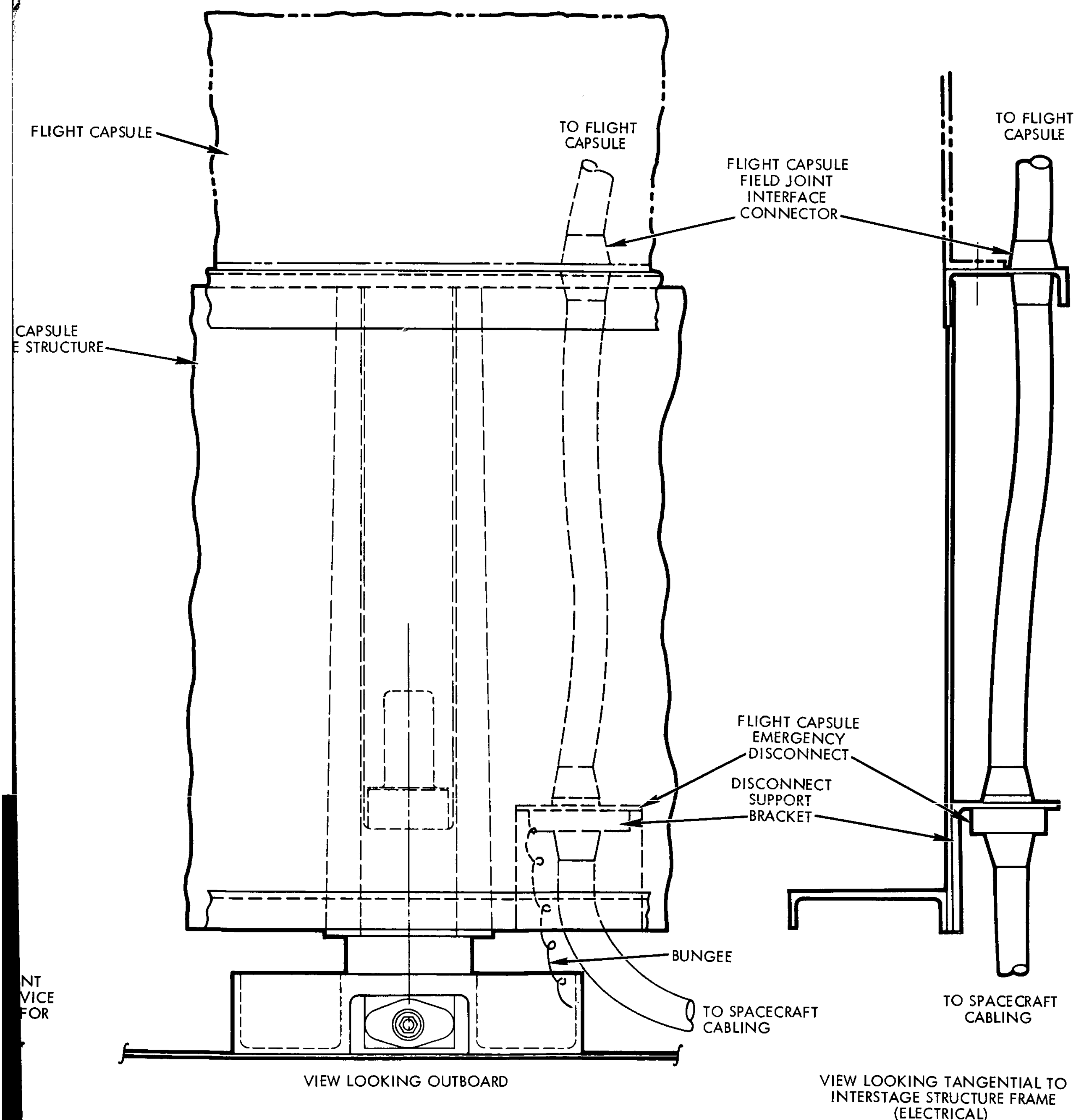


Figure 10. Flight Capsule-Flight Spacecraft Interface

2

7.5.2 Power

a. Power to Flight Capsule

The spacecraft will provide the flight capsule with up to 200 watts of raw DC power from the spacecraft solar array during periods of spacecraft sun-line orientation. This amount of power will also be supplied to the flight capsule during prelaunch as required.

The distribution of spacecraft power within the capsule will be controlled by the capsule.

The spacecraft will provide a turn-on and turn-off capability for the power supplied to the capsule. No spacecraft switching condition or single failure mode will allow the capsule batteries to supply power to the spacecraft.

For the 1971 mission, the spacecraft will provide AC power to the relay link receiving equipment and tape recorder, as given in 5.5. This power is supplied from initiation of capsule preseparation operations until capsule impact and as required for subsequent recorder playback. Power turn-on and turn-off will be provided by the spacecraft.

7.5.3 Flight Capsule Data

The flight spacecraft will transmit flight capsule data from prelaunch until end of the capsule mission. Relay link data will be transmitted either in real time or by playback from the relay link recorder. All flight capsule data will be transmitted in binary digital form. Three data rates are accommodated as follows (see Figure 11).

a. Very Low Data Rate

During interplanetary cruise the spacecraft will be capable of receiving data from the flight capsule equivalent to 20 to 30 measurements sampled about every 4 hours. This data will be delivered to the spacecraft via a hardline data link at a data rate of 10 bits/sec. The spacecraft will require a clock signal with this data line.

b. Low Data Rate

During the preseparation to terminal descent period, the spacecraft will be capable of receiving measurements and status data from the flight capsule at a data rate of 100 bits/sec. This is provided as an output line from the low data rate relay link receiver (telemetry mode 2). The spacecraft will require a clock signal with this data line.

c. High Data Rate

During terminal descent until impact, the flight capsule will transmit data over the relay receiving link at a rate of 50 to 200 k-bits/sec. This will be recorded by the relay link tape recorder for subsequent playback and transmission by spacecraft telemetry.

7.5.4 Commands to Flight Capsule

The spacecraft will provide the flight capsule with six quantitative commands and 10 discrete commands. Times for activating stored commands will be updatable by ground command. Backup by direct ground command will be provided.

The spacecraft commands to the relay link equipment are depicted schematically in Figure 11.

- Receiving equipment commands

  a) On-off, low data rate

  b) RF attenuation control,* low data rate

  c) On-off, high data rate (recording initiated at a designated elapsed time from separation)

  d) RF attenuation control,* high data rate (for checkout)

- Recorder commands

  The spacecraft commands to the relay link recorder are:

  a) Start-stop write

  b) Start-stop playback

*This allows the receiver to receive capsule RF transmission while the capsule is still attached to or very near the spacecraft.

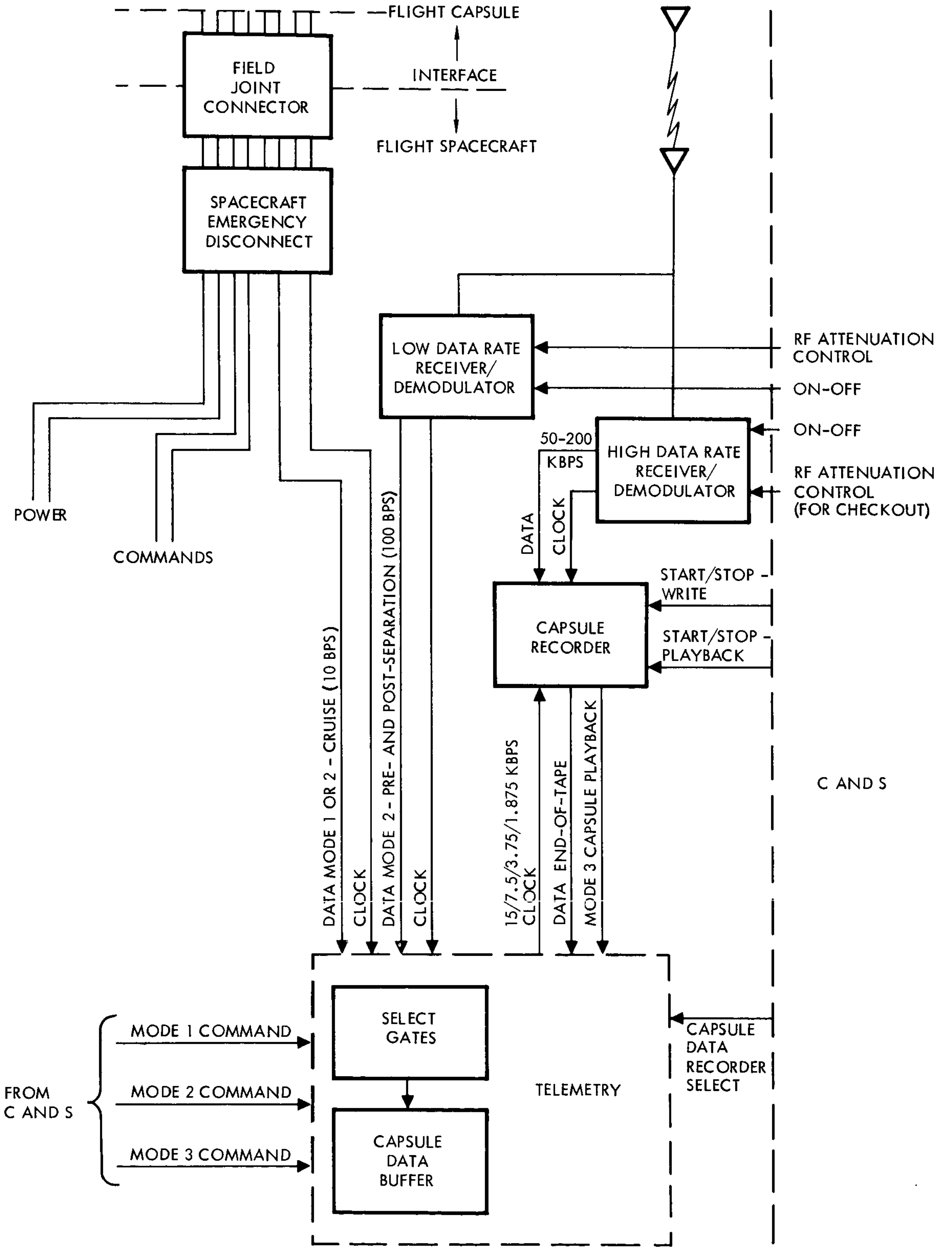


Figure 11. Capsule Signal and Data Handling Interfaces

- Playback commands

  The spacecraft telemetry system will require a command to select the relay link recorder when the recorder data is to be transmitted to earth.

7.5.5 Telemetry

The spacecraft telemetry subsystem will be capable of accepting, selecting, buffering, and modulating the flight capsule and relay link data for transmission to earth. Data rates will correspond to the spacecraft radio link capability as given in 3 of Section II, Volume 1.

The flight capsule and relay link data will be transmitted in the telemetry modes listed below:

- Mode 1: Very low rate capsule data. (This mode contains spacecraft engineering data and flight capsule cruise data.)
- Mode 2: Very low rate and low rate capsule data. (This mode contains Mode 1 plus spacecraft low rate science.) The data rate is adjusted to accommodate the low rate capsule data (100 bits/sec).
- Mode 3: High rate capsule data. (This mode contains the relay link recorder playback of capsule data.)

Clock signals are required as follows:

a) Capsule very low and low data rate clock signals will be transmitted to the spacecraft by the flight capsule and relay link receiver, respectively.

b) The spacecraft telemetry subsystem will provide clock signals to the relay link recorder for playback.

The relay link recorder will transmit an end-of-tape signal to the telemetry subsystem in the write and playback modes.

The telemetry subsystem will provide buffering of capsule data during very low and low rate transmission to the telemetry.

7.5.6 Relay Link Antenna Performance Requirements

The relay link will be operated at a frequency of approximately 400 MHz. The performance required of the relay link antenna system may be described by a parameter G which is defined as follows:

$$G\ (\text{db}) \triangleq 10 \log (\text{antenna gain factor}) - 20 \log (\text{range in thousands of kilometers})$$

The design of the antenna system must be such that G is greater than -5 db from the time of capsule separation to capsule impact. More than one antenna may be required to meet this requirement. If so, switching between antennas may be considered provided there is no interruption of the signal after the capsule reaches an altitude of 800,000 feet. A discussion of descent trajectory considerations is given in Appendix B.

7.6 Capsule Shading

The flight spacecraft will completely shade the flight capsule when the planetary vehicle is sun-stabilized.

7.7 Normal Capsule Separation

Normal separation of the capsule from the spacecraft will occur from Mars orbit. The sequence of events and operations will be as described in 1.12 of Section IV.

The capsule pointing accuracy for separation is given in 2.1.

The capsule separation system will provide a sufficient velocity increment to achieve a 1000-meter separation distance at the time of the de-orbit motor ignition (20 minutes after capsule separation).

8. SPACECRAFT SYSTEM-TRACKING DATA SYSTEM INTERFACE

8.1 General

The interface between the flight spacecraft and the tracking data system (TDS) is the two-way S-band communications link, which is required to provide one-way and two-way doppler tracking, turnaround ranging, command, and telemetry data acquisition for two planetary vehicles.

During certain critical phases of the mission, simultaneous and continuous operation with two planetary vehicles is required. This imposes specific design and performance requirements on the tracking data system which are peculiar to the Voyager program. These requirements are summarized in Table 6, and discussed in detail in the following paragraphs.

8.2 Design Requirements

8.2.1 Multiple Planetary Vehicle Operation

During all phases of the mission, from prelaunch checkout to encounter plus 6 months, two-way communication with each planetary vehicle is required. During certain specific periods in the mission, simultaneous communication with both planetary vehicles within the view capabilities of the DSN stations is required, as follows:

a) Telemetry data acquisition from prelaunch to interplanetary trajectory injection plus 30 days for both planetary vehicles

b) Command of both planetary vehicles from interplanetary trajectory injection to injection plus 30 days

c) Doppler tracking and turnaround ranging of both planetary vehicles from interplanetary trajectory injection to injection plus 30 days.

A similar requirement for simultaneous telemetry data acquisition, command, and tracking of both planetary vehicles exists from encounter plus 5 days to encounter plus 8 days referenced to the first planetary vehicle (encounter minus 5 days to encounter minus 2 days referenced to the second planetary vehicle), if the time separating the arrival of the two vehicles is 10 days.

A further requirement for simultaneous acquisition of telemetry data from the two planetary vehicles exists during orbital operations.

With the exception of these mission phases, time shared communications with the planetary vehicles is acceptable.

Table 6. Tracking Data System Requirements

Mission Phase	Stations	Antenna and Mode	Nominal Gain Transmitter (db)	Nominal Gain Receiver (db)	Transmitter Power	Receiver Noise Temperature	Remarks
Prelaunch	Kennedy Space Center, No. 71	4-ft diplexed	23	25	5 watts	$3,000^{\circ}K$	• Single-channel transmitter used for checkout* • Two-channel receiver required
Launch (L)	Kennedy Space Center, No. 71	4-ft diplexed	23	25	5 watts	$3,000^{\circ}K$	• Two-channel receiver required for both Stations 71 and 72
	Ascension Island, No. 72	30-ft diplexed	42	43		$290^{\circ}K$	
Parking Orbit through Injection (I)	Ascension Island, No. 72	30-ft diplexed	42	43			• Two-channel receiver required at each DSN site • Continuous coverage from DSN not possible in 100-nmi parking orbit • Command through spacecraft command link not possible due to doppler and extended acquisition time • Apollo/Saturn net needed for telemetry through S-IVB in parking orbit and for emergency command if required
	Johannesburg, No. 51 Madrid, No. 61 Woomera, No. 41 Tidbinbilla, No. 42 Goldstone, No. 11/12	3-ft acquisition aid antenna only	19	21		$300^{\circ}K$	
	Apollo Network						
Post Injection to I + 30 days	Ascension Island, No. 72	30-ft diplexed	42	43	10 kilowatts	$290^{\circ}K$	• Station No. 72 only until I + 8 hours, if in view • Simultaneous two-channel transmitter and receiver at each site • Uplink is initially multiplexed via the acquisition aid (or Station 72) until I + 8 hours; after that, option exists for multiplexed operation or separate 85-ft sites for each spacecraft • Two-channel receiver on each antenna
	Johannesburg, No. 51 Madrid, No. 61 Woomera, No. 41 Tidbinbilla, No. 42 Goldstone, No. 11/12	acquistion aid and 85-ft dish	19 50.1	21 52.3	10 kilowatts	$300^{\circ}K$ $55^{\circ}K$	
I + 30 days to Encounter (E) - 5 days	Johannesburg, No. 51 Madrid, No. 61 Woomera, No. 41 Tidbinbilla, No. 42 Goldstone, No. 11/12	85-ft dish	50.1	52.3	10 kilowatts	$55^{\circ}K$	• Time shared uplink required - 100 kw site only required after approximately L + 150 days for a) turnaround ranging via spacecraft high-gain antenna, or b) for emergency command via spacecraft low-gain antenna • Two-channel receiver on each antenna antenna required
	Goldstone, No. 14	210-ft dish	59.8	60.9	100 kilowatts	$35^{\circ}K$***	
E - 5 days to E + 8 days	Johannesburg, No. 51 Madrid, No. 61 Woomera, No. 41 Tidbinbilla, No. 42 Goldstone, No. 11/12	85-ft dish	50.1	52.3	10 kilowatts	$55^{\circ}K$	• Simultaneous two-channel uplink required only when both spacecraft are in their respective E - 5 to E + 8-day period (spacecraft have a minimum arrival separation of 10 days) • 85-ft sites continue to track second spacecraft from E - 5 days of first spacecraft to E - 6 hours of second spacecraft (approximately 15 days) • 210-ft sites establish two-way communication with first planetary vehicle from E - 5 to E + 8 days (for first planetary vehicle) then switch to two-way communication with second planetary vehicle from E - 6 hours to E + 8 days (for second planetary vehicle) while continuing to receive from first planetary vehicle**** • Two-channel receiver on each antenna
	Goldstone, No. 14 Madrid and Australia	210-ft dish**	59.8	60.9	100 kilowatts	$35^{\circ}K$***	
E + 8 days to End of Mission	Goldstone, No. 14 Madrid and Australia	210-ft dish**	59.8	60.9	100 kilowatts***	$35^{\circ}K$***	• Time shared uplink utilized • Two channel receiver required at all sites

*Uplink not established until after injection.

**Only the Mars site is presently equipped with a 210-ft capability. Two additional sites with 210-ft capability are assumed at Madrid and Australia, separated by approximately 120 degrees longitude.

***Diplexed operation is assumed on all sites with a 210-ft capability; this may lead to an increase in system noise temperature.

****A problem may exist if turnaround ranging of second planetary vehicle is required before both spacecraft are in the beamwidth of the 210-ft antenna (about E - 2 days for second spacecraft).

8.2.2 Telemetry Data Acquisition

For compatibility with the mission requirements, each receiving station of the existing and planned DSN, including Stations 71 and 72, must be capable of receiving, processing, and recording data at two different RF frequencies, and at data rates between a few bits per second and a maximum of 15,000 bits per second. Functional or simple block redundancy is to be provided whenever practicable, including, as a minimum, redundant receivers, processing, and recording equipment. Each station must have the capability to receive signals which are either coherent or noncoherent with the uplink carrier.

In the coherent mode of operation, each station should also be capable of simultaneously receiving and demodulating telemetry data and sync and the turnaround ranging code from both planetary vehicles, excepting only periods when the command link is operative.

In parking orbit, continuous acquisition of telemetry data is not possible using the DSN. An additional network such as the Apollo/Saturn network will be required to provide essential monitoring through the launch vehicle telemetry system.

8.2.3 Command

During prelaunch preparations at KSC, the integrity of the command link will require verification via the RF link, using DSIF Station No. 71.

In the parking orbit, command of the planetary vehicles is not possible using the normal RF link, due to the high doppler rates and limited in-view time compared with the sync acquisition time. The TDS must therefore provide commands to the planetary vehicles, if any commands are required in the parking orbit, through the launch vehicle command link and a hard wire interface. Such a capability is not included in the present design.

From interplanetary trajectory injection to the end of the mission, each station of the DSN should be capable of transmitting commands required for interplanetary correction maneuvers, Mars orbit insertion, orbital sequence updating, spacecraft-capsule separation, and as backup for on-board commands to both planetary vehicles simultaneously.

Addressing of two planetary vehicles simultaneously may require discrete RF uplink frequencies for each vehicle, with different values of f_s and orthogonal PN codes for synchronization. At this point in time no decision has been made on the method or combination of methods for simultaneous addressing.

8.2.4 Diplexed Operation

To simplify operational procedures and to enhance the tracking accuracy, diplex operation at each site is highly desirable. The mission profile and communications link capability requires that the 10-kw transmitter should be diplexed on the 85-foot antennas from launch to encounter. Diplex operation on the 210-foot antennas with the 100-kw transmitters is required from encounter minus 5 days to end of mission. This capability is also required after launch +150 days for either emergency command via the spacecraft low-gain antenna or ranging via the spacecraft high-gain antenna. (Although a site with an 85-foot antenna and utilizing 100-kw could be used for this purpose early in the mission, link analysis shows that it is unsatisfactory at the end of the mission. Therefore, the 100-kw 210-ft combination has been selected.)

8.2.5 Tracking Rate

In the parking orbit, a maximum angular rate of approximately 2.5 deg/sec at zenith may be encountered. This exceeds by a factor of more than three the tracking rate capability of the large antennas of the DSN. Those stations of the TDS used for telemetry data acquisition during the parking orbit phase of the mission are required to have beam-width and tracking rate capabilities compatible with this requirement.

8.2.6 Radio Frequency Interference

All systems of the TDS used for the Voyager program are required to be mutually compatible, with no spurious RFI effects between the launch vehicle system, planetary vehicle system, and other near earth or deep space missions operating at the same time.

8.3 Performance Requirements

In establishing the telecommunications design, tracking and data system performance requirements in Table 7 are assumed.

Table 7. Tracking and Data System Performance Requirements

Item	Value
Transmitter power	
10-kw sites	70 dbm + 0.5 db - 0 db
100-kw sites	80 dbm minimum
Transmitter circuit loss	
10-kw sites	-0.4 db $\pm$ 0.1 db
100-kw sites	-0.4 db $\pm$ 0.1 db
Transmitting antenna gain	
Acquisition aid	20 db $\pm$ 2.0 db
85-foot dish	53 db + 1.0 db - 0.5 db
210-foot dish	60.6 db + 1.0 db - 0.5 db
Transmitting antenna pointing loss	
Acquisition aid	0 db
85-foot dish	0 db
210-foot dish	-0.5 db
Receiving antenna gain	
Acquisition aid	21.0 db $\pm$ 1.0 db
DSIF, 71	25.0 db minimum
85-foot dish	53.0 db + 1.0 db - 0.5 db
210-foot dish	61.7 db + 1.0 db - 0.5 db
Receiving antenna pointing loss	
Acquisition aid	0 db
DSIF, 71	0 db
85-foot dish	0 db
210-foot dish	-0.5 db
Ellipticity	
Acquisiton aid	$<$1.5 db
DSIF, 71	Linearly polarized
85-foot dish	0.7 db $\pm$ 0.3 db
210-foot dish	0.3 db $\pm$ 0.2 db
Noise temperature	
Acquisition aid	$270^{o}K \pm 60^{o}K$
DSIF, 71	$3000^{o}K$ maximum
85-foot dish	$55^{o}K + 10^{o}K$
210-foot dish	$30^{o}K \pm 5^{o}K$

9. SCIENCE SUBSYSTEM INTEGRATION

The flight spacecraft will be designed to accommodate and provide support for a science subsystem that is capable of accomplishing the specified science mission. Information and requirements data pertinent to integration of the science subsystem into the spacecraft are given in this section. This material relates to a reference science subsystem that has been defined and studied within the framework of the present preliminary design effort. It is not intended to represent a recommended payload, but to serve as a concrete and representative basis for considering the science integration design problem. In particular, design features and interfaces within the science subsystem are defined only to the degree necessary to consider spacecraft requirements.

9.1 General

9.1.1 Definition

The science subsystem is made up of the following:

a) Science experiment equipment

b) Data automation equipment

c) Science command decoding equipment

d) Science power switching electronics

e) Planetary scan platform and control

f) Science deployment

g) Fixed science packages

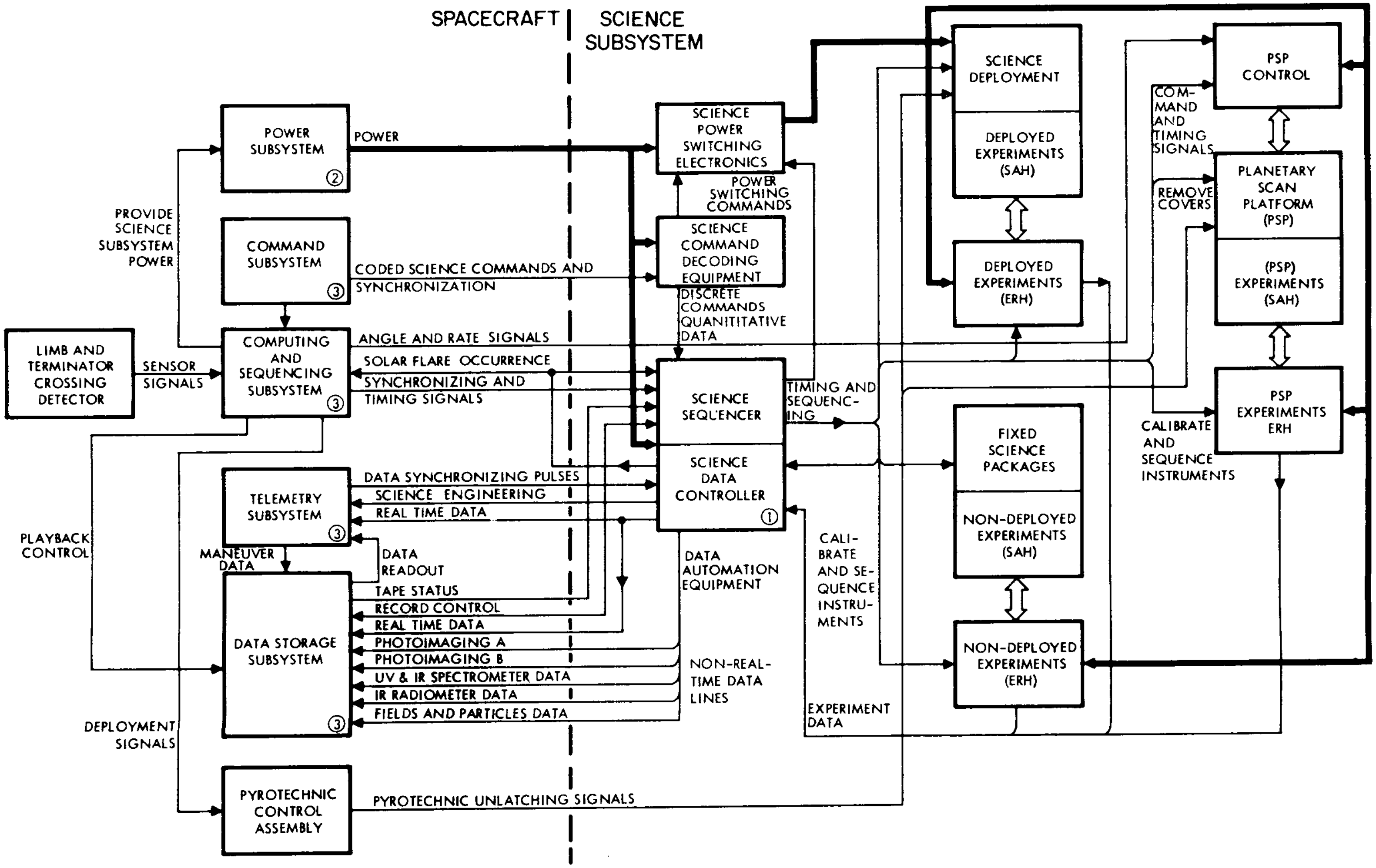
h) Science interconnecting wiring.

9.1.2 Functional Interfaces

Functional interfaces between the science subsystem elements and the spacecraft are to be as shown in Figure 12.

9.1.3 Weight Allocation

The spacecraft is to provide the capability to accommodate a total science subsystem weight of 400 pounds.



① SCIENCE ENGINEERING DATA LINES OMITTED FOR CLARITY

② POWER TO OTHER SUBSYSTEMS OMITTED FOR CLARITY

③ SEQUENCING OF SPACECRAFT SUBSYSTEMS OMITTED FOR CLARITY

SAH SENSORS AND ASSOCIATED HARDWARE

ERH EXPERIMENT REMOTE HARDWARE

⇔ DATA, POWER, AND CONTROL LINES

Figure 12. Spacecraft-Science Subsystem Functional Interfaces

9.1.4 Magnetic Requirements

The maximum magnetic field measured at the science magnetometer sensor due to the demagnetized flight spacecraft is to be less than one gamma (10^{-5} gauss) under all operating conditions. No single assembly is to produce more than 0.2 gamma. After the spacecraft has been exposed to a magnetic field of 25 Oersteds, the field is to be less than 10 gamma.

The spacecraft magnetic field measured at the magnetometer sensor will not change by more than 0.1 gamma due to a change between any two operational modes (after initial transients have died out), or due to the voltage and current changes encountered over the full design range of a subsystem for any individual operating mode.

9.1.5 Science Subsystem Data

The spacecraft is to provide the capability for accepting and transmitting data from the science subsystem as indicated in Table 8.

Table 8. Science Subsystem Data Acceptance Requirements

Phase	Minimum bits/day	Design Goal bits/day
Cruise	2.5×10^6	5×10^6
Orbit (end of life)	5×10^7	10^8

The science subsystem will present to the spacecraft the science data and the science subsystem engineering data in digital binary form and in a format consistent with the data frame formats of the flight spacecraft telemetry and data storage subsystems. Real-time science data will be provided on a real-time data line to the telemetry subsystem, and non real-time data will be provided on non real-time data lines to the data storage subsystem. Science subsystem engineering data will be provided to telemetry on a separate real-time engineering data line to be combined with spacecraft engineering data for transmission.

9.1.6 Sequencing Requirements

The flight spacecraft will have the capability to acquire and transmit planetary/interplanetary observations continuously in cruise and in orbit; to acquire and play back planetary/interplanetary observations during (and after) maneuvers; and to sample and play back planetary/interplanetary observations during solar flares in cruise and in orbit.

a. Orbital Operations

For a minimum of 2 months and preferably for 6 months after orbit is attained, the flight spacecraft will provide the capability to conduct the prescribed experiments and communicate these data to earth via the RF link without ground intervention. During the first two orbits, flight spacecraft sensors will provide the information to establish the time reference for initiation of the orbit sequence. The normal planetary science instrument operations will not be programmed during this period. The flight spacecraft will contain the on-board sensors and logic required to proceed automatically through the orbital data acquisition sequence of events. In addition, ground commands may be utilized to backup or optimize the orbital sequence of events. The orbit sequence of events will involve the selection and pointing of instruments, the initiation of data acquisition, and the storage and subsequent transmission of these data.

b. Orbital Sequence

Each orbital sequence (a data acquisition and playback cycle) will normally last for one orbit but may last for two or more. A typical sequence is discussed in 1 of Section IV. The functions which must be initiated during the orbital sequence will include, but are not limited to, the following:

a) Start the orbital sequence

b) Turn on planetary science power (if required) at least one-half hour before the instruments take data

c) Reposition platform

d) Calibrate instruments (if required)

e) Start planetary science instruments

f) Change data mode for playback of stored data

g) Turn off planetary science power (if required)

h) Change data mode

i) Change data mode for earth occultation.

c. Backup Commands

The C and S subsystem has the capability to perform all of the sequencing and control functions of the DAE on an automatic basis if required as a backup. Also, the command subsystem has the capability to provide backup for a number of discrete commands to permit some sequencing or control based on direct commands from earth. The detailed tradeoffs of reliability versus complexity and weight which will determine the proper backup modes for science have not been completed. Consequently, no backup commands are shown in Figure 12. It is intended that backup capability will be specified when the proper modes have been identified.

9.1.7 Component Design Parameters

Preliminary estimates for volume, weight, power, temperature limits, and magnetic parameters are given in the component design parameters table in 5.5 of Section IV.

9.2 Science Experiment Equipment

Science experiment equipment will be furnished by experimenters for integration into the spacecraft system. This equipment can be subdivided into sensors and directly associated hardware and supporting hardware removed from the sensors. The directly associated hardware will be mounted with the sensors in thermally controlled packages, except for specific sensors such as antennas and the magnetometer and ion chamber sensor, which require special mounting (see Figure 3). Sensors and associated equipment for planetary experiments will be mounted in the planetary scan platform while the remaining associated hardware will be mounted in two fixed science packages.

Remote science hardware will be designed and constructed in accordance with all the requirements of 5 and will be mounted on the spacecraft electronic equipment panels.

A nominal set of experiment equipment is given in Tables 9 and 10, along with associated interface requirements.

9.3 Data Automation Equipment

The data automation equipment (DAE) will be capable of accepting discrete commands and binary coded quantitative data from the science command decoding equipment, and synchronizing and timing signals from the computing and sequencing (C and S) subsystem (including initiation signals derived from the limb and terminator crossing detectors). Based on stored science sequences and commands, the DAE will issue the detailed commands necessary to implement the sequences.

No DC signals will be carried across the spacecraft DAE interfaces. The C and S subsystem will control the science power switching electronics to apply power to the DAE.

9.3.1 Science Sequencer

The science sequencer within the DAE will accept synchronizing and timing signals from the C and S subsystem to establish gross sequencing for science operations. It will generate detailed timing and sequencing signals to control the experiments and to initiate the transfer of non real-time data to the data storage subsystem. The science sequencer will accept revisions to its stored sequences from the science command decoding equipment.

9.3.2 Science Data Controller

The science data controller within the DAE will be synchronized by data sync pulses from telemetry and will perform or control the conversion of science measurements data and engineering data to digital binary form in a format compatible with the spacecraft telemetry and data storage subsystems. It will also sense the occurrence of a solar flare and provide a signal to the C and S subsystem and to the science sequencer. The format for the non real-time data will include gaps

Table 9. Interface Requirements, PSP Mounted Science Equipment

Experiment	Sensors and Associated Hardware (SAH) FOV	Alignment*	Leads	Vol. in.³	Wt. lb	Pwr. watts	Science Remote Hardware (SRH) Vol. in.³	Wt. lb	Pwr. watts	Commands Quant.	Disc.	Aperture Covers	Maximum Data Bits/Orbit
Photoimaging A	9°	0.1°	20	1600	12	4	2000	8	7.5	1	4	In shutter	10^8
Photoimaging B	0.2°	0.1°	20	3200	28	4	4000	14	10.0	3	4	In shutter	10^8
UV Spectrometer	2.5°	1°	12	2000	20	4	750	7	11	-	5	Required	5×10^6
IR Spectrometer	0.5°	0.5°	12	2000	20	4	750	7	10	-	5	Required	5×10^6
IR Radiometer	5°	0.1°	15	2000	20	4	750	7	10	2	16	Required	10^7
TOTAL			79	10,800	100	20	8250	43	48.5	6	34		

*Relative to PSP boresight axis

Table 10. Interface Requirements, Body Mounted Science Equipment

Experiment	Sensors and Associated Hardware (SAH) No.	Total FOV	Alignment	Orientation Clock	Orientation Cone	Deployment	Leads	Vol. in.³	Wt. lb	Pwr. watts	Science Remote Hardware (SRH) Vol. in.³	Wt. lb	Pwr. watts	Discrete Commands
Magnetometer	1	4 ster	1°(1)	-	-	15-20 foot boom	10	180	0.7	1.8	100	5	4	3
Ion chamber	1	4 ster	-	-	-	with magnetometer	3	180	0.7	0.5	100	5	4	2
Cosmic ray telescope	3	45°	5°	- - 270	0 180 90	Protective covers	6 6 6	75 75 75	1 1 1	0.3 0.3 0.3	390	5	4	4
Trapped radiation	3	45°	5°	- - 90	0 180 90	Protective covers	6 6 6	100 100 100	1 1 1	0.3 0.3 0.3	1560	11	4	4
Gamma ray	3	45°	5°	- - 90	0 180 90	Protective covers	6 6 6	75 75 75	1 1 1	0.3 0.3 0.3	1560	11	4	4
Plasma probe	2	20°	3°	- -	0 0	Protective covers	4 4	25 50	1 2	0.5 0.5	160	5	4	4
Cosmic dust detector	4	10°	1°	- - 90 270	0 180 90 90	Protective covers	5 5 5 5	360 360 360 360	1.5 1.5 1.5 1.5	0.2 0.2 0.2 0.2	144	10	1	4
Bistatic radar	1	2 ster	-	-	0°	Two antennas 6 feet	4	-	3	-(2)	390	5	3	2
RF noise detector	1	4 ster	-	Any	Any	60 - 100 foot antenna	2	-	1.5	-	500	4	3	2
Ionosphere sounder	1	4 ster	-	Any	Any	20 foot antenna	2	-	5	-	390	4	3	2
Gravimeter	1	NA	0.1°	S/C axes	S/C axes	No	6	125	1	1	240	2	1.5	2
TOTAL								2750	29.9	7.8	5534	67	35.5	

(1) Alignment of each axis predictable to 1°, alignment to spacecraft axes controlled to 10°.

(2) During deployment only

amounting to 6 per cent, distributed throughout the data. These gaps are to provide time for inserting other telemetry data during playback from the data storage subsystem, as described in 6 of Section II, Volume 2.

The following output data lines will be provided:

a. Non Real-Time Data

1) Photoimaging data A

2) Photoimaging data B

3) UV and IR spectrophotometric data

4) IR radiometric data

5) Non real-time fields and particles data. (This may be the same line as (6), which is switched to the data storage subsystem either during solar flare mode, or for certain maneuvers.)

b. Real-Time Data

6) Fields and particles data

7) Science subsystem engineering data.

9.4 Science Command Decoding Equipment

The science command decoding equipment will accept binary digital command words and synchronizing signals from the command subsystem. It will decode and distribute the science commands within the science subsystem.

9.5 Science Power Switching Electronics

The science power switching electronics will accept discrete commands from the DAE, the science command decoding equipment, and power from the spacecraft power subsystem so as to energize the appropriate elements of the science subsystem. It also will accept direct backup commands from the command subsystem for this functions.

9.6 Planetary Scan Platform and Control

The planetary scan platform will provide suitable mounting interfaces, thermal control, and electrical connections. The PSP will accept angle and angle rate commands from the C and S subsystem, and command and timing signals from the science sequencer. The PSP will also be capable of operating in an automatic tracking mode so as to point its boresight axis toward the center of Mars.

The PSP will be capable of aligning the boresight axis of the platform to within ± 1.0 degree of the center of the planet Mars. This requirement applies when the flight spacecraft is in its cruise attitude control mode, and is in orbit about the planet, and when the sun-spacecraft-planet angle is equal to or greater than 80 degrees.

While tracking, the random angular motion of the PSP will be so limited that the maximum random angular velocity of the optical pointing axis relation to the planet local vertical will be less than 1.3×10^{-4} radian/second during a selected period chosen for a photoimaging experiment exposure. The PSP will be capable of meeting these requirements after being subjected to the acceleration and vibration produced by Mars orbit insertion and orbit trim with the PSP in its deployed position.

The PSP will provide 12 cubic feet of thermally controlled volume to house planetary experiment sensors and associated hardware. It will accommodate equipment weighing up to 100 pounds and dissipating up to 70 watts. The available mounting space perpendicular to the boresight axis of the platform will be a minimum of 6 square feet. Suitable aperture covers will be provided as part of the PSP to protect planetary sensors during cruise and during exposure to sunlight.

9.7 Science Deployment

Various deployment mechanisms will be required to position science sensors to achieve adequate antenna patterns, view angles, or isolation from spacecraft effects. The magnetometer sensor, the ion sensor, and certain low-frequency antennas will require deployment in order to achieve their specified performances. The planetary scan platform will be locked in position during boost flight and will require release.

The spacecraft will provide all signals for firing pyrotechnics associated with science deployment devices. Deployment devices will be designed so as not to interfere with the view angles of any experiment, nor with the performance of other spacecraft subsystems.

9.8 Fixed Science Packages

Two fixed science packages will be provided to furnish attachment, alignment, adequate view angles, and thermal control for all experiments whose requirements can be satisfied by a fixed mounting position on the spacecraft structure. Adequate protective aperture covers will be provided as part of the fixed science package.

The spacecraft will provide suitable attachment points, alignments, and positioning to achieve the required view angles (see Figure 3); and adequate thermal, mechanical, and electrical interfaces for the fixed science packages.

9.9 Science Interconnecting Wiring

The science interconnecting wiring will provide for the electrical interconnections between elements of the science subsystem. The interconnection of the science subsystem to spacecraft subsystems is not included. The science interconnecting wiring will be provided by the spacecraft contractor in keeping with 5.3 and will also satisfy requirements of the appropriate JPL supplied specification.

IV. SPACECRAFT FUNCTIONAL DESCRIPTION

1. SPACECRAFT OPERATIONAL MISSION

This section presents the spacecraft operational mission. It gives the particular events and sequences in narrative form required to carry out the mission profile defined by 4.11 in Section II. A detailed table of events is given in 5.4.

1.1 Prelaunch Phase

Three flight spacecraft will be airlifted to AFETR in sequential overlapping order such that approximately 17 weeks are available on each to carry out prelaunch activities, after previous receipt and checkout of the associated system test complex. The first two flight spacecraft, after checkout in the spacecraft assembly facility (SAF), will continue on through to launch as described below. The third flight spacecraft will be stored in the explosive safe facility (ESF) after its SAF checkout and used as a backup for the other two. All spacecraft will arrive at AFETR fully assembled and in the flight configuration, except for fuel and ordnance. Since the flight spacecraft is designed to function independently of the flight capsule it may be flown with a dummy capsule or ballast in place of the flight capsule if necessary.

1.1.1 Spacecraft Assembly Facility Operations

While at the SAF each spacecraft will undergo receiving inspection, functional spacecraft tests, and integrated systems tests (IST).

1.1.2 Explosive Safe Facility Operations

After moving from the SAF to the ESF, the spacecraft will receive alignment, weight, and center of mass checks followed by a separation and release test of the appendages using live ordnance. After these tests, ordnance simulators will be installed, the flight capsule will be mated to the spacecraft, and a spacecraft-capsule compatibility test will be performed. Following this test an integrated systems test, science quiet test, and spacecraft-capsule vertical alignment test will be performed. The planetary vehicle will then be mated to the flight shroud, after which a shroud cooling system test, a modified IST, and a mock countdown will be performed. The shroud sections with encapsulated planetary vehicles will then be separated and moved individually to the vertical assembly building (VAB).

1.1.3 Vertical Assembly Building Operations

While at the VAB, flight shroud-booster alignment will be performed followed by planetary vehicle IST, RFI tests, flight readiness demonstration test, mock countdown, and a simulated flight test.

1.1.4 Explosive Safe Facility Operations

After the encapsulated planetary vehicles have been returned to the ESF, all systems are flight pressurized, flight ordnance is installed, spacecraft propellants are loaded, leak tests are performed, and an IST is conducted. Following the IST, the planetary vehicle surface sterilization and sealing process takes place, prior to its movement to the launch pad.

1.1.5 Pad Prelaunch Operations

At the launch pad, the encapsulated planetary vehicles are mated to the booster and a flight shroud-booster alignment check is performed. A planetary vehicle IST is then conducted, after which RFI tests, joint flight acceptance composite test (J-FACT), flight readiness demonstration tests, and a mock countdown are accomplished. The pre-countdown is then initiated, during which time the launch vehicle is fueled and prepared for launch.

1.2 Launch and Injection Phase

1.2.1 Launch Countdown

The launch countdown initiates with the start of the formal time-controlled launch sequence and ends with the Saturn V hold-down release. The spacecraft operating status and configuration at launch is as follows:

- S-band communications: The up and down RF links are operative. The low-gain antenna is in a side pointing position and is being utilized. The steerable high-gain and medium-gain antennas are locked in a stowed position. The 1-watt transmitter is operating; the 50-watt transmitters are turned off. The "maximum coverage" receiving mode is operative.

- Relay link: Not operating.
- Command subsystem: Operating.
- Computing and sequencing: Operating; the mission time clock is initiated late in the launch countdown.
- Telemetry: Operating in Mode 1 (engineering data) at a 234 bit rate.
- Data storage: Not operating.
- Guidance and control: Only the gyros operating (in caged or rate output configuration).
- Power: Power equipment operative with power supplied from batteries.
- Cabling (electrical distribution): The preflight disconnect (with circuits via the launch vehicle to the launch complex equipment) will be disconnected by remote control late in the countdown and the disconnection verified.
- Temperature control: Heaters are operative as required by the associated automatic thermostat controls.
- Pyrotechnics: Power to the spacecraft pyrotechnics will be blocked by the normally open contacts of a safe/arm device.
- Planetary scan platform: Not operating and stowed in a locked position.

1.2.2 Powered Boost Flight

Powered boost flight begins with launch vehicle holddown release and ends after injection into the parking orbit. Primary events are given in 5.4.

1.2.3 Parking Orbit Phase

The parking orbit initiates with the end of powered boost flight and ends with the signal to restart the SIVB engine. Characteristics are as follows:

- Altitude: 185 km (100 n mi)
- Coast time: 2-90 min depending on day and time of launch
- Launch vehicle orientation: Pitch and roll axes are along the local horizontal with the roll axis in the direction of flight.

1.2.4 Powered Injection Flight

Powered injection flight initiates with restart of the SIVB and ends after injection into the interplanetary transit trajectory (generation of the separation initiate signal). Launch vehicle attitude is approximately the same as during the parking orbit phase.

1.2.5 Launch Vehicle-Planetary Vehicle Separation

Upon injection, the launch vehicle sends firing signals directly to fire the pyrotechnic separation devices of the forward planetary vehicle, which activate to release this vehicle. The separation springs impart a velocity of approximately 0.5 feet per second relative to the launch vehicle. Separation switches are activated to implement the following items:

- Signal C and S to initiate the post-separation program.
- Switch G and C to attitude hold mode.
- Arm spacecraft main pyrotechnic bus (backed up by C and S).

The appropriate individual pyrotechnic busses are armed during the mission before each event requiring pyrotechnics.

After a time delay to prevent interference with the forward planetary vehicle, the launch vehicle folds back the mid-fairing section to uncover the aft planetary vehicle. At a time that has allowed the forward planetary vehicle to achieve a separation distance of at least 100 feet, the launch vehicle sends a signal to the aft planetary vehicle separation devices to initiate separation of the aft planetary vehicle in the same manner as for the forward planetary vehicle. Separation switches on the aft planetary vehicle implement the same functions as described for the forward planetary vehicle. At a time that has allowed the aft planetary vehicle to achieve a suitable separation distance, the launch vehicle retrothrusts as required.

1.3 Celestial Reference Acquisition and Preparation for Cruise

1.3.1 Establish Spacecraft Configuration

Upon separation, the Guidance and Control subsystem stabilizes the planetary vehicle against any initial disturbing angular velocity. The attitude deviation of the planetary vehicle relative to the orientation before separation will be controlled to within 10 degrees for the period from separation until the initiation of sun acquisition, so as to allow communication with ground stations.

Although some of the appendages and articulated elements are not required until later in the mission, all are to be deployed or released soon after separation for enhanced reliability:

- Deploy science items; magnetometer boom, ionosphere antenna, RF noise antenna, bistatic radar antennas, and the planetary scan platform
- Release medium-gain antenna
- Release high-gain antenna
- Turn on science equipment for cruise operation
- Enter telemetry data Mode 2 (engineering and science) at 234 bit rate.

At the optimum time (depending on time of launch) the low-gain antenna will be deployed to the aft pointing position to obtain the best earth coverage.

1.3.2 Sun Acquisition

Depending upon day and time of launch the spacecraft will either be in earth eclipse or will enter earth eclipse shortly after injection. In the latter case, sun acquisition is not enabled until the planetary vehicle is in earth eclipse, to avoid an incomplete acquisition maneuver. At the time the sun is sensed, the guidance and control subsystem automatically begins solar acquisition. Acquisition is accomplished by using a gas jet reaction system in conjunction with a coarse sun sensor, with lock-on maintained by a fine sun sensor.

With sun acquisition complete, the planetary vehicle aft roll axis is pointing directly at the sun. Power transfer from batteries to the solar array is accomplished automatically as sun illumination of the solar cells is achieved.

1.3.3 Canopus Acquisition

At the same time that control reverts to the fine sun sensor, the C and S signals the G and C to initiate a roll maneuver. The first phase of this consists of a 0.22-degree per second roll spin lasting 50 minutes for magnetometer calibration and evaluation of the Canopus sensor track. After 50 minutes a 0.1-degree per second Canopus roll search begins. When a star of sufficient brightness appears in the field of view of the Canopus sensor the star acquisition gate is activated and the G and C automatically switches to hold the Canopus roll reference, turn off gyros and switch in the pitch/yaw derived rate networks to the cruise configuration.

The roll gas jet firing signals to establish the required roll rate are electrically integrated, and after a jet firing time slightly in excess of normal, the input signal to the switching amplifier is interrupted. This serves as a backup to prevent gas exhaustion and continued spin up should the roll gyro fail to provide the normal rate signal to null the spin or search rate command.

Canopus acquisition may be verified by prepositioning the high-gain antenna so that the beam intersects earth with the spacecraft in the nominal cruise attitude. For the initial acquisition, a near-earth detector, which is illuminated by earth-shine in the nominal cruise attitude, provides a confirmation signal to telemetry. The intensity of the Canopus sensor output, telemetered to earth, provides additional verification of acquisition, as well as the telemetered Canopus sensor star track during search. In the case of competing star-like objects it is possible, by ground command, to override the roll-reference lock-on and to command a series of 2-degree roll increments to achieve a true Canopus lock-on.

After completion of celestial acquisition, the high-gain and medium-gain antennas are positioned to be earth pointing if this has not been accomplished previously. This function is commanded by the C and S from stored data.

1.4 Interplanetary Cruise Phase (Prior to First Correction)

During initial interplanetary cruise, engineering and science data is being transmitted to earth by the low power transmitter and low-gain antenna. Approximately 16 hours after liftoff, the 1-watt transmitter is switched off and the 50-watt transmitter is switched on. Later in the flight, at an appropriate time, transmission is switched from the low-gain antenna to the high-gain antenna. Sometime before the first mid-course correction, a boresight calibration of the high-gain and medium-gain antennas may be accomplished. In addition, the high-gain and medium-gain antennas are repositioned approximately every 9 hours using the function generator for the cone angle determination. The Canopus sensor direction is updated as required throughout the flight.

During cruise the solar flare mode of the science equipment is automatic with flare occurrence.

The solar array for the spacecraft has been sized to accommodate the combined maximum average loads for all operating modes during the Voyager mission. If peak loads at any time exceed the capability of the solar array, a preprogrammed sequence of switching off certain functions is automatically accomplished to prevent battery drain. The first item to be switched off is the reaction gas heating system; the second is capsule power. These items as well as others can be switched on or off by commands from the ground.

1.5 First Interplanetary Trajectory Correction

The first interplanetary trajectory correction is to be performed at a time selected by the MOS between 2 and 10 days after launch. It consists of inertially (gyro) controlled turns which orient the thrust axis in a selected direction, followed by a propulsion burn to achieve a velocity increment of selected magnitude. This correction is to account for trajectory dispersion due to injection by the launch vehicle as well as a trajectory alteration to achieve the desired separation in the Mars arrival dates of the two planetary vehicles.

Prior to the maneuver, the magnitudes and direction of the maneuver turns and the magnitude of the velocity increment will be transmitted from the MOS via the DSN to the flight spacecraft. The magnitudes will be computed from information obtained by ground-based tracking and orbit determination. In addition, high-gain antenna pointing data will also be transmitted from the MOS to the spacecraft.

Upon receipt of the command data, the spacecraft will read out this data for verification by the MOS, and following verification the MOS will transmit an enable for the reorientation maneuver preparations. During these preparations, the following activities take place:

- The gyros are turned on.
- The thrust vector control system is tested by commanding engine gimballing and verifying the action.
- The telemetry is switched to engineering only, remaining at a 234 bit rate.
- S-band transmission is switched to the low-gain antenna and upon verification by the MOS that the switch has taken place, the MOS enables repositioning of the high-gain antenna so that it will be earth pointing, with the planetary vehicle in the thrusting attitude. Upon completion of the repositioning the MOS verification of the antenna position by means of gimbal angle data, which has been telemetered over the low-gain antenna, the MOS enables the spacecraft reorientation. It also enables an extra roll turn for enhanced coverage by the medium-gain antenna if this antenna is to be used.

Upon receipt of the reorientation enable, the C and S switches the G and C to maneuver mode and fine range attitude control. A roll turn followed by a pitch turn at 0.2 degree per second is implemented in accordance with the command data previously received from MOS. After reorientation, S-band transmission is switched from the low-gain antenna to the high-gain antenna, which is now earth pointing.

Based on transmission by the high-gain antenna and playback of recorded data for the reorientation, the MOS verifies the spacecraft attitude and enables engine operation. If an enable is not received after a certain period of time, the C and S commands reacquisition, and the spacecraft reacquires the Sun and Canopus as described in 1.3. After receipt of the enable, at the proper time the C and S issues a signal for engine start at low thrust. When integration of an accelerometer corresponds to the stored value for the velocity increment desired, the C and S signals engine cutoff. A backup shutdown signal based on a timer, set for a time slightly longer than the expected engine firing time, is also supplied by the C and S.

From this point in the flight, many of the operations of the spacecraft are functionally repeated. Such repeat operations have been designed to be as closely alike as possible; however, there are some differences in operations and sequencing.

Following engine cutoff the high-gain antenna is repositioned to be earth pointing after celestial reference acquisition, and the C and S switches G and C to the sun acquisition mode. The G and C aligns the spacecraft roll axis with the sun as before. After this alignment, the fine sun sensor automatically takes over and the coarse sun sensor is switched off. Upon completion of solar acquisition, a roll search maneuver is automatically initiated by the G and C. This maneuver, a 0.1-deg/sec roll until Canopus has been acquired, is the same as described in 1.3.3 except that the 0.22-deg/sec roll for magnetometer calibration and Canopus sensor track is not conducted. Upon completing celestial reference acquisition, cruise operation is reestablished by the C and S as follows:

- S-band transmission is switched to the low-gain mode.
- Telemetry is switched to Mode 3 to transmit data stored during engine firing and spacecraft orientation.
- Upon completion of Mode 3, telemetry is switched to Mode 2 to transmit engineering and science information at a 234 bit rate.

1.6 Interplanetary Cruise (Between First and Second ITC'S)

During interplanetary cruise, the spacecraft continues to acquire science and engineering data and telemeter this to earth. Every 9 hours during cruise, the high-gain and medium-gain antennas are repositioned using the function generator for cone angle, and the Canopus sensor is updated as required. At 75 days after injection the communications reception is switched to the maximum gain mode for the remainder of the mission.

1.7 Second Interplanetary Trajectory Correction

At a time determined by the MOS a second trajectory correction will be made, in the same manner as was described in Section 1.5, except that the medium-gain antenna is utilized for transmission rather than the low-gain antenna during maneuver preparations, and a second set of explosive valves in propulsion is utilized for opening and sealing propellant lines to the engine. After celestial reference acquisition, the high-gain antenna is utilized for transmission rather than the low-gain antenna, which was previously used during cruise condition.

The nominal mission is established with two interplanetary trajectory corrections, the second one occurring approximately 1 month before Mars encounter. In the event that a third correction is required, it can be done following the sequence just described.

1.8 Interplanetary Cruise After Final ITC

During this period of interplanetary cruise, the spacecraft operation again consists of science data acquisition with transmission of this and engineering data to the MOS as in the previous cruise condition. The high- and medium-gain antennas are repositioned as before about every 9 hours with Canopus sensor updating as required.

1.9 Mars Orbit Insertion

At a time selected by the MOS, the spacecraft will be inserted into Mars orbit. This maneuver consists of the same operations as required for the midcourse corrections. The necessary commands to accomplish the maneuver at a designated time will be transmitted from the MOS to the spacecraft.

After the C and S has read out the command data for verification, the MOS enables the maneuver preparation. Again, these preparations (consisting of turning on the gyros, testing the TVC system, and positioning the high-gain antenna) are carried out as described in Section 1.5, except that the medium-gain antenna is utilized during maneuver preparations instead of the low-gain antenna. The spacecraft is reoriented by a roll turn and pitch turn, as described in Section 1.5 and the capsule canister lid is jettisoned just prior to reorientation.

The engine operates at high thrust for Mars orbit insertion, and the sequencing therefore is somewhat different for this firing. The helium tank isolation valve is opened, which causes a diaphragm to burst and for the first time unseals the helium supply for pressurization of engine propellants. The engine is started similarly to previous starts utilizing a new set of explosive valves in the high-flow propellant lines. When the proper velocity increment is sensed by the accelerometer integrator, the C and S commands normally-open explosive valves to close, thereby shutting down the engine. As with the previous firings, a time signal from the C and S is utilized as a backup to shut down the engine.

Celestial reference is again acquired in the same manner as in Section 1.5, except that telemetry is switched to a rate of 7500 bits/sec for each spacecraft. In the event that only one spacecraft is operational at this time, the bit rate would be switched to 15,000 bits/sec.

1.10 Mars Orbit Cruise (Prior to Orbit Trim Maneuver)

The spacecraft will orbit for several days prior to the orbit trim maneuver. During the initial period the terminator and limb crossing sensors are turned on and the crossing signals are utilized by the C and S to establish the science orbital program. The stored orbital sequence is activated by the C and S as appropriate, following celestial acquisition. The planetary scan platform is also turned on and the acquisition angles are supplied by the C and S so that the PSP can begin tracking Mars. Following warm-up and a calibration of the planetary experiments, the science subsystem acquires data and telemeters it to earth.

During Mars orbit, the high-gain and medium-gain antennas are repositioned every 9 hours with Canopus sensor updating occurring as required.

1.11 Mars Orbit Trim Maneuver

At a time selected by the MOS, a Mars orbit trim maneuver will be conducted, in the same manner as described for an interplanetary trajectory correction in Section 1.5, except that the C and S inhibits the planetary scan package during the maneuver and communication transmission is switched to the high-gain mode rather than maximum coverage after the engine firing. Following these operations the spacecraft acquires science and engineering data and telemeters this to earth in the same manner as before the trim maneuver.

1.12 Capsule Separation and Flight Phase

At a time selected by the MOS approximately 3 to 10 days after Mars orbit insertion, the capsule is separated from the spacecraft. For this maneuver, the MOS sends spacecraft orientation and timing commands in the same manner as for engine firings, except that the commands include a capsule separation time rather than engine start and shutdown data. After the C and S has read out its memory for verification of the commands, the MOS enables the maneuver preparations. The spacecraft is reoriented in the same manner as for previous maneuvers, as described in Section 1.5, except that communications are switched to the medium-gain antenna for transmission during maneuver preparations and the planetary scan package is again inhibited during the capsule separation operations. The C and S energizes the capsule and sequences the capsule preseparation operations as necessary. The capsule relay receiver is tested and switched to telemetry. After the time set by the MOS is reached, the C and S signals capsule separation. This separation is accomplished by separation devices within the flight capsule. Telemetry is in mode 2 at an appropriate data rate and spacecraft and capsule data is telemetered in real-time until terminal entry of the capsule. During terminal entry, capsule data is recorded aboard the spacecraft by the relay link recorder for later transmission to earth. At an appropriate time following separation, the C and S commands the spacecraft to reorient to a celestial reference mode. This is done in the same manner as described in Section 1.5 except the communications are switched to the high-gain antenna rather than high coverage mode.

Following celestial reference acquisition the spacecraft is sequenced by the C and S into an orbit cruise operation the same as that preceding capsule separation. Additional discussion of data acquisition is given in 2.3 below.

2. SYSTEM FUNCTIONS AND INTERRELATIONSHIPS

The functions and relationships of the elements of the flight spacecraft system which provide the capability to execute the flight sequence described in 1 are discussed here. In some cases, the functions are discussed in groupings which are different from the subsystem areas around which the hardware breakdown is organized. This has been done to bring out the essential interactions underlying the over-all mission functions. An over-all spacecraft functional diagram is shown in Figure 13.

2.1 Communications

Communications with the DSN are carried out on the spacecraft primarily by elements of the radio subsystem and the command subsystem for uplink communications, and by elements of the radio subsystem and the telemetry subsystem for downlink communications. Three antennas (low, medium, and high gain) are provided and can be utilized for both up and downlinks.

The low-gain antenna provides sufficient gain for command reception for the entire mission, with the spacecraft in a cruise attitude. This antenna is fixed, once it has been deployed shortly after planetary vehicle separation. The medium-gain antenna has a single degree of freedom. It has a fan beam oriented so that rotation about the single axis can provide moderate gain along the spacecraft-earth line with the spacecraft in the cruise attitude. The high-gain antenna has two degrees of freedom, and in principle is capable of continuously pointing at earth. However, in the interests of system simplification, earth pointing is not required during spacecraft reorientations.

2.1.1 Uplink Communications

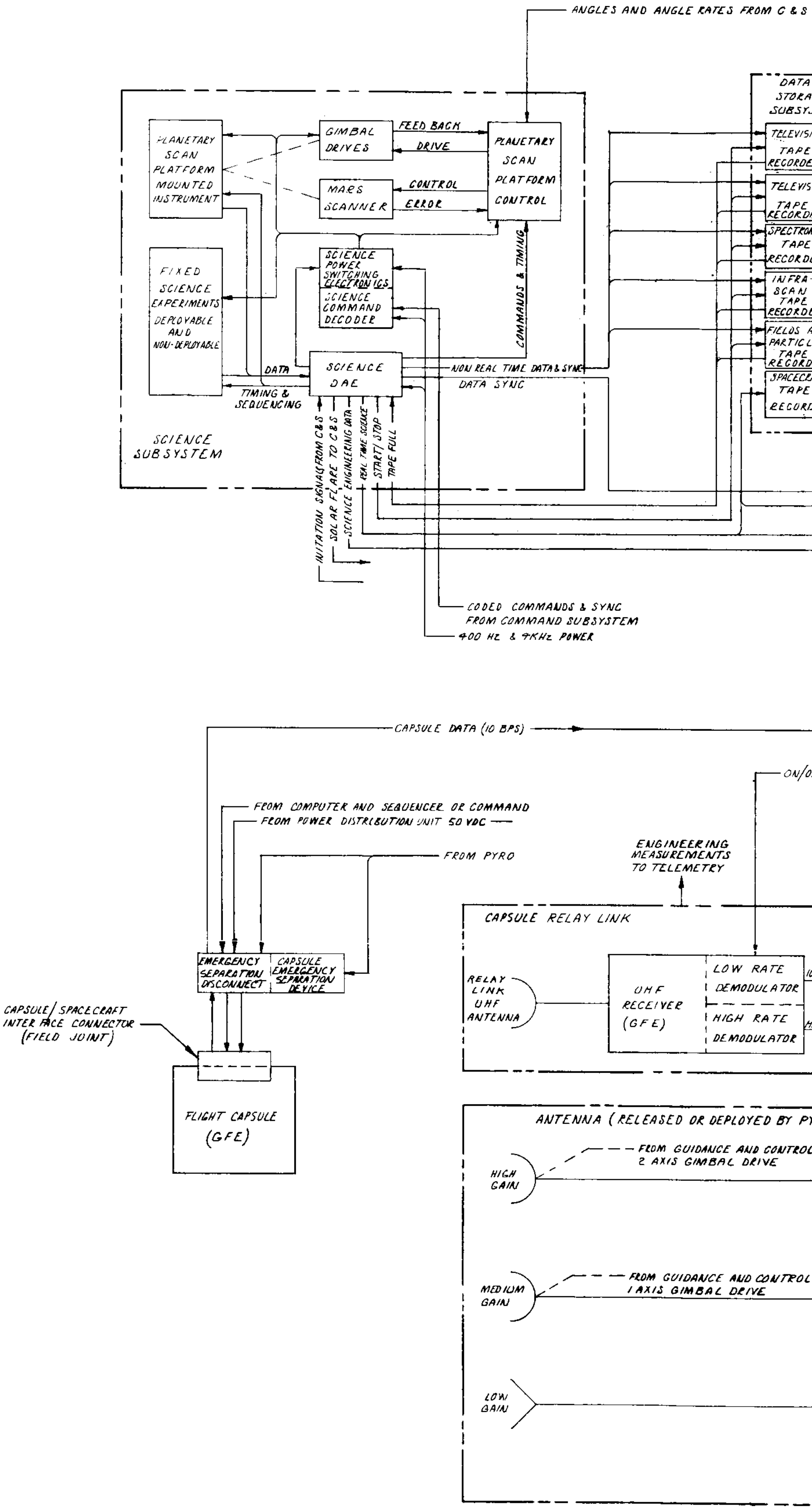
Uplink communication provides for transmission of both quantitative and discrete ground commands to the spacecraft. In conjunction with the downlink, it also provides a coherent carrier for two-way doppler and a range code turnaround channel for range measurements. Each antenna is connected to a separate receiver through a diplexer. Each receiver having adequate signal strength locks on the carrier and provides a coherent reference signal. The PN range code is an output when turnaround ranging is being used; a composite signal made up of the command data and command synchronizing subcarriers is supplied when command transmissions are being sent. It is assumed that there is no operational requirement for command and ranging transmission at the same time.

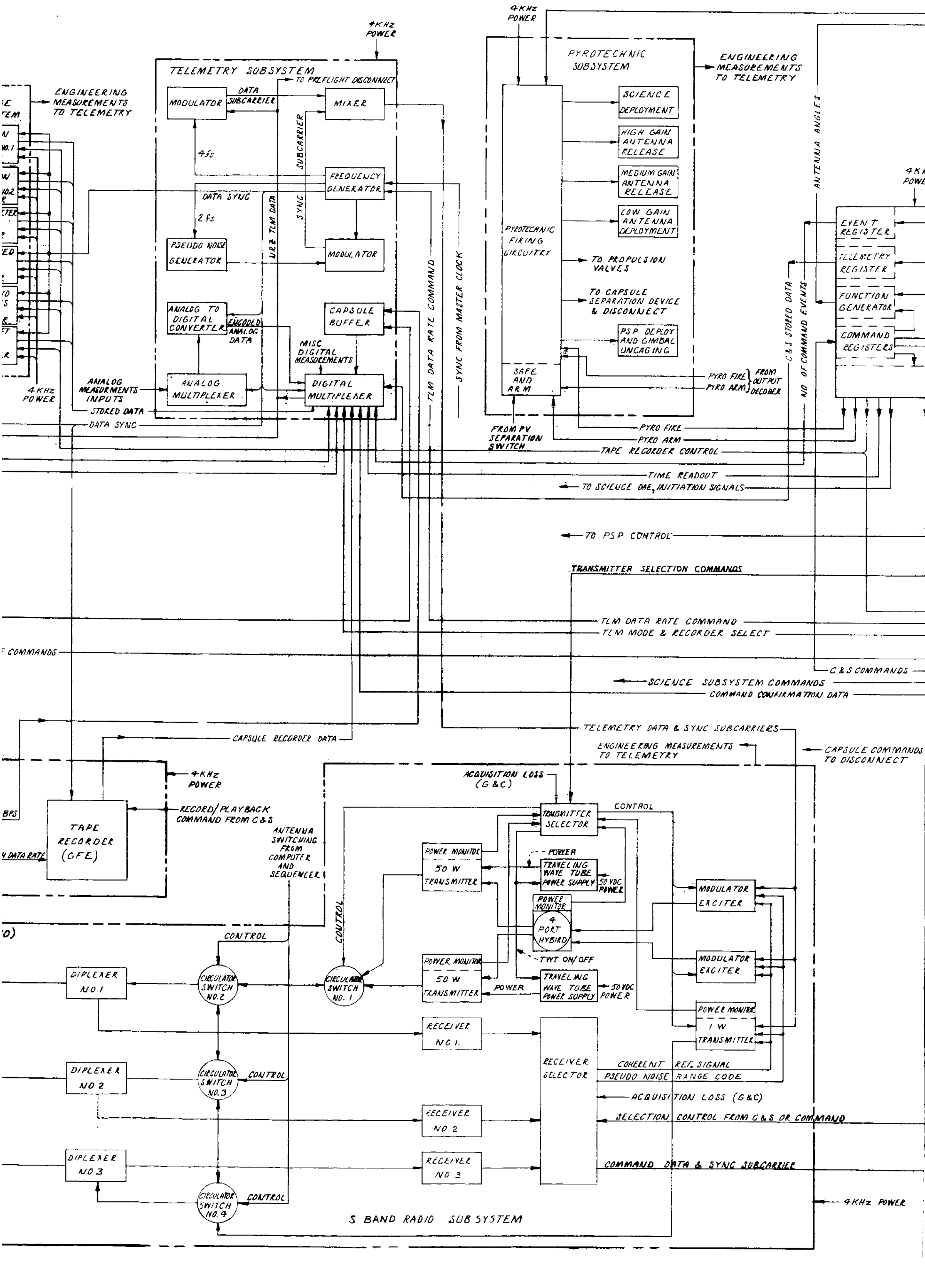
A receiver selector delivers the output of one of the receivers to the command detector and to the modulator exciters, as shown in Figure 13. This receiver selector operates in one of two modes subject to command override. The modes are:

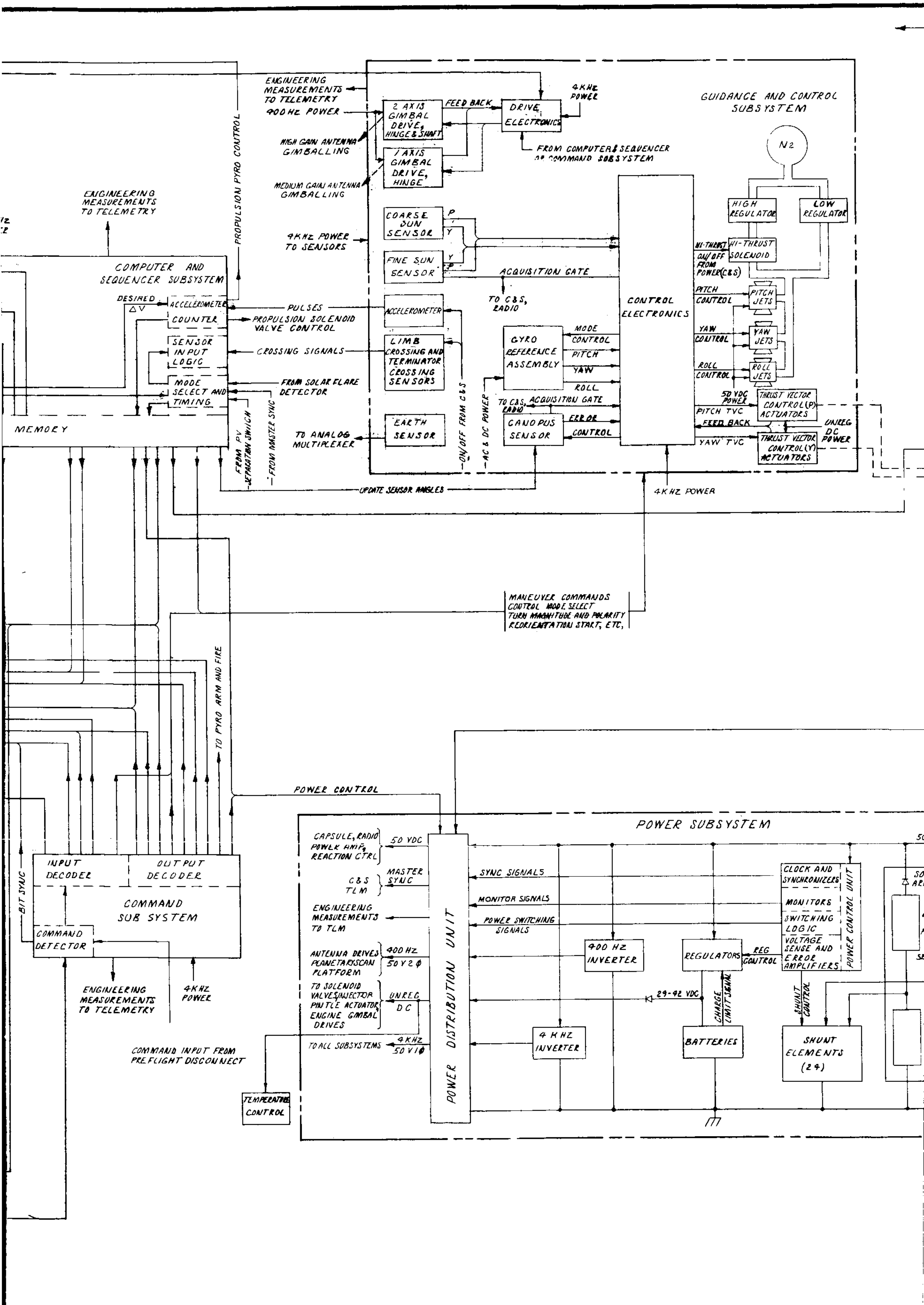
- Maximum gain: in this mode a receiver is selected by priority logic in the order of high-gain, medium-gain, and low-gain
- Maximum coverage: in this mode, a receiver is selected by priority logic in the order low-gain, medium-gain, and high-gain

On ground command, the receiver outputs can be shut off one at a time, to permit evaluation of the various receiver outputs. One additional ground command turns on all receivers to restore the selector to automatic operation.

The command detector establishes bit synchronization with the incoming synch subcarrier signal and develops a coherent reference signal for data demodulation. The detector in-lock signal and the data bit stream are delivered to the command decoder. These signals are also sent to the telemetry subsystem for transmission, to allow confirmation of correct receipt of commands.


ANGLES AND ANGLE RATES FROM C & S
DATA STORA SUBSYS
PLANETARY SCAN PLATFORM MOUNTED INSTRUMENT
GIMBAL DRIVES
FEED BACK
DRIVE
PLANETARY SCAN PLATFORM CONTROL
MARS SCANNER
CONTROL
ERROR
FIXED SCIENCE EXPERIMENTS DEPLOYABLE AND NON-DEPLOYABLE
SCIENCE POWER SWITCHING ELECTRONICS
SCIENCE COMMAND DECODER
COMMANDS & TIMING
DATA
SCIENCE DAE
TIMING & SEQUENCING
NON REAL TIME DATA & SYNC
DATA SYNC
SCIENCE SUBSYSTEM
INITIATION SIGNALS FROM C&S
SOLAR FLARE TO C&S
SCIENCE ENGINEERING DATA
REAL TIME SCIENCE
START/STOP
TAPE FULL
TELEVISI TAPE RECORDE
TELEVIS TAPE RECORD
SPECTROM TAPE RECORD
INFRA SCAN TAPE RECORD
FIELDS A PARTICL TAPE RECORD
SPACECR TAPE RECORD
CODED COMMANDS & SYNC FROM COMMAND SUBSYSTEM
400 HZ & 4KHZ POWER
CAPSULE DATA (10 BPS)
ON/O
FROM COMPUTER AND SEQUENCER OR COMMAND
FROM POWER DISTRIBUTION UNIT 50 VDC
FROM PYRO
ENGINEERING MEASUREMENTS TO TELEMETRY
CAPSULE RELAY LINK
EMERGENCY SEPARATION DISCONNECT
CAPSULE EMERGENCY SEPARATION DEVICE
CAPSULE/SPACECRAFT INTERFACE CONNECTOR (FIELD JOINT)
RELAY LINK UHF ANTENNA
UHF RECEIVER (GFE)
LOW RATE DEMODULATOR
HIGH RATE DEMODULATOR
FLIGHT CAPSULE (GFE)
ANTENNA (RELEASED OR DEPLOYED BY PY
FROM GUIDANCE AND CONTROL 2 AXIS GIMBAL DRIVE
HIGH GAIN
FROM GUIDANCE AND CONTROL 1 AXIS GIMBAL DRIVE
MEDIUM GAIN
LOW GAIN


TELEMETRY SUBSYSTEM
4KHz POWER
TO PREFLIGHT DISCONNECT
MODULATOR
DATA SUBCARRIER
MIXER
SUBCARRIER
4fs
FREQUENCY GENERATOR
DATA SYNC
SYNC
2fs
UZZ TLM DATA
PSEUDO NOISE GENERATOR
MODULATOR
ANALOG TO DIGITAL CONVERTER
ENCODED ANALOG DATA
CAPSULE BUFFER
MISC DIGITAL MEASUREMENTS
ANALOG MEASUREMENTS INPUTS
ANALOG MULTIPLEXER
DIGITAL MULTIPLEXER
STORED DATA
DATA SYNC
ENGINEERING MEASUREMENTS TO TELEMETRY
TLM DATA RATE COMMAND
SYNC FROM MASTER CLOCK
PYROTECHNIC SUBSYSTEM
SCIENCE DEPLOYMENT
HIGH GAIN ANTENNA RELEASE
MEDIUM GAIN ANTENNA RELEASE
LOW GAIN ANTENNA DEPLOYMENT
PYROTECHNIC FIRING CIRCUITRY
TO PROPULSION VALVES
TO CAPSULE SEPARATION DEVICE & DISCONNECT
PSP DEPLOY AND GIMBAL UNCAGING
SAFE AND ARM
PYRO FIRE
PYRO ARM
FROM OUTPUT DECODER
FROM PV SEPARATION SWITCH
TAPE RECORDER CONTROL
TIME READOUT
TO SCIENCE DAE, INITIATION SIGNALS
TO PSP CONTROL
ANTENNA ANGLES
C & S STORED DATA
NO OF COMMAND EVENTS
EVENT REGISTER
TELEMETRY REGISTER
FUNCTION GENERATOR
COMMAND REGISTERS
TRANSMITTER SELECTION COMMANDS
TLM DATA RATE COMMAND
TLM MODE & RECORDER SELECT
COMMANDS
C & S COMMANDS
SCIENCE SUBSYSTEM COMMANDS
COMMAND CONFIRMATION DATA
TELEMETRY DATA & SYNC SUBCARRIERS
CAPSULE RECORDER DATA
ENGINEERING MEASUREMENTS TO TELEMETRY
CAPSULE COMMANDS TO DISCONNECT
TAPE RECORDER (GFE)
RECORD/PLAYBACK COMMAND FROM C&S
BPS
DATA RATE
ANTENNA SWITCHING FROM COMPUTER AND SEQUENCER
ACQUISITION LOSS (G&C)
TRANSMITTER SELECTOR
CONTROL
POWER MONITOR
50 W TRANSMITTER
POWER
TRAVELING WAVE TUBE POWER SUPPLY
50 VDC POWER
4 PORT HYBRID
TWT ON/OFF
MODULATOR EXCITER
CIRCULATOR SWITCH NO. 1
CIRCULATOR SWITCH NO. 2
CIRCULATOR SWITCH NO. 3
CIRCULATOR SWITCH NO. 4
DIPLEXER NO. 1
DIPLEXER NO 2
DIPLEXER NO 3
RECEIVER NO 1
RECEIVER NO 2
RECEIVER NO 3
RECEIVER SELECTOR
1 W TRANSMITTER
COHERENT REF. SIGNAL
PSEUDO NOISE RANGE CODE
ACQUISITION LOSS (G&C)
SELECTION CONTROL FROM C&S OR COMMAND
COMMAND DATA & SYNC SUBCARRIER
4KHz POWER
S BAND RADIO SUB SYSTEM


GUIDANCE AND CONTROL SUBSYSTEM
ENGINEERING MEASUREMENTS TO TELEMETRY
400 HZ POWER
2 AXIS GIMBAL DRIVE, HINGE & SHAFT
FEED BACK
DRIVE ELECTRONICS
4 KHZ POWER
HIGH GAIN ANTENNA GIMBALLING
1 AXIS GIMBAL DRIVE, HINGE
FROM COMPUTER & SEQUENCER OR COMMAND SUBSYSTEM
MEDIUM GAIN ANTENNA GIMBALLING
N2
HIGH REGULATOR
LOW REGULATOR
COARSE SUN SENSOR
4 KHZ POWER TO SENSORS
FINE SUN SENSOR
ACQUISITION GATE
HI-THRUST ON/OFF FROM POWER (C&S)
HI-THRUST SOLENOID
PROPULSION PYRO CONTROL
ENGINEERING MEASUREMENTS TO TELEMETRY
COMPUTER AND SEQUENCER SUBSYSTEM
DESIRED ΔV
ACCELEROMETER
COUNTER
PULSES
PROPULSION SOLENOID VALVE CONTROL
TO C&S, RADIO
CONTROL ELECTRONICS
PITCH CONTROL
PITCH JETS
SENSOR INPUT LOGIC
CROSSING SIGNALS
LIMB CROSSING AND TERMINATOR CROSSING SENSORS
GYRO REFERENCE ASSEMBLY
MODE CONTROL
PITCH
YAW
ROLL
YAW CONTROL
YAW JETS
ROLL CONTROL
ROLL JETS
MODE SELECT AND TIMING
FROM SOLAR FLARE DETECTOR
ON/OFF FROM C&S
AC & DC POWER
TO C&S, RADIO
ACQUISITION GATE
50 VDC POWER
THRUST VECTOR CONTROL (P) ACTUATORS
PITCH TVC
FEED BACK
UNREG DC POWER
MEMORY
FROM PV SEPARATION SWITCH
FROM MASTER SYNC
TO ANALOG MULTIPLEXER
EARTH SENSOR
CANOPUS SENSOR
ERROR
CONTROL
YAW TVC
THRUST VECTOR CONTROL (Y) ACTUATORS
UPDATE SENSOR ANGLES
4 KHZ POWER
MANEUVER COMMANDS
CONTROL MODE SELECT
TURN MAGNITUDE AND POLARITY
REORIENTATION START, ETC.
TO PYRO ARM AND FIRE
POWER CONTROL
POWER SUBSYSTEM
CAPSULE, RADIO POWER AMP, REACTION CTRL
50 VDC
MASTER SYNC
C&S TLM
SYNC SIGNALS
CLOCK AND SYNCHRONIZERS
MONITOR SIGNALS
MONITORS
ENGINEERING MEASUREMENTS TO TLM
POWER SWITCHING SIGNALS
SWITCHING LOGIC
VOLTAGE SENSE AND ERROR AMPLIFIERS
POWER CONTROL UNIT
INPUT DECODER
OUTPUT DECODER
COMMAND SUB SYSTEM
BIT SYNC
COMMAND DETECTOR
ANTENNA DRIVES PLANETARY SCAN PLATFORM
400 HZ
50 V 2 φ
400 HZ INVERTER
REGULATORS
REG CONTROL
TO SOLENOID VALVES/INJECTOR PINTLE ACTUATOR ENGINE GIMBAL DRIVES
UNREG DC
29-42 VDC
CHARGE LIMIT SIGNAL
SHUNT CONTROL
ENGINEERING MEASUREMENTS TO TELEMETRY
4 KHZ POWER
TO ALL SUBSYSTEMS
4 KHZ 50 V 1 φ
POWER DISTRIBUTION UNIT
4 KHZ INVERTER
BATTERIES
SHUNT ELEMENTS (24)
COMMAND INPUT FROM PREFLIGHT DISCONNECT
TEMPERATURE CONTROL

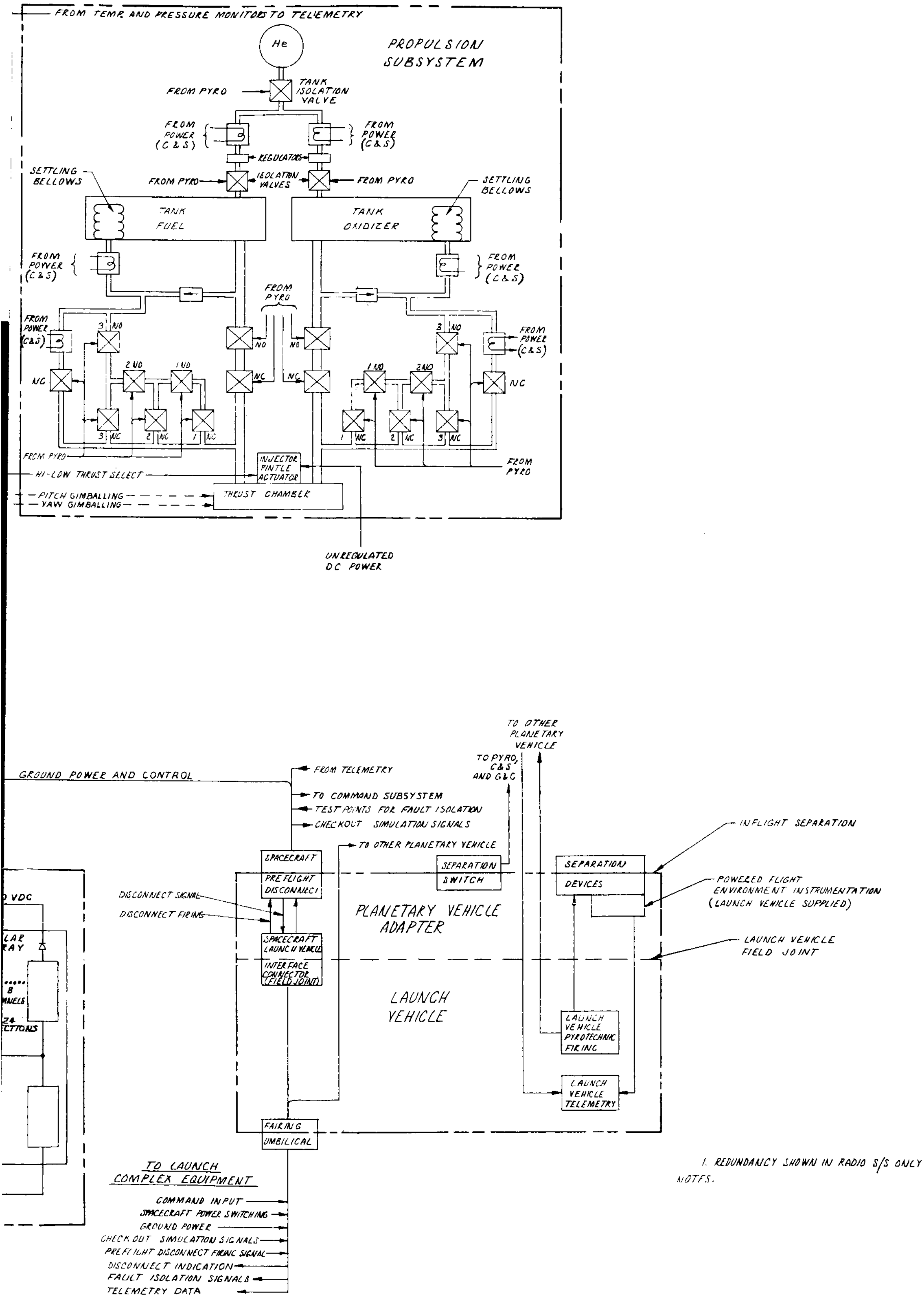


Figure 13. Over-all Functional Block Diagram

The command subsystem decoder is organized in two sections. The first is the input decoder which determines whether the digital word being processed is a discrete command for immediate execution or a quantitative to be routed to a storage register in some subsystem. Many of the words processed as quantitatives at this point are discretes tagged for delayed execution (usually by the C and S subsystem). Quantitatives are shifted at the command bit rate (≈ 1 bit/sec) to the subsystem identified in the address. Discretes are delivered to the output decoder, which supplies an actuating signal on the proper line.

2.1.2 Downlink Communications

The downlink telemeters engineering and scientific data to earth and provides a coherent carrier signal and PN range modulation for range and range rate measurements.

Three transmitters are available. Two have 50-watt power output and one has 1-watt power output. In normal operation, the choice between 1-watt and 50-watt and the choice among antennas is programmed through the computing and sequencing subsystem, backed up by direct radio commands. The choice between 50-watt transmitters and the two modulator exciters is controlled by the transmitter selector, with internal logic operating on signals from the power monitors so as to transfer operation from a unit when its RF output falls below a given threshold value. The internal logic is established by the computer and sequencer but can be overridden by signals from the command subsystem.

The modulator exciters (the 1-watt transmitter is basically also a modulator exciter) receive a composite modulating signal. The telemetry composite signal made up of the telemetry data and synchronizing subcarriers is always present; the reference signal from the selected receiver is provided for coherent operation; and the PN range code is supplied when ranging is active. Only the telemetry signal is supplied for noncoherent operations, with the downlink carrier signal being generated from a crystal oscillator within the modulator exciter.

2.1.3 Communications Capabilities

Throughout launch, injection, and early interplanetary cruise, the low-power transmitter is turned on. The low-gain antenna is switched in and the receivers are in the maximum coverage mode; 234 bits/sec downlink capacity is available directly from each spacecraft.

Shortly after injection, the DSIF station in view establishes two-way lock with both planetary vehicles using the acquisition aid antenna and frequency-multiplexed transmission. Then the DSIF station switches reception from the acquisition aid to the 85-foot dish. Frequency-multiplexed uplink communication is maintained via the acquisition aid antenna. The second DSIF station to acquire after injection uses the 85-foot dish for both up and downlink communications, with 10-kw, frequency-multiplexed transmission.

Approximately 16 hours after launch, the selected 50-watt transmitter is turned on and the 1-watt transmitter is switched off. The low-gain antenna continues to be utilized and the receivers continue in the maximum coverage mode. Turnaround ranging is actuated by ground command and operates simultaneously with 234 bits/sec telemetry. The DSIF stations use 85-foot antennas and with 10-kw transmitters can maintain continuous two-way communication with both spacecraft. The capability includes both planetary vehicles within view of a single station employing frequency-multiplexed operation or separate DSIF stations. For turnaround ranging after 15 days, the high-gain antenna is switched in.

After approximately 75 days, both spacecraft switch the 50-watt transmitters to the high-gain antennas, and switch receivers to the maximum gain mode. This condition permits communication with the 10-kw, 85-foot dish sites at the cruise bit rate until Mars encounter.

After 150 days, the 100-kw ground transmitter with the 85-foot dish (or 210-foot dish) is required for turnaround ranging.

Five days before Mars encounter (E -5 days) by the first spacecraft, the DSIF sites are switched for this spacecraft from the 85-foot to the 210-foot facility. The second spacecraft continues to operate with the 85-foot sites until E -6 hours. Continuous two-way communication is thus provided to both spacecraft for the period of E -5 to E +10 days. Although the 210-foot facility need not be used until after the bit rate is increased, it is activated at E -5 days for the first spacecraft to provide a transition period. After encounter, the first spacecraft transmits at 7500 bits/sec to support the high data rate planetary science. At E -6 hours for the second spacecraft, two-way communications is switched from the 85-foot to the 210-foot sites. The 210-foot facility also continues to receive from the first spacecraft. After encounter, the second spacecraft also transmits at 7500 bits/sec. The 210-foot sites then receive simultaneously from both spacecraft at this bit rate until the end of the mission. The uplink operates on a time-shared basis using the 100-kw transmitters on the 210-foot antennas. If only a single spacecraft is operational during the early Mars orbital phase, the spacecraft data rate is increased to 15,000 bits/sec.

2.2 Spacecraft Sequencing

The basic sequencing of the spacecraft is accomplished automatically based on command data stored prior to launch in the computing and sequencing (C and S) subsystem. The mission commands from the C and S are organized into distinct timed sequences controlled by on-board clock references. The sequences are initiated individually at times determined either by ground command, stored commands, or specific mission events, such as launch, separation, or terminator crossing. The stored commands in each sequence are time-referenced to the start of the sequence and are issued at the indicated elapsed time from the point of initiation of the associated counter controlling the sequence. An enable bit is associated with a stored sequence which can be used to inhibit or enable the sequence, or to synchronize the operation of the sequence to specific events, such as terminator or limb crossings.

Long term events, such as Canopus sensor update, are timed from a mission clock, which is initiated just prior to launch. Some sequences of events, such as the Mars orbit insertion sequence, are initiated at a designated mission clock time stored before launch, but can be updated to correct for trajectory dispersions. Other maneuver sequences such as interplanetary trajectory corrections and orbit trim, are initiated by radio command as the need arises. Orbit sequences are timed from Mars arrival, from the completion of a previous sequence, from a terminator or limb crossing signal, from the occurrence of a solar flare, or from a ground command. All sequences or events can be updated or inhibited by ground command as desired.

2.3 Data Handling

The telemetry subsystem provides the focus for the flow of data, carries out the detailed interleaving of data from the various sources, and generates the downlink subcarriers and their modulations. The C and S subsystem directs traffic and sets the telemetry mode. There are three sources of data: the spacecraft engineering data, the spacecraft science data, and the capsule data. The methods of treating this data are significantly different in interplanetary cruise, during capsule flight, and during orbital operations.

To match telemetry data transmission to the mission requirements, four telemetry modes are utilized as follows:

- Mode 1 includes engineering data only. Both spacecraft and low rate capsule engineering data are transmitted. This mode is used during launch, maneuvers, and when required for failure analysis.
- Mode 2 is a combination of spacecraft and capsule real-time science data, such as field and particle measurements. This mode is used during cruise, including the flight operation phase for the separated capsule.
- Mode 3 is the playback of non-real-time data from the tape recorders. This includes stored science instrument data, stored capsule data, and data recorded during maneuvers.

- Mode 4 is selected engineering data plus C and S memory readout. This mode is used prior to maneuvers for verification of maneuver commands and for checkout of guidance and control equipment.

The telemetry equipment samples analog and digital inputs in a fixed sequence at a variety of rates. All analog measurements are converted into 7-bit binary words (or partial words). All digital measurements are broken into 7-bit segments. These 7-bit words plus suitable synchronizing words are modulated on the data subcarrier at a variety of rates as indicated in Table 11. The telemetry bit rate is programmed by the computer and sequencer or by direct radio command.

Table 11. Telemetry Data Rates

Mode		Data Rate (bits/sec)
Emergency		7.1
Cruise		234
High Rate	1.	1,875
	2.	3,750
	3.	7,500
	4.	15,000

2.3.1 Data Acquisition for Orbital Operations

The management of science data is carried out by the DAE (within the science subsystem) based on detailed stored sequences which are initiated by the C and S subsystem, and updated via the command subsystem (using the science command decoder). The initiation signals are derived from signals during cruise based on the C and S mission time clock. During orbital operations the initiation signals are generally derived from terminator and limb crossing sensor signals augmented by timed signals and ground commanded changes.

The C and S subsystem also provides pointing data to the planetary scan platform (PSP) to cause its boresight axis to point in the appropriate direction. Three modes of PSP operation are provided. PSP Mode 1 involves pointing the boresight axis in a fixed direction. This mode may provide acquisition for Mars tracking or may provide for special experiment requirements. Mode 2 involves tracking the center of Mars. This mode is entered from Mode 1. It is the nominal orbital science mode. Under the contamination constraints, a part of the sterilization container may remain attached to the flight spacecraft so as to partially block the view of the PSP. Thus sequencing between Mode 1 and Mode 2 may be required each orbit. PSP Mode 3 involves open loop image motion compensation. This mode is entered from Mode 2. Command rates which cause the boresight axis to point at a fixed point on the planet's surface are executed. Considerable reduction in image motion can be achieved by using the capability of the spacecraft to maintain a fixed attitude from which to develop accurate scanning rates.

2.3.2 Cruise

The nominal data rate throughout cruise provides adequate capacity for meeting the normal requirements for spacecraft and flight capsule engineering data and interplanetary science data. The ranges and ground antenna requirements for various combinations of power level, antenna, and data rates are discussed in 2.1 above. In the event a solar flare occurs, the DAE will signal the C and S, and either change experiment scale factors or record the data by the fields and particles tape recorder, or both. After the flare, the recorded data can be mixed with real-time data and transmitted in the cruise mode at one of the high data rates. It may prove desirable to transmit solar flare data in real time utilizing one of the high data rates.

Data collected during maneuvers must be stored because the simple antenna pointing methods used may cause the downlink to be interrupted while the spacecraft is turning. Also, while the spacecraft is in its maneuver attitude, part of the telemetry capacity is taken up with verifying the execution of commands. Playback of this data is at the cruise telemetry rate at the expense of some real-time capacity.

2.3.3 Orbital Operations

Gathering orbital science data is concentrated in a relatively short part of the orbit for the elliptical orbits studied. A number of tape recorders are provided in the data storage subsystem to permit rapid collection of data during the period of interest, with playback and transmission during the less active times. The DAE prepares the data in a format consistent with the telemetry and records it on the tape recorders provided: one each for two TV experiments, one for two spectrometer experiments, one for an IR radiometer experiment. In each case, the DAE leaves frequent gaps amounting to about 6 per cent of the tape. These gaps are utilized to permit the telemetry system to automatically interleave real-time engineering and science data with the recorded data, when the C and S causes one of the tape recorders to readout to telemetry.

The telemetry subsystem provides synchronizing signals for the tape playback operations. The tape recorders provide signals when the data gaps are present to cause the telemetry subsystem to sample other data.

2.4 Attitude Control

The spacecraft generally operates in a fully attitude stabilized mode utilizing the Sun and Canopus as celestial references. At times, such as planetary vehicle separation, celestial reference acquisition, propulsion maneuvers, and capsule separation, the spacecraft operates in a controlled maneuvering mode utilizing on-board inertial references. Control reverts automatically to an inertial holding mode during eclipse and occultation conditions, or at any time when the celestial references are lost.

Control torque is generally obtained from a redundant reaction gas jet system that provides two levels of thrust. During engine firings, control torque in pitch and yaw is obtained by gimballing of the single thrust chamber about two axes. Roll control is obtained at all times by the reaction gas system.

2.4.1 Planetary Vehicle Separation

For the period immediately following planetary vehicle separation, the spacecraft orientation is to be maintained approximately the same as it was prior to separation, to allow continued communication during the post-separation eclipse period until sun acquisition. At separation, a rate-nulling mode is activated which utilizes the reaction control high thrust level to control the initial tip-off angular motion. After about 10 seconds, the system is switched to an inertial attitude hold mode utilizing the gyros for an attitude reference. The low thrust level is also switched in to reduce propellant consumption.

2.4.2 Celestial Reference Acquisition

When the spacecraft sun sensor is activated, after the post-separation eclipse, attitude control reverts automatically from attitude hold to the sun acquisition mode. Pitch and yaw angular rate signals are obtained from gyros in the rate mode (caged), with position errors obtained from the coarse sun sensor. Illumination of the fine sun sensor occurs when the aft roll axis is within 10 degrees of the sun line, and an acquisition gate then switches from the coarse to the fine sun sensor.

For some missions, planetary vehicle separation may occur in sunlight, with an eclipse following shortly thereafter. The sun acquisition operation is then locked out for this initial post-separation sun period to avoid an incomplete acquisition.

When both the pitch and yaw fine sun sensors are illuminated during the initial post-separation acquisition, a controlled spin rate about the roll axis is automatically commanded. This spin maneuver is to obtain a star track as seen by the Canopus sensor and to calibrate an on-board magnetometer if desired. After an appropriate time, the system automatically slows the roll rate to allow search and acquisition of Canopus for roll reference.

For the initial acquisition, a near-earth detector, which is illuminated by earth-shine in the nominal cruise attitude, verifies proper Canopus acquisition. In addition, Canopus acquisition can be verified by positioning the high-gain antenna so that the beam intersects earth with the spacecraft in the proper cruise attitude. The correlation of the telemetered star track and roll angle data also provides verification of proper acquisition.

2.4.3 Cruise

During the cruise mode, the spacecraft attitude is maintained in a limit cycle about the Sun-Canopus reference orientation. The guidance and control system generally operates with low thrust reaction control and maintains a gross accuracy of 0.5 degree about each axis. A fine attitude control corresponding to an accuracy of 0.25 degree is also provided. This is utilized prior to an inertial maneuver and during planetary science data gathering (such as TV exposures).

At appropriate intervals, the C and S updates the Canopus sensor line of sight cone angle to maintain Canopus within the field of view.

2.4.4 Inertial Control

An inertial maneuvering mode is provided for reorientations and attitude hold as required. The C and S activates the gyro reference assembly and switches to fine attitude control and high thrust reaction control before each maneuver. The reorientation maneuvers are carried out by torquing the gyros so as to turn the spacecraft around one axis at a time until the integrated gyro outputs indicate the proper angle. The desired attitude is then held under inertial hold operation.

When the inertial control mode signal is removed, control reverts automatically to an acquisition operation. Roll acquisition follows directly after sun acquisition without an intervening roll spin maneuver as carried out for the initial post-separation phase.

2.5 Power

Power is provided in suitable forms for distribution to electrical equipment on board the flight spacecraft, and to the flight capsule until its separation. Primary power is derived from the sun by means of silicon photovoltaic cells, mounted on a fixed solar array. Secondary silver-cadmium batteries are used whenever the solar array is incapable of supporting the loads, as during launch, maneuvers, and eclipses. Appropriate controls are provided to maintain proper functioning of the subsystem.

The solar array output is limited to 50 VDC ±1 per cent by shunt regulation of a portion of each series string of solar cell modules. Power dissipation in the shunt elements assembly is minimized by a special sequential shunt configuration. The shunt elements are controlled from bus voltage sensing and error signal amplifier circuitry in the power control unit (PCU).

The batteries are charged from the 50-volt bus through simple dissipative current regulators. Charging is terminated by a control signal from individual temperature-compensated cell voltage sensors mounted on the battery cells. When the highest cell voltage decreases below a present level, constant current charging is again initiated. When the solar array is incapable of supporting the system load, the battery discharges through a boost regulator to maintain the bus at 50 volts. A switching type boost regulator is used which is controlled by the PCU voltage sensor and error amplifier circuits.

The two main outputs from the system are the regulated 50-VDC ±1 per cent bus and a 50-VAC ±2 per cent, 4-kHz, single-phase, square-wave bus. A simple unregulated inverter is used to supply the 4-kHz-AC output. Sequential inverter redundancy is provided by sensing AC bus undervoltage and switching to a standby inverter in the event of inverter failure. This sensing and switching function is performed by the power distribution unit. Additional 400-Hz two-phase power is provided to supply AC power to the antenna and PSP drive motors. Sequentially redundant units are provided in the same manner as in the case of the 4-kHz inverter. High-current, short-duration requirements for propulsion are supplied directly from the battery bus. These include solenoid valve and gimbal motor power.

A spacecraft synchronizing signal generator is provided in the PCU to generate sync frequencies. Some of these are used internally for synchronization of the boost regulators and inverters. The remainder of the synchronization frequencies are distributed throughout the spacecraft.

All power, including regulated and unregulated DC, 4-kHz and 400-Hz AC, plus synchronization frequencies, is distributed throughout the spacecraft from the power distribution unit. In addition, all power switching for programmed loads is performed in the power distribution unit. Control of power switching is generally under the programmed control of the C and S, with backup control by ground command.

2.6 Propulsion

The propulsion function is provided by a single LEM descent engine utilizing storable hypergolic liquid propellants, fed by a stored gas pressurization system. The engine is designed with two thrust levels: a high thrust of 7750 pounds for Mars orbit insertion and a low thrust of approximately 1040 pounds for midcourse corrections and orbit trim firings.

The fuel is a mixture of hydrazine and UDMH in equal proportions, and the oxidizer is nitrogen tetroxide. These propellants are effectively sealed from the engine before and between engine firings by means of explosive and solenoid valves as depicted in Figure 13. These valves are so arranged that a separate series pair of normally open and normally closed valves are utilized for an engine firing. These are provided for three midcourse corrections and for start of Mars orbit insertion, with subsequent engine control, including Mars orbit trim, accomplished by solenoid valves.

Start tanks for each propellant are utilized until the spacecraft acceleration insures propellant settling in the main feed system. These are non-rechargeable tanks utilizing bellows, and are mounted within the main tanks.

Prior to launch, the ullage space in the propellant tanks is pressurized with helium gas, and this initial pressurization provides blowdown operation of the propellant feed system during the low thrust midcourse engine firings. For the Mars orbit insertion firing at high thrust, regulated pressurization gas is supplied to the tanks from a high-pressure helium supply vessel. This vessel is sealed before launch and remains sealed until arrival at Mars, to insure that there is no loss during the interplanetary flight. The helium supply system is brought into use by actuation of the helium tank isolation valve (NC explosive valve) shown in Figure 13. Actuation of downstream normally closed explosive valves at the same time allows the two propellant tanks to be pressurized from the single helium tank through redundant check valves and regulators. The pyrotechnic downstream isolation valves are used to seal off propellant tank vapors from the pressurization components during the interplanetary flight.

The engine utilizes an ablatively cooled nozzle; thrust vector control is accomplished by two-axis engine gimballing under control of the G and C subsystem.

2.7 Temperature Control

The temperature control function ensures that spacecraft components will experience satisfactory temperatures throughout the Voyager mission. The design features include 1) surface finishes to attain desired radiometric properties, particularly on external equipment, 2) appropriate distribution of electronic components, and 3) structural design to achieve various degrees of thermal coupling (generally close coupling within the main compartment and poor coupling between the main compartment, solar array, and capsule, and between the external equipment and the solar array backup structure). Temperature control hardware includes multilayer aluminized Mylar insulation, bimetal-actuated louver assemblies, and thermostatically-controlled heaters.

Temperature control of the main compartment is achieved by insulation on the external surface, outriggers, and capsule adapter, and by louvers covering the uninsulated radiating areas on the equipment panels. The louvers regulate the temperature of the electronic equipment, while the insulation reduces the temperature gradients within the compartment, thereby providing a proper thermal environment for the remaining internal equipment.

Insulation covering the back side of the solar array reduces the cooling rate during eclipses, thereby permitting survival of longer eclipses. Appropriate radiometric properties of the front side limit maximum array temperatures to acceptable values.

External equipment such as antenna drive motors will use a combination of insulation and thermostatically controlled heaters for temperature control. Much of the external equipment will be passively controlled using appropriate surface finishes alone or in combination with insulation.

3. SYSTEM DESIGN DISCUSSION

This section presents a description of the selected spacecraft configuration that provides the design features necessary to satisfy the mission objectives, constraints, and requirements set forth in the preceding parts of this Volume. This section also contains design considerations which have sufficient importance to warrant separate discussion. Several basic drawings are included to illustrate the significant characteristics of the system. These are as follows: Figure 14, which shows the outboard profile of the operational in-flight configuration; Figure 15, which pictures the launch-ready spacecraft without the flight capsule, sectioned to exhibit equipment arrangement; Figures 16 and 17, which present isometric views of the spacecraft bus; Figure 18, which delineates the placement of all appendages and sensors. The detailed descriptions and related analyses for each spacecraft subsystem are contained in Volume 2. A detailed structural drawing is included in 9 of Section II, Volume 2.

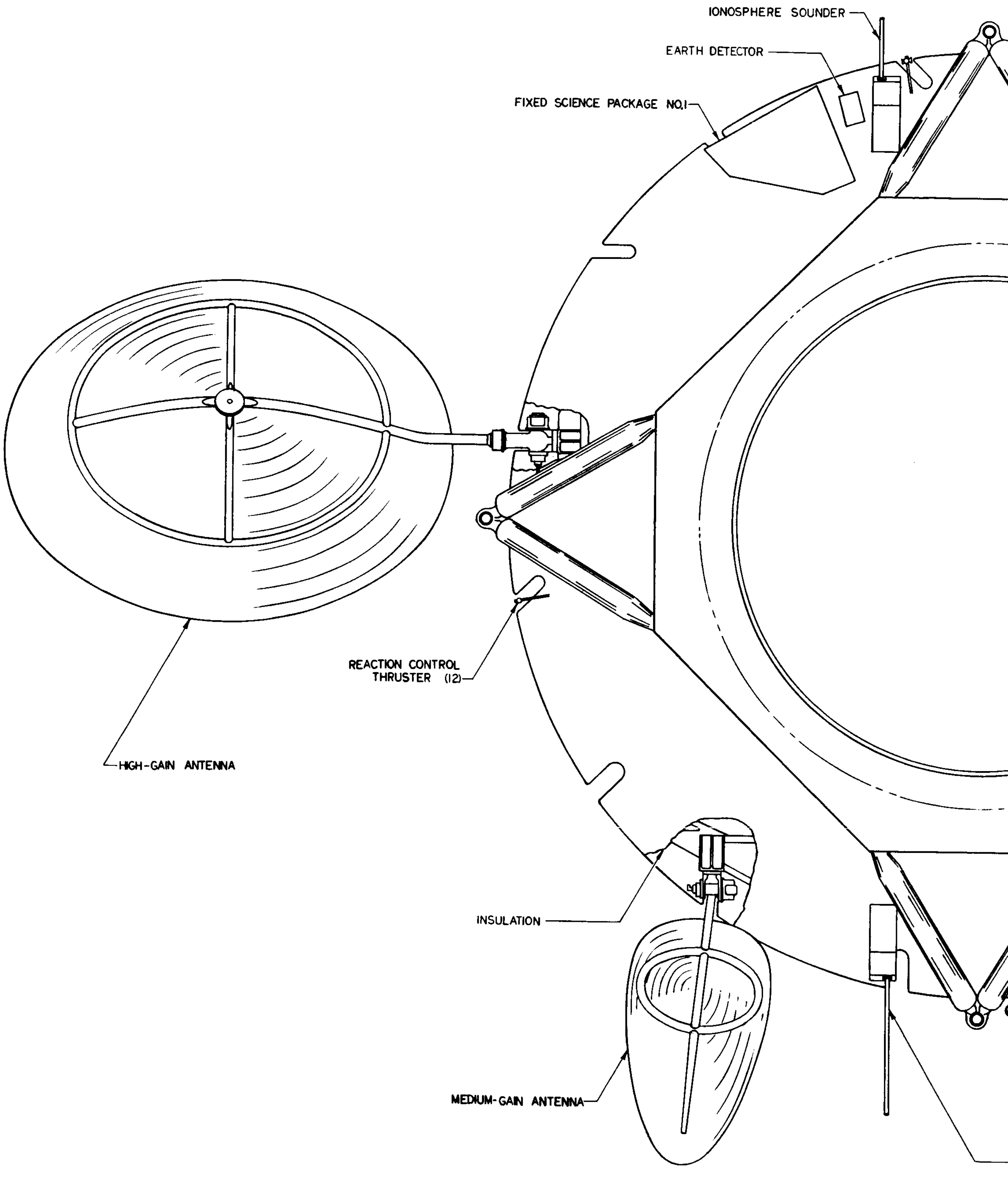
3.1 General Arrangement

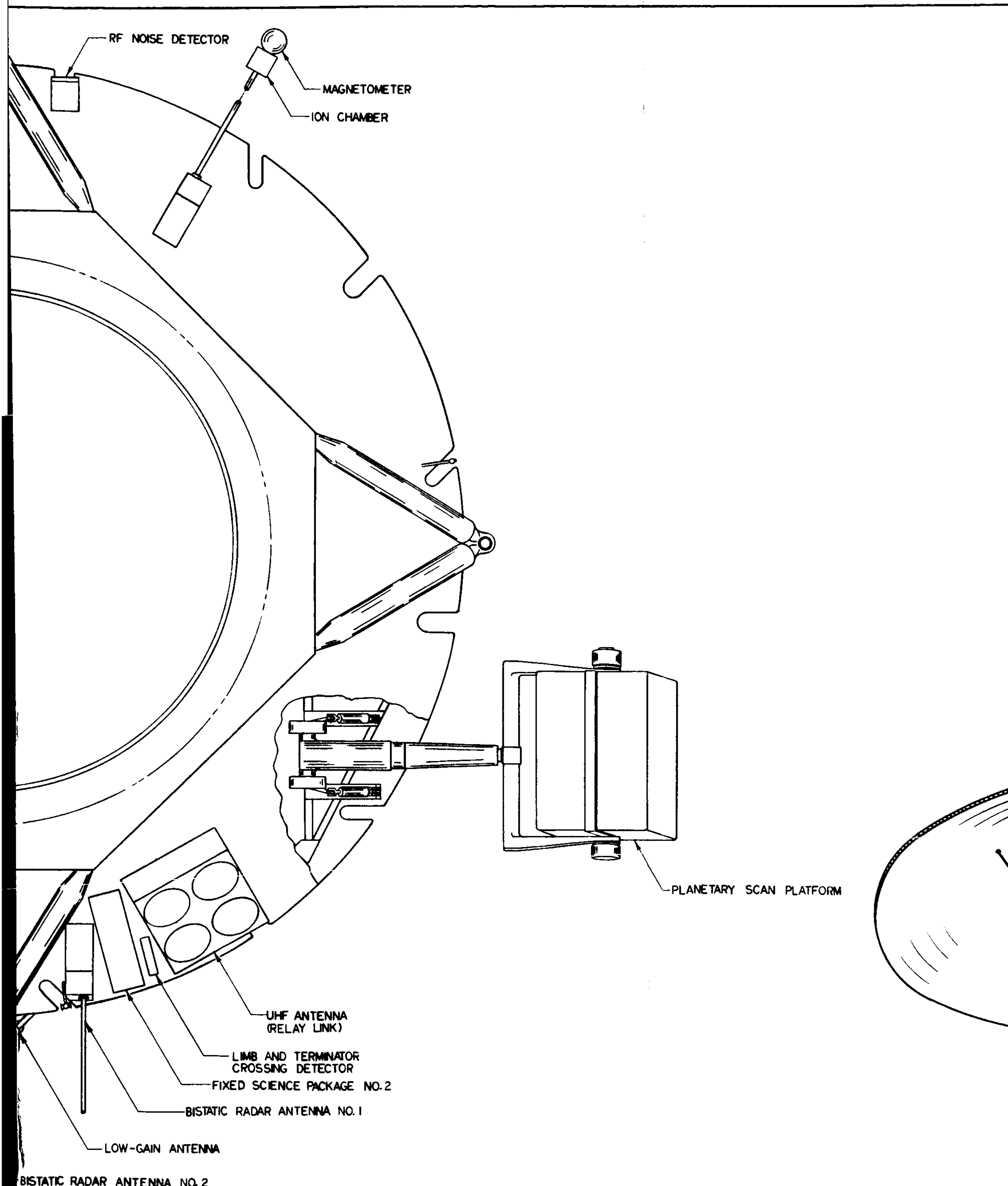
The selected configuration for the Voyager flight spacecraft, as shown in the referenced figures, utilizes a modified lunar excursion module (LEM) descent stage to provide the propulsion system for interplanetary trajectory corrections, orbit injection, and orbit trim maneuvers. This module is combined with bus structural elements to achieve the integration of the various spacecraft subsystems into a unified system.

The basic LEM descent stage structure is used with minor modifications. It consists of two pairs of transverse beams arranged in a cruciform together with forward and aft bulkhead closures. As shown in Figure 15, the four main outboard compartments contain the oxidizer and fuel tanks of the existing LEM module. The central compartment serves to redistribute the LEM descent engine thrust and actuator loads, which are introduced into the transverse beams at the mid and aft frames. Octagonal symmetry is completed by the addition of four panels, which form the corner prismatic compartments. The corner compartment faced with panel I provides the enclosure and support for the single 40.9-inch-diameter 6Al-4VA titanium alloy pressure vessel, which contains the helium used for pressurizing the propellant tanks. The diagonally opposite compartment serves as an environmental closure for the redundant pressure vessels of the reaction control subsystem. The two remaining compartments contain the majority of the spacecraft electronic equipment, as will be discussed in 3.2.5.

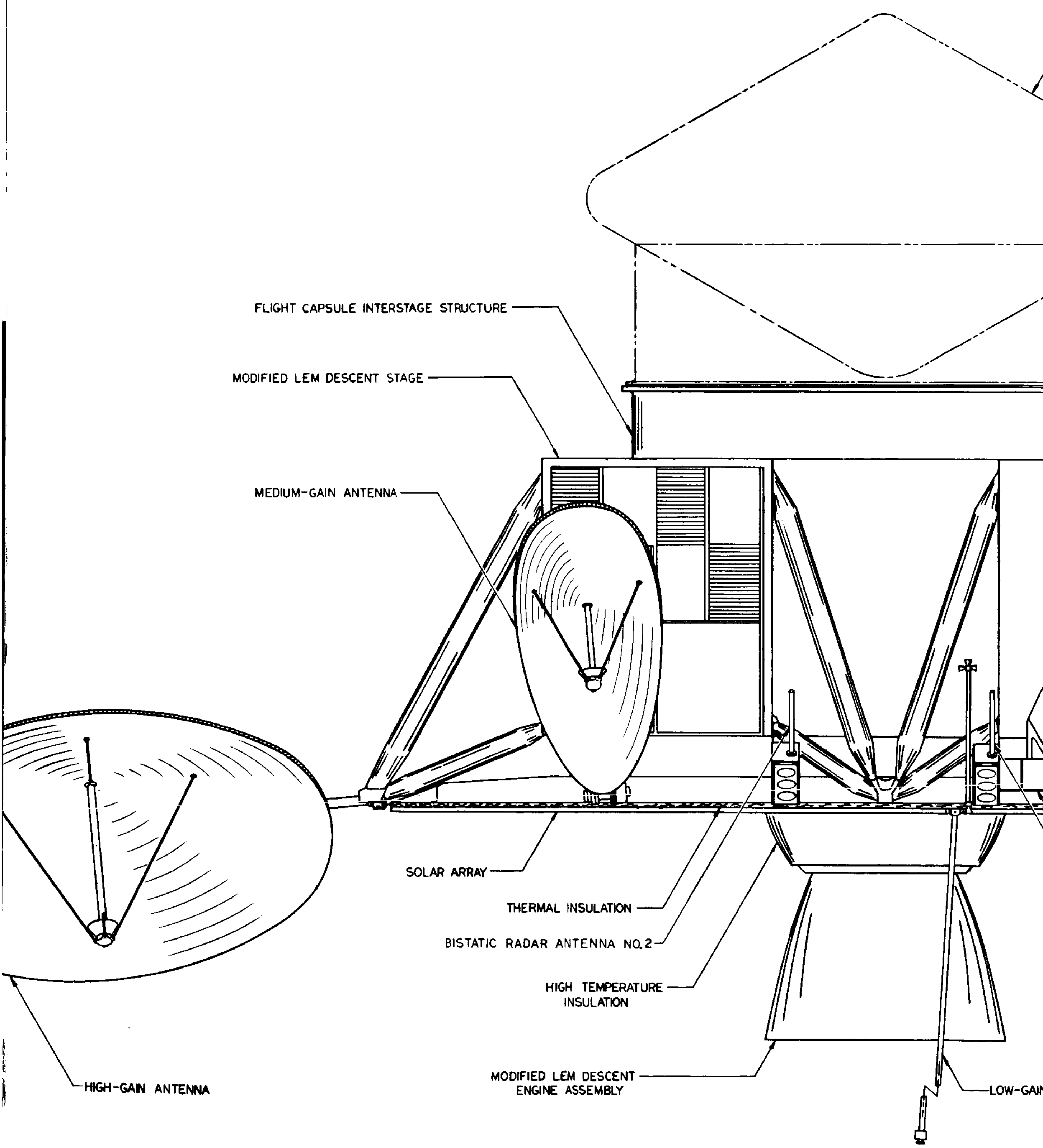
The LEM module structure serves also as the support for the flight capsule. The capsule interstage, which is a 10-foot-diameter semi-monocoque aluminum cylinder, distributes the capsule inertia loads into eight machined fittings which are attached to the caps of the transverse beams mentioned previously. This interface is referred to as the capsule emergency separation joint and includes provisions for structural continuity as well as the required mechanical and electrical separation system mechanisms.

The flight capsule inertia loads, in addition to spacecraft inertia loads, are sheared into the transverse beams and are carried outboard to the four outriggers, which have been specifically tailored for the Voyager spacecraft. Each outrigger consists of four aluminum truss members which extend from the four outer corners of the cruciform structure and converge to a single pad at the interface with the planetary vehicle adapter. This adapter consists of four short machined aluminum fittings and provisions to accept the separation mechanisms and instrumentation. These fittings transmit the planetary vehicle loads efficiently into the nose fairing at the nose fairing field joint. The axial load is introduced into four main longerons, which are integral with the fairing, and is uniformly distributed in shear to the reinforced, shear resistance


IONOSPHERE SOUNDER
EARTH DETECTOR
FIXED SCIENCE PACKAGE NO.1
REACTION CONTROL THRUSTER (12)
HIGH-GAIN ANTENNA
INSULATION
MEDIUM-GAIN ANTENNA


RF NOISE DETECTOR
MAGNETOMETER
ION CHAMBER
PLANETARY SCAN PLATFORM
UHF ANTENNA (RELAY LINK)
LIMB AND TERMINATOR CROSSING DETECTOR
FIXED SCIENCE PACKAGE NO. 2
BISTATIC RADAR ANTENNA NO. 1
LOW-GAIN ANTENNA
BISTATIC RADAR ANTENNA NO. 2

2


FLIGHT CAPSULE INTERSTAGE STRUCTURE
MODIFIED LEM DESCENT STAGE
MEDIUM-GAIN ANTENNA
SOLAR ARRAY
THERMAL INSULATION
BISTATIC RADAR ANTENNA NO. 2
HIGH TEMPERATURE INSULATION
MODIFIED LEM DESCENT ENGINE ASSEMBLY
HIGH-GAIN ANTENNA

3

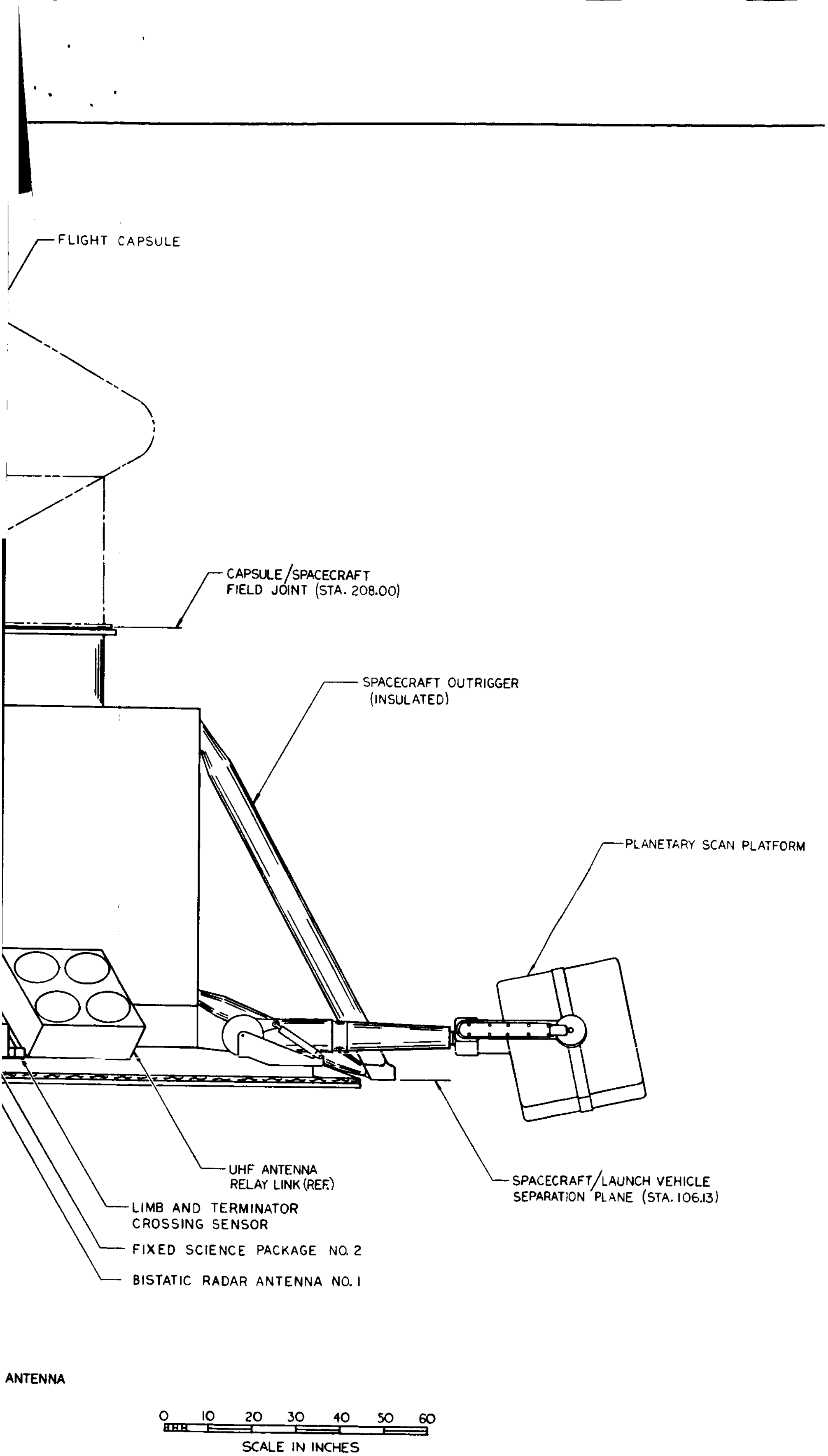
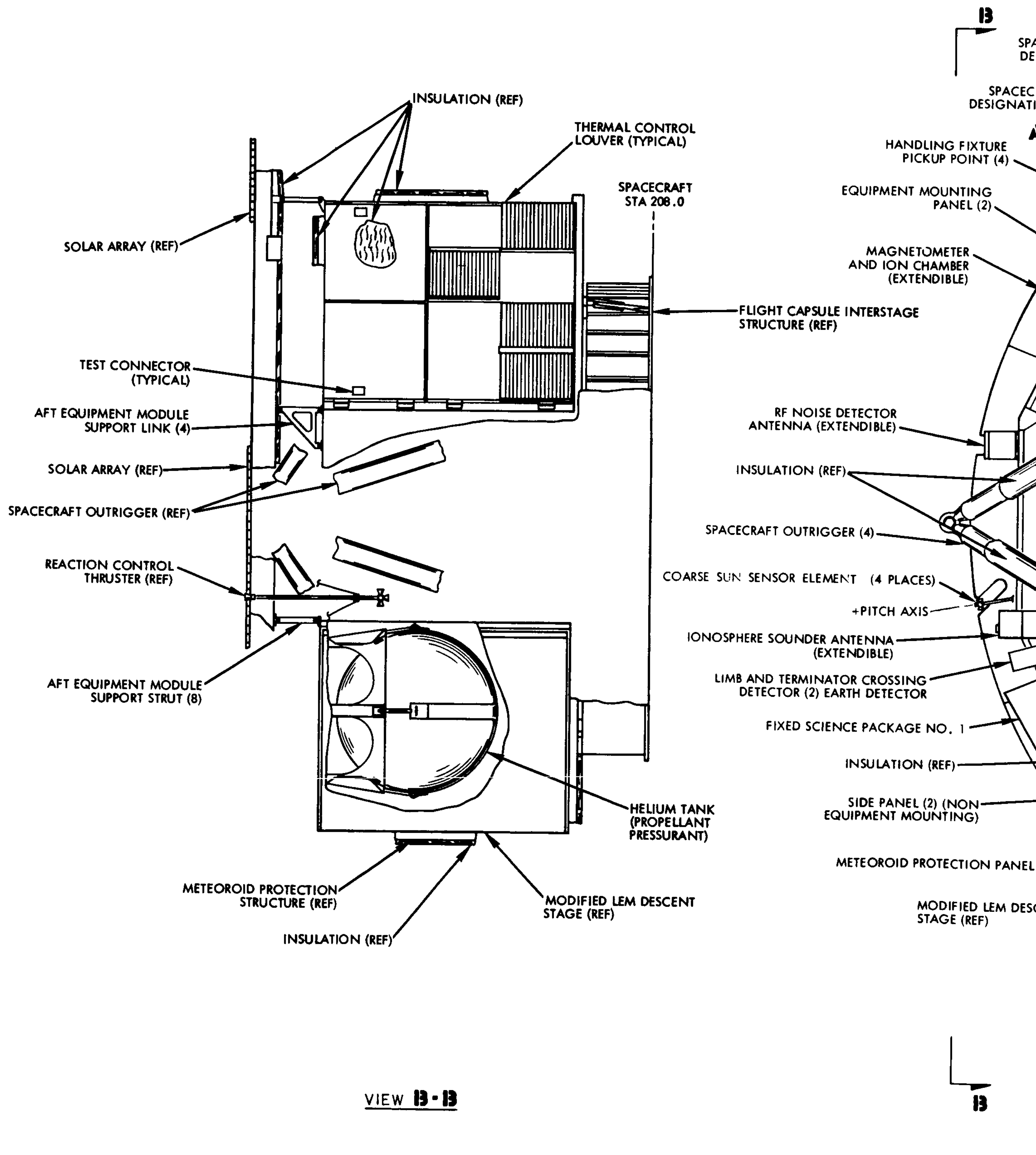
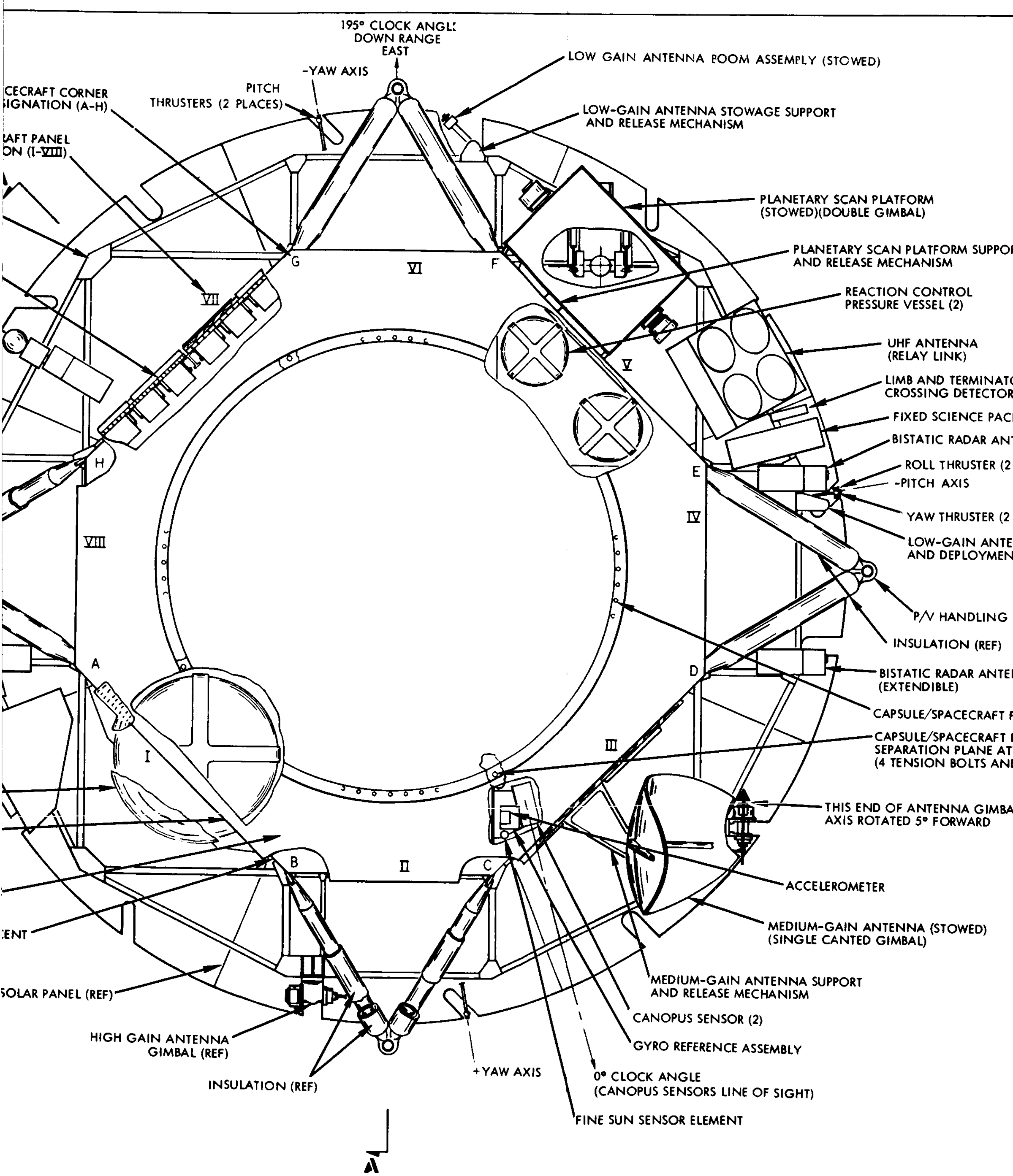


Figure 14. 1971 Voyager Spacecraft—
Outboard Profile

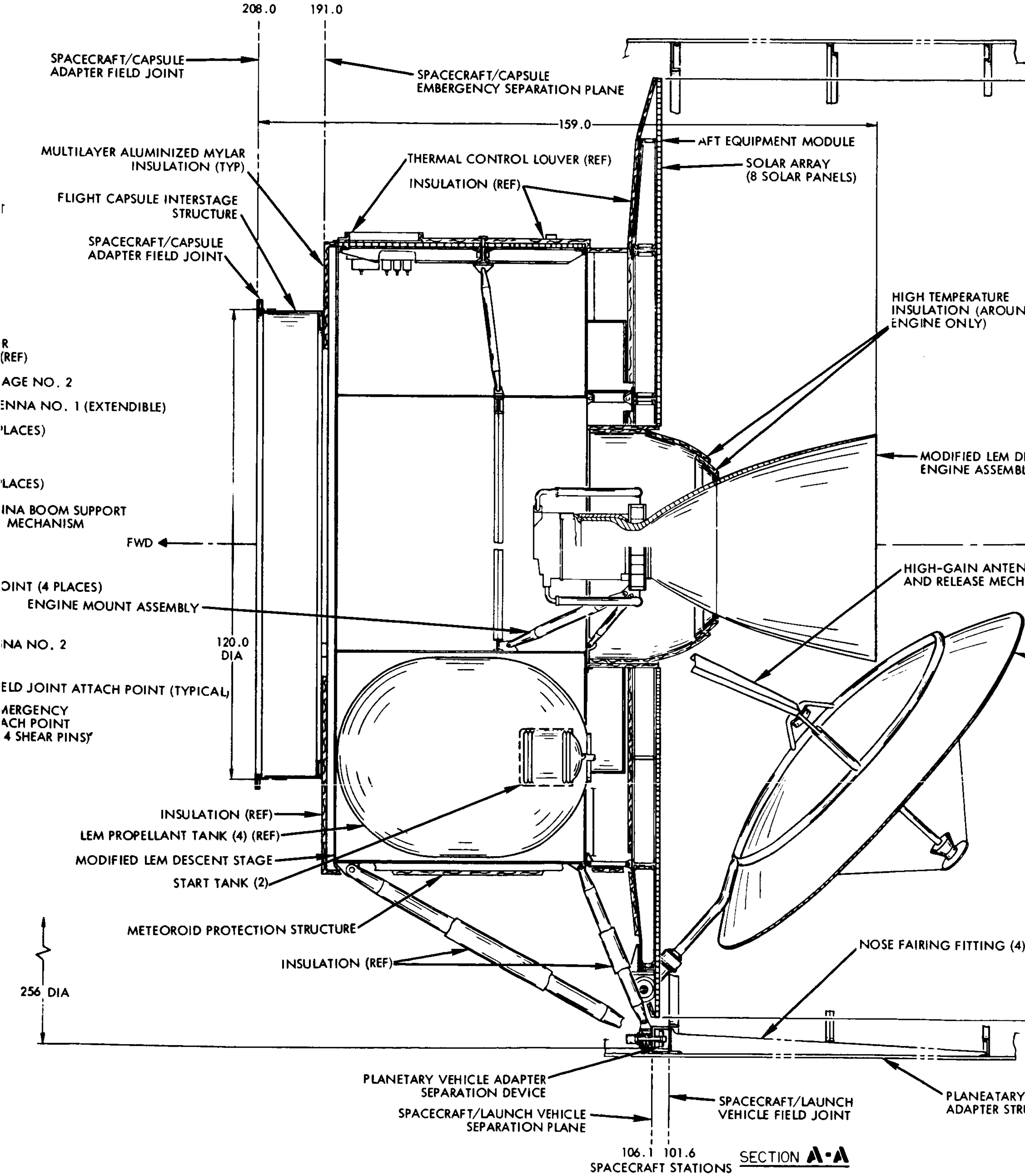

B
SPA
DE
SPACEC
DESIGNATI
INSULATION (REF)
THERMAL CONTROL LOUVER (TYPICAL)
SPACECRAFT STA 208.0
HANDLING FIXTURE PICKUP POINT (4)
EQUIPMENT MOUNTING PANEL (2)
MAGNETOMETER AND ION CHAMBER (EXTENDIBLE)
SOLAR ARRAY (REF)
FLIGHT CAPSULE INTERSTAGE STRUCTURE (REF)
TEST CONNECTOR (TYPICAL)
AFT EQUIPMENT MODULE SUPPORT LINK (4)
RF NOISE DETECTOR ANTENNA (EXTENDIBLE)
SOLAR ARRAY (REF)
INSULATION (REF)
SPACECRAFT OUTRIGGER (REF)
SPACECRAFT OUTRIGGER (4)
REACTION CONTROL THRUSTER (REF)
COARSE SUN SENSOR ELEMENT (4 PLACES)
+PITCH AXIS
IONOSPHERE SOUNDER ANTENNA (EXTENDIBLE)
AFT EQUIPMENT MODULE SUPPORT STRUT (8)
LIMB AND TERMINATOR CROSSING DETECTOR (2) EARTH DETECTOR
FIXED SCIENCE PACKAGE NO. 1
INSULATION (REF)
HELIUM TANK (PROPELLANT PRESSURANT)
SIDE PANEL (2) (NON EQUIPMENT MOUNTING)
METEOROID PROTECTION PANEL
METEOROID PROTECTION STRUCTURE (REF)
MODIFIED LEM DESCENT STAGE (REF)
MODIFIED LEM DESC STAGE (REF)
INSULATION (REF)
B

VIEW B-B



195° CLOCK ANGLE
DOWN RANGE
EAST
-YAW AXIS
LOW GAIN ANTENNA BOOM ASSEMBLY (STOWED)
PITCH
THRUSTERS (2 PLACES)
LOW-GAIN ANTENNA STOWAGE SUPPORT
AND RELEASE MECHANISM
PLANETARY SCAN PLATFORM
(STOWED)(DOUBLE GIMBAL)
REACTION CONTROL
PRESSURE VESSEL (2)
UHF ANTENNA
(RELAY LINK)
-PITCH AXIS
INSULATION (REF)
ACCELEROMETER
MEDIUM-GAIN ANTENNA (STOWED)
(SINGLE CANTED GIMBAL)
MEDIUM-GAIN ANTENNA SUPPORT
AND RELEASE MECHANISM
CANOPUS SENSOR (2)
GYRO REFERENCE ASSEMBLY
0° CLOCK ANGLE
(CANOPUS SENSORS LINE OF SIGHT)
FINE SUN SENSOR ELEMENT
+YAW AXIS
INSULATION (REF)
HIGH GAIN ANTENNA
GIMBAL (REF)
SOLAR PANEL (REF)
THIS END OF ANTENNA GIMBAL
AXIS ROTATED 5° FORWARD
BISTATIC RADAR ANTENNA
(EXTENDIBLE)
A
B
C
D
E
F
G
H
I
II
III
IV
V
VI
VII
VIII

2


208.0
191.0
SPACECRAFT/CAPSULE ADAPTER FIELD JOINT
SPACECRAFT/CAPSULE EMBERGENCY SEPARATION PLANE
159.0
MULTILAYER ALUMINIZED MYLAR INSULATION (TYP)
THERMAL CONTROL LOUVER (REF)
AFT EQUIPMENT MODULE
SOLAR ARRAY (8 SOLAR PANELS)
INSULATION (REF)
FLIGHT CAPSULE INTERSTAGE STRUCTURE
SPACECRAFT/CAPSULE ADAPTER FIELD JOINT
HIGH TEMPERATURE INSULATION (AROUN
ENGINE ONLY)
R
(REF)
AGE NO. 2
ENNA NO. 1 (EXTENDIBLE)
LACES)
LACES)
MODIFIED LEM D
ENGINE ASSEMB
INA BOOM SUPPORT MECHANISM
FWD
OINT (4 PLACES)
HIGH-GAIN ANTEN
AND RELEASE MECH
ENGINE MOUNT ASSEMBLY
INA NO. 2
120.0 DIA
ELD JOINT ATTACH POINT (TYPICAL)
MERGENCY
ACH POINT
4 SHEAR PINS)
INSULATION (REF)
LEM PROPELLANT TANK (4) (REF)
MODIFIED LEM DESCENT STAGE
START TANK (2)
METEOROID PROTECTION STRUCTURE
INSULATION (REF)
NOSE FAIRING FITTING (4)
256 DIA
PLANETARY VEHICLE ADAPTER SEPARATION DEVICE
SPACECRAFT/LAUNCH VEHICLE SEPARATION PLANE
SPACECRAFT/LAUNCH VEHICLE FIELD JOINT
PLANEATARY
ADAPTER STR
106.1 101.6
SPACECRAFT STATIONS
SECTION A-A

3

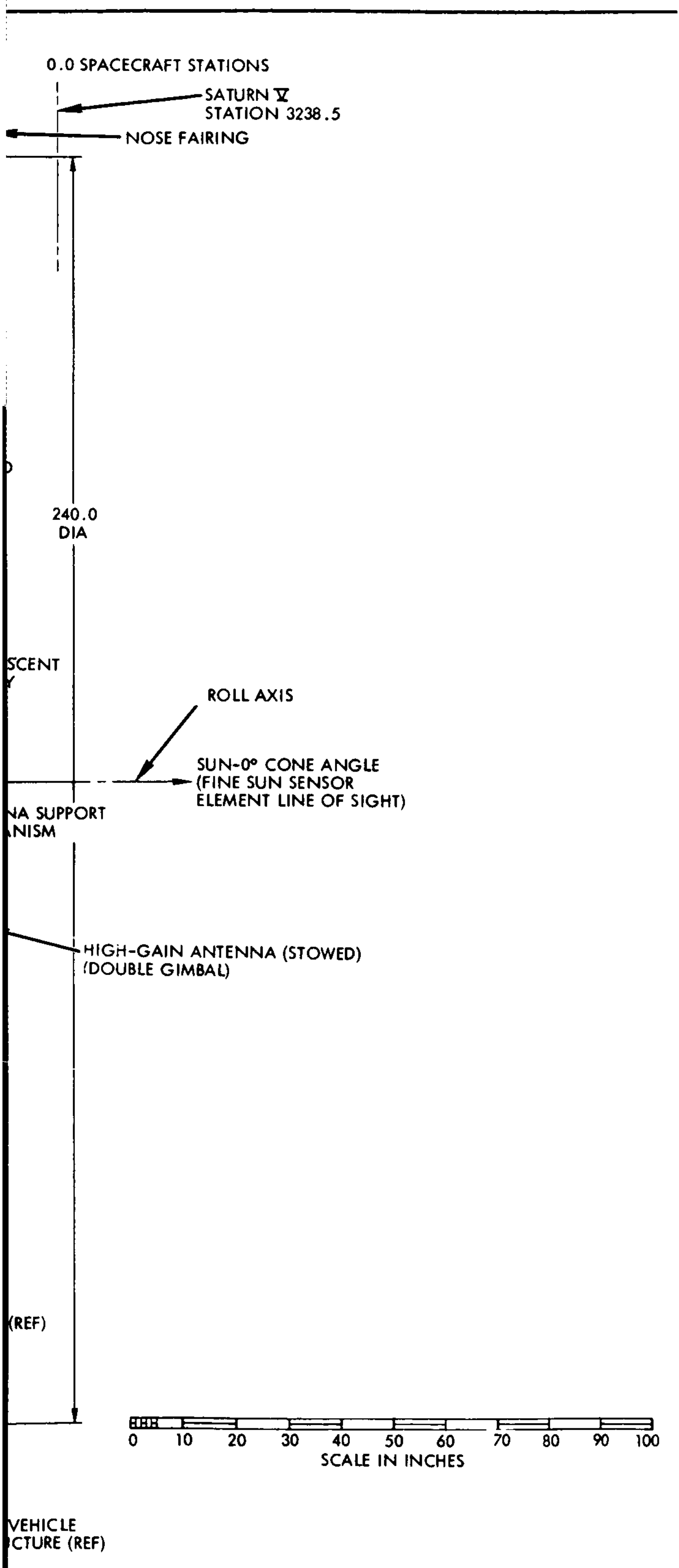
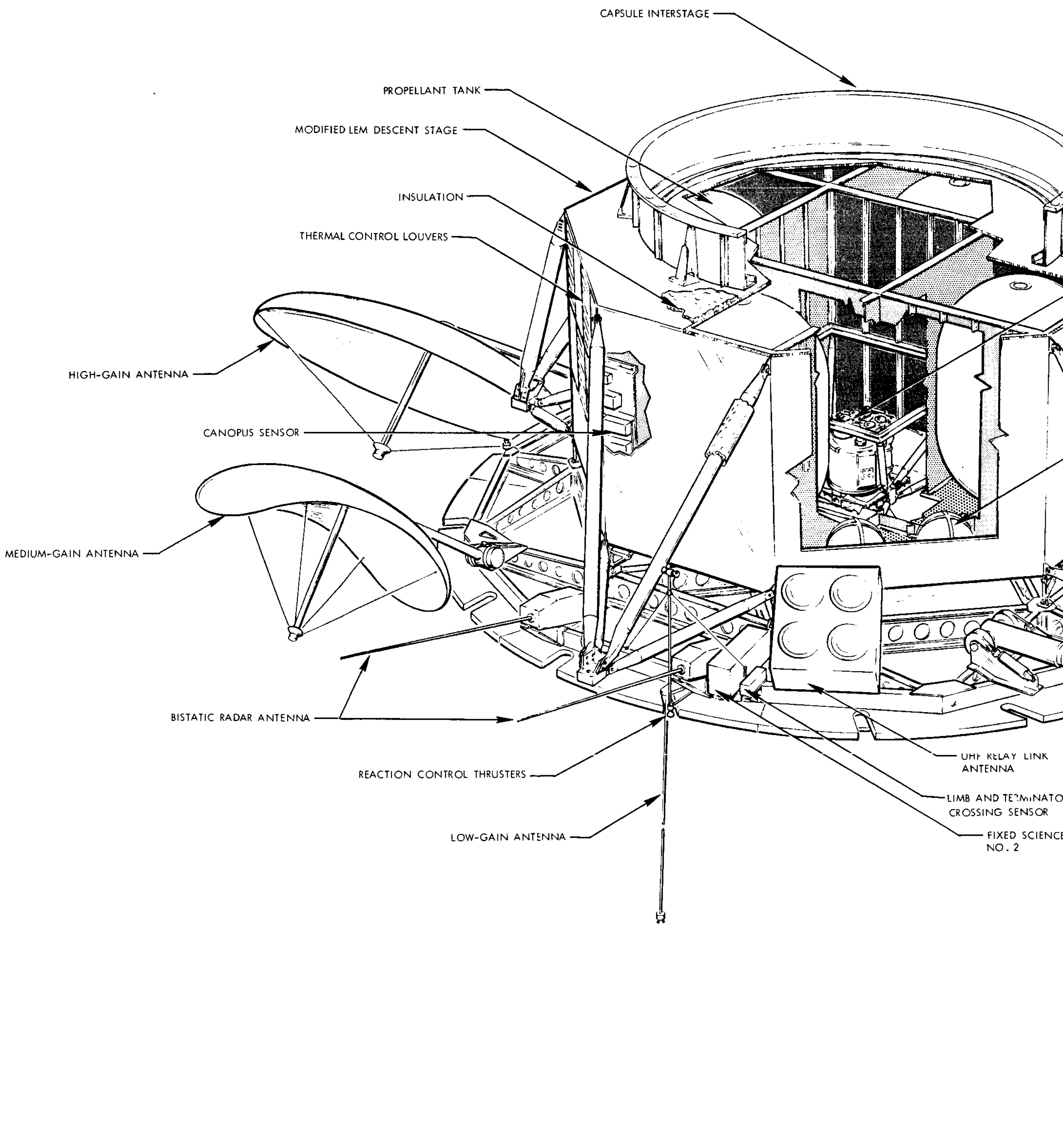
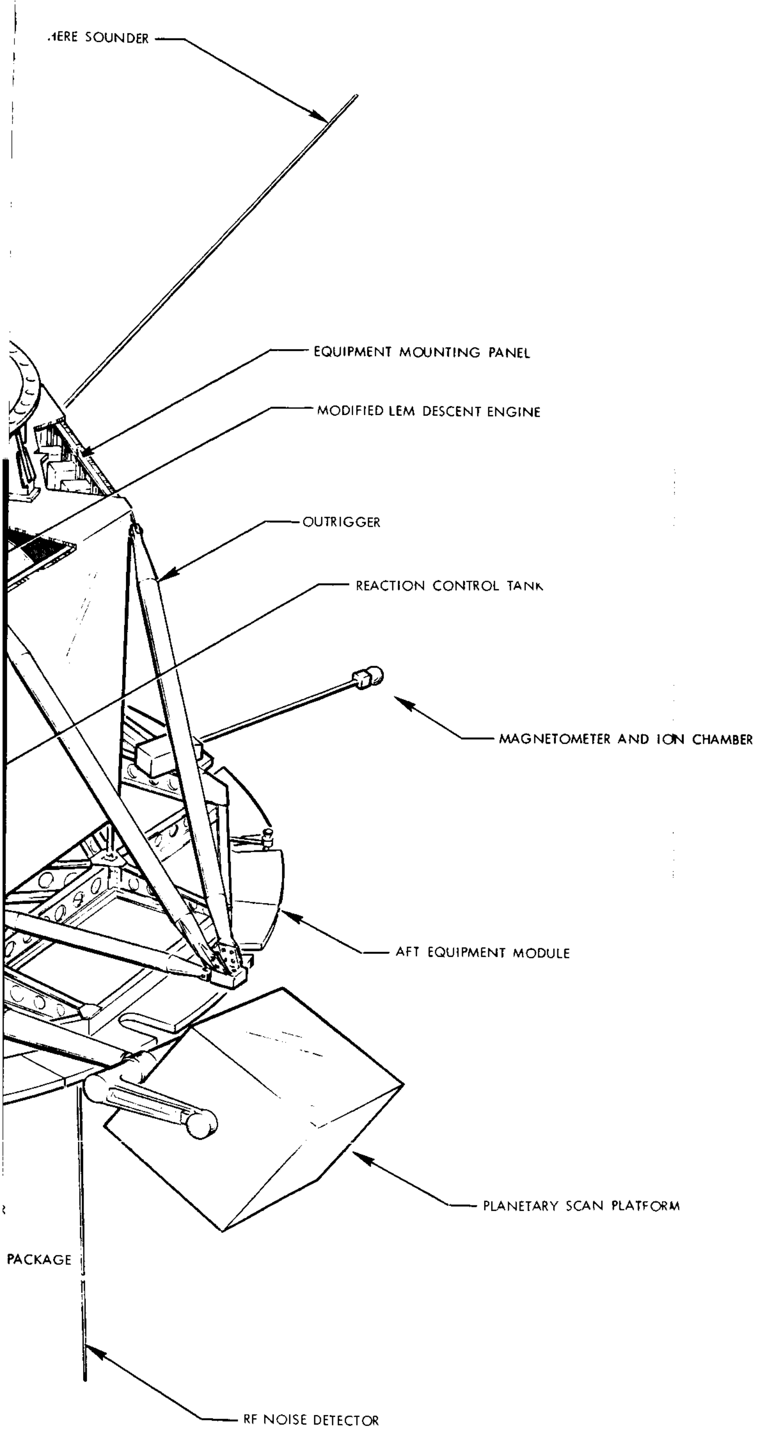


Figure 15. 1971 Voyager Spacecraft—Inboard Profile


CAPSULE INTERSTAGE
PROPELLANT TANK
MODIFIED LEM DESCENT STAGE
INSULATION
THERMAL CONTROL LOUVERS
HIGH-GAIN ANTENNA
CANOPUS SENSOR
MEDIUM-GAIN ANTENNA
BISTATIC RADAR ANTENNA
REACTION CONTROL THRUSTERS
LOW-GAIN ANTENNA
UHF RELAY LINK ANTENNA
CROSSING SENSOR
NO. 2



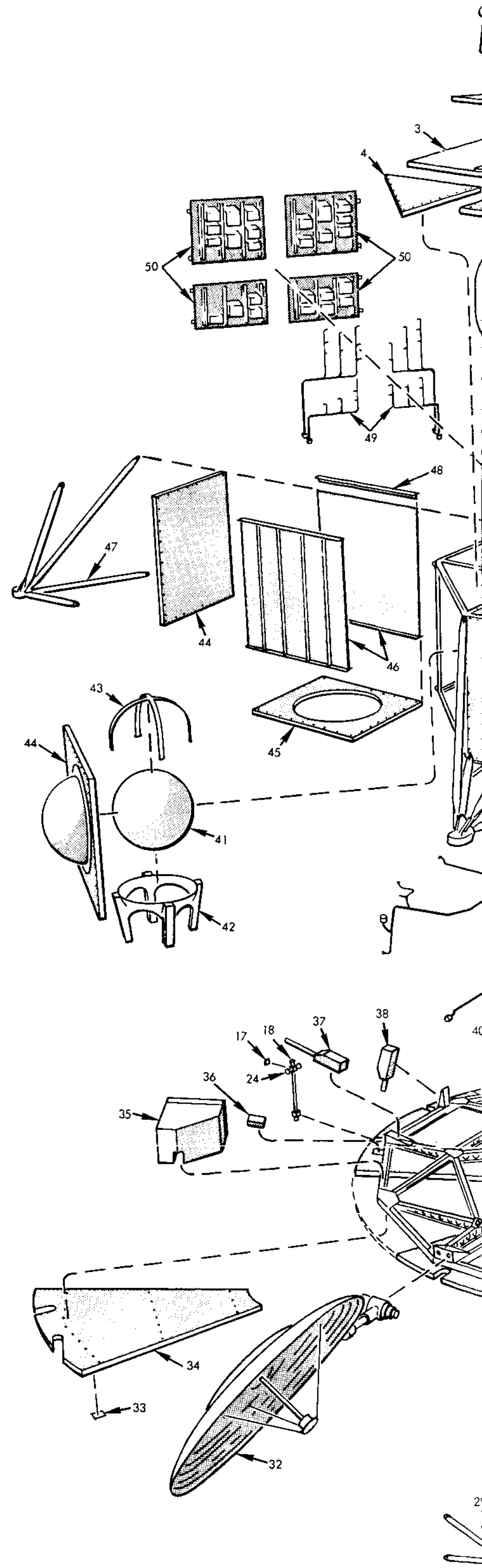
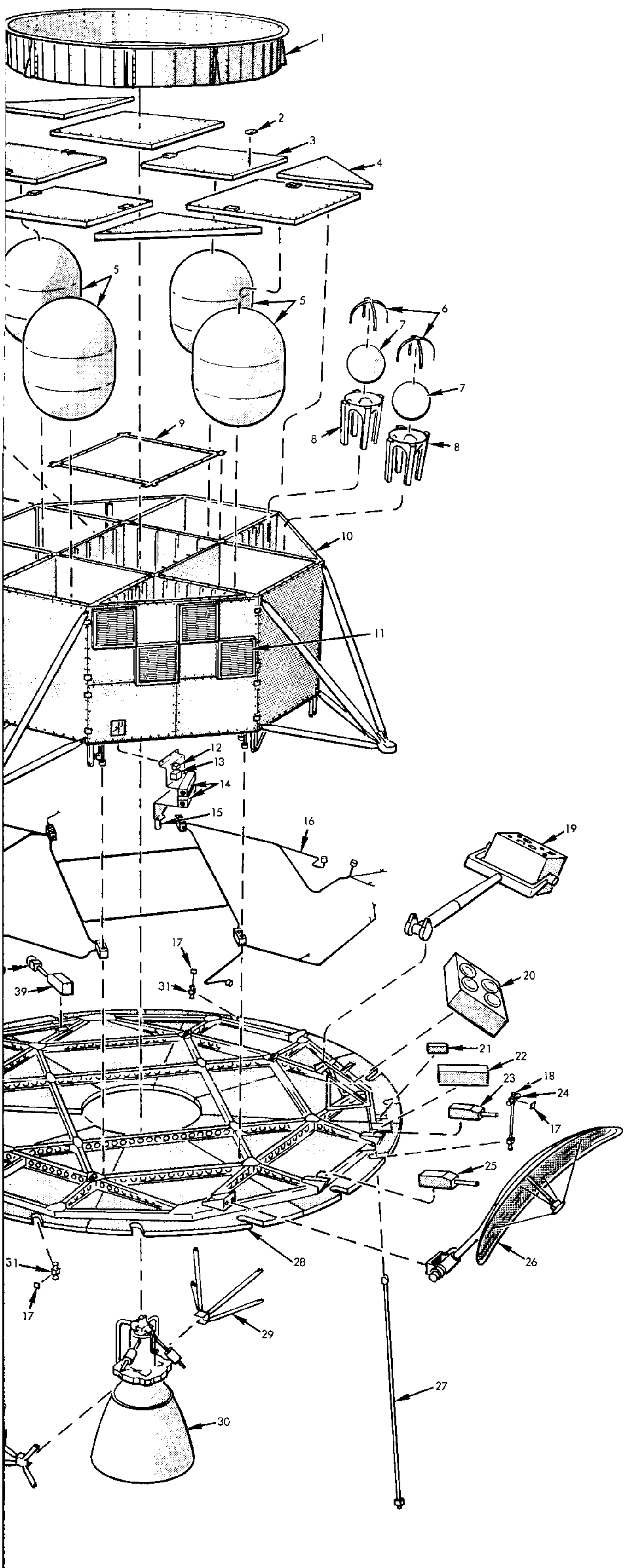


Figure 16. 1971 Voyager Spacecraft—In-Flight Configuration Isometric Drawing

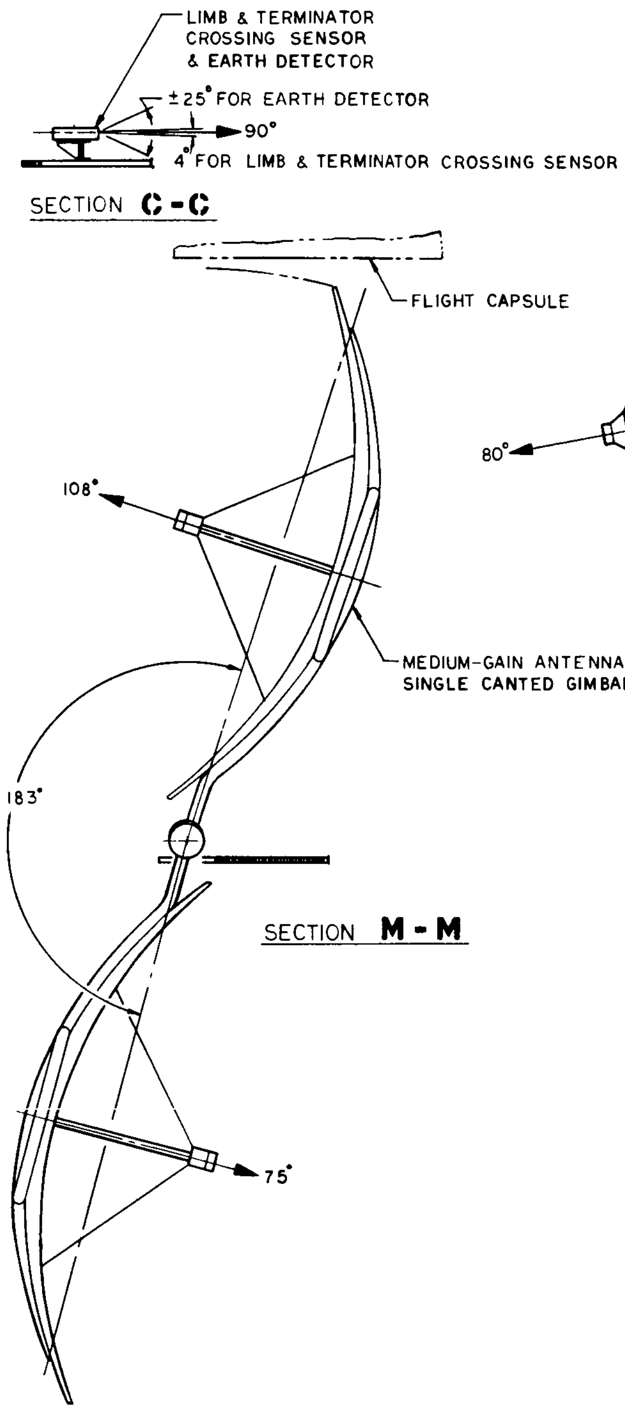
2

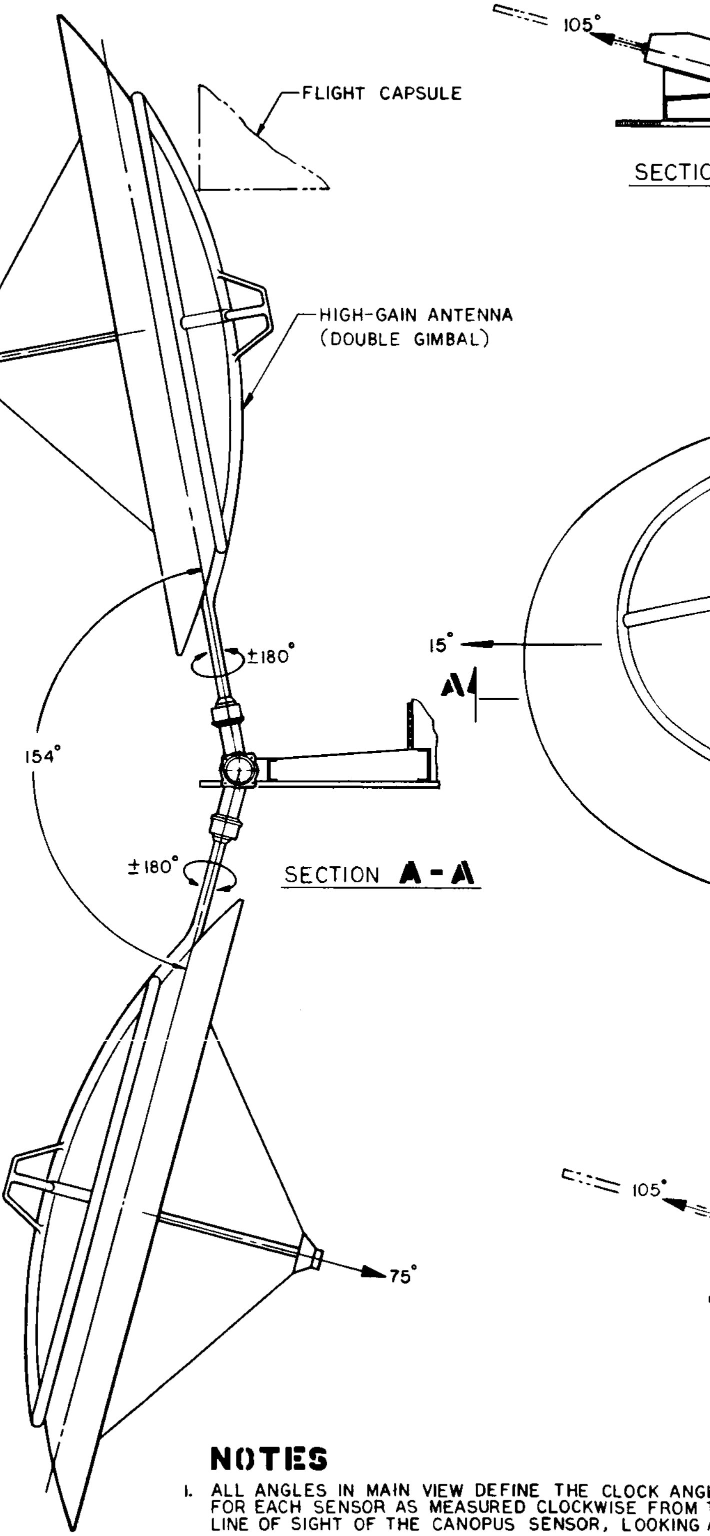


1	FLIGHT CAPSULE INTERSTAGE STRUCTURE
2	CAPSULE INTERSTAGE SUPPORT FITTING (8)
3	FORWARD MICROMETEOROID PROTECTION PANEL (5)
4	FORWARD CORNER PANEL (4)
5	LEM PROPELLANT TANK (4)
6	PRESSURE VESSEL SUPPORT STRAP (2)
7	REACTION CONTROL PRESSURE VESSEL (2)
8	PRESSURE VESSEL SUPPORT (2)
9	BEAM CAP ASSEMBLY
10	MODIFIED LEM PRIMARY STRUCTURE
11	THERMAL CONTROL LOUVER BANKS (8)
12	ACCELEROMETER
13	GYRO REFERENCE ASSEMBLY
14	CANOPUS SENSOR (2)
15	FINE SUN SENSOR
16	ELECTRICAL CABLING AND JUNCTION BOXES
17	COARSE SUN SENSOR ELEMENT (4)
18	YAW THRUSTERS
19	PLANETARY SCAN PLATFORM
20	UHF ANTENNA (RELAY LINK)
21	LIMB AND TERMINATOR CROSSING DETECTOR
22	FIXED SCIENCE PACKAGE NO. 2
23	BISTATIC RADAR ANTENNA NO. 1
24	ROLL THRUSTERS
25	BISTATIC RADAR ANTENNA NO. 2
26	MEDIUM-GAIN ANTENNA AND DRIVE ASSEMBLY
27	LOW-GAIN ANTENNA BOOM ASSEMBLY
28	AFT EQUIPMENT MODULE
29	ENGINE MOUNT ASSEMBLY (2)
30	MODIFIED LEM DESCENT ENGINE
31	PITCH THRUSTERS
32	HIGH GAIN ANTENNA AND DRIVE ASSEMBLY
33	SOLAR CELL MODULE
34	SOLAR ARRAY PANEL (8)
35	FIXED SCIENCE PACKAGE NO. 1
36	LIMB AND TERMINATOR CROSSING DETECTOR AND EARTH DETECTOR
37	IONOSPHERE SOUNDER ANTENNA
38	RF NOISE DETECTOR ANTENNA
39	MAGNETOMETER BOOM ASSEMBLY
40	MAGNETOMETER AND ION CHAMBER SENSORS
41	PROPELLANT PRESSURANT VESSEL
42	PRESSURE VESSEL SUPPORT STRUCTURE
43	PRESSURE VESSEL SUPPORT STRAP
44	MICROMETEORITE PROTECTION SIDE PANEL
45	LOWER DECK PANEL (4)
46	CRUCIFORM BEAM WEBS
47	SPACECRAFT OUTRIGGER (4)
48	BEAM CAP
49	EQUIPMENT PANEL ELECTRICAL CABLING
50	EQUIPMENT MOUNTING PANELS

Figure 17. 1971 Voyager Spacecraft—Exploded View

3



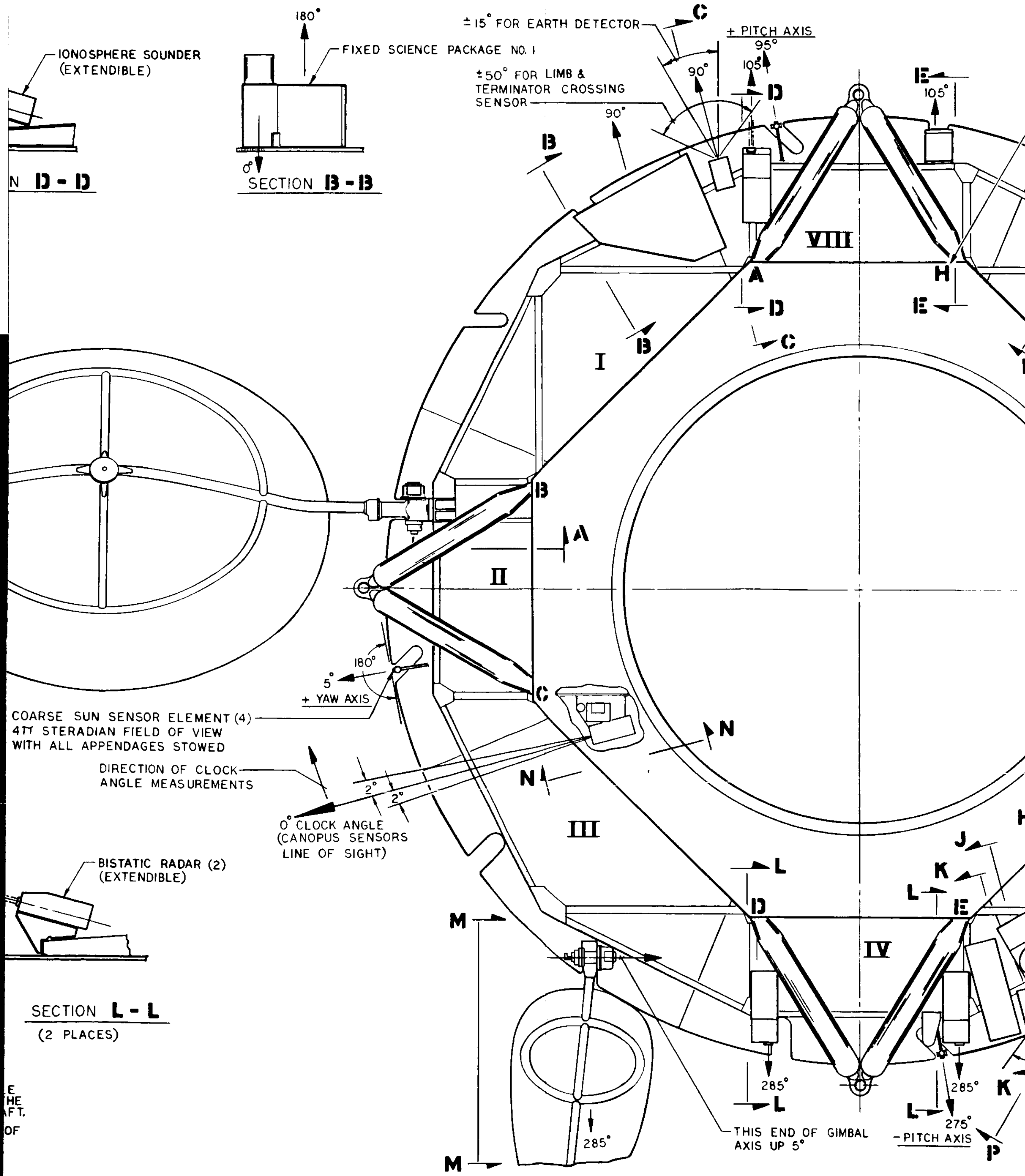


NOTES

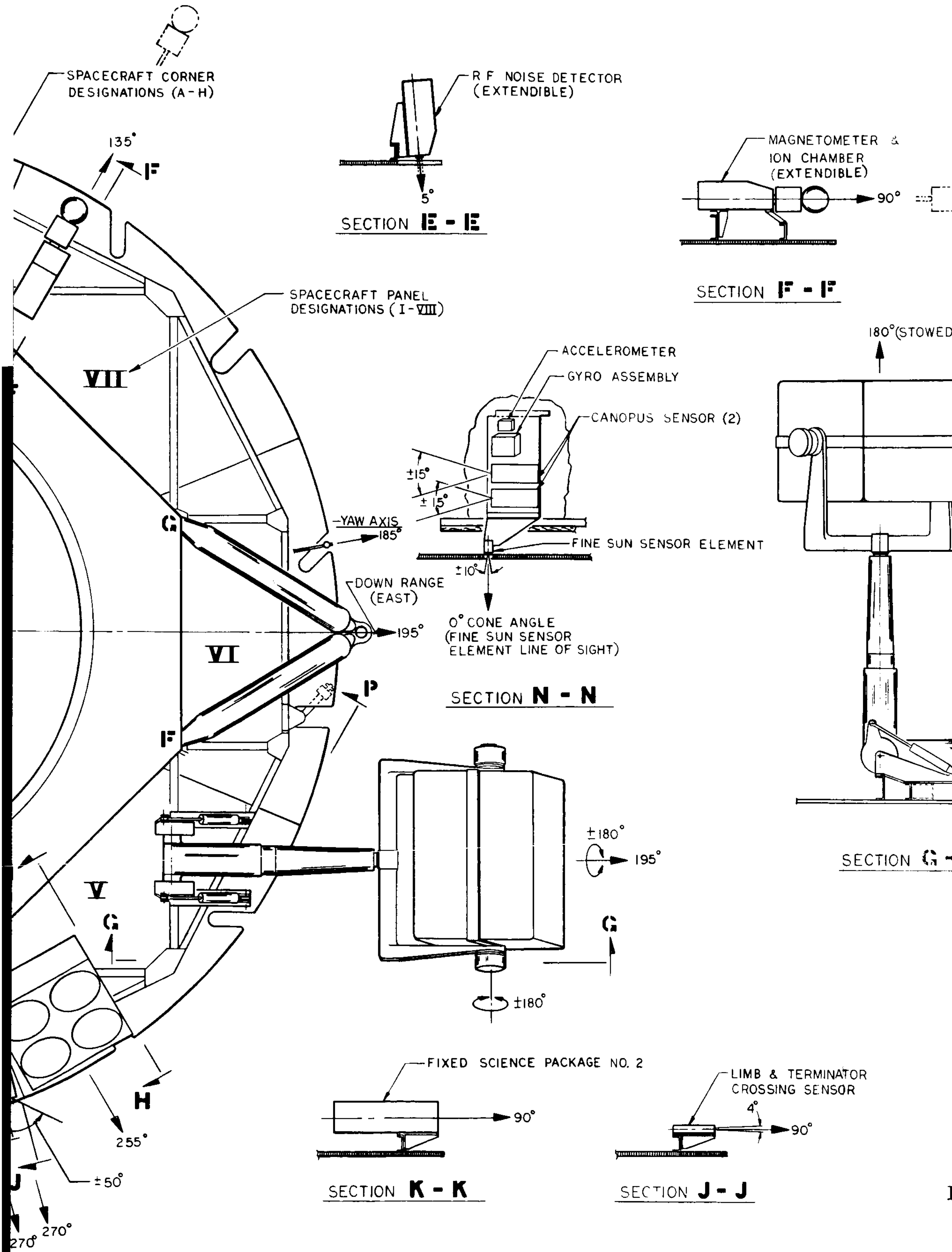
1. ALL ANGLES SHOWN IN ALL SECTIONS DEFINE THE CONE ANGLE AXIS FOR EACH SENSOR.
2. 0° CONE ANGLE IS DEFINED AS A LINE THRU THE SPACECPAFT POINTING AFT— AT THE SUN.
3. SECTIONS SHOWN HAVE BEEN ROTATED FOR CLARITY.

NOTES

1. ALL ANGLES IN MAIN VIEW DEFINE THE CLOCK ANG FOR EACH SENSOR AS MEASURED CLOCKWISE FROM LINE OF SIGHT OF THE CANOPUS SENSOR, LOOKING
2. 0° CLOCK ANGLE IS DEFINED AS THE LINE OF SIGHT THE CANOPUS SENSOR.


IONOSPHERE SOUNDER
(EXTENDIBLE)
N D - D
180°
FIXED SCIENCE PACKAGE NO. 1
SECTION B - B
±15° FOR EARTH DETECTOR
C
+ PITCH AXIS
95°
105°
90°
±50° FOR LIMB & TERMINATOR CROSSING SENSOR
90°
D
E
105°
B
VIII
A
H
D
E
C
B
I
B
A
II
C
180°
5°
+ YAW AXIS
COARSE SUN SENSOR ELEMENT (4)
4π STERADIAN FIELD OF VIEW
WITH ALL APPENDAGES STOWED
DIRECTION OF CLOCK ANGLE MEASUREMENTS
N
N
2°
2°
0° CLOCK ANGLE
(CANOPUS SENSORS LINE OF SIGHT)
III
J
K
BISTATIC RADAR (2)
(EXTENDIBLE)
L
L
D
E
M
IV
SECTION L - L
(2 PLACES)
285°
L
285°
K
L
275°
- PITCH AXIS
P
285°
M
THIS END OF GIMBAL AXIS UP 5°
E
HE
AFT.
OF

2



SPACECRAFT CORNER DESIGNATIONS (A-H)
135°
F
SPACECRAFT PANEL DESIGNATIONS (I-VIII)
VII
VI
V
G
F
YAW AXIS
185°
DOWN RANGE (EAST)
195°
P
G
H
255°
±50°
J
270°
270°
R F NOISE DETECTOR (EXTENDIBLE)
5°
SECTION E - E
MAGNETOMETER & ION CHAMBER (EXTENDIBLE)
90°
SECTION F - F
ACCELEROMETER
GYRO ASSEMBLY
CANOPUS SENSOR (2)
±15°
±15°
FINE SUN SENSOR ELEMENT
±10°
0° CONE ANGLE (FINE SUN SENSOR ELEMENT LINE OF SIGHT)
SECTION N - N
180°(STOWED
±180°
195°
G
±180°
SECTION G
FIXED SCIENCE PACKAGE NO. 2
90°
SECTION K - K
LIMB & TERMINATOR CROSSING SENSOR
4°
90°
SECTION J - J

3

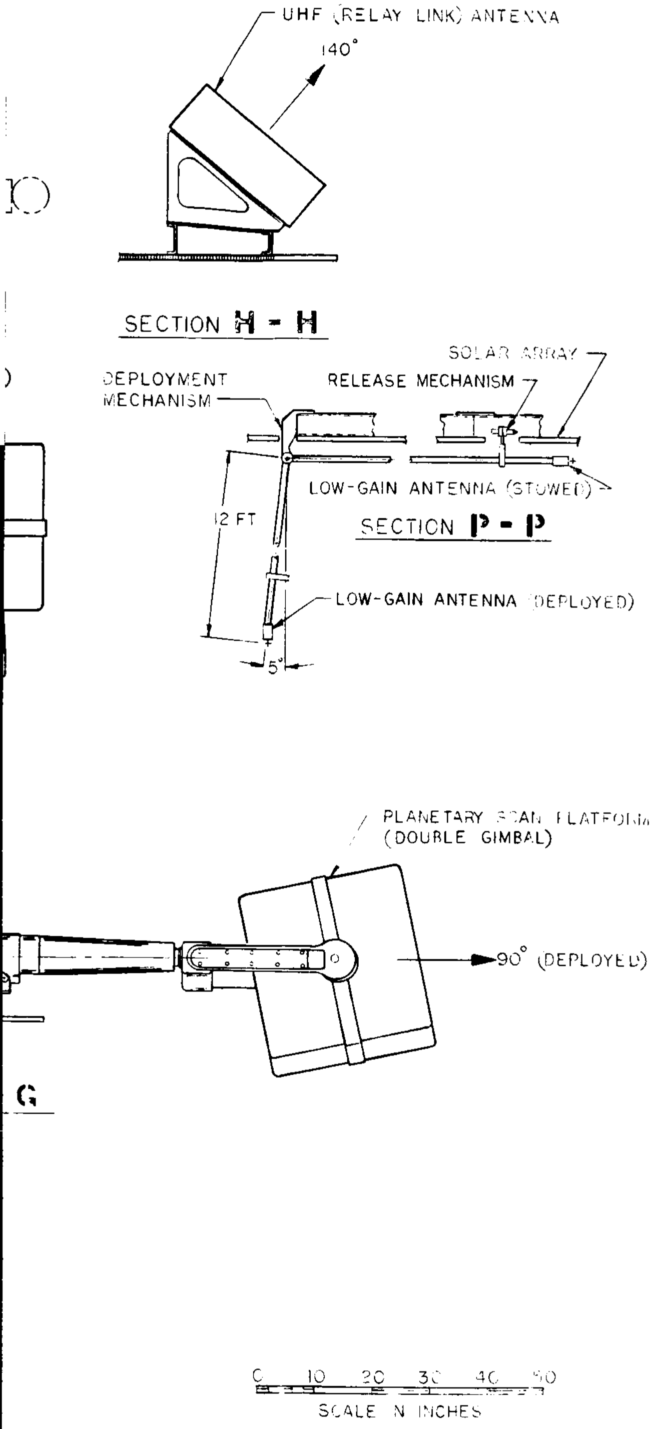


Figure 18. 1971 Voyager Spacecraft— Sensor and Appendages Geometry

4

panels of the fairing. Torsional and lateral loads are reacted by the main circumferential frame at the field joint located at the aft face of the adapter.

The efficient utilization of the fairing effects a substantial weight savings of approximately 1000 pounds, facilitates fairing separation, and permits the maximum utilization of the allowable dynamic envelope in the vicinity of the rigidized planetary vehicle adapter. Advantage is taken of this envelope to accommodate a fixed solar array support structure, which is referred to as the aft equipment module. Eight identical solar array panels are affixed to this module to form the sun-oriented face of the spacecraft and provide 290 square feet of area, which satisfies the power subsystem design goal.

As shown in Figure 15, the basic LEM structure is used also to support the aft equipment module mentioned in the preceding paragraph. Twelve structural pin-ended link assemblies interconnect the two structures and are arranged to provide a rigid load path for all axial, lateral and torsional loads. The interface loads are carried into an interlaced arrangement of 6-inch deep beams which comprise the frame of this stable module. The beam system geometry readily accommodates all spacecraft appendages, the electrical harness with interface junction boxes, the reaction control system, the relay link antenna, science equipment and the solar array. In addition, it features handling provisions for the AHSE and MOSE so that all operations and tests can be accomplished in a facile and expeditious manner.

The S-band antennas of the radio subsystem that provide the two-way spacecraft-to-earth link are shown in Figures 14 and 15 in the normal inflight and stowed attitudes, respectively. The gimballed high- and medium-gain antennas are oriented to permit optimum coverage for normal pointing conditions and maximum data transmission throughout the mission. The double-gimballed high-gain antenna is attached to the aft equipment module and is stowed during the boost phase in a position adjacent to the LEM engine nozzle, aft of the solar array. The single-gimballed medium-gain antenna is stowed forward of the equipment module and is retained adjacent to panel III. Since both antennas, after release, are driven across the plane of the array, local cutouts at the array periphery are provided for base gimbal clearance. The stowed configuration of these antennas establishes the axial envelope of the spacecraft bus.

The local cutouts, which are incorporated in each of the eight identical solar panels, facilitate the installation of the low-gain antenna and the reaction control jets. The low-gain antenna is hinged from and retained by the aft equipment module but is stowed beneath the solar array. Prior to nose fairing separation, the antenna provides telemetry coverage through a coupling aperture in the fairing, and thereby establishes the downrange orientation of the planetary vehicle when installed to the launch vehicle.

The symmetry in the configuration of cutouts provides the flexibility necessary for the mounting of the control nozzles on or near the principal axes of the system. Presently, the nozzles are located as shown in Figures 15 and 16 and establish the pitch and yaw axes. Two independent pneumatic systems are used, each feeding heated nitrogen to six nozzles, and provide for redundant coupled control about each major axis. The roll nozzles, which provide the control torques about the roll axis (vehicle thrust axis), are elevated approximately 3 feet above the aft equipment module at a compromised c.g. station so that in the roll control failure mode, a major unbalancing torque about the pitch axis would not be generated.

The boom-mounted planetary scan platform is stowed forward of the aft equipment module and retained adjacent to panel V. Subsequent to release, completely redundant linear actuators deploy and latch the appendage at a 90-degree cone angle. The yoke and shaft gimbals provide the capability to satisfy Mars visibility requirements.

A DeHavilland STEM (storable tubular extendible member) system, used to support and extend the magnetometer and ion chamber experiments, is mounted on the aft equipment module outboard of panel VII; however, should the STEM unit fail to meet alignment accuracies, a two-element insulated boom assembly can be employed. Such a boom assembly could be supported and retained by the aft equipment module in the space provided between the two outriggers adjacent to panel VII.

As shown in Figure 18, the lines of direction associated with sensors and appendages are given in terms of clock and cone angles, which are defined in 3.9.2 in Section III. The two redundant Canopus sensors, the accelerometer, the gyro reference assembly, and fine sun sensor are packaged together. This assembly is rigidly attached to the LEM cruciform structure to achieve the best possible alignment control relative to each other and to the engine mount. The viewport for the fine sun sensor is located at the division between adjacent solar array panels while the Canopus viewport is described as an elongated slot in panel III.

As mentioned previously, panels III and VII are the faces for the two, diagonally opposite, prismatic equipment compartments. The equipment attached to these panels is arranged in accordance with the subsystems matrix and is intended to satisfy thermal and mass properties requirements. As discussed in 3.3.5, panel III supports the guidance and control, command, computing and sequencing, S-band radio, telemetry and data storage subsystems equipment. Panel VII supports the power, pyrotechnic, relay link, and spacecraft science subsystems equipment.

Actually, each of the above panels is split into four panels, which are hinged along the outside edges at the corner longerons of the LEM cruciform structure. These equipment and thermal radiation panels are constructed of a stiffened sandwich consisting of 0.032 aluminum skins bonded to a truss grid core with auxiliary stiffening provided by 3-inch deep hat sections which serve also as the equipment mounting rails. Auxiliary support members are added to support the free edges of these panels.

Micrometeorite protection is provided for all pressurized units and sensitive electronic equipment within the spacecraft. Resistance to penetration is inherent in the structural aluminum sandwich panels of the equipment mounting doors and the solar array. All other exposed units are protected by a shield, constructed of 0.020-inch aluminum skins separated by a 2-inch thick low density core, which will effectively disperse the particles encountered.

To minimize the uncontrolled radiant energy interchange between the main compartment and solar array, an aluminized mylar insulation blanket is utilized. This blanket envelopes the exposed truss members, is installed on the back side of the solar array, and is tied to the external surface of all micrometeoroid shields. To actively regulate the radiant energy interchange between the main compartment and its environment, two bi-metal actuated louver banks are attached to each of the aforementioned upper equipment mounting doors, with the exception that one bank will be installed on the panel VII lower door that mounts some of the power subsystem equipment. The equipment mounting panels otherwise and all other irregular protrusions and seams are suitably insulated to minimize heat leaks.

3.2 Design Considerations

The synthesis of the selected spacecraft configuration described above has been developed on the basis of a logical sequence of design decisions. Several of the major underlying design considerations are identified and discussed below.

3.2.1 Solar Array Design

The selection of a solar array concept logically precedes the other design decision areas discussed in this section. It involves a tradeoff between fixed and deployable arrays, the adequacy of a fixed array for the required power profile, survival at the end of the maximum Martian elipse period, and growth potential.

Factors which favor the fixed solar array concept include: simplicity of the resulting structure, with its attendant reliability; ease with which appendages and science equipment can be mounted on the back side of the supporting equipment module; resulting unobstructed sensor fields of view in the launch-ready and inflight configurations; and lower boost phase design limit loads. The concomitant opposing factors include the less efficient solar cell arrangement which utilizes an annulus rather than a deployable rectangular panel, and the limited growth potential. However, since there appears to be a straightforward solution to the cell packing problem and the available area satisfies power requirements for all planned missions, the fixed solar array concept has been chosen as a basis for the subsequent steps in the design synthesis process.

The adequacy of the fixed array for the required power profile has been investigated with regard to available area, operating temperature, and type approval upper temperature limits. The power subsystem presentation which is given in Volume 2, indicates that the critical requirement of 719 watts prior to flight capsule separation can be satisfied with approximately 260 of the 290 square feet that is available with a fixed array. However, as discussed later, insulation has been added to the back side of the array to minimize the uncontrolled radiant energy interchange between the array and the main equipment compartment, and also to accommodate a longer Martian orbit solar eclipse. The insulation leads to higher solar array temperatures, which affect the cell efficiency (I-V characteristics). This results in a requirement for 284 square feet of cell area, but this is still within the 290 square feet available from the fixed array.

With the back side of the array insulated, the near-earth array temperature approaches 118°C for an absorptivity-emissivity ratio (α/ϵ) of 1.0. Actually, the maximum anticipated temperature is 116°C for an α/ϵ of 0.95, and a 98°C minimum anticipated temperature for an α/ϵ of 0.85. An operating temperature limit of 115°C had been established early in the design phase based on the premise that the upper temperature uncertainty is small for a measured α/ϵ, so that a type approval test requirement 25°C above 115°C is realistic. The recommended 40°C increase for type approval when applied to the 118°C appears to be beyond the state of the art. Perhaps future improvements in cell bonding techniques may raise the limit above 140°C; however, additional study is required to justify either the use of a lower safety margin or the higher type approval test limit.

To prevent excessively low solar array temperatures, which would occur at the end of the Martian orbit solar eclipse, insulation has been added to the back side of the aft equipment module. Again, a thermo-mechanical problem is identified. For an eclipse period of 2.3 hours, with an array specific weight of 1.0 pound per square foot as used in the study, a conservative calculation indicates that the array temperature approaches $-133^{\circ}C$, which is very nearly the state-of-the-art temperature limit. However, to minimize array weight, advantage has been taken of the thermal mass of the aft equipment module frame structure. Since the array modules are not in intimate contact with the aft equipment module, the frames, as well as the back side insulation, serve to control the rate of cooling. This results in an array temperature at the end of a 2.3-hour eclipse which is above $-120^{\circ}C$. Thus, it is estimated that careful attention to interface design detail and the judicious use of insulation can effect a weight saving of approximately 30 per cent.

Therefore, as shown by the above discussion, a fixed solar array is considered adequate for the mission, and superior to deployable arrays.

3.2.2 Modularity

During the initial phase of the Task B study, when "competing characteristics" of the assigned alternate systems were being assessed, it was felt that the modularity of the LEM descent stage was not one of its distinguishing features. Later, it was recognized that modularity should not be considered only in terms of complete subsystems. It then became evident that many benefits could be derived from the versatility and assembly interface simplicity of the LEM module. The fact that LEM does not have an integral or separable propulsion assembly is advantageous in that the basic engine can be modified with little, if any, effect on the rest of the module. LEM features also a versatile structure, which can be adapted to a large variety of missions, and a propellant storage capacity well suited for Voyager applications. The interface simplicity facilitates the integration and test of the subassemblies.

The octagonal symmetry of the LEM module presents an attractive configuration in that the four corner prismatic compartments can accommodate a variety of subsystem arrangements. In addition, the LEM module will readily accept many types of external structures.

As illustrated by the exploded isometric in Figure 17, the selected spacecraft configuration takes advantage of these features. The electronics equipment mounting panels, discussed in 3.2.5, are separable modules with an integral harness which does not require routing through the LEM module for attachment to the aft equipment module. This aft equipment module, which is simply linked to LEM, supports the remainder of the spacecraft equipment, including eight identical solar array panels, numerous science sensors and antennas, the primary spacecraft antennas, the complete reaction control system and junction boxes—all of which can be tested separately. At the next higher assembly level, all nonpropulsive functions can be implemented by supporting the electronics equipment panels, with or without the modularized thermal louver banks, on jigs contiguous with the integrating aft equipment module, thus eliminating the immediate need for the LEM module. At final assembly, the aft equipment module can be elevated into position with the engine installed. As a result, the normal serial time expended in the assembly and test of a flight spacecraft, or even an engineering model spacecraft, is substantially reduced and effects an attractive cost saving.

3.2.3 Thermal Control Concept

Continuous temperature control of the Voyager spacecraft from lift-off to mission end presents an interesting challenge to the thermal design team, especially since the exposed surface area is large, the internal heat generation is small and external heat flux is variable.

As in Task A, an over-all heat balance calculation indicates that a completely passive design to provide an acceptable thermal environment for spacecraft equipment is not feasible. This is particularly so when the design must allow for undefined variations in material properties, sensor openings, component heat generation, and structural interface conductances. Therefore a design approach has been selected that utilizes thermal insulation to minimize the uncontrolled energy interchange between the main compartment and its appendages, in conjunction with louver control to actively regulate the energy interchange between the main compartment and its environment. A heat balance calculation is carried out to determine the requirement for area with modulated radiation. This calculation uses conservative postulates and considers the interplay between the principal assemblies of the main compartment, the aft equipment module, and the propulsion system. Further refinement of the mathematical model has led to the conclusion that the temperature of electronic components housed within the main compartment can be held within acceptable limits, 30-110°F, with approximately 18 ft^2 of modulated radiation area. This assumes passive design features that provide for an effectively insulated external surface, a reasonable balance or distribution of heat generating equipment, and appropriate radiant and conductive interchange both within and external to the compartment.

Although the selected temperature control design, which is detailed in Volume 2, provides satisfactory temperature control, it may be desirable to enhance the probability of mission success by complementing the simple basic mechanical system with auxiliary surface heaters which could be controlled from earth. An associated design problem has been identified that relates to propellant temperature control. During Martian orbit maneuvers at periods of low power dissipation, the lower limit of 50°F for the propellant might be reached or exceeded. To prevent this, heaters are proposed that could utilize the surplus power that necessarily is available under such a condition.

The thermal investigations conducted during the study included also an evaluation of the influence of the LEM descent engine on the system. It was found that the radiation effects of the nozzle and its extension would be deleterious unless circumvented. Thus, the solar array, which attaches to the aft equipment module, requires that a suitable barrier be provided. This was accomplished by a change to an ablatively cooled nozzle extension with a high temperature insulation over the nozzle. The latter also minimizes the heat loss from the central engine compartment.

In addition to the main compartment insulation previously discussed, it has been found necessary to incorporate insulation on the back side of the aft equipment module to minimize the uncontrolled radiation between this structure and the main compartment. In particular, this was utilized to satisfy the 2.3-hour Martian eclipse period as discussed in 3.2.1.

3.2.4 Appendage Installation

Figure 18 is the principal drawing used herein for reference, and delineates the placement and alignment of all sensors and appendages of the selected Voyager configuration.

At the inception of the configuration design activity, the challenge at hand was to satisfy the orientation and field of view requirements of the many sensors and antennas. To be considered also were the nose fairing envelope constraints, structural requirements for the retention and support of the appendages, alignment accuracy, design simplicity, preservation of modularity, and symmetry. The resulting design is presented in Figure 18 and was evolved in the following manner.

It became apparent early in the design phase that the relative locations of the planetary scan platform (PSP), the high-gain antenna (HGA), the medium-gain antenna (MGA) and the Canopus sensor would play a significant role in the placement of all appendages. It was desired to position the PSP, with two-axis gimballing capability, 180 degrees from the Canopus sensor, so that the visibility of Mars would not be obscured. With the spacecraft oriented to the Sun-Canopus references, it was recognized that the HGA and MGA must be gimballed to permit optimum coverage of the earth during interplanetary cruise and after orbit injection. In order to avoid shadowing the solar array and creating an interference with the outrigger structure during articulation, the HGA, which is the 114-inch-diameter circular paraboloid, and the MGA, which is the 36 by 94-inch elliptical paraboloid, were positioned at clock angles 15 and 285 degrees, respectively.

The prismatic equipment bay faced with panel III appeared to be a convenient place to mount the Canopus sensors since the HGA and MGA reflectors would not obstruct the field of view in that location. Furthermore, since the fine sun sensor, the axis of which establishes the zero cone angle reference, is packaged with the Canopus sensor, the sun sensor view port could be located at the split line between the solar array panels, which is a line intercepting panel III. Lastly, the PSP was positioned beyond the desired 180-degree clock angle, outboard of panel V, in order to preclude interference with the outrigger structure and to facilitate storage and retention during boost. As a result, the Martian surface may become slightly obscured by the capsule sterilization canister toward the end of the mission.

The capsule relay link antenna, which is an array of four cavity-backed archimedean spiral elements, was conveniently placed adjacent to the PSP on the aft equipment module with a clock angle of 255 degrees. This arrangement permits the area of the aft equipment module outboard of panel VII to be reserved for the storage and retention of a multi-element magnetometer boom should the DeHavilland STEM system fail to meet the axes alignment requirement of 1 degree.

The area of the aft equipment module outboard of panel I accommodates the fixed science package and the remaining science support antennas. Again, the experiment view ports coincide with the split line between solar array panels. However, should the view port requirements increase, it would seem advisable to mount the complete package at the extremity of a deployable boom in order to minimize the adverse effect on the power subsystem. DeHavilland extendible boom units will be employed for science antennas. The 20-foot ionosphere sounder antenna must be retracted during the retropropulsion maneuver whereas the short 6-foot bistatic radar antennas may remain deployed. The 60- to 100-foot RF noise detector antenna is extended at a 5-degree cone angle, which allows it to remain extended during deboost, because the thrust induces a stabilizing tensile load in the element.

The low-gain antenna, which is a cup-backed turnstile, is mounted on the end of a 12-foot aluminum tubular boom and stowed beneath the solar array. The cutouts in the periphery of the solar array panels facilitate its installation and provide for the hinge and retention joints. Prior to nose fairing separation, this wide-beam antenna provides telemetry coverage through a coupling aperture in the metallic nose fairing. This interface is identified as an area which should receive particular attention during the next phase of the Voyager program.

3.2.5 Packaging

The octagonal geometric symmetry of the LEM module presents an attractive configuration for the accommodation of electronics equipment. The four corner prismatic compartments provide the flexibility necessary for the utilization of volume and mounting surface to achieve an arrangement compatible with spacecraft system and subsystem requirements. These requirements demand consideration of mass properties control, modularity, the balance of power and power dissipation, accessibility, structural efficiency, growth accommodation, standardization, electrical harness routing, and meteoroid protection.

A gross inertia balance dictated the utilization of the diagonally opposite bays faced with panels I and V for the installation of the propulsion and reaction control gas storage vessels, as shown in Figure 15. An assessment of the volume and mounting area provisions of the two remaining prismatic compartments, faced with panels III and VII, indicated that subsystem assemblies could be grouped and installed with little difficulty.

The next step in the design process was to establish the structural configuration of the equipment panels. It was desired to create an integrated structure which allows subassemblies and their mounting provisions to contribute to the strength and rigidity of the assembly, panel and spacecraft structure. The combination of the selected truss grid core-aluminum sandwich with longitudinal stiffeners appeared to be the most efficient construction and satisfied the requirements associated with weight, thermal conductivity, micrometeorite protection, and standardization of assembly sizes.

Consistent with strength and rigidity requirements, the panels were subdivided in four smaller panels, hinged along the outside vertical edges, to satisfy the objectives of reasonable manufacturing practicability, modularity, and serviceability.

Subsequent iterations in equipment placement to satisfy mass properties, power and power dissipation requirements may lead to a different equipment arrangement; however, present design requirements are satisfied with the equipment installation delineated in Figures 19 and 20.

4. RELIABILITY DESIGN OBJECTIVES

This section provides a consolidated statement of Voyager spacecraft reliability goals as needed to achieve maximum mission reliability, consistent with established mission and spacecraft constraints. The presentation includes a breakdown at the subsystem level along with a discussion of over-all system considerations. General supporting data is given and discussed, with references provided to the detailed data and analysis for each subsystem in Volume 2.

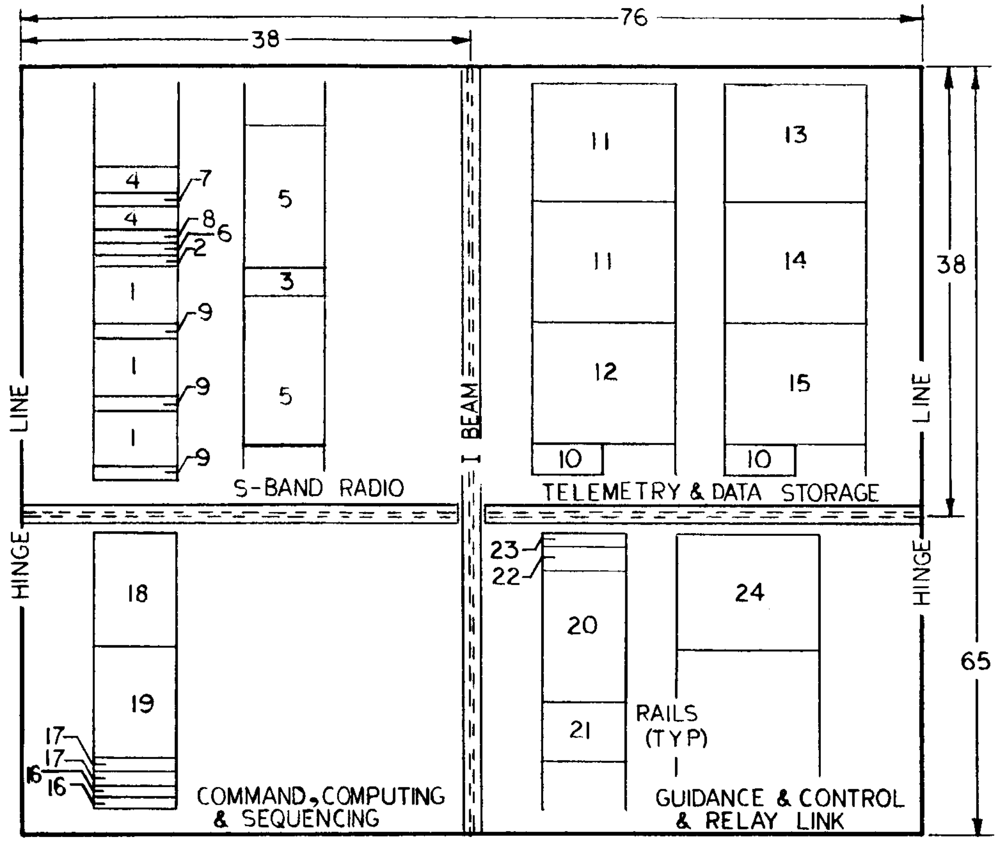
4.1 Reliability Ground Rules

4.1.1 Assessments

In deriving the present reliability assessment numerics, a consistent level of conservatism has been maintained for all components, so that system tradeoff decisions are not seriously biased by assessment errors. A standard of design conservatism is prescribed in terms of derating and stress constraints consistent with the 1971 Voyager mission objectives. Component reliability goals, as apportioned, appear to be realistically achievable under the design constraints specified.

4.1.2 Mission Model

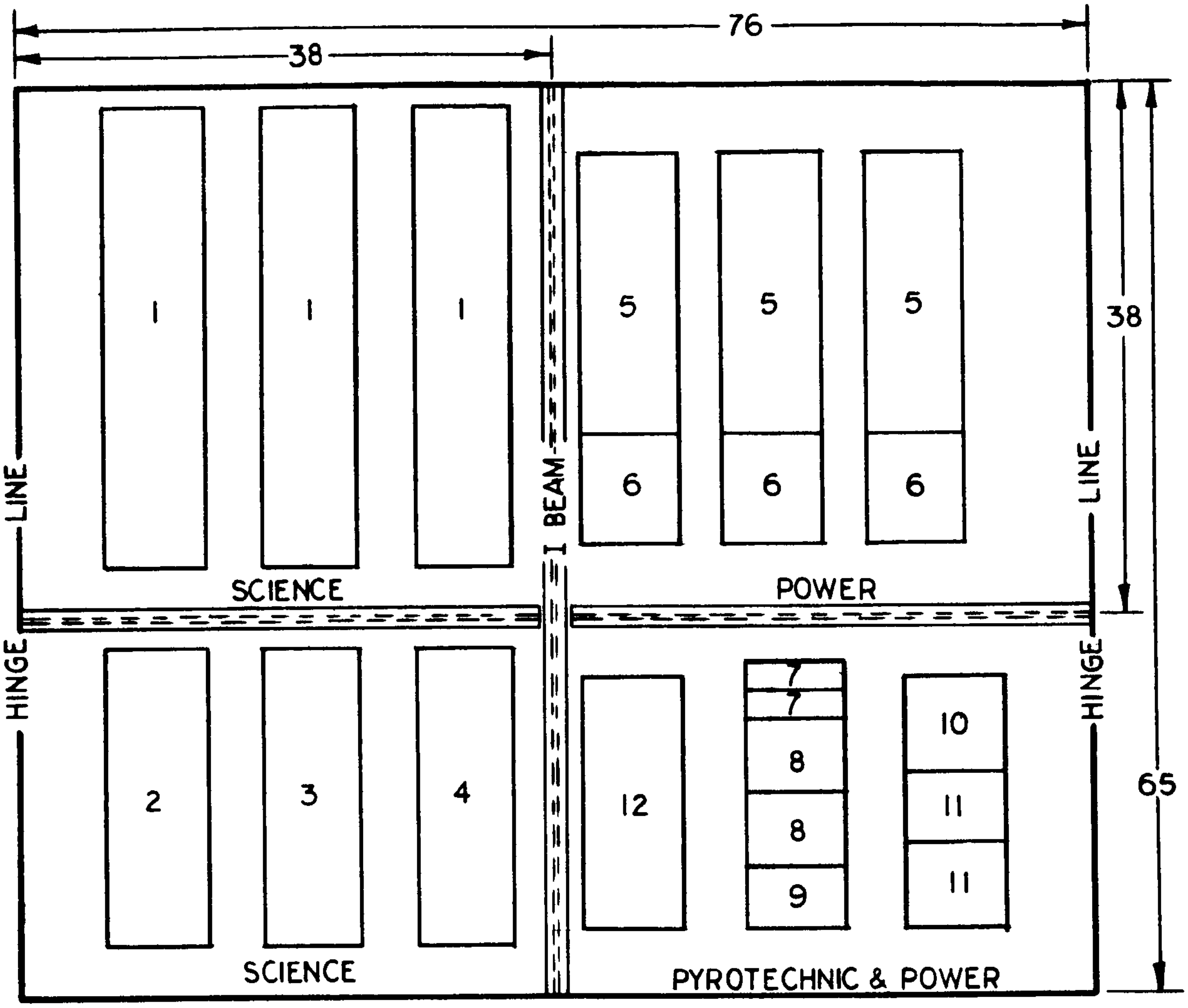
For the purpose of establishing reliability design goals a representative Voyager mission is established as a basis for quantitative analysis. Reliability design and test goals are derived from subsystem success probability requirements for time intervals and events corresponding to the representative mission phases given below:



	ITEM	QUANTITY	SIZE EACH	WEIGHT/ UNIT	TOTAL WEIGHT
S-BAND RADIO					48.4
1	S-BAND RECEIVER	3	7×6×4.7	5.0	15.0
2	RECEIVER SELECTOR	1	7×6×1.0	1.0	1.0
3	TRANSMITTER & MONITOR	1	7×6×2.2	3.5	3.5
4	MODULATOR EXCITER	2	7×6×2.2	3.0	6.0
5	PWR AMPL, SUPPLY & MONITER	2	7×6×12.0	7.8	15.6
6	TRANSMITTER SELECTOR	1	7×6×1.0	1.0	1.0
7	HYBRID RING & PWR MONITER	1	7×6×1.0	0.6	0.6
8	CIRCULATOR SWITCH-4 IN BOX	1	7×6×1.0	1.8	1.8
9	DIPLEXER	3	7×6×1.2	1.3	3.9
TELEMETRY					8.0
10	PCM ENCODER	2	7×6×2.7	4.0	8.0
DATA STORAGE					72.0
11	TV TAPE RECORDER	2	12×10×5.0	12.0	24.0
12	SPECTROMETER RECORDER	1	12×10×5.0	12.0	12.0
13	IR SCANNER RECORDER	1	12×10×5.0	12.0	12.0
14	SPACECRAFT RECORDER	1	12×10×5.0	12.0	12.0
15	FIELD/PARTICLE RECORDER	1	12×10×5.0	12.0	12.0
COMMAND					10.4
16	COMMAND DETECTOR	2	7×6×1.0	2.6	5.2
17	DECODER	2	7×6×1.2	2.6	5.2
COMPUTING & SEQUENCING					36.0
18	COMPUTER & SEQUENCER	1	7×6×7.8	19.0	19.0
19	BACKUP COMPTR & SEQUENCER	1	7×6×7.8	17.0	17.0
GUIDANCE & CONTROL					18.0
20	GUIDANCE & CONTROL ELECT.	1	7×6×11.0	13.0	13.0
21	ANTENNA DRIVE ELECT.	1	7×6×5.0	5.0	5.0
RELAY LINK					16.8
22	RECEIVER-2 IN BOX	2	7×6×1.0	2.0	4.0
23	DEMODULATOR-2 IN BOX	2	7×6×1.0	0.4	0.8
24	TAPE RECORDER	1	12×10×5.0	12.0	12.0

PANEL III EQUIPMENT INSTALLATION

Figure 19. Panel III Equipment Installation



		ITEM	QUANITITY	SIZE EACH	WEIGHT/ UNIT	TOTAL WEIGHT
SCIENCE						233.0
	1	REMOTE HARDWARE P.S.P.	3	7×6×33.0	43.0	129.0
	2	DATA AUTOMATION	1	7×6×22.0	37.0	37.0
	3	REMOTE HRDW. DEPLOY. SEN.	1	7×6×22.0	23.0	23.0
	4	" " NONDEPLOY. "	1	7×6×22.0	44.0	44.0
POWER						232.1
	5	BATTERY	3	7×8×19.5	46.0	138.0
	6	BATTERY REGULATOR	3	7×8×7.2	14.0	42.0
	7	400 CPS INVERTER	2	7×6×2.0	3.0	6.0
	8	SHUNT ELEMENT	2	7×6×5.2	8.0	16.0
	9	POWER CONTROL UNIT	1	7×6×4.5	8.0	8.0
	10	POWER DISTR. & SWITCHING	1	7×6×7.0	7.5	7.5
	11	4.0 KC INVERTER	2	7×6×5.2	7.3	14.6
PYROTECHNIC						25.0
	12	PYROTECHNIC CONTROL	1	7×6×17.0	25.0	25.0

PANEL VII EQUIPMENT INSTALLATION

Figure 20. Panel VII Equipment Installation

a) Phase 1A (.3 hour) (2 vehicles) — The operations and events starting at liftoff and continuing through boost, parking orbit insertion, coast, and injection of two planetary vehicles into suitable planetary transfer trajectories.

b) Phase 1B (4280 hours) (1 or 2 vehicles) — The operations and events starting from the completion of injection of two planetary vehicles and continuing through successful separation, celestial reference acquisition, interplanetary cruise, midcourse corrections, and jettisoning of the flight capsule canister lid.

c) Phase 2A (1 or 2 vehicles) — The operations and events associated with achievement of planetary vehicle Mars orbit insertion by means of retro propulsion, including completion of all preparatory events and sequences.

d) Phase 2B (168 hours) (1 or 2 vehicles) — The operations and events after Mars orbit injection and continuing through reacquisition of celestial attitude references, successful accomplishment of one or more orbit trim maneuvers, and performance of orbital operation of spacecraft science.

e) Phase 3A (1 or 2 vehicles) — The operations and events constituting successful flight capsule separation, after completion of all preparatory events and sequences.

f) Phase 3B (15 hours) (1 or 2 vehicles) — The operations and events after flight capsule separation and continuing through planetary vehicle relay of capsule data to earth (1971).

g) Phase 4 (540 hours) (1 or 2 vehicles) — The operations and events after relay of capsule data to earth through completion of a total of one month of operation of the flight spacecraft in Mars orbit.

The execution of two planetary vehicle flights is evaluated in terms of the reliability of two independent missions after Phase 1A, where success is defined as the successful boost, parking orbit insertion, and injection of two planetary vehicles. During Phase 1B, each separating vehicle is responsible for its own separation from the launch vehicle as an independent event. The small difference in flight calendar time for the two planetary vehicles does not necessitate a significant adjustment of the defined mission phases.

4.1.3 Failure Modes and Criticality

The Voyager flight sequence of events as given in 5.4 constitutes an event-by-event identification of failure mode potentials for the flight spacecraft. The criticality of these events is dependent on their relation to system operation and the corresponding competing characteristic criteria.

Each defined subsystem is equally critical (i.e., in-line as a reliability risk) to the probability of success for the mission. Failure mode recognition within individual subsystems is provided in terms of incorporated redundancy in each subsystem. In nonredundant subsystems, each contributing equipment is of equal criticality for that subsystem function. In this case, mission criticality for recognized equipment failure modes is equivalent to the loss of the corresponding subsystem function.

In the spacecraft design, significant levels of equipment redundancy are employed. The application of redundancy includes (1) cooperative multichannel designs, (2) functional redundancy, and (3) ordinary block redundancy to preclude the known failure modes. The selection of these redundant configurations is based upon the need to preserve individual functional capabilities of subsystems or portions of subsystems, in the event of foreseen internal failures. The magnitude of these failure probabilities, as determined from individual equipment reliability assessments, prescribes the incentive for redundant equipment configurations within subsystems and equipment. Furthermore, reliability assessments are based upon conservative usage of design materiel including proven practices of part level redundancy, which are in turn based upon known failure modes and their relative probabilities of occurrence.

4.1.4 Mathematical Models and Distributions

a. Exponential Representation

Reliability models for Voyager system and subsystem analyses are based upon an assumed exponential representation for all nonredundant electronic equipment called upon to operate over extended periods. In all but a few instances, the electrical stress levels are subjected to design derating so that electronic equipment failure rates are essentially the same in their unenergized state as in their energized state. The projected probabilities of survival for such equipment is, therefore, essentially independent of the duty factors and characterized only by the duration of the individual and cumulative mission phases. This conservative limiting case also provides that the reliability prediction for sequentially redundant equipment (i.e., those energized separately) converges to the more conservative case of parallel redundancy where all elements are energized. In reliability versus weight analyses, the actual consumption of power in parallel redundant cases was accounted for as an equivalent weight increase incurred by the power subsystem but attributed to the redundant subsystem.

b. Redundancy Representations

Reliability functions of a nonexponential form are used to characterize redundant configurations at equipment level. These R(t) functions are derived (except for "one-shot" events) from the equivalent parallel redundant models using exponential model constituents. In all instances where equipment redundancy entails the use of sensing and switching functions with significant risk magnitudes, model adjustments are made to include series-risk increases.

c. Mission Survival

The computation of mission survival for each subsystem employs R(t) for each mission time at mission-phase termination. Computation of mission survival for the complete system combines subsystem reliabilities as a simple product because of their probabilistic independence as found during failure mode and effect analyses. Computation of the probabilities of successfully completing each separate mission phase, for each subsystem, is provided by dividing the R(t) probability for the time of phase termination by the R(t) probability for completing the preceding phase of operation.

d. Boost-Success Model

The reliability models used to depict probability of success for Mission Phase 1A are based upon a modified interpretation of the exponential R(t) function. Recourse is taken in the estimate of a significantly increased environmental ambient for all equipment for the short Phase 1A time period of 0.3 hour. This estimate provides an equivalent failure rate multiplier of 10^3 as an operator upon the nominal environment failure rate for individual equipment. This concept affords the net effect whereby equipment level redundancies yield a safeguard (i.e., better than nonredundant reliability) consistent with random equipment exposure and response to the launch phase environmental profile.

e. Time-Independent Probabilities

Reliability models for event success probabilities, employ selectively the binomial distribution. Analysis of one-shot devices and other actuation (rather than time dependent) probabilities has been treated in this manner. Selected use is also made of reliability assessment models where survival is a function of cumulative events rather than time. These probability numerics are inserted within the mathematical model for mission phases in accordance with their time of expected occurrence throughout the flight sequence for the Voyager representative mission.

f. Design Limits and Wear-Out Modes

Reliability estimates for some subsystem components necessitate the probabilistic evaluation of violating an important design limit, rather than an irreversible chance failure. For identified useful life, reliability objectives are set in terms of 3-sigma limits for a sufficient period of operating time to allow for both ground testing and extended missions. For solar array modeling, the power-reserve characteristic of mission Phase 4 is a limiting criteria, since the long cruise and orbital phases are significant factors for possible cell failures. In contrast, operation after mission Phase 4 places maximum power demand on the solar array, thus lowering the threshold to failure and significantly reducing the probability of surviving extended periods of orbital operation.

g. Use of Weight and Power Reserves

The reliability goals established for spacecraft subsystems are based upon the allocation of weight reserves (and power as equivalent weight reserves) to those subsystems or subsystem elements achieving the maximum reliability improvement. The details of this process were discussed as part of the iteration of system design for reliability given in the Phase 1A Task A study.

h. Command, Switching and Redundancy

Selection of alternate spacecraft functional modes and alternate equipment operation (from a redundant set) for a given functional mode requires the continued capability to receive and execute commands. Particular attention has been given to the problem of assuring the success of modes which are conditional to the reliability of uplink communications as a high priority mode. This places a conditional-probability emphasis upon onboard redundancy management and accomplishment of uplink command communications to assure downlink telemetry with a high level of reliability. These supporting modes have been considered in the establishment of weighted reliability objectives for the spacecraft radio and command subsystems and the spacecraft computing and sequencing subsystem. The reliability risks entailed in the management of redundancy have been incorporated in the reliability models established for spacecraft subsystems.

4.1.5 Environments and Derating Policies

The thermal environment conditions assumed for reliability computations are based upon estimated capabilities of the temperature control subsystem using a conservative upper limit. The nominal operating temperature assumed for all electronic equipments is 50°C, which is essentially a worst-case expectation.

Electronic equipment design criteria include a derating policy to 40 per cent of rated electrical stress for electronic parts in analog functions and 10 per cent of rated stress for digital functions. No provisions are made to assess the probabilistic influence of overpowering environmental factors of unknown limits.

Mechanical structures are assessed for reliability potential in terms of rupture probabilities after assuring a conservative "margin of safety" value as the design criterion for all structural stress analysis. Micrometeorite protection has been dealt with as discussed in 9 of Section II, Volume 2.

Mechanical and pneumatic component reliability values are based upon estimated performance degradations as a function of mission time. Thus, critical factors such as reserve strength or gas supply are accounted for as depletion probability parameters pertinent to the mission phase(s) in which the corresponding component is critical.

4.1.6 Electronic Packaging Policies

In the utilization of reserve spacecraft weight and power to gain increased system reliability (through redundancy), the weight and power demands for basic nonredundant equipment are critical factors. For the determination of electronic package characteristics it was established that discrete part types used for repetitive functions would be replaced by selected integrated circuit packages. The design constraints applicable to this usage are as follows:

- Each integrated circuit function incorporated is judged to be more amenable to the circuit tolerance controls characteristic of integrated circuit technology.
- Weight reductions inherent in a transition to integrated circuit packaging allow multichannel, functional and block redundancies within the constraints of subsystem weights for equivalent nonredundant discrete part assemblies.
- The number of integrated circuit types are severely limited and each type kept in its most simplified configuration.

4.2 Subsystem Reliability Models and Apportionments

Using the ground rules given in 4.1, each mission-critical subsystem of the spacecraft is assigned a reliability objective. Table 2 lists the reliability objectives for each subsystem and the complete spacecraft (equipment under cognizance of the spacecraft contractor only) both by individual mission phases and cumulatively. Table 12 provides a more detailed reliability objective listing for the components of the subsystems.

Each component of each subsystem has been apportioned a reliability goal based upon the subsystem reliability models. These models, representing the selected design, have been developed for each subsystem and are presented in the corresponding writeup in Volume 2. The redundant configurations established as a basis for the design are illustrated. Reliability values are given for subsystem components and the appropriate mathematical reliability function is given for each model. These models are the basis for the reliability apportionments given in Table 12.

4.3 Reliability Demonstration Requirements

The planning of reliability demonstration tests for individual subsystem components will utilize their MTBF objectives as appropriate statistical design hypothesis. Table 13 provides a summary of these MTBF values as determined directly from their failure rate characteristics as utilized in the subsystem reliability models.

4.3.1 Truncated-Sequential Test Plan

The Voyager life-test quantities and time intervals are based upon a minimal truncated-sequential statistical test criterion. Truncation is provided at 0.6 MTBF and two failures. For zero failures a decision is reached at 0.33 MTBF. These criteria are provided to establish an equitable allocation of life test resources rather than to demonstrate numerical MTBF levels at a prescribed confidence.

Table 12 Reliability Design Objectives

Spacecraft Function	Spacecraft Reliability by Individual Mission Phase							Spacecraft Reliability for Cumulative Mission Phases						
	1A	1B	2A	2B	3A	3B	4	1A	1A and B	1 and 2A	1 and 2	1, 2, and 3A	1, 2, and 3	1, 2, 3, and 4
1.0 Structural and Mechanical	.99961	.97889	.99999	.99996	.99999	.99795	.99385	.99961	.97851	.97850	.97846	.97845	.97645	.97044
1.1 Structure	.99961	.99999	.99999	.99999	.99999	.99999	.99999	.99961	.99960	.99959	.99958	.99957	.99956	.99955
1.2 Meteoroid protection		.97900				.99796	.99386		.97900	.97900	.97900	.97900	.97700	.97100
1.3 Low-gain antenna deployment		.99998												
1.4 High-gain antenna release		.99998							.99998	.99998	.99998	.99998	.99998	.99998
1.5 Medium-gain antenna release		.99998							.99998	.99998	.99998	.99998	.99998	.99998
1.6 Low-gain antenna release														
1.7 Planetary platform release		.99997							.99997	.99997	.99997	.99997	.99997	.99997
1.8 Flight capsule emergency separation				.99997							.99997	.99997	.99997	.99997
1.9 Planetary vehicle separation		.99999							.99999	.99999	.99999	.99999	.99999	.99999
2.0 Command	.99998	.99582		.99969		.99997	.99895	.99998	.99584	.99584	.99554	.99554	.99552	.99448
2.1 Command detector	.99999	.99969		.99997			.99991	.99999	.99969	.99969	.99967	.99967	.99967	.99959
2.2 Decoder	.99526	.93445		.99734		.99976	.99148	.99526	.93003	.93003	.92756	.92756	.92734	.91944
3.0 Propulsion	.99989	.99148	.99901	.99905				.99989	.99138	.99039	.98945			
3.1 Cabling	.99999	.99999	.99999	.99999				.99999	.99998	.99997	.99996			
3.2 Engine	.99999	.99276	.99905	.99961				.99999	.99275	.99181	.99143			
3.3 Propellant Feed														
3.4 Thrust vector control actuator	.99991	.99872	.99997	.99945				.99991	.99864	.99861	.99806			
4.0 Cabling	.99999							.99999	.99999	.99999	.99999	.99999	.99999	.99999
4.1 Junction box														
4.2 Interconnecting cables														
4.3 Planetary vehicle adapter, electrical														
5.0 Computing and Sequencing	.99865	.95451		.99730		.99975	.99097	.99865	.95323	.95323	.95066	.95066	.95043	.94185
5.1 Primary computer and sequencer	.98636	.82205		.99232		.99931	.97557	.98636	.81084	.81084	.80462	.80462	.80407	.78443
5.2 Backup computer and sequencer	.98693	.82886		.99265		.99933	.97659	.98693	.81803	.81803	.81202	.81202	.81148	.79249
6.0 S-Band Radio	.99222	.96688		.99541		.99976	.99170	.99222	.95936	.95936	.95496	.95496	.95474	.94682
6.1 Low-gain antenna	.99998	.99974		.99998		.99998	.99996	.99998	.99973	.99973	.99972	.99972	.99971	.99968
6.2 S-band receiver	.99484	.92892		.99711		.99973	.99074	.99484	.92413	.92413	.92146	.92146	.92122	.91269
6.3 Receiver selector	.99963	.99479		.99979		.99997	.99934	.99963	.99443	.99443	.99423	.99423	.99421	.99356
6.4 One-watt transmitter and power monitor	.99999	.99948		.99995			.99986	.99999	.99948	.99948	.99944	.99944	.99944	.99931
6.5 Modulator exciter	.99999	.99948		.99995			.99986	.99999	.99948	.99948	.99944	.99944	.99944	.99931
6.6 Power amplifier, supply and monitor	.99999	.99541		.99966		.99996	.99886	.99999	.99541	.99541	.99508	.99508	.99505	.99392
6.7 Transmitter selector	.99922	.98897		.99956		.99995	.99860	.99922	.98820	.98820	.98777	.98777	.98773	.98635
6.8 Four-port hybrid ring and monitor	.99993	.99892		.99994			.99986	.99993	.99886	.99886	.99881	.99881	.99881	.99868
6.9 Circulator switch	.99993	.99892		.99994			.99986	.99993	.99886	.99886	.99881	.99881	.99881	.99864
6.10 Diplexer	.99993	.99892		.99994			.99986	.99993	.99886	.99886	.99881	.99881	.99881	.99868
7.0 Guidance and Control	.99700	.94400	.99994	.99999	.99995	.99999	.99400	.99700	.94117	.94111	.94110	.94105	.94104	.93539
7.1 Gyro reference assembly	.97510	.99289	.99999	.99970	.99999		.99911	.97510	.96816	.96815	.96785	.96784	.96784	.96697
7.2 Accelerometer														
7.3 Guidance and control electronics	.99810	.97521		.99999		.99999	.99700	.99810	.97335	.97335	.97334	.97334	.97333	.97041
7.4 Canopus sensor	.99805	.97154	.99999	.99878	.99999	.99999	.99642	.99805	.96964	.96963	.96844	.96843	.96842	.96495
7.5 Sun sensor	.99999	.99995	.99999	.99999	.99999	.99999	.99999	.99999	.99994	.99993	.99992	.99991	.99990	.99989
7.6 Earth detector														
7.7 Limb and terminator detector	.99999	.99995	.99999	.99999	.99999	.99999	.99999	.99999	.99994	.99993	.99992	.99991	.99990	.99989
7.8 Reaction control	.99929	.97077	.99998	.99875	.99999	.99999	.99632	.99929	.97008	.97006	.96884	.96883	.96882	.96525
7.9 High-gain antenna, feed and drive assembly	.98975	.86329		.99425		.99948	.98160	.98975	.85444	.85444	.84953	.84953	.84909	.83317
7.10 Medium-gain antenna, feed and drive assembly	.99486	.92914		.99712		.99974	.99078	.99486	.92436	.92436	.92170	.92170	.92146	.91296
7.11 Antenna drive electronics	.99633	.94889		.99794		.99982	.99341	.99633	.94541	.94541	.94346	.94346	.94329	.93707
8.0 Telemetry	.99995	.99039		.99931		.99992	.99764	.99995	.99035	.99035	.98967	.98967	.98960	.98727
8.1 PCM encoder	.99325	.90789		.99621		.99973	.98782	.99325	.90176	.90176	.89834	.89834	.89810	.88716
9.0 Temperature Control	.99998	.99939		.99998			.99990	.99998	.99937	.99937	.99935	.99935	.99935	.99929
9.1 Heaters and thermostats	.99999	.99985		.99999			.99998	.99999	.99984	.99984	.99983	.99983	.99983	.99981
9.2 Louvers	.99999	.99954		.99999			.99992	.99999	.99953	.99953	.99952	.99952	.99952	.99948
10.0 Power	.99997	.99631		.99973		.99996	.99908	.99997	.99629	.99629	.99603	.99603	.99600	.99509
10.1 Solar array	.99995							.99995	.99995	.99995	.99995	.99995	.99995	.99995
10.2 Power control unit	.99999	.99977		.99999		.99999	.99997	.99999	.99978	.99978	.99977	.99977	.99976	.99973
10.3 Shunt elements	.99999						.99999	.99999	.99999	.99999	.99999	.99999	.99999	.99999
10.4 Battery	.99784	.96965		.99879		.99990	.99612	.99784	.96756	.96756	.96639	.96639	.96629	.96254
10.5 Battery regulator	.99993	.99892		.99996		.99999	.99987	.99993	.99885	.99885	.99881	.99881	.99880	.99867
10.6 4 kc inverter	.99999	.99997					.99999	.99999	.99996	.99996	.99996	.99996	.99996	.99995
10.7 400 cps inverter	.99999	.99990		.99999			.99997	.99999	.99989	.99989	.99988	.99988	.99988	.99985
10.8 Power distribution unit	.99999	.99999					.99999	.99999	.99998	.99998	.99998	.99998	.99998	.99997
11.0 Data Storage	.98529	.80713		.99163		.99914	.97295	.98529	.79526	.79526	.78860	.78860	.78792	.76661
11.1 TV tape recorder	.99631	.94855		.99794		.99981	.99336	.99631	.94505	.94505	.94310	.94310	.94292	.93666
11.2 Spectrometer tape recorder	.99631	.94855		.99794		.99981	.99336	.99631	.94505	.94505	.94310	.94310	.94292	.93666
11.3 IR scanner tape recorder	.99631	.94855		.99794		.99981	.99336	.99631	.94505	.94505	.94310	.94310	.94292	.93666
11.4 Spacecraft tape recorder	.99631	.94855		.99794		.99981	.99336	.99631	.94505	.94505	.94310	.94310	.94292	.93666
11.5 Fields/particles tape recorder	.99631	.94855		.99794		.99981	.99336	.99631	.94505	.94505	.94310	.94310	.94292	.93666
12.0 Pyrotechnic	.97733							.97733	.97733	.97733	.97733	.97733	.97733	.97733
12.1 Launch vehicle separation														
12.2 Capsule emergency jettison														
12.3 Umbilical disconnect	.99998							.99998	.99998	.99998	.99998	.99998	.99998	.99998
12.4 Capsule emergency disconnect														
12.5 High-gain antenna release														
12.6 Medium-gain antenna release														
12.7 Low-gain antenna release														
12.8 Explosive valve														
12.9 Pyrotechnic control	.97735							.97735	.97735	.97735	.97735	.97735	.97735	.97735
Mission Success Probability (one planetary vehicle)	.95071	.67020	.99894	.98212	.99994	.99644	.94040	.95071	.63717	.63649	.62511	.62507	.62285	.58574
Mission Success Probability (at least one of two planetary vehicles)								.90385*	.86835	.86786	.85946	.85943	.85776	.82839

*In Mission Phase 1A, success is defined as the successful completion of phase events for both planetary vehicles

Table 13. Design Reliability Requirements for Long-Life Spacecraft Components

No. in Table 12	Operating Component	R_3	MTBF, hours
2.1	Command detector	.99969	14,300,000
2.2	Decoder	.93003	63,500
5.1	Primary C and S	.81084	23,500
5.2	Backup C and S	.81803	24,400
6.2	S-band receiver	.92413	58,600
6.3	Receiver selector	.99443	798,000
6.4	1-watt transmitter and power monitor	.99948	8,550,000
6.5	Modulator exciter	.99948	8,550,000
6.6	Power amplifier, supply and monitor	.99541	969,000
6.7	Transmitter selector	.98820	376,000
7.1	Gyro reference assembly	.96815	139,000
7.3	G and C electronics	.97335	166,000
7.4	Canopus sensor	.96963	146,000
7.5	Sun sensor	.99993	63,500,000
7.7	Limb and terminator detector	.99993	63,500,000
7.8	Reaction control	.97006	148,000
7.9	High-gain antenna feed and drive	.85444	30,500
7.10	Medium-gain antenna, feed and drive	.92436	58,800
7.11	Antenna drive electronics	.94541	81,400
8.1	PCM encoder	.99035	460,000
10.2	Power control unit	.99977	19,300,000
10.3	Shunt elements	.99999	444,000,000
10.4	Battery	.96639	132,000
10.5	Battery regulator	.99881	3,730,000
10.6	4 kc inverter	.99996	111,000,000
10.7	400 cps inverter	.99988	37,000,000
10.8	Power distribution unit	.99998	222,000,000

$\mathrm{MTBF} = \frac{4448}{1-R_3}$ hours

R_3 = Survival probability for 4448 hours at 30°C

4.3.2 Life Test Acceleration Factors

In order to expedite life test schedules, and reduce the number of required test articles, elevated environments (temperature increases above the flight nominal) are prescribed for the spacecraft and critical subsystem components. The expected reliability degradation resulting from elevated temperatures is based upon an Arrhenius constant of 0.0693 (failure rate doubles per $10^{o}C$ increments above $30^{o}C$) applicable to complex conductor-insulation systems.

4.3.3 Spacecraft Life-Test Interval

With a Phase 1 and Phase 2 mission of 4448 hours and a flight spacecraft reliability of approximately 0.656 a projected (equivalent exponential) MTBF test hypothesis is established at 10,550 hours. The prescribed truncated sequential test will necessitate a test time of 10,550 or 3520 hours with no failures allowed and for no environmental acceleration. With a temperature increment of $10^{o}C$ (test temperature at $40^{o}C$) an acceleration factor of two is provided establishing a calendar test interval of $(\frac{3520}{2})$ or 1750 hours.

4.3.4 Subsystem Component Life-Test Intervals

The life-test interval for individual critical components is established using their assessed MTBF as test hypotheses. A zero failure test criteria is employed at the one-third MTBF test interval. Test quantities are selected to provide an accumulated test time-quantity product (as required to fulfill the one-third MTBF criterion) when adjusted for the effect of acceleration factors. Acceleration of life tests are not employed for complex test articles (i.e., those with MTBF less than 10^5 hours) and kept less than 32 for any test article. The calendar spans prescribed are (1) 6 months for MTBF less than 10^5 hours, (2) 1 year for MTBF between 10^5 and 10^6 hours, and (3) 18 months for MTBF greater than 10^6 hours. Required test article quantities are adjusted to meet the prescribed one-third MTBF reliability test criterion.

5. SYSTEM DATA

5.1 Weights and Mass Properties

5.1.1 Weights Data

a. Weight Summary

Table 14 lists a planetary vehicle weight summary and also gives subsystem totals. It indicates which of the weights are in the spacecraft bus, spacecraft propulsion, and flight capsule. The total weight is within the 1971 mission weight allocation given in 2.3 of Section III. The total combined weight for the spacecraft propulsion and bus meets the allocation of 17,100 pounds (not including science). However, the propulsion and the bus weights cannot be represented strictly as 15,000 pounds and 2100 pounds individually, because of the difficulty in establishing a clear demarcation between the propulsion module subsystem and the spacecraft bus.

b. Detailed Weight Statement

Tables 15 and 16 contain the detailed breakdown for the summary weights in Table 14.

c. Weight Sequence

The weight sequence for the 1971 mission is given in Table 17.

d. Flight Capsule Weights

The total allocated weight for the flight capsule is divided nominally as follows:

	Weight, lb
Separated capsule vehicle	2490
Jettisoned capsule canister lid	260
Unseparated flight capsule weight*	250
Total	3000

*The maximum allocated weight above the field joint which may remain with the spacecraft.

Table 14. Voyager Planetary Vehicle Weight Summary

Item	Weights, lb	
Spacecraft Bus Inert Weight		2,386
Structural and mechanical	777	
Pyrotechnics	51	
Temperature control	111	
Radio	126	
Relay link	25	
Data storage	72	
Telemetry	8	
Command	11	
Computing and sequencing	36	
Cabling	229	
Power	522	
Guidance and control	268	
Balance weights	15	
Contingency	135	
Spacecraft Science Payload		400
Spacecraft Propulsion		14,714
Dry weight	2,863	
Inert fluids	477	
Consumable propellants	11,374	
Flight Spacecraft		17,500
Planetary Vehicle Adapter		422
Flight Capsule		3,000
Planetary Vehicle with Adapter		20,922
Total Allocated Weight		22,000
Weight Margin (adapter)		1,078

Table 15. Detailed Weight Statement, Spacecraft Bus Weight

Item	Weight, lb	
Structural and Mechanical		777
Equipment latches	2.4	
Equipment panel hinges	2.2	
Miscellaneous supports	8.7	
Spacecraft outriggers (4)	180.0	
Aft equipment module	176.1	
Equipment mounting panels (8)	106.3	
Equipment mounting rails	78.0	
Solar array support linkage system	30.0	
Attachments and miscellaneous	44.0	
Flight capsule interstage	149.0	
Pyrotechnics		51
Release and deployment system	7.7	
Flight capsule emergency separation	13.7	
Electrical connectors	2.2	
Explosive valve pyrotechnics (18)	0.6	
Pyrotechnic control assembly	25.0	
Attachments and miscellaneous	1.5	
Temperature Control		111
Insulation	85.3	
Louvers	17.1	
Heaters and thermostats	2.0	
Attachments and miscellaneous	6.3	
Radio		125
Modulator-exciter (2)	6.0	
4 port hybrid ring and power monitor	0.6	
1-watt transmitter and power monitor	3.5	
Power amplifier (50w), power supply, and RF power monitor (2)	15.6	
Transmitter selector	1.0	

Table 15. Detailed Weight Statement, Spacecraft Bus Weight (Continued)

Item	Weight, lb	
Radio (Continued)		
S-band receiver (3)	15.0	
Receiver selector	1.0	
Circulator switch (4)	7.3	
Diplexer (3)	3.9	
Low-gain antenna	3.0	
Medium-gain antenna*	13.4	
High-gain antenna*	55.2	
Relay Link		25
UHF antenna	8.0	
Receiver (2)	4.0	
Tape recorder	12.0	
Demodulator (2)	0.8	
Data Storage		72
Tape recorders (6)	72.0	
Telemetry		8
PCM encoder (2)	8.0	
Command		11
Decoder (2)	5.3	
Command detector (2)	5.2	
Computing and Sequencing		36
Cabling		229
Interconnecting cables	190.0	
Junction box (4)	20.0	

*Includes structural support arm

Table 15. Detailed Weight Statement, Spacecraft Bus Weight (Continued)

Item	Weight, lb	
Cabling (Continued)		
Umbilical	8. 0	
Cabling channels	10. 9	
Power		522
Solar array	290. 0	
Batteries (3)	138. 0	
Inverter 250 w, 4 kc (2)	14. 6	
Inverter 50 w, 400 cy (2)	6. 0	
Battery regulator (3)	42. 0	
Power control unit	8. 0	
Shunt element assembly (2)	16. 0	
Power distribution box	7. 5	
Guidance and Control		268
Gyro reference assembly	10. 0	
Accelerometer	1. 0	
Guidance and control electronics	13. 0	
Canopus sensor (2)	12. 0	
Fine sun sensor Coarse sun sensor (4)	1. 0	
Earth detector	0. 3	
Reaction control assembly	140. 0	
High-gain drive assembly	32. 0	
Medium gain drive assembly	17. 0	
TVC actuator (2)	36. 0	
Limb and terminator crossing detector (2)	1. 2	
Antenna drive electronics	5. 0	

Table 16. Spacecraft Propulsion Weight

Item	Weights, lb	
Structure and Mechanical		923.6
Primary structure	671.5	
Reaction control supports	11.7	
Meteoroid protection	226.1	
Attachments and miscellaneous	14.3	
Temperature Control		73.9
Insulation	68.8	
Heaters and thermostats	2.0	
Attachments and miscellaneous	3.1	
Engine and Valves		586.1
Injector	29.3	
Combustion chamber assembly	421.6	
Injector pintle actuator	4.0	
Propellant lines and ducts	13.9	
Cabling set	11.0	
Instrumentation	7.4	
Gimbal assembly	27.2	
Hardware-engine integration	9.4	
Trim orifice-fuel	1.8	
Trim orifice-oxidizer	1.8	
Cavitating venturis (2)	2.0	
1-inch explosive valves (4)	12.0	
3/8-inch explosive valves (14)	11.2	
Solenoid valves (16)	28.0	
Quad check valves (2)	1.0	
Trim orifice (2)	0.5	
Filter (4)	4.0	
Propellant Feed System		528.6
Pressurization System		438.4
Valves, regulators, etc.	33.2	
Plumbing and fill and vent	13.2	
Tank (1)	392.0	

Table 16. Spacecraft Propulsion Weight (Continued)

Item	Weights, lb	
Tank and Engine Supports		161.8
Start Tank Assembly		22.6
Contingency		128.0
Spacecraft Propulsion Dry Weight		(2863)
Spacecraft Propulsion Inert Fluids		(476)
Residuals, propellant feed	424.1	
Residuals, start tank	12.2	
Helium	40.0	
Spacecraft Propulsion Inert Weight		3339

Table 17. Weight Sequence for the 1971 Mission

Item		Weights, lb
Planetary vehicle with adapter		20,922
Planetary vehicle adapter		-422
Separated planetary vehicle		20,500
Propellant for interplanetary trajectory correction (200 meter/sec)		-1,400
Planetary vehicle prior to Mars arrival		19,100
Jettisoned capsule canister lid		- 260
Planetary vehicle at initiation of orbit insertion		18,840
Propellant for Mars orbit insertion (2.16 km/sec)		-9,654
Planetary vehicle at orbit insertion burnout		9,186
Planetary vehicle orbit trim propellant (100 meter/sec)		- 320
Planetary vehicle prior to capsule separation		8,866
Separated capsule vehicle		-2,490
Flight spacecraft in orbit		6,376
Unseparated flight capsule	250	
Spacecraft propulsion inert weight	3,340	
Spacecraft science payload	400	
Spacecraft bus inert weight	2,386	
	6,376	

e. Science Weights

The allocated weight for the spacecraft science subsystem is divided nominally as follows. Additional weights data breakdown is given in 9 of Section III.

	Weight, lb
Science experiments on the PSP	100
Science mounted on the bus	140
Science support	160
Total	400

f. Jettison Weight for Emergency Capsule Separation

The weight forward of the emergency capsule separation joint is jettisoned by the emergency capsule separation device if commanded. This consists of the following:

Flight capsule	3,000
Capsule interstage	149
	3,149

g. Planetary Vehicle Adapter

The weight for the planetary vehicle adapter is estimated to be as follows:

	Weight, lb
Umbilical system	8
Telemetry sensors and attachments	5
Separation system	32
Structural and mechanical subsystem	349
Doublers (shroud)	28
Total estimated weight	422
PV adapter allocation	1500
Weight margin	1078

5.1.2 Mass Properties

The time histories for the centroidal moments of inertia have been determined computationally for the flight system from planetary vehicle separation to end of mission. Table 18 lists the moments of inertia about the pitch, yaw, and roll axes shown in Figure 4. Also included in Table 18 are longitudinal center of mass values which are measured from spacecraft station 0 (20 inches below the launch vehicle/planetary vehicle field joint).

Table 18. Mass Properties

Condition	Weight, lb	Longitudinal* Center of Mass Station	Moments of Inertia, slug-ft^2		
			I_x	I_y	I_z
Flight spacecraft in orbit	6,376	141	5,486	4,518	7,761
Planetary vehicle after orbit insertion	9,186	190	20,061	19,094	10,676
Planetary vehicle prior to Mars arrival	19,100	170	26,596	25,623	21,090
Planetary vehicle at separation	20,500	170	27,211	26,157	21,279
Total weight installed	20,922	168	28,551	28,420	21,840

*Measured from spacecraft station 0.

The spacecraft center of mass has been calculated using weights data given in 5.1.1. The externally mounted equipment locations are shown in Figure 18 and layout data for equipment panels is given in 3.2.5 above. Figure 21 shows the radial center of mass offset during the flight. A 0.5-inch tolerance is required at retropropulsion burnout for attitude control performance, and this will be realized by established mass balance control techniques.

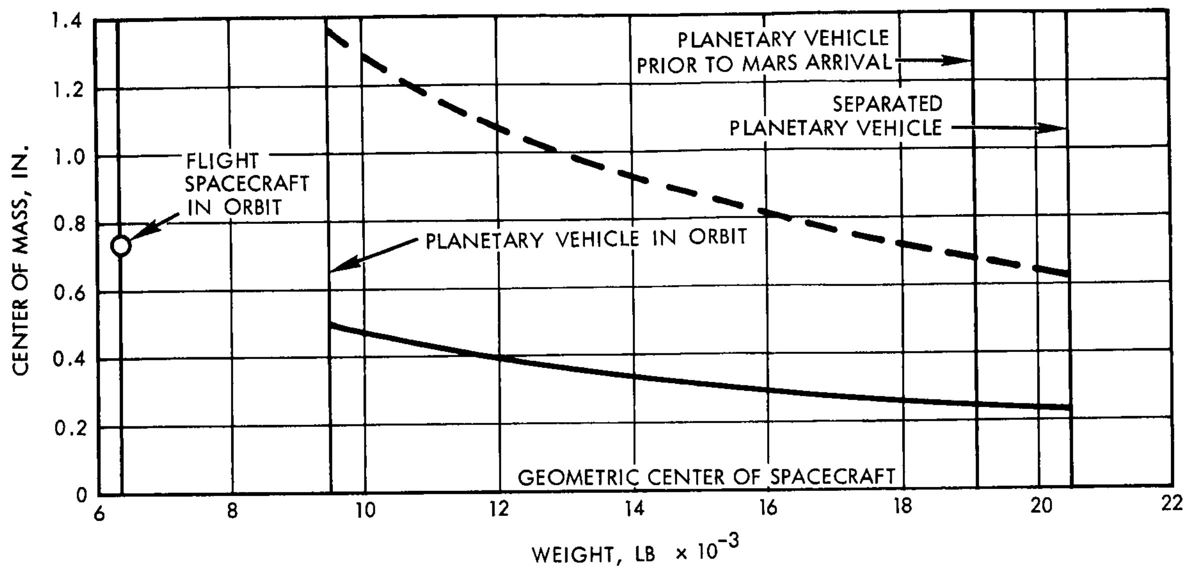


Figure 21. Voyager Radial Center of Mass Envelope

5.1.3 Propellants

a. Interplanetary Trajectory Correction and Mars Orbit Trim Propulsion

The interplanetary trajectory correction and Mars orbit trim propulsion utilizes the LEM descent engine, propellant feed system and propellants from the main Mars orbit insertion propulsion subsystem. The propellant weights are based on an I_{sp} = 285 seconds and ΔV = 200 meter/sec for interplanetary trajectory correction and ΔV = 100 meter/sec for Mars orbit trim.

b. Mars Orbit Insertion Propellants

The propellant weight of 9654 pounds for Mars orbit insertion is based on maximum tank loading within the flight spacecraft total weight allocation. This will yield a ΔV of 2.16 km/sec for a specific impulse of 305 seconds, with an initial weight of 18,840 pounds as shown in Table 17. The spacecraft tankage volume capability allows for an additional propellant load of approximately 6150 pounds.

c. Propellant Residuals

Trapped propellants for the current LEM descent propulsion system amount to 455 pounds. The trapped propellants for the Voyager spacecraft propulsion have been reduced from this by 30 pounds as follows:

Changes in tank outflow lines	10 pounds
Elimination of heat exchanger	5 pounds
Elimination of tank baffles	15 pounds

5.1.4 Balance Weights

Although equipment has been placed to minimize the center of mass offset, an estimate of 15 pounds for balance weights is included in the spacecraft bus inert weight of Table 14 to account for possible design changes after equipment locations are fixed, and to reduce the center of mass travel when appendages are moved from the stowed to the deployed location during spacecraft flight.

5.1.5 Contingency

A contingency has been added to the spacecraft bus and propulsion subsystem nominal weights. The contingency allows for uncertainties in weight estimation techniques and slight modifications of the design. It also includes an allowance for expected weight growth during design completion and the development phase of the spacecraft. The allocated contingency has been taken as 6 per cent for the bus and 4 per cent for the spacecraft propulsion. These contingencies reflect the over-all level of confidence of the weight estimates and are consistent with the current level of design. The contingency allocated to the propulsion inert weight is less than that required for the bus because many of the propulsion components are in an advanced state of development.

5.2 Power Profile

Estimated average power load requirements are listed in Table 19 as a function of mission phases. Peak power requirements are listed in Table 20.

Table 19. Estimated Average Power Profile (watts)

Subsystem Identification	Loads	Form	Prelaunch	Launch and Injection	Acquisition	Cruise (after battery charging)	Trajectory Corrections	Orbit Insertion	Planetary Vehicle Orbital Operations (with battery charging)	Planetary Vehicle Orbit Trim	Capsule Separation	Flight Spacecraft Orbit Trim	Flight Spacecraft Orbital Operations: Sunlight	Flight Spacecraft Orbital Operations: Eclipse	NOTES
Radio	Electronics	4 kc	14	20	20	12	12	12	12	12	14	12	12	12	
Telemetry	"	"	5	5	5	5	5	5	5	5	5	5	5	5	
Command	"	"	13	13	13	13	13	13	13	13	13	13	13	13	
C and S	"	"	20	20	20	20	20	20	20	20	20	20	20	20	
Data storage	Tape recorders	"	17	0	0	7	4	4	17	4	17	4	17	17	33% duty cycle (maneuvers)
Science	Instr. and DAE	"	14	14	14	57	57	57	78	78	78	78	78	78	
GCS	Electronics	"	27	27	29	27	33	37	27	32	29	32	29	29	
Subtotal	4 kHz Inv. output	"	110	99	101	141	144	148	172	164	176	164	174	174	
	4 kHz Inv. input	50 VDC	128	118	120	165	168	173	201	192	206	192	203	203	85% max. eff.
Thermal Control	Heaters	"	0	5	15	44	54	54	24	34	34	34	34	34	
	RCS Heaters	"	0	36	36	(128)	36	36	(128)	36	36	36	(128)	36	128 W nonessential
Capsule	Unspecified	"	0	0	0	200	0	0	200	0	0	0	0	0	
Radio	TWTA	"	0	0	0	150	150	150	150	150	150	150	150	150	
GCS	Gyro Inv. input	"	0	0	8	0	8	8	0	8	8	8	8	8	
CCS	400 Hz Inv. input	"	1	1	2	2	10	56	3	3	3	3	3	3	
Propulsion	Gimbal actuators		0	0	0	0	(18.5 w-hr)	(70 w-hr)	0	(4.3 w-hr)	0	(3 w-hr)	0	0	Direct from batt. bus Energy in w-hr
	Valves	29-42 VDC	0	0	0	0	(0.2 w-hr)	(0.6 w-hr)	0	(1.5 w-hr)	0	(1 w-hr)	0	0	
Total	Loads	50 VDC	129	160	181	561	426	477	578	423	437	423	398	434	Excluding gas heaters and propulsion spikes
Power	Batt. + Chg. Reg.	"	0	0	0	0	0	0	125	0	0	0	125	0	Cont. batt. charging
	Boost Reg.	"	32	40	45	6	106	119	6	106	109	106	6	108	80% efficiency
	PCU	"	10	10	10	10	10	10	10	10	10	10	10	10	
Solar array	Output required	"	-	-	-	577	-	-	719	-	-	-	539	-	Excl. gas heaters
Battery	Output required	29-42 VDC	171	210	236	-	542	606	-	539	556	539	-	55-2	Excl. propulsion spikes
Battery Output	With science off							524						440	Excl. propulsion spikes

Table 20. Estimated Peak Power Profile (watts)

Subsystem Identification	Loads	Form	Prelaunch	Launch and Injection	Acquisition	Cruise (with battery charging)	Trajectory Corrections	Orbit Insertion	Planetary Vehicle Orbital Operations (with battery charging)	Planetary Vehicle Orbit Trim	Capsule Separation	Flight Spacecraft Orbit Trim	Flight Spacecraft Orbital Operations Sunlight	Flight Spacecraft Orbital Operations Eclipse	Notes
Radio	Electronics	4 kc	23	20	20	12	12	12	12	12	14	12	12	12	
Telemetry	"	"	5	5	5	5	5	5	5	5	5	5	5	5	
Command	"	"	13	13	13	13	13	13	13	13	13	13	13	13	
C and S	"	"	20	20	20	20	20	20	20	20	20	20	20	20	
Data storage	Tape recorders	"	60	0	0	12	12	12	60	12	60	12	60	60	
Science	Instr. and DAE	"	135	14	14	57	57	57	135	135	135	135	135	135	
GCS	Electronics	"	69	69	74	72	80	80	72	80	74	80	74	74	
Subtotal	4 kHz Inv. output	"	325	141	146	191	199	199	317	277	321	277	319	319	
	4 kHz Inv. input	50 VDC	382	167	173	224	233	233	372	325	376	325	374	374	
Thermal Control	Heaters	"	0	10	20	49	59	59	49	59	49	59	59	59	
	RCS Heaters	"	0	36	36	(128)	36	36	(128;	36	36	36	(128)	36	128 W nonessential
Capsule	Unspecified	"	0	0	0	200	0	0	200	0	0	0	0	0	
Radio	TWTA	"	0	0	0	150	150	150	150	150	150	150	150	150	
GCS	Gyro Inv. input	"	0	0	8	0	8	8	0	8	8	8	8	8	
GCS	400 Hz Inv. input	"	1	1	38	38	66	66	53	53	53	53	53	53	
GCS	Valves	"	0	0	48	48	118	118	48	118	48	118	48	48	
Total	Loads	50 VDC	383	214	323	709	670	670	872	749	720	749	692	728	Excluding propulsion spikes
Power	Batt. + Chg. Reg.	50 VDC	0	0	0	125	0	0	125	0	0	0	125	0	
	Boost Reg.		97	54	81	6	170	170	6	187	180	187	6	182	
	PCU	50 VDC	10	10	10	10	10	10	10	10	10	10	10	10	
Solar array	Output required	50 VDC	-	-	-	850	-	-	1013	-	-	-	833	-	Excluding gas heaters
Battery output	Required	29-42 VDC	490	278	414	-	850	850	-	946	910	946	-	920	Excluding propulsion spikes

The power requirement of 90 watts for reaction control gas heaters is shown as a nonessential load and is not included in the total figures. The control subsystem gas heaters are not essential to the success of the mission and are used to increase lifetime only when the solar array capability exceeds the essential load requirements by a suitable margin.

5.3 Engineering Measurements

This section compiles and describes the engineering measurements which will be transmitted from the spacecraft via the TDS to the MOS. The same test points will also be available for ground testing via the telemetry link. Scientific data to evaluate the experiments, and engineering data to evaluate the spacecraft will be transmitted either separately or combined, depending on the telemetry data mode. The available modes and their switching are described in 5 of Section II, Volume 2.

5.3.1 Measurement Priorities

Because of the limitation on number and sampling rates of subcarrier channels, a priority basis for selecting measurement has to be established. Four categories are utilized in the present discussion:

a) Measurements required for the performance of flight operations. These include measurements for diagnostic purposes to select either commanded alternate modes of operation or a degraded mission.

b) Measurements required to verify the performance of specific flight spacecraft and subsystem functions

c) Measurements to indicate the effect of the space environment on the flight spacecraft performance

d) Data to evaluate critical components, in particular those newly developed for Voyager.

The most important engineering measurements function is to report anomalies to the earth, which can be corrected from there. In most cases, this will involve switching in redundant equipment by ground command. Other cases may involve thrust or antenna angle changes. Second telemetry priority is assigned to system diagnosis. If a component or subsystem degradation is diagnosed at an early state, it may be possible to select an alternate mode from the ground to save the over-all mission, such as temporary shutoff of a circuit, etc.

5.3.2 Summary of Channel Requirements

Table 21 shows a summary of the required subcarrier channels listed by subsystem, signal types, and sampling rates. It is based on telemetry Mode I (engineering data only) at a bit rate of 234 bits per second. At this bit rate, commutation and subcommutation results in the sampling rates shown. (One sample every 1.2 seconds, 24 seconds, 4 minutes, and 8 minutes, respectively.) The telemetry system as presently configured can accommodate 27, 70, 126, and 209 channels of the four sampling rates. It can be seen that some surplus of fast channels is yet available. Every fast channel can be subcommutated into up to 20 slow channels, so that the excess required of the latter is not critical.

An additional telemetry mode is proposed to combine selected engineering data into one frame, which could be sampled fast, but only during short periods, such as before maneuvers, separation, etc.

5.3.3 Telemetry Lists

The preliminary list of all spacecraft telemetry points is shown in Table 22. It shows each subsystem's requirements further broken down to the assembly level. The type of telemetry can be either analog (voltages, temperatures, etc.), binary (encoders, memory, etc.), or discretes (switches, on-off, etc.). Binary telemetry is done by 7-bit words, so that longer words, memory for instance, use more than one subchannel. The use of 7 bits per subchannel on the other hand, allows telemetry of seven discrete functions on one subchannel. The type of telemetry [(analog (A), binary (B), or discrete (D)] is shown in the third column. The fourth column lists the sampling rate, based on telemetry mode I (engineering data) at 234 bits per second. At other bit rates or at the engineering-science mix modes, these sampling rates will, of course, differ from the four rates shown. The fifth column lists the numbers of channels required, including separate channels for redundant hardware. The discrete subchannels can be added up later for full efficiency, but presently there has been no effort to mix subsystems on discrete subchannels. The sixth column shows redundant hardware or functions, as applicable. The last column lists the priority assigned to the telemetry point as defined above.

Table 21. Telemetry Summary, Engineering Measurements List

	Sample every 1.2 sec			Sample every 24 sec			Sample every 4 min			Sample every 8 min		
	A	B	D	A	B	D	A	B	D	A	B	D
S-Band Radio	3			3		1 (3)	5		3 (21)	32		
Power				15			38		5 (32)	13		
G and C	7	1	1 (3)	10		3 (19)	10	7	2 (9)	27		1 (3)
TM and DS						2 (12)			2 (12)	24		
C and S					8					12	12	
Command		4			4					12		
Propulsion	3			4		3 (19)	30		3 (20)			
Pyro						2 (8)			2 (13)	2		
Thermal									1 (1)	41		
Structure							10					
	13	5	1	32	12	11	91	7	18	163	12	1

Note: 1) A = analog, B = binary, D = discrete

2) One D channel can accommodate up to seven functions (bracketed)

3) Sampling rates listed are at 234 bits/second in Mode I (engineering data only)

Table 22. Engineering Measurements, Flight Performance

Subsystem	Measurement Description					Redundant Function (or Degraded Mission)	Priority
	Function or Unit		Type (see Note 1)	Sampling Rate	No. of Channels (See Note 2)		
S-band radio	1-watt transmitter and power monitor	output	A	1/4m	1		a
		voltage (2)	A	1/8m	2		b
	Modulator exciter (2)	output	A	1/4m	2	Other exciter	a
		voltage (2)	A	1/8m	4		b
	Power amplifier, power supply and RF power monitor	output	A	1/4m	2	Other amplifier or Low power transmitter	a
		helix current	A	1/8m	2		b
		temperature (2)	A	1/8m	4		c and d
		voltage (2)	A	1/8m	4		c and d
	S-band receiver (3)	in-lock	D	1/24s	3^{x}	Other receiver	a
		signal strength	A	1/24s	3		b
		loop stress	A	1/1.2s	3		a
		VCO temperature (freq)	A	1/8m	3		b
		temperature	A	1/8m	3		c
		voltage (2)	A	1/8m	6		c
	Receiver selector	mode	D	1/4m	5^{x}		a
		voltage (2)	A	1/8m	2		c
	Transmitter selector	mode	D	1/4m	8^{x}		a
		voltage (2)	A	1/8m	2		c
	Circulator switch (4)	state	D	1/4m	8^{x}		a
Power	Power control unit	oscillator voltage (2)	A	1/24s	2	Standby oscillator	b
		voltage (6)	A	1/8m	6		a
		current (2)	A	1/8m	2		c and d
		oscillator, main-standby	D	1/4m	1^{x}		c and d
	Battery (3)	voltage	A	1/24s	3	Disconnect defective battery	a
		current	A	1/24s	3		a
		current sign	D	1/4m	3^{x}		a
		temperature (2)	A	1/4m	6		b
		charge control	D	1/4m	3^{x}		
	Battery regulator (3)	current	A	1/4m	3	Two charge rates	a
		voltage (2)	A	1/4m	6		b
	4.0-kc and 400 cps inverters (2x2)	current in	A	1/4m	4	Redundant pairs	b
		output voltage	A	1/4m	6		a
		output current	A	1/24s	6		a
		temperature	A	1/8m	4		b and c
		sync	D	1/4m	1^{x}		a
	Power distribution box	switch state	D	1/4m	27^{x}		a
		voltage (4)	A	1/4m	4		b
		temperature	A	1/8m	1		c
	Solar array	current	A	1/24s	1		b
		temperature (8)	A	1/4m	8		b
		shunt current	A	1/4m	1		c
Guidance and Control	Gyro (3)	rate or position	A	1/1.2s	3	Derived rate, free roll	a
		motor	D	1/4m	3^{x}		b
		temperature	A	1/8m	3		c&d
	Gyro Torquer	On/Off	D	1/4m	1^{x}		a
	Sun sensor	pitch output	A	1/1.2s	1		a
		yaw output	A	1/1.2s	1		a
		fine/coarse gate	D	1/24s	1^{x}		a
		temperature (2)	A	1/8m	1		c&d
	Canopus sensor	roll error	A	1/1.2s	1	Roll override, acquisition gate override	a
		star intensity	A	1/1.2s	1		a
		sun present	D	1/24s	1^{x}		b
		star present	D	1/24s	1^{x}		b
		Tracking position	B	1/4m	1		b
		temperature	A	1/8m	1		c&d
	Accelerometer	output	B	1/1.2s	1		a
		voltage (4)	A	1/8m	4		c&d
	Derived rate filter (3)	control signal	A	1/24s	3		a
	Switching amplifier (3)	output	D	1/1.2s	3^{x}		a
	Mode control monitor	mode (4)	D	1/24s	4^{x}		a
	Earth sensor	output	D	1/4m	1^{x}		a
		voltage (2)	A	1/8m	2		c
		temperature	A	1/8m	1		d
	Limb and Terminator crossing sensor (2)	output	D	1/4m	2^{x}		a
		voltage (2)	A	1/8m	4		c
		temperature	A	1/8m	2		d
	Roll Integrator	output	A	1/24s	1		b
		voltage (2)	A	1/8m	2		c&d
	Pressure vessel (2)	temperature	A	1/4m	2	2 redundant	c&d
		pressure	A	1/24s	2		a
		regulated pressure	A	1/4m	2		b
	TVC Actuator (2)	position	A	1/24s	2	Half acceleration	b
		control	A	1/24s	2		b
	Thrusters (12)	on/off	D	1/24s	12^{x}		b
	Thruster level	high/low	D	1/4m	1^{x}		c

Table 22. Engineering Measurements, Flight Performance (Continued)

Subsystem	Measurement Description: Function or Unit		Type (see Note 1)	Sampling Rate	No. of Channels (see Note 2)	Redundant Function (or Degraded Mission)	Priority
Guidance and Control (cont)	High Gain Drive Assembly	shaft position	B	1/4m	2	Other antenna	a
		hinge position	B	1/4m	2		a
		actuator power (2)	D	1/8m	2^{x}		c
		actuator temperature (2)	A	1/8m	2		d
	Medium Gain drive assembly	hinge position	B	1/4m	2	Low-gain antenna	a
		actuator power	D	1/8m	1^{x}		c
		actuator temperature	A	1/8m	1		d
Computer and Sequencer (2)	Timer and register (3)	time and mode	B (21 bits)	1/24s for every 21 bit position	6	Two redundant	a*
	Memory (256)	word	B (21 bits)				a
		output occurrence	B	1/24s	2		a
		function generator word	B (21 bits)	1/8m	3		c
		supply voltage (4)	A	1/8m	8		c&d
		current	A	1/8m	2		c&d
		temperature	A	1/8m	2		c&d
		velocity control word	B (21 bits)	1/8m	6		b
Telemetry	Digital unit (2)	calibration voltage (2)	A	1/8m	4	Two redundant	c
		temperature	A	1/8m	2		d
	Analog unit (2)	voltage (3)	A	1/8m	6		c
Data Storage	Recorder (6)	tape in motion	D	1/24s	6^{x}		a
		read mode	D	1/4m	6^{x}		a
		write mode	D	1/4m	6^{x}		a
		end of tape	D	1/24s	6^{x}		a
		temperature	A	1/8m	6		c
		pressure	A	1/8m	6		c
Command	Command detector (2)	bit sync	B (10 bits)	1/24s	4		a
		voltage (2)	A	1/8m	4		c&d
		temperature	A	1/8m	2		c&d
	Decoder (2)	voltage (2)	A	1/8m	4		c
		temperature	A	1/8m	2		c&d
		telemetry buffer	B (12 bits)	1/1.2s	4		a
Propulsion Module	Injector inlet pressures (2)		A	1/1.2s	2		a
	Chamber pressure		A	1/1.2s	1		a
	Engine inlet pressures (4)		A	1/24s	4		b
	Other pressures (11)		A	1/4m	11		b
	Temperatures (19)		A	1/4m	19		a
	Insector pintle actuator	position	D	1/24s	1^{x}		b
		current	D	1/4m	1^{x}		
Propulsion Module (cont)	Explosive valve (1)	position	D	1/4m	1^{x}	Automatic or ground command	a
	Explosive valve (2)	position	D	1/4m	2^{x}		a
	Explosive valve (18)	position	D	1/24s	18^{x}		a
	Solenoid valve assembly (4 x 4)	position	D	1/4m	16^{x}		a
Temperature Control	Temperatures (40)		A	1/8m	40		b
	Capsule current	on-off	D	1/4m	1^{x}		b
		current	A	1/8m	1		a
Pyro	Safe (2)	open/close	D	1/4m	2^{x}	Separation switch, C and S, command	a
	Firing (8)	voltage	D	1/24s	8^{x}		a
	LV Separation	switch state (3)	D	1/4m	3^{x}		a
	Capsule Emergency Separation	switch state (3)	D	1/4m	3^{x}		a
	Science deployed	switch states	D	1/4m	4^{x}		a
	Magnetometer deployed	switch state	D	1/4m	1^{x}		a
	Low gain antenna deployed	switch state	D	1/4m	1^{x}		a
	Temperatures (2)		A	1/8m	2		c
Structure and Mechanical	Strain Gage (10)	current	A	1/4m	10		c

Note: 1) A = analog B = binary D = discrete

2) One telemetry channel can handle up to seven D functions

*C and S provides logic to change words for commutator through memory + register + code

Table 23 lists the major mission events and the manner by which they will be telemetered. It complements the detailed mission sequence (see 5.4). Every mission phase is detailed into its major events and the telemetry of start and completion of these events is described.

In an alternative column, the redundancy is listed. There are different redundancy possibilities and the redundancies listed are classified in three categories: function (F), measurement (M), initiation (I). If a redundant function or initiation is available, it implies an alternate telemetry point.

5.3.3 Design Features

No single rule can be set on how to telemeter spacecraft events. Wherever possible, a normal equipment function is being used to telemeter significant event changes. A prime example of this class is engine start and stop which can be observed by telemetering chamber pressure. Other events need special telemetry transducers, for instance the deployment of the low-gain antenna. The initiation of this event can be telemetered by verification of arming and firing, but the completion of deployment needs at least one displacement switch at a well chosen boom position. The telemetering of fast, one-time events, for instance firing, requires additional implementation. It is impossible to sample so rapidly, that no firing current can be overlooked, in spite of its duration of only 50 milliseconds. Therefore, buffering and storage of the firing current is provided so that the event is sure to show up as a step function when the telemetry commutator samples this point every 24 seconds in Mode I.

Initiation of an event is generally telemetered by the output indication of the computer and sequencer. This is a binary word which shifts one digit for every executed command. In addition, most event initiation functions have their own telemetry points, such as firing circuit current on pyrotechnic devices, engine pressure on propulsion, etc.

Table 23. Engineering Measurements, On-Board Event Verification

No.	Mission Phase	Unit, Function, or Event	Start or Complete	Measurement Description: Source	Measurement Description: Type	Alternatives: Redundant Function (F) Measurement (M) or Initiation (I)
1.	Launch vehicle separation	PV No. 1 separation (PV No. 2 identical)	S	L/V telemetry	D (discrete)	
2.			C	PV separation switch	D	
3.		Computer and Sequencer enable	S	Separation switch	D	Command (I)
4.			C	C and S action and verification	B (binary)	
5.		Actuate "safe" circuit		C and S	B	Separation (I), command (I)
6.		Arm science and antenna EED's	S	C and S	B	Command (I)
7.			C	Charging circuit	A (analog)	
8.		Deadband fine/coarse	S	C and S		Command (I)
9.			C	gyros	A	Derived rate (F)
10.		Attitude hold	S	separation switch	D	C and S (I)
11.			C	gyros	A	Derived rate (F)
12.		Thruster level (high/low)	S	C and S	B	Command (I)
13.			C	high/low switch	D	
14.		Thruster firing	S	switching amplifier	D	
15.			C	jet transducer	D	Dual system (F)
16.		Low gain antenna deploy	S	firing circuit	D	C and S or command (I)
17.			C	antenna switch	D	
18.		Science deploy	S	firing circuits	D	C and S or command (I)
19.			C	position switches	D	
20.		Release high- and medium-gain antennas	S	firing circuits	D	
21.			C	antenna drives	B	Position switches (M)
22.	Celestial reference acquisition	Sun detect coarse sensor		detector outputs	A (pitch and yaw)	
23.		Sun acquire fine sensor	S	acquisition gate	D	
24.			C	detector outputs	A	
25.		Solar array use		current monitor	A	Battery charge monitor (M)
26.		Roll spin mode (0.22 deg/sec)	S	sun acquisition	A	Command (I)
27.			C	C and S (time)	D	Intermittent jet firing (F)
28.		Canopus acquisition (0.1 deg/sec roll)	S	Canopus tracking on	D	C and S or Command (I)
29.			C	earth detector	D	Canopus sensor error (M)
30.		Uncage Planetary Scan Platform	S	C and S	B	Command (I)
31.			C	position switch	D	
32.	Cruise	Science Mode Switch	S	C and S	B	Command (I)
33.			C	resp. instrument	D	
34.		Arm emergency capsule separation		S and A	D	C and S or Command (I)
35.		Power amplifier on	S	C and S	B	Command (I)
36.			C	power monitor	A	Second amplifier (F)
37.		Calibrate science	S	C and S	B	
38.			C	Science TM		
39.	Interplanetary trajectory correction	Start: MOS commands	S	C and S verify	B	gyros ON (M)
40.		Telemetry mode switch	C	PCM	D	Second PCM (F)
41.		Switch to low gain antenna	C	Circulator switch	D	Signal level on ground (M)
42.		Reorient high gain antenna	S	Command	B	
43.			C	Antenna encoder	B	C and S verify (M)
44.		Arm propulsion EED's		S and A	D	
45.		Engine start	S	firing circuits	D	C and S (I)
46.			C	Chamber pressure	A	
47.		Recorder start write	S	C and S	D	Command (I)
48.			C	Recorder	D	
49.		Roll/Pitch turn	S	Torquer	D	
50.			C	Gyro	A	switching amplifier (M)
51.		High/low thrust (Low/high gain)	S	Command verify	B	
52			C	Pintle and gimbal actuator	D	
53.		Sun and star acquisition	same as above			
54.		Stop: Engine stop	S	Accelerometer counter	B	firing circuit (M)
55.			C	chamber pressure	A	
56.		Down link to directional antenna	S	C and S	B	Command (I)
57.		Recorder start playback	S	C and S	B	
58.			C	Recorder	D	
59.		Limb and Terminator crossing sensor on	S	C and S	B	Command (I)
60.			C	Sensor output	D	
61.	Capsule separation	Preseparation commands		C and S verify	B	
62.		Switch relay link	S	C and S	B	
63.			C	Capsule TM	D	
64.		Separation	S	C and S	B	Command (I)
65.			C	Separation switches	D	Capsule TM (M)

NOTE: Events are shown only once, even though they may repeat (cruise, orbit trim, etc.).

5.4 Mission Sequence

A detailed presentation of the sequence for the 1971 Voyager mission is given in Table 24 to complement the narrative discussion presented in 1 above. The table presents mission events in chronological sequence with related times, signal source, signal destination, and signal backup as appropriate. Remarks are added for explanation as required.

5.5 Component Design Parameters

The flight spacecraft components are listed in Table 25 along with associated design parameters. These parameters include weight, volume, electrical power requirements, allowable operating and nonoperating temperature limits, and a magnetic properties allocation as appropriate. The components are listed against an item number that has been defined for general use throughout the spacecraft project, and in particular for program planning identificantion and correlation.

Table 24. 1971 Mission Sequence for the Spacecraft System

Event	Time	Signal Source	Signal Destination	Signal Backup	Remarks
1. Prelaunch					Two of the three spacecraft delivered to AFETR are processed as shown. The third spacecraft is checked out but not mated to the launch vehicle. An STC associated with each spacecraft has been received and checked out previously. STC denotes systems test complex
1.1 Spacecraft Assembly Facility (SAF) Operations	T - 17 wk				
a) Receive spacecraft at AFETR Air Strip					
b) Transport spacecraft to SAF					
c) Perform spacecraft receiving inspection					
d) Mate spacecraft to vertical spacecraft dolly					
e) Install test capsule					
f) Perform spacecraft functional test					
g) Perform battery capacity tests					
h) Perform solar array tests					
i) Perform IST and critique					
j) Remove test capsule					
k) Prepare spacecraft for movement outside SAF					
1.2 Explosive Safe Facility (ESF) Operations	T - 14 wk				
a) Move spacecraft to ESF					
b) Verify G&C alignments					
c) Conduct spacecraft weight and center of mass checks					
d) Perform separation and release tests of spacecraft appendages					Using live ordnance
e) Install ordnance simulators					
f) Install flight capsule					
g) Perform spacecraft-capsule compatibility test					
h) Perform IST and critique					
i) Perform science quiet test					
j) Perform spacecraft-capsule vertical alignment test					
k) Perform final in-hanger button-up operations					
l) Mate planetary vehicle to flight shroud section					
m) Perform shroud cooling system test					
n) Perform modified IST and compatibility tests					
o) Perform mock countdown					
p) Prepare encapsulated planetary vehicles for movement outside ESF					
1.3 Vertical Assembly Building (VAB) Operations	T - 9 wk				
a) Move individual encapsulated planetary vehicles to VAB					
b) Mate encapsulated planetary vehicles to booster					
c) Perform flight shroud-booster alignment					
d) Perform planetary vehicle IST					
e) Perform RFI tests					
f) Perform flight readiness demonstration test					
g) Perform mock countdown test					
h) Perform simulated flight test					
i) Demate and prepare encapsulated planetary vehicles for movement outside VAB					
1.4 Explosive Safe Facility (ESF) Operations	T - 5 wk				
a) Move individual encapsulated planetary vehicles to ESF					
b) Flight pressurize all systems					
c) Install flight ordnance					
d) Load spacecraft propellants					
e) Perform leak tests					
f) Perform IST and critique					
g) Perform planetary vehicle surface sterilization and seal					
h) Prepare encapsulated planetary vehicles for movement outside of ESF					

Table 24. 1971 Mission Sequence for the Spacecraft System (cont.)

Event	Time	Signal Source	Signal Destination	Signal Backup	Remarks
1.5 Pad Prelaunch Operations	T - 2 wk				
a) Move individual encapsulated planetary vehicles to pad					
b) Mate encapsulated planetary vehicle to booster					
c) Perform flight shroud-booster alignment check					
d) Perform planetary vehicle IST					
e) Perform RFI tests					
f) Perform J-FACT					
g) Perform flight readiness demonstration test					
h) Perform mock countdown					
i) Perform pre-countdown					
2. Launch and Injection					
2.1 Launch Countdown					
a) Set launch configuration for both PV's		LCE	Flight spacecraft		Launch complex equipment (LCE) establishes all spacecraft systems in launch configuration (see paragraph 1.2.1)
b) Disconnect umbilicals from planetary vehicle to launch complex equipment		LCE	Flight spacecraft		
2.2 Powered Boost Flight					Times represent a typical flight only
a) First stage (SIC) ignition					
b) Hold down release	T				T is time of holddown release
c) Start gravity turn	T + 12 sec				
d) First stage center engine cutoff	T + 146 sec				
e) First stage outboard engines cutoff	T + 151 sec				
f) Jettison SIC	T + 152.5 sec				
g) Second stage (SII) ignition	T + 152.5 sec				
h) Jettison SIC/SII interstage	T + 167.5 min				
i) Jettison forward fairing section (approximately 350,000 ft)	T + 203 sec				
j) Second stage cutoff	T + 546.5 sec				
k) Jettison second stage	T + 548 sec				
l) Third stage (SIV) ignition	T = 548 sec				
m) Third stage cutoff	T + 701.4 sec				
2.3 Earth Parking Orbit Coast Period					2 to 90 minutes depending on launch day and time
2.4 Powered Injection Flight					
a) Third stage re-ignition					
b) Third stage cutoff					
2.5 LV-PV Separation					
2.5.1 Forward PV separation					
a) Transmit separation firing signal	S_1	Launch vehicle	Forward PV separation device		S_1 is time of forward PV separation
b) Initiate post separation program in C&S	S_1	Separation switch	C&S	MOS	C&S is computing and sequencing subsystem
c) Enable G&C rate control	S_1	Separation switch	G&C	C&S	G&C is guidance and control
d) Arm spacecraft main pyrotechnic bus	S_1	Separation switch	Pyrotechnic subsystem	C&S	
e) Enable G&C attitude hold	S_1 + 10 sec	C&S	G&C		
2.5.2 Mid-fairing foldback to uncover aft PV					
2.5.3 Aft PV separation	S_1 + 200 sec	Launch vehicle	Aft PV separation device		After allowing time for PV to clear Repeat 2.5.1
2.5.4 Third stage (SIVB) retrothrusts					If required
3. Celestial Reference Acquisition					The remaining sequence is accomplished independently for each spacecraft. S is separation time for each spacecraft.
3.1 Establish Spacecraft Configuration					
a) Deploy magnetometer boom	S+	C&S	DAE	MOS	DAE is data automation equipment in the science subsystem
b) Deploy science antennas	S+	C&S	DAE	MOS	
c) Release and deploy planetary scan platform (PSP)	S+	C&S	Pyrotechnic subsystem and DAE	MOS	
d) Release medium-gain antenna	S+	C&S	Pyrotechnic subsystem	MOS	
e) Release high-gain antenna	S+	C&S	Pyrotechnic subsystem	MOS	
f) Turn on power to science subsystem	S+	C&S	Power distribution unit	MOS	
g) Initiate cruise science sequence	S+	C&S	DAE	MOS	Solar flare mode automatic with flare occurrence
h) Enter telemetry mode 2 (engineering and science) 234 bit rate	S+	C&S	Telemetry subsystem	MOS	

Table 24. 1971 Mission Sequence for the Spacecraft System (cont.)

Event	Time	Signal Source	Signal Destination	Signal Backup	Remarks
i) Release and deploy low-gain antenna (aft pointing)		C&S	Pyrotechnic subsystem and G&C	MOS	
j) Position high-gain and medium-gain antennas to be earth pointing after celestial reference acquisition		C&S	G&C	MOS	
3.2 Sun Acquisition					
a) G&C initiates sun acquisition					G&C switches automatically to sun acquisition mode when sun is sensed after leaving eclipse; This mode is enabled only after eclipse has been entered to avoid incomplete acquisition.
b) Sun orientation complete (switch to fine sun sensor control and initiate roll maneuver program)					If magnetometer calibration pitch and yaw rotations are required, they would be accomplished at this time.
3.3 Roll Maneuver	R				R is time of roll maneuver start
a) Earth detector turned on	R	C&S	G&C	MOS	
b) Canopus sensor turned on	R	C&S	G&C	MOS	
c) Begin 0.22 deg/sec roll-spin	R	C&S	G&C		For evaluation of Canopus sensor track; magnetometer calibration accomplished as well.
d) Begin 0.1 deg/sec roll-search	R + 50 min	C&S	G&C		For Canopus acquisition
e) Acquire Canopus and verify by Earth detector		G&C	Telemetry/MOS		Near earth detector used to verify Canopus acquisition
f) Turn gyros off		C&S	G&C	MOS	
g) Switch control jets to low thrust		C&S	G&C	MOS	
4. Interplanetary Cruise (prior to first correction)					
4.1 Turn off Earth Detector		C&S	G&C	MOS	
4.2 Switch Communications to High Power (high-power — low-gain mode)	T + 16 hr	C&S	S-band radio	MOS	50-watt transmitter turned on
4.3 Boresight Calibration of High-Gain and Medium-Gain Antennas		MOS	G&C		Only if required
4.4 Reposition High-Gain and Medium-gain Antennas Periodically		C&S	G&C	MOS	Uses function generator for cone angle (function generator is inhibited during maneuver)
4.5 Canopus Sensor Updating as Required		C&S	G&C	MOS	
5. First Interplanetary Trajectory Correction (ITC)					Between 2 and 10 days after launch. Time selected by MOS
5.1 Send Commands as Follows:		MOS	C&S		
a) Command to initiate maneuver subprogram at designated time					
b) Roll turn start and stop times and polarity					
c) Pitch turn start and stop times and polarity					
d) Engine start time					
e) Velocity increment magnitude					
f) Backup engine shut down time					
g) High-gain antenna pointing data					
5.2 Readout C and S Memory for Data Verification					
a) Switch to telemetry mode 4		C&S	MOS		Rapid readout of C&S data for verification
b) Switch to telemetry mode 2		C&S	MOS		
5.3 Enable Maneuver Preparations		MOS	C&S		
5.4 Execute Maneuver Preparations					
a) Turn on gyros	M_1	C&S	G&C	MOS	M_1 is time of initiation of maneuver subprogram
b) TVC to high gain		C&S	G&C	MOS	TVC is thrust vector control
c) Test TVC system		C&S	G&C	MOS	
d) Switch to telemetry mode 1 (engineering) at 234 bit rate	M_1 + 60 min	C&S	Telemetry	MOS	
e) Enable high-gain antenna pointing command		MOS	C&S		
f) Position high-gain antenna to be earth pointing after PV reorientation		C&S	G&C	MOS	
g) Verify high-gain antenna position		Telemetry	MOS		Based on antenna gimbal angles
h) Enable PV reorientation		MOS	C&S		
5.5 Reorient PV					
a) Switch to fine range attitude control		C&S	G&C	MOS	
b) Switch control jets to high thrust		C&S	G&C	MOS	
c) Switch to maneuver mode		C&S	G&C	MOS	To disable the sun and Canopus seeking function
d) Start roll turn (0.2 deg/sec)		C&S	G&C		
e) Stop roll turn		C&S	G&C		
f) Start pitch turn (0.2 deg/sec)		C&S	G&C		
g) Stop pitch turn		C&S	G&C		
h) Perform additional roll-turn if required		C&S	G&C		If the additional roll condition has been enabled

Table 24. 1971 Mission Sequence for the Spacecraft System (cont.)

Event	Time	Signal Source	Signal Destination	Signal Backup	Remarks
5.6 Switch Communications to High-Gain Antenna (high power — high gain mode)		C&S	S-band Radio	MOS	Received signal via high gain antenna verifies PV orientation
5.7 Verify PV attitude and enable engine operation		MOS	C&S		If enable is not received C&S commands reacquisition after a predetermined elapsed time
6. Engine Firing					At time previously stored
a) Command pintle to low thrust position		C&S	Propulsion	MOS	
b) Turn on TVC system and accelerometer		C&S	G&C	MOS	Disable pitch and yaw thrusters and enable TVC actuators
c) Open start tank solenoid valves		C&S	Propulsion via power subsystem		
d) Open "normally closed" explosive valves (engine start)		C&S	Propulsion via pyrotechnic subsystem		Set No. 1 low-flow propulsion explosive valves used
e) Close start tank valves 6 sec after (d)		C&S	Propulsion via power subsystem		Propellant continues to flow from main tank
f) Close "normally open" explosive valves (engine cutoff)		C&S	Propulsion via pyrotechnic subsystem		Signal based on accelerometer integration
g) Close "normally open" explosive valves (engine cutoff)		C&S	Propulsion via pyrotechnic subsystem		Backup signal based on timer
h) Turn off TVC system and accelerometer		C&S	G&C	MOS	Enable pitch and yaw thrusters
i) Switch communications transmission to low-gain antenna (high power — low gain mode)		C&S	S-band radio	MOS	
7. Re-establish Cruise Operations					
7.1 Sun Acquisition					
a) Position high-gain antenna to be earth pointing after celestial reference acquisition		C&S	G&C	MOS	
b) Switch G&C to sun acquisition mode and coarse range attitude control		C&S	G&C	MOS	
c) Complete sun orientation, switch from sun coarse sensor to fine sensor, and initiate roll maneuver program					Automatic
7.2 Canopus Acquisition					
a) Execute 0.1 deg/sec roll maneuver					
b) Acquire Canopus		G&C	C&S		
c) Turn gyros off		C&S	G&C	MOS	
d) Switch control jets to low thrust		C&S	G&C	MOS	
7.3 Establish Cruise Operation					
a) Switch to telemetry mode 3 to transmit data stored during firing and orientation		C&S	Telemetry subsystem	MOS	
b) Switch to telemetry mode 2		C&S	Telemetry subsystem	MOS	
8. Interplanetary Cruise (between interplanetary trajectory corrections)					
8.1 Reposition High-Gain and Medium-Gain Antennas Periodically		C&S	G&C	MOS	For turn-around ranging after 15 days the high-gain antenna is utilized. The low-gain antenna is utilized otherwise up to 75 days
8.2 Canopus Sensor Updating as Required		C&S	G&C	MOS	
8.3 Switch Communication Transmission to High Power — High Gain Mode and Reception to High Gain Mode	T + 75 days	C&S	S-band radio	MOS	
9. Second Interplanetary Trajectory Correction (ITC) and Third ITC if Required					Time selected by MOS; perform in similar manner to 5, 6, 7; medium-gain antenna is utilized prior to and during reorientation; set No. 2 low-flow propulsion explosive valves utilized. The set No. 3 valves are used for a third ITC if required.
10. Interplanetary Cruise (after final ITC)					Similar to 8.
11. Mars Orbit Insertion Preparation and Orientation					
11.1 Send Commands as Follows:		MOS	C&S		
a) Command to initiate insertion program at designated time					
b) Roll turn start and stop times and polarity					
c) Pitch turn start and stop times and polarity					
d) Engine start time					
e) Velocity increment magnitude					
f) Backup engine shut down time					
g) High-gain antenna pointing data					

Table 24. 1971 Mission Sequence for the Spacecraft System (cont.)

Event	Time	Signal Source	Signal Destination	Signal Backup	Remarks
11.2					
a) Switch telemetry to mode 4		C&S	MOS		
b) Switch telemetry to mode 2		C&S	MOS		
11.3 Enable Maneuver Preparations		MOS	C&S		
11.4 Execute Maneuver Preparations	D				D is time of initiation of maneuver preparations
a) Turn on gyros	D	C&S	G&C	MOS	
b) Switch TVC to low gain	D	C&S	G&C	MOS	
c) Test thrust vector control system		C&S	G&C	MOS	
d) Switch to telemetry mode 1 at 234 bit rate	D+60 min	C&S	Telemetry subsystem	MOS	
e) Switch communication to medium-gain antenna (high power-medium gain mode)		C&S	S-band radio	MOS	
f) Enable high-gain antenna pointing command		MOS	C&S		Based on verification of switch to low gain
g) Position high-gain antenna to be earth pointing after PV re-orientation		C&S	G&C		
h) Verify high-gain antenna position		Telemetry	MOS		Based on antenna gimbal angles
i) Enable PV reorientation		MOS	C&S		
j) Enable extra-roll-turn if necessary		MOS	C&S		For certain antenna conditions or failure
11.5 Jettison Capsule Canister Lid		C&S	Capsule		
11.6 Reorient PV					
a) Switch to fine range attitude control		C&S	G&C	MOS	
b) Switch control jets to high thrust		C&S	G&C	MOS	
c) Switch to maneuver mode		C&S	G&C	MOS	To disable the sun and Canopus seeking function
d) Start roll turn (0.2 deg/sec)		C&S	G&C		
e) Stop roll turn		C&S	G&C		
f) Start pitch turn (0.2 deg/sec)		C&S	G&C		
g) Stop pitch turn		C&S	G&C		
h) Perform additional roll-turn if required		C&S	G&C		If the additional roll condition has been enabled
11.7 Switch Communication to High-Gain Antenna (High Power-High Gain Mode)		C&S	S-band radio	MOS	Received signal via high gain antenna verifies PV orientation
11.8 Verify PV Attitude and Enable Engine Operation		MOS	C&S		If enable is not received C&S commands reacquisition
12. Engine Firing for Mars Orbit Insertion					
a) Open helium tank isolation explosive valve and component isolation explosive valves		C&S	Propulsion via pyrotechnic subsystem	MOS	
b) Turn on TVC system and accelerometer		C&S	G&C	MOS	Disable pitch and yaw thrusters and enable TVC actuators
c) Open start tank solenoid valves		C&S	Propulsion via power subsystem		
d) Open "normally closed" tank pressurization explosive valves		C&S	Propulsion via pyrotechnic subsystem		
e) Open low-flow solenoid valves (engine start)		C&S	Propulsion via power subsystem		
f) Close start tank valves 6 sec after (e)		C&S	Propulsion via power subsystem		
g) Open large "normally closed" explosive valves		C&S	Propulsion via pyrotechnic subsystem		
h) Move pintle to high thrust position		C&S	Propulsion	MOS	
i) Close low-flow solenoid valves (opened at (e)) 3 sec after (g)		C&S	Propulsion via power subsystem		
j) Close "main tank" large "normally open" explosive valves (engine cutoff)		C&S	Propulsion via pyrotechnic subsystem		Based on integration of acceleration output to achieve stored velocity increment
k) Close main tank large "normally open" explosive valves (engine cutoff)		C&S (timer backup)	Propulsion via pyrotechnic subsystem		
l) Turn off TVC system and accelerometer		C&S	G&C	MOS	Enable pitch and yaw thrusters
m) Switch communication transmission to medium gain antenna		C&S	S-band radio		
13. Re-establish Cruise Operations					Similar to 7. Cruise communication is high power-high gain transmission at 7500 bits/sec and high gain reception
14. Planetary Vehicle Orbital Operations (pre-trim)					
14.1 Turn on Terminator and Limb Crossing Sensors		C&S	DAE	MOS	

Table 24. 1971 Mission Sequence for the Spacecraft System (cont.)

Event	Time	Signal Source	Signal Destination	Signal Backup	Remarks
14.2 Switch Terminator and Limb Crossing Sensors to Telemetry		C&S	DAE	MOS	
14.3 Switch Terminator and Limb Crossing Sensors to DAE		C&S	DAE	MOS	
14.4 Enable DAE		C&S	DAE	MOS	
14.5 Prepare PSP Operations					
a) Turn on PSP	LDT +	C&S	DAE	MOS	LDT is time of crossing light-dark terminator
b) Set acquisition angles and begin tracking Mars		C&S	DAE	MOS	
c) Warm up planetary experiments		C&S	DAE		
14.6 Calibrate Planetary Experiments					
14.7					
a) Start IR spectrometer subsequence	DLT +				DLT is time of crossing dark-light terminator
b) Start UV spectrometer subsequence					
c) Start IR radiometer subsequence					
d) Start and stop tape recorder					Per desired sequence
14.8 Set Color Filter for Photoimaging					
14.9 Set Shutter Speeds					
14.10 Set IMC Inputs					
14.11 Switch G&C to Fine Range Attitude Control		C&S	G&C	MOS	
14.12 Start Photoimaging Subsequence	LC_{DL}				LC_{DL} is first limb crossing after DL terminator crossing
a) Actuate image motion compensation					Repeat (a) through (f) sequence as desired
b) Actuate shutters					
c) Start tape recorder					
d) Read camera No. 1					
e) Read camera No. 2					
f) Stop tape recorder					
14.13					
a) Set tape recorders read speeds for telemetry rate					
b) Telemeter spectrometer tape	DLT	C&S	Data storage		
c) Turn off spectrometer tape					
d) Telemeter radiometer tape					
e) Turn off radiometer tape					
f) Telemeter photoimaging tape No. 1					
g) Turn off photoimaging tape No. 1					
h) Telemeter photoimaging tape No. 2					
i) Turn off photoimaging tape No. 2					
14.14 Switch G&C to Coarse Range Attitude Control		C&S	G&C	MOS	
14.15 Reposition High-Gain and Medium-Gain Antennas Periodically		C&S	G&C	MOS	
14.16 Canopus Sensor Updating as Required		C&S	G&C	MOS	
15. Carry Out Mars Orbit Trim					Similar to 5, 6, 7; medium-gain antenna is utilized prior to and during reorientation; low-flow propulsion solenoid valves utilized; the PSP is inhibited during the maneuver.
16. Planetary Vehicle Orbital Operations (Post-Trim)					Continue operations as in 14.
17. Spacecraft-Capsule Separation Preparation and Orientation					
17.1 Send Commands as Follows:		MOS	C&S		
a) Command to initiate maneuver as designated time					
b) Roll turn start and stop times and polarity					
c) Pitch turn start and stop times and polarity					
d) Capsule separation time					
e) Start time for relay link recorder					
f) High-gain antenna pointing data					
17.2 Switch Telemetry to Mode 4		C&S	MOS		
17.3 Enable Maneuver Preparations		MOS	C&S		

Table 24. 1971 Mission Sequence for the Spacecraft System (cont.)

	Event	Time	Signal Source	Signal Destination	Signal Backup	Remarks
17.4	Execute Maneuver Preparations	C	MOS	C&S		C is time of initiation of capsule separation maneuver preparations
	a) Turn on spacecraft gyros	C	C&S	G&C		
	b) Inhibit PSP		C&S	DAE	MOS	
	c) Switch to telemetry mode 1		C&S	Telemetry subsystem	MOS	
	d) Switch communication to medium-gain antenna (high power-medium gain mode)	C+60 min	C&S	S-band radio	MOS	
	e) Enable high-gain antenna pointing command		MOS	C&S		Based on verification of switch to medium-gain antenna
	f) Position high-gain antenna for earth pointing after PV re-orientation		C&S	G&C		
	g) Verify high-gain antenna position		Telemetry	MOS		Based on antenna gimbal anbles
	h) Enable PV reorientation		MOS	C&S		
	i) Perform extra roll turn if necessary		C&S	G&C		For certain antenna conditions or failure
17.5	Activate Capsule Preseparation Phase					
	a) Turn on capsule relay link receiver		C&S	VHF radio	MOS	
	b) Test capsule relay link operation		C&S	VHF radio		
	c) Switch capsule relay link receiver outputs to telemetry (telemetry mode 4)		C&S	Telemetry	MOS	Spacecraft and (100 bps) capsule data in real time until terminal entry. RF attenuation is utilized until a suitable separation distance is achieved.
17.6	Reorient PV					Similar to 5.5
17.7	Switch Communication to High-Gain Antenna (High Power-High Gain Mode)		C&S	S-band radio	MOS	
17.8	Verify PV Attitude and Enable Capsule Separation		MOS	C&S		
18.	Capsule Separation and Relay Link Operations					
18.1	Command Capsule Separation		C&S	Capsule	MOS	
18.2	Reestablish Cruise					Similar to 7, with telemetry remaining in mode 2
18.3	Switch out RF Attenuation		C&S	Relay link receiver		See 17.5 c
18.4	Start Relay Link Recorder		C&S	Relay link recorder		Based on time from capsule separation
18.5	Stop Relay Link Recorder and turn off receiver		C&S	Relay link recorder		End of tape will act as backup to stop recorder
18.6	Playback Relay Link Recorder		C&S	Telemetry and relay link recorder	MOS	Relay link recorder receives clock from telemetry subsystem
19.	Spacecraft Orbital Operations					Similar to 14.

Table 25. Flight Spacecraft Component Design Parameters

Item No.	Component	No. of Items	Weight lb (Each)	Volume in.^{3} (Each)	Electrical Power and Sources: Average	Peak	Primary Power Source	Allowable Operating Temperature, °F: Min.	Max.	Allowable Nonoperating Temperature, °F: Min.	Max.	Magnetic Allocation* Nonoperating/Operating
0100	STRUCTURAL AND MECHANICAL SUBSYSTEM											
0122	High-gain antenna release mechanism	1	-	-	-	-	-	-250	250	-250	250	0/0
0123	Medium-gain antenna release mechanism	1	-	-	-	-	-	-250	250	-250	250	0/0
0124	Low-gain antenna release mechanism	1	-	-	-	-	-	-250	250	-250	250	0/0
0125	Planetary scan platform release mechanism	1	-	-	-	-	-	-250	250	-250	250	0/0
0130	Flight capsule emergency separation	1	13.7	-	-	-	-	-250	250	-250	250	0/0
0200	COMMAND SUBSYSTEM											
0201	Command detector	2	2.6	40	1.5	1.5	4 kHz	30	110	-25	175	500/500
0202	Decoder	2	2.6	50	4.7	5.0	4 kHz	30	110	-25	175	500/500
0300	PROPULSION MODULE SUBSYSTEM											
0302	Cabling set	1	11.0	-	-	-	-	-65	160	-65	160	0/0
0311	Quad solenoid assembly	4	7.0	-	-	18	Battery	-65	160	-65	160	100/1000
0312	Explosive valve, 1/2-inch	14	0.8	-	-	-	-	-65	160	-65	160	0/0
0314	Injector pintle actuator	1	4.0	-	-	50	Battery	-65	160	-65	160	500/500
0316	Check valve, propellant feed	8	.1	-	-	-	-	-65	160	-65	160	0/0
0319	Explosive valve, 1-inch	4	3.0	-	-	150	Battery	-65	160	-65	160	0/0
0333	Explosive valve, pressurization	3		-	-	150	Battery	-65	160	-65	160	0/0
0335	Latching solenoid valve	2		-	-	90	Battery	-65	160	-65	160	100/1000
0336	Pressure regulator	2		-	-	-	-	-65	160	-65	160	0/0
0337	Check valve, pressurization	8		-	-	-	-	-65	160	-65	160	0/0
0340	Relief valve and burst disk, oxygen	1		-	-	-	-	-65	160	-65	160	0/0
0358	Relief valve and burst disk, fuel	1		-	-	-	-	-65	160	-65	160	0/0
0400	CABLING SUBSYSTEM											
0401	Junction box	4	5.0	275	-	-	-	-65	160	-65	160	0/0
0410	Interconnecting cables		208.9	-	-	-	-	-250	250	-250	250	0/0
0500	COMPUTING AND SEQUENCING SUBSYSTEM											
0501	Computer and sequencer	1	19	325	33	33	4 kHz	30	110	-30	160	500/500
0502	Backup computer and sequencer	1	17	325	26	26	4 kHz	30	110	-30	160	500/500
0600	S-BAND RADIO SUBSYSTEM											
0601	Low-gain antenna	1	3.0	-	-	-	-	-350	360	-350	360	
0610	High gain antenna	1	55.2	-	-	-	-	-350	360	-350	360	0/0
0620	Medium gain antenna	1	13.4	-	-	-	-	-350	360	-350	360	0/0
0641	S-band receiver	3	5.0	190	2.5	2.5	4 kHz	30	110	-25	175	500/500
0642	Receiver selector	1	1.0	36	0.8	0.8	4 kHz	30	110	-25	175	500/500
0643	1 watt transmitter and power monitor	1	3.5	90	10	10	4 kHz	30	110	-25	175	500/500
0644	Modulator exciter	2	3.0	90	2	2	4 kHz	30	110	-25	175	500/500
0645	Power amplifier, power supply and RF power monitor	2	7.8	500	150	150	50 VDC	30	185	-25	250	500/500
0646	Transmitter selector	1	1.0	36	0.8	0.8	4 kHz	30	110	-25	175	500/500
0647	4-port hybrid ring and power monitor	1	0.6	30	-	-	-	30	110	-25	175	500/500
0648	Circulator switch	4	1.8	36	0.25	1.0	4 kHz	30	110	-25	175	500/500
0649	Diplexer	3	1.3	50	-	-	-	30	110	-25	175	500/500
0700	GUIDANCE AND CONTROL SUBSYSTEM											
0701	Gyro reference assembly	1	10.0	180	17.5	38.5	(**)	30	130	-22	180	500/500
0702	Accelerometer	1	1.0	13	3	3	4 kHz	30	130	-22	180	500/500
0703	Guidance and control electronics	1	13.0	450	8/24	10/28	4 kHz / 50 VAC	-30	130	-30	200	500/500
0711	Canopus sensor	2	6.0	248	1.8	6.0	4 kHz	-30	100	-30	100	500/500
0712	Sun sensor	1	1.0	34	0.7	0.7	4 kHz	30	130	-20	160	500/500
0713	Earth detector	1	0.3	-	0.15	0.15	4 kHz	30	130	-20	160	500/500
0714	Limb and terminator crossing detector	2	0.6	-	0.4	0.4	4 kHz	30	130	-20	160	500/500
0722	Regulator	4	1.0	-	-	-	-	0	140	-0	200	0/0
0723	Solenoid valve	16	1.25	-	-	120	29 VDC	-65	250	-65	250	100/1000
0724	Transducer	4	0.2	-				0	140	0	200	500/500
0726	Relief valve	2		-	-	-	-	0	140	0	200	0/0
0728	Thrusters	4	1.0	-	-	32(***)	50 VDC	-200	250	-200	250	0/0
0731	High-gain drive assembly	1	32.0	-	4	33	400 Hz	0	140	0	160	1000/1000
0732	Medium-gain drive assembly	1	17.0	-	0.3	7	400 Hz	0	140	0	160	500/500
0733	Drive electronics	1	5.0	200	6	8	4 k Hz	-20	130	-20	200	500/500
0741	Actuator	2	18.0	-	400	650	Battery	-30	200	-65	275	500/500

*Field in gamma at 1 foot of an equivalent dipole.

**Power sources: 4 kHz, 50 VDC, 50 VAC, 29 VDC

***Surplus power, if available

Table 25. Flight Spacecraft Component Design Parameters (cont.)

Item No.	Component	No. of Items	Weight lb (Each)	Volume in.^{3} (Each)	Electrical Power and Sources: Average	Peak	Primary Power Source	Allowable Operating Temperature, °F: Min.	Max.	Allowable Nonoperating Temperature, °F: Min.	Max.	Magnetic Allocation* Nonoperating/Operating
1200	TELEMETRY SUBSYSTEM											
1201	PCM encoder	2	4.0	115	5	5	4 kHz	30	110	-25	175	500/500
1300	TEMPERATURE CONTROL SUBSYSTEM											
1310	Insulation - Mylar							-300	300	-300	300	
	- Refrasil							-300	2000	-300	2000	
1320	Heaters and thermostats		2.0	-	44	54	50 VDC	-100	300	-100	300	0/0
1330	Louvers		17.1					-100	300	-100	300	
1400	POWER SUBSYSTEM											
1411	Solar panels	8	36.7	-	-	-	-	-184	248	-184	248	
1421	Power control unit	1	8.0	215	10	20	50 VDC	-20	120	-50	200	500/500
1422	Shunt element assembly	2	8.0	216	-	-	-	-20	150	-50	200	500/500
1423	Battery	3	46.0	1100	-	-	-	50	90	50	90	500/500
1424	Battery regulator	3	14.0	400	-	-	-	-20	120	-50	200	500/500
1425	4.0 - kHz inverter	2	7.3	216	224	392	50 VDC	-20	120	-50	200	500/500
1426	400 Hz inverter	2	3.0	75	3	66	50 VDC	-20	120	-50	200	500/500
1427	Power distribution unit	1	7.5	288	0.5	4	50 VDC	-20	120	-50	200	500/500
1900	DATA STORAGE SUBSYSTEM											
1901	TV tape recorder	2	12.0	600	5	12	4 kHz	30	110	-25	175	2000/2000
1902	Spectrometer tape recorder	1	12.0	600	5	12	4 kHz	30	110	-25	175	2000/2000
1903	IR scanner tape recorder	1	12.0	600	5	12	4 kHz	30	110	-25	175	2000/2000
1904	Spacecraft tape recorder	1	12.0	600	5	12	4 kHz	30	110	-25	175	2000/2000
1905	Fields/particles tape recorder	1	12.0	600	5	12	4 kHz	30	110	-25	175	2000/2000
2000	PLANETARY VEHICLE ADAPTER											
2030	Planetary vehicle adapter electrical			-	-	-	-	-65	165	-300	165	0/0
2050	Separation device			-	-	-	-	-250	250	-250	250	0/0
2100	RELAY LINK SUBSYSTEM											
2101	UHF antenna	1	8.0	-	-	-	-	-350	360	-350	360	0/0
2102	Receiver	2	2.0	40	1.5	3	4 kHz	30	110	-25	175	500/500
2103	Demodulator	2	0.4	40	1.0	2	4 kHz	30	110	-25	175	500/500
2104	Tape recorder	1	12.0	600	5	12	4 kHz	30	110	-25	175	2000/2000
2200	PYROTECHNIC SUBSYSTEM											
2201	Launch vehicle separation pyrotechnic	4		-	-	-	(**)					
2202	Capsule emergency jettisoning pyrotechnic	4		-	-	-	(**)					
2203	Umbilical disconnect pyrotechnic	1		-	-	-	(**)					0/0
2204	Capsule emergency disconnect pyrotechnic	1	0.6	-	-	-	(**)					
2205	High-gain antenna pyrotechnic	1		-	-	-	(**)					
2206	Medium-gain antenna pyrotechnic	1		-	-	-	(**)	-250	250	-250	250	
2207	Low-gain antenna pyrotechnic	1		-	-	-	(**)					
2208	Explosive valve pyrotechnic (engine)	18		-	-	-	(**)					
2211	Pyrotechnic control assembly	1	25.0	720	-	-	-					500/500
2221	Planetary scan platform pyrotechnic	1	0.6	-	-	-	(**)					0/0
2222	Science cover release pyrotechnic	1	0.6	-	-	-	(**)					0/0
2300	SPACECRAFT SCIENCE SUBSYSTEM											
2311	Planetary scan platform	1	102	(internal) 20736	-	14	400 Hz	-250	240	-250	240	0/0
2312	Planetary scan control	1	8	-	25	25	4 kHz	-250	240	-250	240	1000/1000
2321	Data automation equipment	1	37	850	21	21	4 kHz	0	140	-25	175	500/500
2322	Science power switching electronics	1	3	75	2	2	4 kHz	0	140	-25	175	500/500
2323	Science command decoding equipment	1	3	75	3	3	4 kHz	0	140	-25	175	500/500
2330	PSP - Mounted instruments (sensors)		100.0	10800	20.0	20.0	4 kHz	0	140	-25	175	500/500
	(Remote electronics)		43.0	1350	48.5	48.5	4 kHz	0	140	-25	175	500/500
2340	Nondeployable sensor instruments (sensors)		19.0	2290	5.5	5.5	4 kHz	0	140	-25	175	500/500
	(Remote electronics)		44.0	900	18.5	18.5	4 kHz	0	140	-25	175	500/500
2350	Deployable sensor instruments (sensors)		10.9	360	2.3	2.3	4 kHz	0	140	-25	175	500/500 (1)
	(Remote electronics)		23.0	900	17.0	17.0	4 kHz	0	140	-25	175	500/500
2361	Fixed science package structure	2	20.0	(internal) 8000	-	-	-	-250	250	-250	250	0/0
2362	Fixed science package internal cabling	2	10.0	-	-	-	-	-250	250	-250	250	
2363	Fixed science package temperature control assembly	2		-	2.0	2.0	4 kHz					
2371	Magnetometer boom (motor)	1	10.0	-	(***)		4 kHz	-250	250	-250	250	500/500
2372	RF noise antenna deployable mechanism	1	10.0	-	(***)		4 kHz	-65	250	-65	250	500/500
2373	Ionosphere experiment deployable mechanism	1	10.0	-	(***)		4 kHz	0	140	-25	175	500/500
2374	Bistatic radar deployable mechanism 1	1	10.0	-	(***)		4 kHz	-65	250	-65	250	500/500
2375	Bistatic radar deployable mechanism 2	1	10.0	-	(***)		4 kHz	-65	250	-65	250	500/500

*Field in gamma at 1 foot of an equivalent dipole.

**Fired by charged capacitors.

***Drive motors

1 Except magnetometer and ionization sensors.

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5. "Voyager Spacecraft System, Phase IA, Task B, Additional Capsule Data - Capsule Relay Link," "Appendix to Additional Capsule Data - Capsule Relay Link," JPL, 22 November 1965, and "Transcription of Concurrent Contractor Briefing for Phase IA Extension," JPL, 12 November 1965.

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APPENDIX A

TRAJECTORY CONSIDERATIONS

1. INTRODUCTION

The purpose of this appendix is to summarize the requirements imposed on the Voyager spacecraft and its subsystems by trajectory considerations, particularly insofar as these requirements differ in Task B of the Phase IA Study from those of Task A. The physical characteristics of Earth-Mars trajectories which are possible during the time of the Voyager program, of course, have undergone no change. However, because certain constraints on the parameters associated with these trajectories have been revised, the range of launch opportunities available must be revised.

The principal trajectory areas in which the Task B mission description differs from the Task A mission are:

a) Increased launch azimuth range for launch vehicle

b) Increased C_3 capability of launch vehicle

c) Simultaneous launching of two planetary vehicles scheduled for arrival dates at least 10 days apart

d) Changing of the lander separation operation from the approach part of the interplanetary phase to the orbiting phase

e) Reduced velocity increment available for orbit insertion in 1975-1977, as a result of the weight allocations to spacecraft, capsule, and propulsion.

The implications of these changes are summarized in the following sections.

2. LAUNCH WINDOWS

The increased launch azimuth range and the increase in the C_3 launch capability from 18 to 25 km^2/sec^2 tend to increase the number of possible launch days in a launch opportunity, but this may be offset by the requirement that 10 days' separation of the two planetary vehicles at arrival be achieved within the 200 meters/sec midcourse velocity increment prescribed for each vehicle.

The launch azimuth range, coupled with the 2-hour daily launch window requirement has the following effect on DLA (declination of launch asymptote):

Opportunity	Task	Launch Azimuth Limits	Maximum Permitted |DLA|
1971	A	90-114	34.8°*
1971	B	60-115	38.4°
1973-77	B**	35-120	56.8°

By thus converting the limits in launch azimuth to a DLA constraints, its effects can be evaluated in the same manner as other trajectory constraints. Table A-1 shows the launch days available in the different opportunities corresponding to the three options above (with the increased C_3), and for one option with $C_3 = 18\ km^2/sec^2$. For all cases $V_{\infty} \leq 4.5$ km/sec.

Table A-1. Number of Days of Launch Opportunity

Maximum C_3 (km^2/sec^2)		25	25	25	25	18
Permitted Range of DLA		-33° to $+10^{\circ}$	-33° to $+33^{\circ}$	-38.4° to $+38.4^{\circ}$	-56.8° to $+56.8^{\circ}$	-56.8° to $+56.8^{\circ}$
		← days →				
Launch Opportunity and Type of Interplanetary Trajectory	1971 Type I	93	94	104	108	91
	1973 Type I	0	70	70	70	37
	1975 Type I	0	0	7	40	0
	1975 Type II	77	90	90	90	64

It is seen that the increases in launch azimuth and coast time proposed for Task B are indeed timely for exploitation of the Mars opportunities, and that the increase in C_3 capability over that of Task A is necessary for the

*DLA for the 1971 mission is further limited in Task A to about $-33^{\circ} < DLA < +10^{\circ}$ by the 25-minute limit on the duration of the Centaur parking orbit.

**From the 1971 Preliminary Mission Description: "Subsequent Voyager Missions may require launch azimuths from 35 to 120° East of North."

1975 (Type I) opportunity. Although the 1975 (Type II) opportunity requires neither the maximum increase in DLA range nor the increase in C_3, it should be noted that flight times of 300 to 350 days result, compared with 190 to 250 for Type I. The characteristics for 1977 are similar to those of 1975.

For the 1971 opportunity the launch window corresponding to the Task B constraints is illustrated in Figure A-1. Constraints are shown on a plot of arrival date versus launch date, and represent the following restrictions:

a) Declination of launch asymptote (DLA)

$$5^\circ < |DLA| < 38.4^\circ$$

The lower limit (Reference A-1, page 22) is established to aid in radio tracking of the spacecraft in interplanetary flight. The upper limit, increased from 33 degrees for Task A, is discussed above.

b) Launch vehicle capability (C_3)

$$C_3 < 25 \text{ km}^2/\text{sec}^2$$

This limit, raised from 18 km^2/sec^2 in Task A, is based on the Saturn V capability of 63,000 pounds at $C_3 = 25 \text{ km}^2/\text{sec}^2$ (Reference A-1, pages 22, 31).

c) Only Type I trajectories are considered (Reference A-1, page 22).

d) Inclination of orbit to ecliptic

$$|INC| > 0.1^\circ$$

(Reference A-1, page 22).

e) Hyperbolic excess velocity at Mars

$$V_\infty < 4.5 \text{ km/sec}$$

(Reference A-1, page 22).

f) Vector difference between launch asymptote velocities of the two planetary vehicles

$$\Delta V < 300 \text{ m/sec}$$

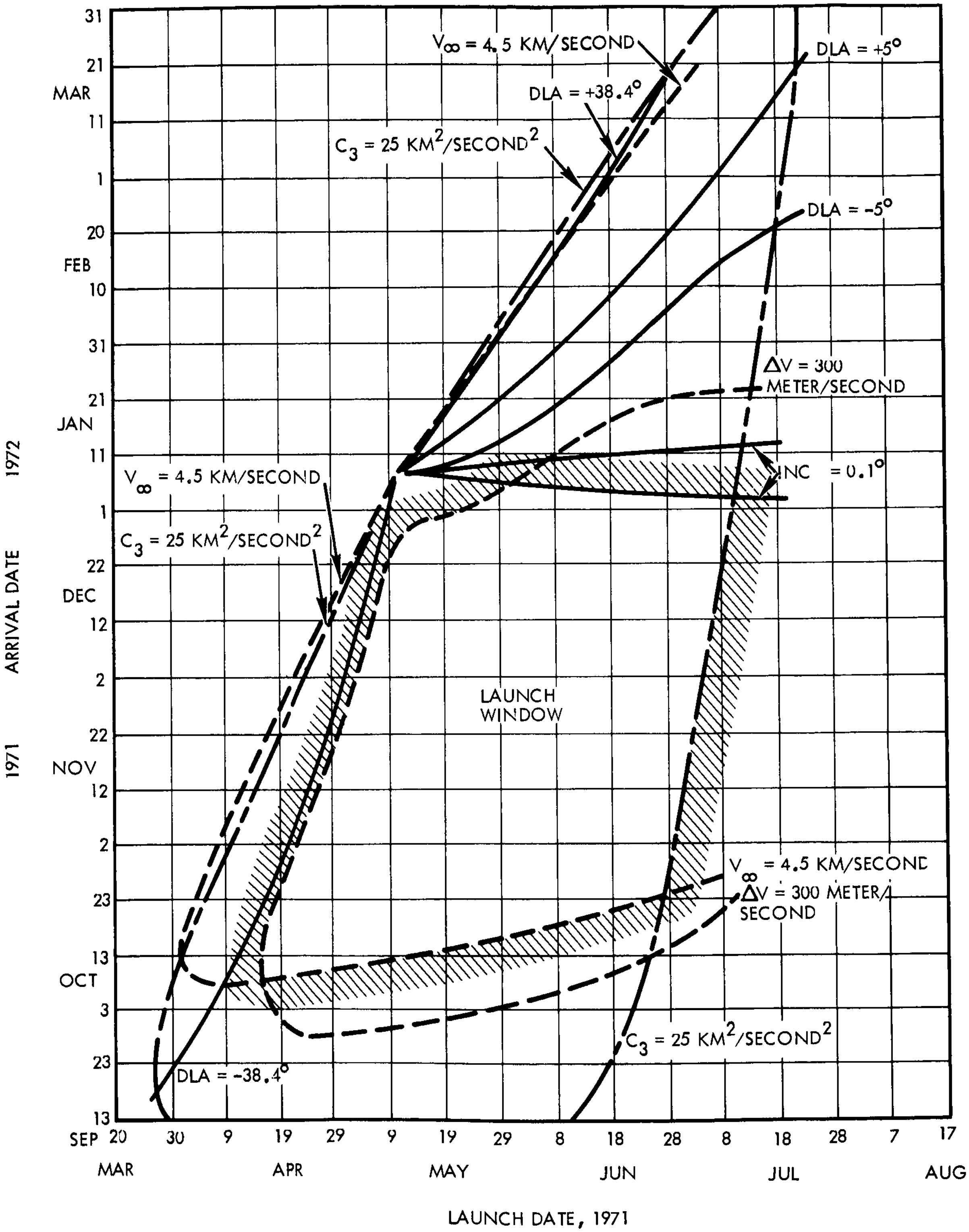

Figure A-1. Constraints on Launch Opportunities, Earth-to-Mars, 1971

This is an assumed restriction between the velocities of spacecraft destined to arrive at Mars 5 days earlier and 5 days later than the indicated point. It is based on the use of up to 150 meters/sec of the allocated 200 meters/sec midcourse correction capability for each spacecraft's half of the ΔV required for arrival separation, and 50 meters/sec remaining for the removal of injection dispersion in the first midcourse correction, and of trajectory biasing and dispersions in subsequent midcourse corrections.

g) Arrival dates after January 8, 1972, were relinquished in the TRW Task A Study, as being incompatible with the single-gimbal mechanization chosen for the medium-gain antenna. This constraint is observed also for Task B, although it represents a very small loss of available launch opportunity.

The launch window remaining, after imposing the above constraints, is shown in Figure A-1. The extreme ranges of trajectory parameters which are possible if the entire remaining window is exploited are listed in Table A-2.

Table A-2. Extreme Ranges of Trajectory Parameters*
1971 Launch Window of Figure A-1

Launch Dates	April 16, 1971	to	July 9, 1971
Arrival Dates	October 8, 1971	to	January 5, 1972
Flight Times (days)	119		230
Communication Distance (km)			
At arrival	80,000,000	to	180,000,000
1 month after arrival	120,000,000	to	230,000,000
6 months after arrival	310,000,000	to	385,000,000
Launch Asymptote Geometry			
C_3 (km^2/sec^2)	7.94	to	25
DLA (declination) (deg)	-13	to	-37
ZAL (initial cone angle of earth) (deg)	22	to	123
Arrival Asymptote Geometry			
V_∞ (km/sec)	2.82	to	4.5
LVI (declination re Mars) (deg)	-26	to	+1
ZAP (angle between approach asymptote and Mars-Sun line)	72	to	144

*Data in the table corresponds to the 1971 launch window of Figure A-1.

Note that it is not necessarily true that more than one quantity may have its extreme value for any one Earth-Mars trajectory. Also, it is quite possible that additional constraints will be imposed. For example, it is likely that the orientation as well as the size of the selected 1971 orbit will be prescribed so as to cater to the role played in delivering the lander to its entry. This, plus the finite ΔV limitation for orbit insertion, may effectively constrain the V_{∞} for arrival at Mars to less than 4.5 km/sec.

3. DEPARTURE FROM EARTH: ECLIPSE CHARACTERISTICS

The launch, injection, and departure of the Voyager planetary vehicle from the earth necessarily involves passage through the shadow of the earth. The timing and duration of this period of eclipse is pertinent to the design of spacecraft subsystems, particularly the radio, electric power, thermal control, and guidance and control subsystems.

However, the eclipse characteristics are a function of launch date and Mars arrival date. In this section, they are considered over the entire 1971 launch opportunity illustrated in Figure A-1.

Figure A-2 shows the time of entry of the spacecraft into the eclipse, the time of exit from the eclipse, and the distance of the spacecraft from the earth's center at the time of emergence from the shadow. The times of entry into and exit from eclipse are measured from injection, which is defined as the end of the second (final) burn of the launch period. Eclipse entry and exit was taken as the instant when the earth's limb coincides with the center of the sun's disk, as seen from the spacecraft.

The eclipse characteristics are highly dependent upon launch date, and only slightly dependent on arrival date. For this reason, the ordinate in Figure A-2 is launch date, with arrival date indicated by separate curves in each family labeled "Earliest," and "Latest." Reference to Figure A-1 shows that earliest arrival dates range from October 8 to October 23, 1971, and latest arrival dates range from December 31, 1971 to January 5, 1972.

The "Intermediate" curve was introduced to show the extreme eclipse characteristics existing for a given launch date, where this extreme is not associated with either the earliest or latest arrival date.

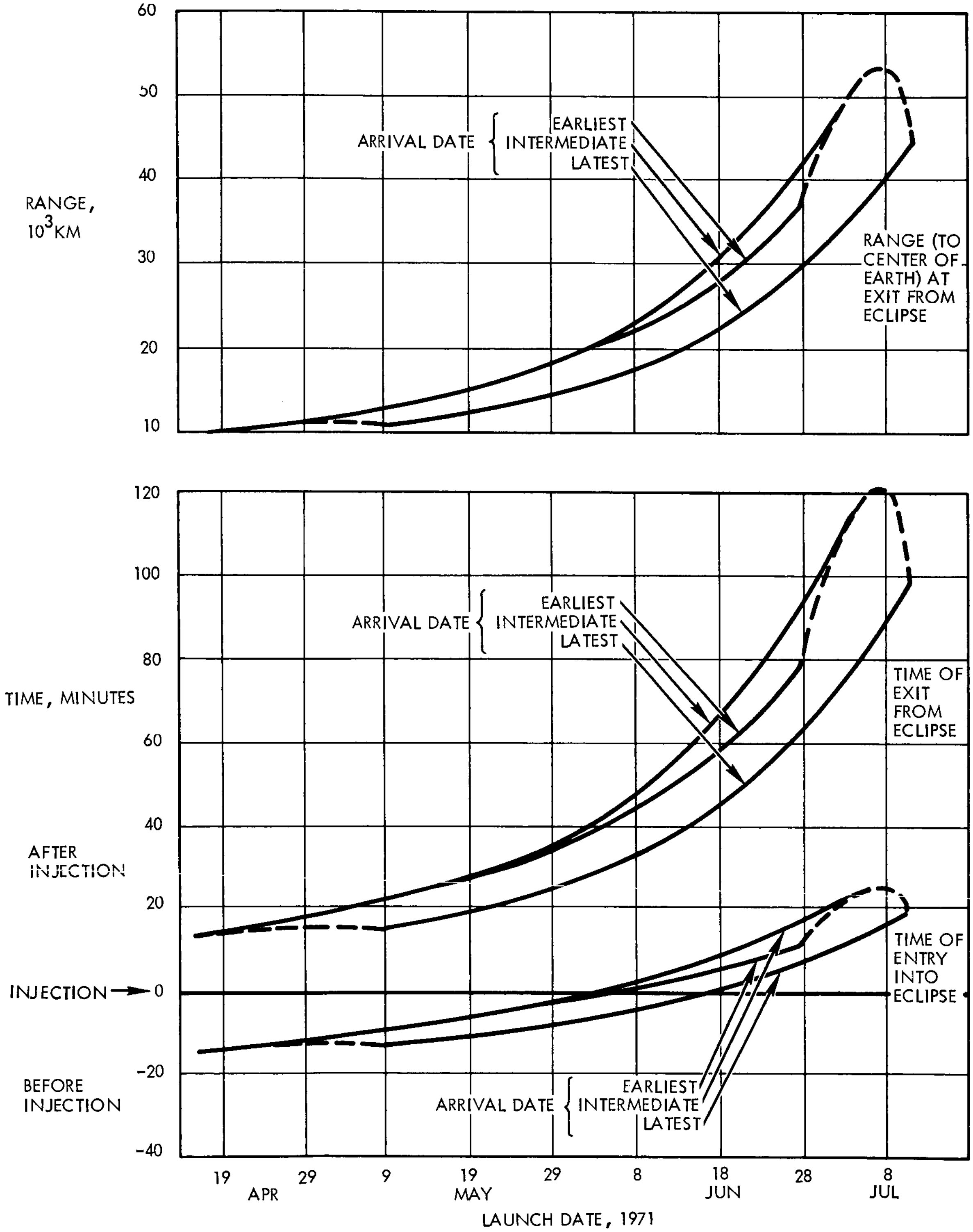

Figure A-2. Eclipse Characteristics of Voyager Injection and Departure from Earth as a Function of Launch Date, 1971

The most severe eclipse condition possible under the 1971 mission constraints given in the preceding section is associated with the following Earth-Mars trajectory:

Launch date: July 6, 1971

Arrival date: December 12, 1971 ("intermediate")

Time of entry into eclipse: 25 minutes after injection

Time of exit from eclipse: 121 minutes after injection

Duration of eclipse: 96 minutes

Range at exit from eclipse: 53,000 km (to center of earth).

In addition to the variations in eclipse characteristics with launch and arrival dates, there is a variation with the time of day of the launch, within the daily launch window. However, it is anticipated that for 1971 launches, the variation of characteristics with time of day will never cause them to vary greatly from the extreme characteristics (i.e., when the departure trajectory plane is parallel to the sun-earth line, and the spacecraft passes through the greatest dimension of the earth's shadow). These extreme characteristics are plotted in Figure A-2.

The results shown in Figure A-2 are based on the following data:

Values of C_3 and ZAL: Reference A-2

Radius of earth: 6378 km

Perigee of departure asymptote: 6563 km radius (185 km altitude)

True anomaly at injection: 12 degrees.

4. EARTH-MARS TRAJECTORY GEOMETRY

In Reference A-3, Appendix D, page 338, six sample Earth-Mars trajectories for 1971 are identified. All but one of these (1, 2, 3, 5, 6) lie within the window outlined in Figure A-1. Therefore, they remain candidates for 1971 trajectories, and their properties are still germane. In particular, the geometrical properties of these trajectories given in Reference A-4, Figures 3-76, 3-78, 3-80, and 3-81, are appropriate to Task B.

The "nominal" trajectory of Reference A-3, Appendix D, remains an attractive sample of the permitted range of trajectories available, and the detailed description of the geometrical characteristics of this trajectory is applicable to Task B.

5. ORBITS ABOUT MARS

The Phase IA Study (Task A) resulted in a proposed design which would be able to deliver a velocity increment (ΔV) for orbit insertion of 2.00 to 2.05 km/sec. This capability was considered liberal, in view of the requirements. For example, from an interplanetary trajectory arriving at Mars at a V_∞ of 3.25 km/sec (characteristic of the TRW nominal trajectory, and possible throughout more than 50 days of the launch opportunity) it is possible to enter an orbit of 2,000 x 20,000 km altitude (the TRW nominal orbit) with a ΔV of 1.54 km/sec. The ΔV requirement is increased if V_∞ rises, or if the desired orbit is more circular.

The Task B Preliminary Mission Description (1971) requires a ΔV capability of 2.0 km/sec for orbit insertion, with a design goal of 2.2 km/sec. Thus it appears that the class of orbits considered in Task A are equally feasible for Task B. Specifically, the "nominal" orbit described in Reference A-4, Appendix D, is still valid as a nominal orbit.

JPL has indicated (contractor concurrent briefing, November 12, 1965) that Martian upper atmospheric densities are considered to be reduced below the figures given for Task A. The quarantine restriction on orbit sizes may be relaxed from that employed in Task A. (See Reference A-4, Section 5.3.2.) Use of the "realistic" instead of the "conservative" will permit the minimum permissible periapsis altitude (for orbits with apoapsis altitude of 20,000 km) to be reduced from 1800 km to about 500 km. However, as implied in the Task B mission profile, it is possible that one of the planetary vehicle components put into (essentially) the same orbit as the spacecraft is a half of the capsule canister. (See Appendix E of this volume.) With the ballistic coefficient, m/C_DA, of this component estimated at 0.025 slug/ft^2, the minimum permissible periapsis altitude would have to be raised to about 1000 km to insure a 50-year orbit life. Of course the "nominal" orbit, into which it is intended to place the spacecraft, must be sufficiently above the minimum permissible orbit to account for tracking errors and orbit insertion execution errors, and sufficiently far above Mars so that the approach trajectory itself has a sufficiently low probability of impact.

6. ORBIT INSERTION

It is also implied by Reference A-1, and indicated by analysis of capsule descent trajectory constraints (see Appendix B), that it is desirable to have a capability to rotate the line of apsides of the elliptical orbit from its "optimum" orientation, which corresponds to a minimum energy orbit insertion. The amount of this rotation could be 60 degrees, or possibly as high as 90 degrees. Of course, such an orbital insertion maneuver carries with it a penalty on the velocity increment required. An analysis of the velocity increment for orbit insertion for varying apsidal rotation angles has been made for one case, the case outlined for orbit insertion accuracy as defined in Reference A-1. The results are given in Figure A-3. For each value of apsidal rotation angle (from the minimum impulse orientation), the magnitude of the vector $\bar{B}$* for the approach asymptote is chosen for minimum orbit insertion ΔV.

For the 1973 opportunity the orbit insertion capability will be the same as in 1971, so orbits of the same class will be feasible.

For the 1975 and 1977 missions, the allocation of weight to the spacecraft, capsule, and propulsion has changed so as to drastically reduce the orbit insertion capability to below 1.5 km/sec. This capability is so low, compared with 1971-73, that not only must orbits be restricted to those of high eccentricity and minimal apsidal rotation, but the launch opportunity must be restricted so that approaches to Mars conform to lowest values of V_∞. Launch windows thus become substantially shorter

*The vector $\bar{B}$ is from the center of Mars perpendicularly to the areocentric approach asymptote.

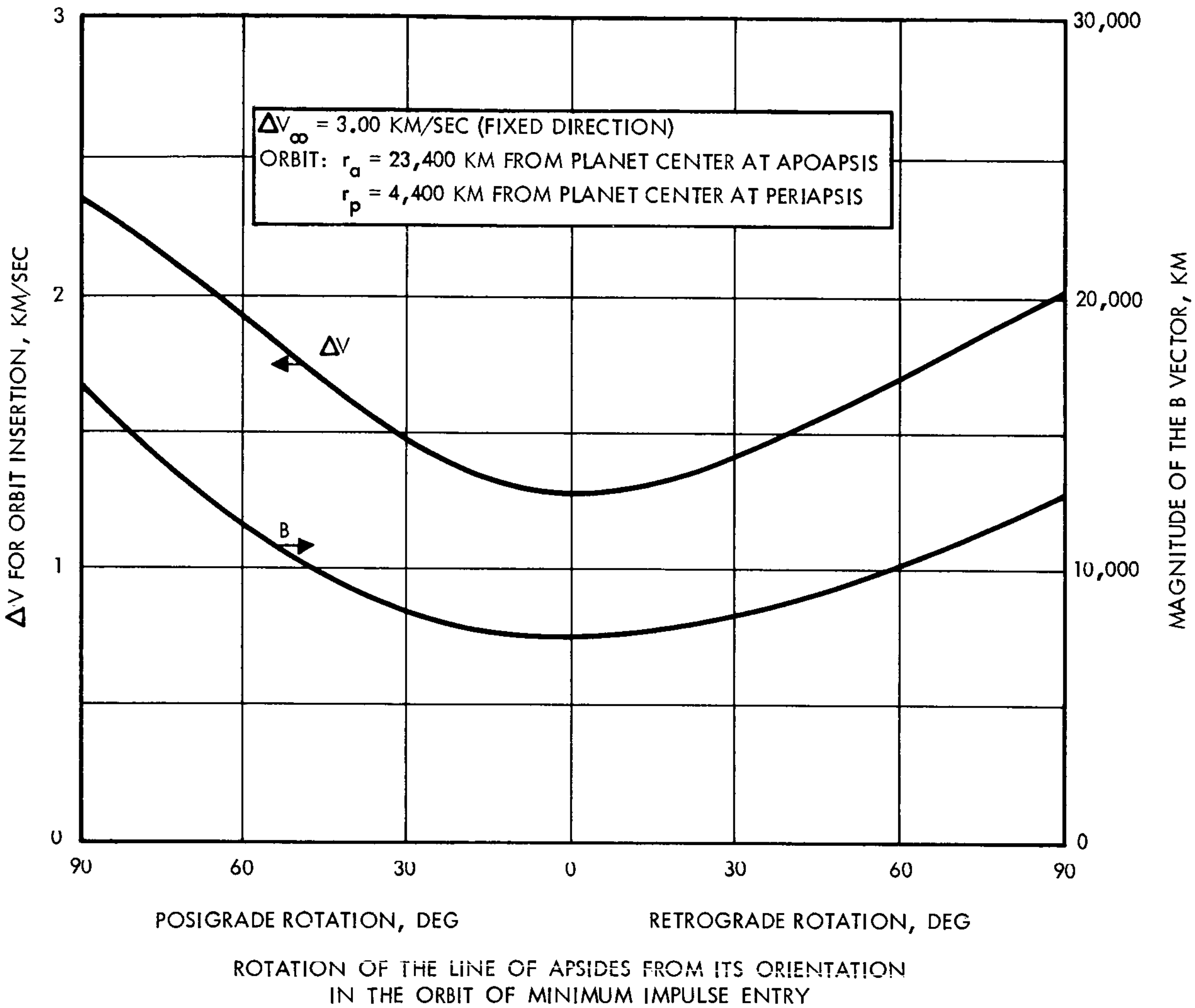

Figure A-3. ΔV and B for Optimum Orbit Insertion for Rotation of the Line of Apsides

than those given in Section 2 above. A more detailed analysis of vehicle orbit insertion capabilities, ΔV requirements, and launch periods for 1975-77 is given in Volume 5, Section 2.1.

A detailed discussion of the capsule orbital descent problem is presented in Appendix B.

REFERENCES

A-1. "Voyager 1971 Preliminary Mission Description," (JPL document governing Phase IA Study, Task B) 15 October 1965.

A-2. "Trajectory Selection Considerations for Voyager Missions to Mars During the 1971-1977 Time Period," JPL Engineering Planning Document No. 281, 15 September 1965.

A-3. Phase IA Study Report, Voyager Spacecraft, Volume 4 Appendices, TRW Systems Report 5410-0004-RU-001, 30 July 1965.

A-4. Phase IA Study Report, Voyager Spacecraft, Volume 4, TRW Systems Report 5410-0004-RU-000, 30 July 1965.

APPENDIX B
FLIGHT CAPSULE DESCENT TRAJECTORY AND REQUIREMENTS OF THE RELAY LINK

This appendix defines the geometrical requirements of the capsule-to-spacecraft relay link in terms of communication range and aim angle of the spacecraft-mounted fixed array UHF antenna. These results have been used in the configuration layout for the UHF antenna and in the analysis of communication link performance characteristics (see 4 of Section II, Volume 2).

1. CHOICE OF CAPSULE DESCENT ORBIT

The choice of an appropriate capsule descent orbit is subject to constraints imposed by (1) mission considerations, (2) characteristics of the Mars atmosphere (as presently postulated), (3) design simplicity of the orbiter and capsule system, and (4) by the objectives of maximum data return and high reliability of operation. These constraints are described in detail in Reference 5. Table B-1 lists the principal constraints and gives an interpretation of operational and design implications affecting the relay link and the capsule descent trajectory.

Constraints on orbit characteristics dictate the choice of the capsule deorbit point, the dimensions of the descent orbit, the central angle from deorbit to entry, and the position of the entry and landing points. Factors such as required deorbit velocity increment and achievable guidance accuracy must be considered (see Table B-1).

Mars surface lighting conditions for capsule TV operation (i.e., location of the terminators) and earth visibility of the capsule and/or the orbiter during the entire descent phase dictate the timing of deorbit entry and landing and influence the choice of the spacecraft orbit apsidal orientation relative to the sun line.

Primary candidates for capsule descent orbits are of two types as illustrated in Figures B-1a and B-1b. The first class of orbits (designated Class I) yield a central angle from separation to landing between 90 and 180 degrees; the second class of orbits (Class II) yields a central

Table B-1. Capsule Descent Orbit Considerations

Constraint on Capsule Descent Orbit and Orbit Parameters	Constraint Item No. in JPL Ref.*	Implication on Capsule Design and/or Operation	Implications on Spacecraft Design and/or Operation
1) Capsule deorbit maneuver ΔV not to exceed 550 meter/sec	3a	Deorbit point must fall within $\sim \pm 90$ degrees of apoapsis of spacecraft orbit.	
2) Capsule descent time 50 min $\leq T_d \leq$ 12 hr	3b	Long capsule descent period favors choice of Class II descent orbit (central angle > 180°). Long capsule descent has adverse guidance accuracy implications.	Long descent times (orbits of Class II) require reorientation of spacecraft to cruise attitude prior to capsule entry.
3) Event timing:			
a) Visibility from Goldstone during separation and deorbit maneuver	3e	Requires careful choice of capsule deorbit point and landing site.	Affects orbit orientation of spacecraft.
b) No occultation from earth of orbiter during capsule descent	5d		
4) Entry angle between vacuum skip-out and 20 degrees	3d	Limited entry corridor dimension. Constraint on characteristics of descent orbit and impact point selection.	
(Entry altitude nominally at 800, 000 ft)	3b		
5) Angle of attack less than 60 degrees if no capsule reorientation prior to atmospheric entry	3c	Strong constraint on selection of deorbit point relative to entry point. Favors use of Class II descent trajectory.	
6) Range and look angle constraints of relay antenna	(Not imposed by JPL)	a) Landing point near orbiter periapsis desirable to reduce terminal range during high data rate operation.	a) -
		b) Choice of capsule deorbit point and landing site determines overall line-of-sight angle variation during descent.	b) Fixed relay antenna favored by small line-of-sight angle variation ($\sim$ 30 degrees). Prelaunch adjustment of spacecraft relay antenna aim angle permits mission flexibility.
	2	c) -	c) Maximum communication range of 5600 km consistent with antenna gain and with performance requirement, item (2) in Ref.
7) Maximum atmospheric descent time less than 900 sec	5f	Limits look angle variation during terminal phase	Favors use of single fixed antenna
8) Orbiter looks down nearly vertically when capsule has reached terminal descent phase	(Not imposed by JPL)	Simplifies capsule mounted relay antenna design problem. (Antenna beam oriented parallel to capsule roll axis.) Eliminates roll attitude control requirement.	
9) a) Spacecraft may not be able to perform orbit trim	5e	Affects capsule landing point accuracy only	Affects maneuver backup mode and maneuver sequence of spacecraft.
b) Postponement of capsule landing attempt	5b		

* "Additional Capsule Data-Capsule Relay Link," Voyager Spacecraft System Phase IA, Task B, JPL correspondence dated 22 November 1965.

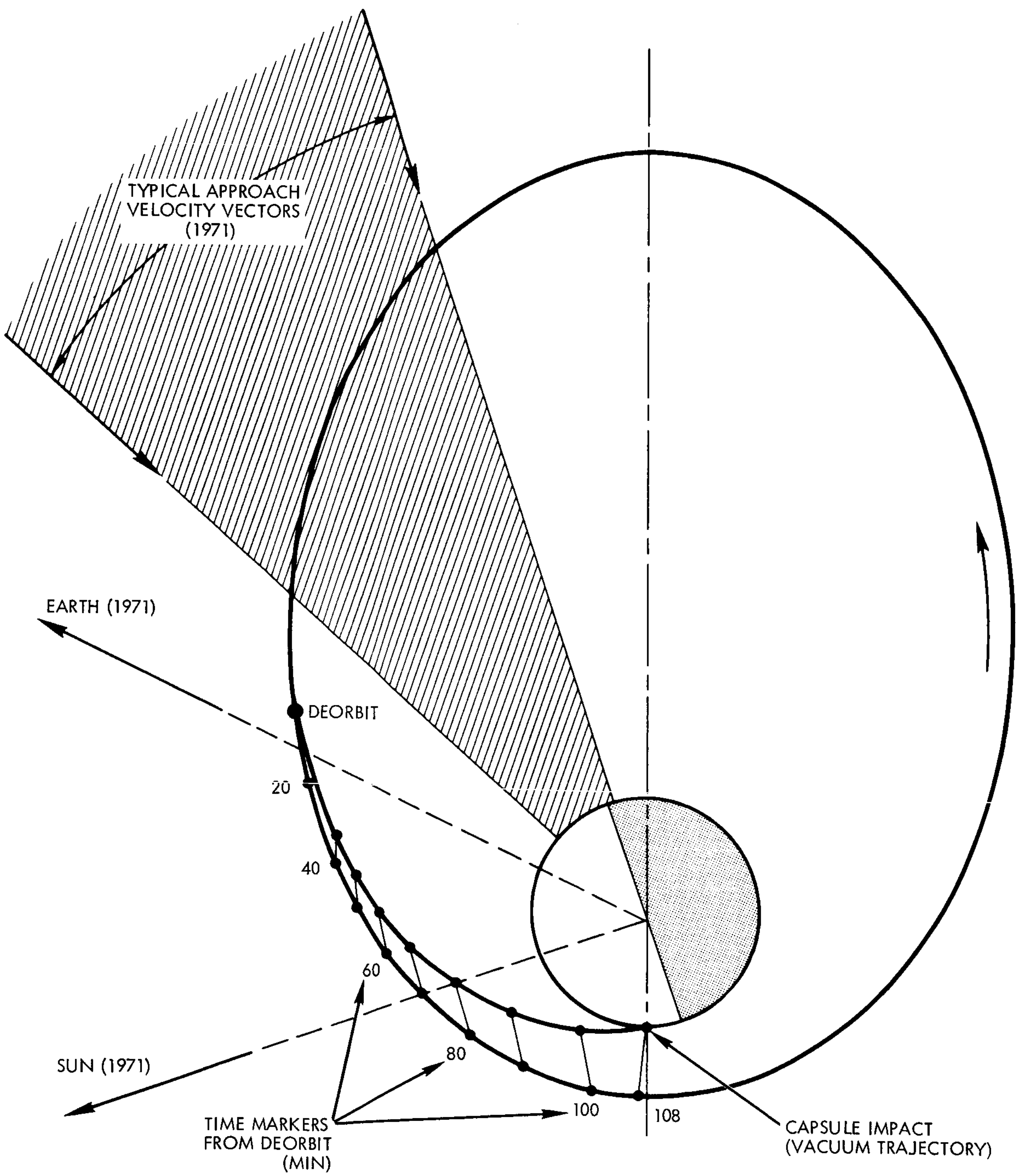

Figure B-1a. Spacecraft Orbit and Short Capsule Descent (Class I)

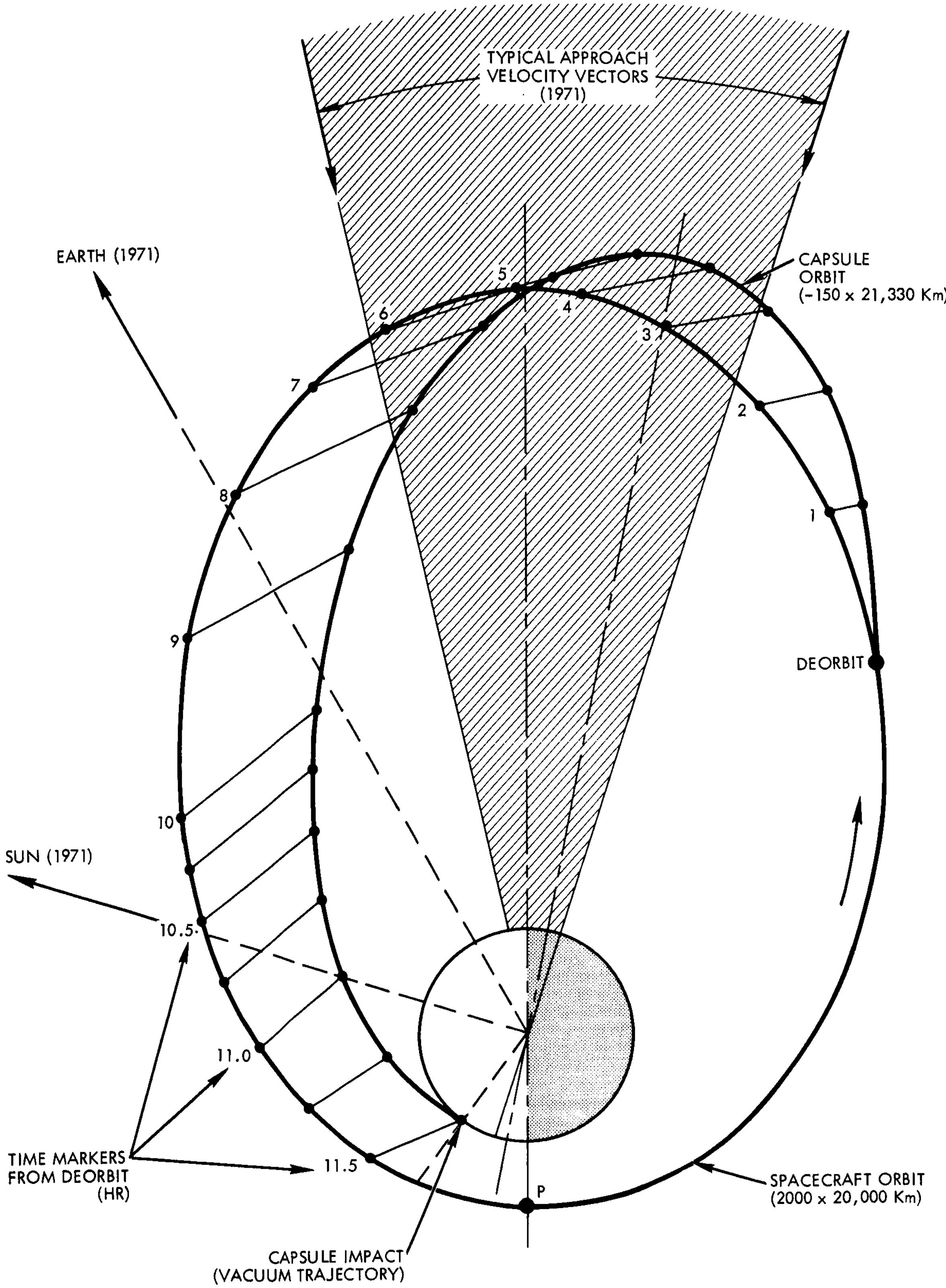

Figure B-1b. Spacecraft Orbit and Long Capsule Descent (Class II) angle approximately equal to or greater than 180 degrees. As Figure B-1 indicates, both classes of orbits meet the requirement of a favorable look angle and range from spacecraft to capsule at the time of atmospheric descent and landing, i.e., at the time of maximum data acquisition by the capsule and of maximum data rate of the relay link.

The capsule impacts Mars approximately at the time when the orbiter passes overhead. The impact point is chosen close to the orbiter's subperiapsis point to assure a short communication distance during the atmospheric descent phase. Look angle directions vary over only a small angular range during the entire descent, thus making feasible the use of a single, fixed relay antenna. This is shown in Figures B-1a and B-1b, or in Figure B-2, a polar plot of loci of relative ranges and look angles as seen from the spacecraft for descent orbits of Class I and II.

Deorbit points at a central angle of less than 90 degrees from the orbiter periapsis tend to impose excessive ΔV requirements for the deorbit maneuver (greater than 550 meter/sec) and have therefore been ruled out. Deorbit points close to the apoapsis, while yielding the smallest deorbit ΔV, correspond to unsuitable descent paths for both Class I and II orbits. This is due to unfavorable spacecraft-to-capsule look angles at the time of atmospheric descent (for Class I), and to unfavorable deorbit maneuver orientation of the capsule (Class I and II).

Table B-2 compares typical characteristics of Class I and II orbits from a standpoint of meeting the imposed mission, operational and design constraints. Class II orbits are preferred since only in this descent mode can the capsule remain in the separation attitude until entry, e.g., by use of spin stabilization, and achieve entry angles of attack not exceeding 60 degrees. Descent orbits of Class I cannot meet this constraint. However, for a given error in executing the deorbit maneuver the capsule landing point accuracy with the shorter Class I descent orbits would be greater than for Class II orbits.

The preferred class of descent orbits (Class II) meets all constraints, but with only a narrow margin in some instances. To assure the desired intervals for capsule and orbiter visibility from earth during

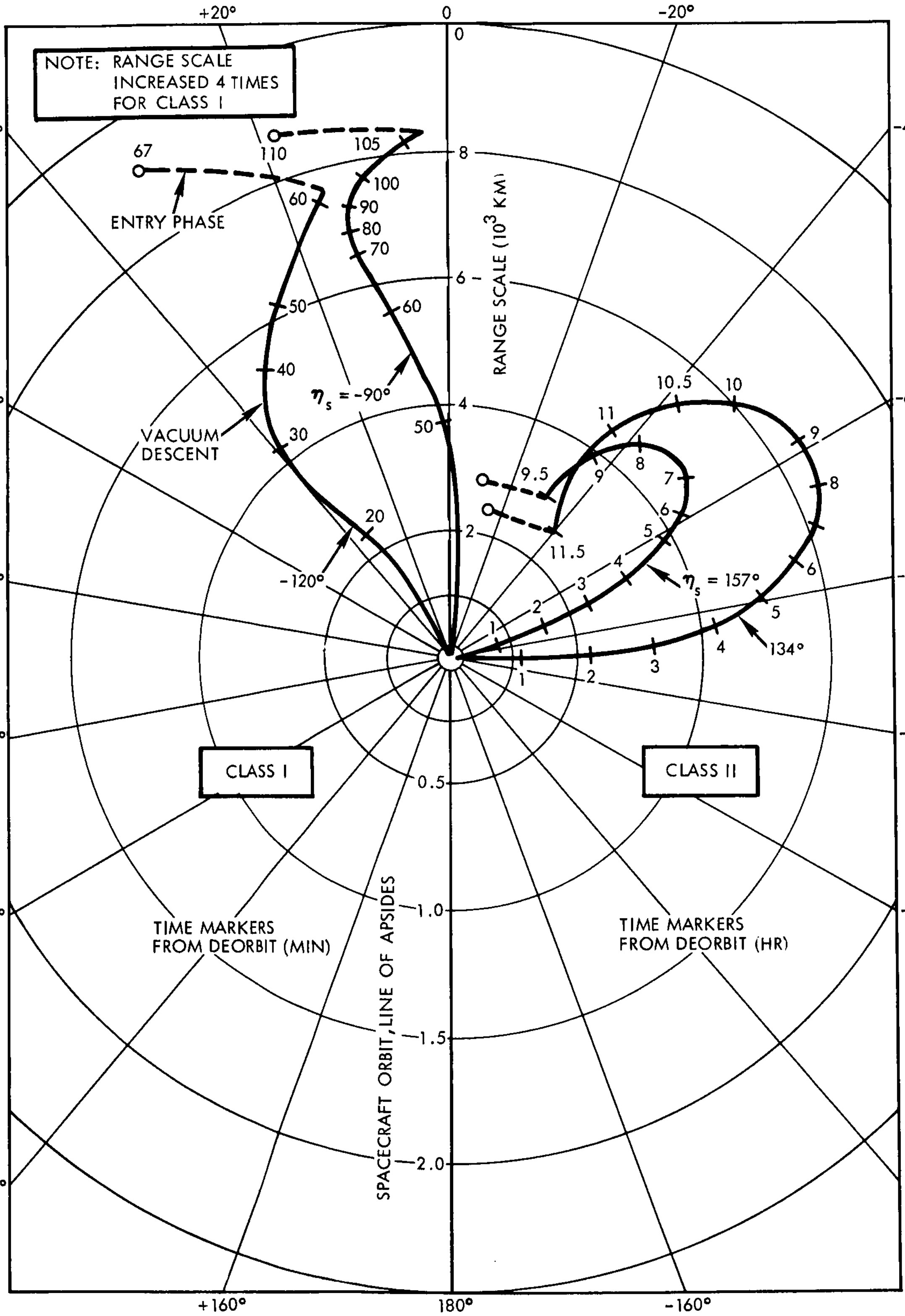

Figure B-2. Polar Plot of Look Angle and Range for Typical Class I and II Descent Trajectories

Table B-2. Comparison of Class I and II Orbits

Criterion	Capsule Descent Orbit		
	Class I	Class I and II	Class II
Deorbit relative to orbiter periapsis	100-130 deg prior to periapsis		140-150 deg past periapsis
Capsule deorbit ΔV requirement		Similar ~ 200-400 meter/sec	
Capsule flight time	Short (80-120 minutes)		Very long (10 to 12 hr)
Capsule landing point accuracy for given deorbit maneuver execution error	Higher accuracy*		Lower accuracy (orbit has higher error sensitivity)
Capsule entry velocity	*		Higher by 5 to 15 percent due to required greater apoapsis distance
Entry corridor	*		Reduced due to greater apoapsis distance
Maximum communication range	2000 km or less during capsule atmosphere descent*		2000 km (or less) during capsule atmosphere descent. 5000-6000 km in midcourse.
Near-vertical look-down during capsule descent		Achievable in both cases	
Orbiter visibility from earth during capsule descent; separation visibility from Goldstone	*	Achievable in both cases (more easily in Class II)	
Landing in 15-30 deg zone near a terminator		Achievable in both cases	
Roll axis TV camera oriented away from sun		Achievable in both cases	
Angle of attack at entry if deorbit maneuver attitude is maintained	Exceeds 60 deg		Below 60 deg if deorbit and entry points are suitably chosen*
Constraints on planetary vehicle Mars orbit injection	None; can use periapsis-to-periapsis (minimum energy) injection*		Yes. To meet constraints 3 and 4 (Table B-1) must reorient apsidal line by ~ 35 deg from minimum ΔV injection condition, in 1971 (~ 20 percent higher ΔV required; not serious in 1971, 1973)
Constraints on spacecraft attitude during capsule descent	Short descent time may require spacecraft to remain in separation attitude until capsule impact		Long descent time permits spacecraft reorientation to cruise attitude before capsule impact*

*Indicates relative advantage.

the descent phase and to assure landing in the 15 to 30 degree zone adjacent to one of the terminators, it is necessary to adopt apsidal orientations for the orbiter and the capsule orbits as shown in Figure B-1b. Determination of the desired orbit must proceed in two steps: first, the deorbit, entry and landing points relative to the spacecraft orbit are selected so as to meet the orbital mechanics constraints; second, the resulting spacecraft and capsule orbit orientations are rotated relative to the earth and sun line so as to meet the lighting angle and earth visibility constraints. The procedure may have to be iterated. For the case depicted in Figure B-1b it was necessary to apply a counterclockwise rotation of the spacecraft orbit apsidal line by approximately 35 degrees from the orientation, which permits a minimum-energy orbit injection, i.e., periapsis-to-periapsis.* However, the ΔV penalty of the modified orbit injection maneuver is sufficiently small (e.g., $\Delta V = 1.47$ km/sec for a 30-degree apsidal rotation in 1,000 by 20,000 km orbit, where $\Delta V_{min} = 1.29$ km/sec) to justify the use of this adaptation technique for the spacecraft orbit in 1971 and 1973.

The above results are derived from a preliminary analysis conducted for a nominal Mars orbit of 2,000 by 20,000 km (apsidal altitudes) at typical arrival times in 1971 and 1973. A more extensive study of the effect of spacecraft orbit dimensions and of the influence of ZAP, ZAE, ETS, and ETE angles as function of launch and arrival time will be necessary before firming up the capsule orbit selection and the relay link design requirements. It is also noted that the analysis approximates the true orbit orientation by a projection into the Mars ecliptic plane.

The following data describes characteristics of the selected Class II capsule descent orbit under conditions of a nominal 2,000 by 20,000 km spacecraft orbit and arrival time of 10 November 1971.

*Voyager Spacecraft Phase IA Study Report, Volume 4, Appendix D, "Nominal 1971 Trajectory and Orbit," TRW Systems Document No. 5410-0004-RU-001, pp. 338-345, 30 July 1965.

• Apsidal line shift of descent ellipse relative to spacecraft orbit	-4.0 degrees*
• True anomaly of deorbit point (relative to orbiter periapsis)	157 degrees
• True anomaly of capsule entry point	318 degrees
• True anomaly of capsule impact (vacuum trajectory)	333 degrees
• Time of descent, deorbit to entry	9.85 hours
• Average line-of-sight orientation prior to entry (relative to apsidal line)	-50 degrees
• Line-of-sight angle maximum variation	65 degrees
• Maximum communication distance	4800 km
• Maximum communication distance in atmospheric entry phase	2600 km
• Line-of-sight angle at capsule impact:	
vacuum trajectory	-32 degrees
atmospheric trajectory	-10 degrees
• Margin of orbiter visibility from earth after capsule landing (central angle)	27 degrees
• Capsule impact position from evening terminator (vacuum impact)	9 degrees
• Capsule deorbit ΔV	227 meter/sec

Capsule descent orbits of Class II for various positions (true anomalies) of the deorbit point in the vicinity of the above example have been investigated under the condition that the orbiter pass over the capsule impact site at the time of capsule impact. (This condition is met approximately by letting the capsule vacuum trajectory intersect the planetary surface 350 seconds before spacecraft crossover, thereby taking into account the effect of atmospheric deceleration.) The resulting variations in capsule descent time, required velocity increment for the deorbit

*Counterclockwise rotation is designated by positive angles.

maneuver, the angle of attack at entry, and the relative apsidal orientation of the capsule orbit are shown in Figure B-3. It is noted that the angle of attack constraint ($\alpha < 60$ degrees) and the time-of-flight constraint ($T_c <$ 12 hours) are more critical in the selection of the deorbit point than the ΔV constraint ($\Delta V < 550$ meter/sec). A true anomaly $\eta \geq 130$ degrees at deorbit satisfies these requirements. The selected descent orbit described by the characteristics listed above is shown in Figure B-3.

2. EFFECT OF ATMOSPHERIC ENTRY ON RELAY LINK GEOMETRY

The above relay link geometry data was derived by approximating the terminal descent in terms of a vacuum trajectory. A more accurate analysis of relay link geometry must take the effect of atmospheric drag deceleration into account. A major difference due to atmospheric drag is the increase by several minutes of the atmospheric descent time, hence an increase in line-of-sight variation in the terminal phase. A lesser effect is the reduction of downrange distance of the capsule landing point compared to the vacuum trajectory.

Atmospheric descent trajectories have been studied for selected entry conditions representing Class I trajectories with entry angles from 12.9 to 17.5 degrees. Four Mars model atmospheres, VM-3, 4, 7, and 8, have been assumed in this study, and various capsule ballistic coefficients m/C_DA between 0.2 and 0.3 were postulated for typical blunt sphere-cone capsule entry body configurations. A subsonic parachute sized to reduce impact velocities to less than 100 ft/sec was assumed to be deployed at 15,000 feet altitude. Figures B-4 and B-5 show the resulting descent times and downrange distances between entry, assumed at 200 km altitude, and impact for the densest (VM-3) and thinnest (VM-8) Mars model atmospheres, versus entry angle. Table B-3 compares the atmospheric descent times and downrange distances obtained for the densest model atmosphere (VM-3) having 10 mb surface pressure and 47,000 feet scale height, with and without parachute deployment, against the results of the vacuum trajectory computation. The table also shows the resulting terminal range and look angle for each case. The results indicate that the spacecraft/capsule relative geometry during the last

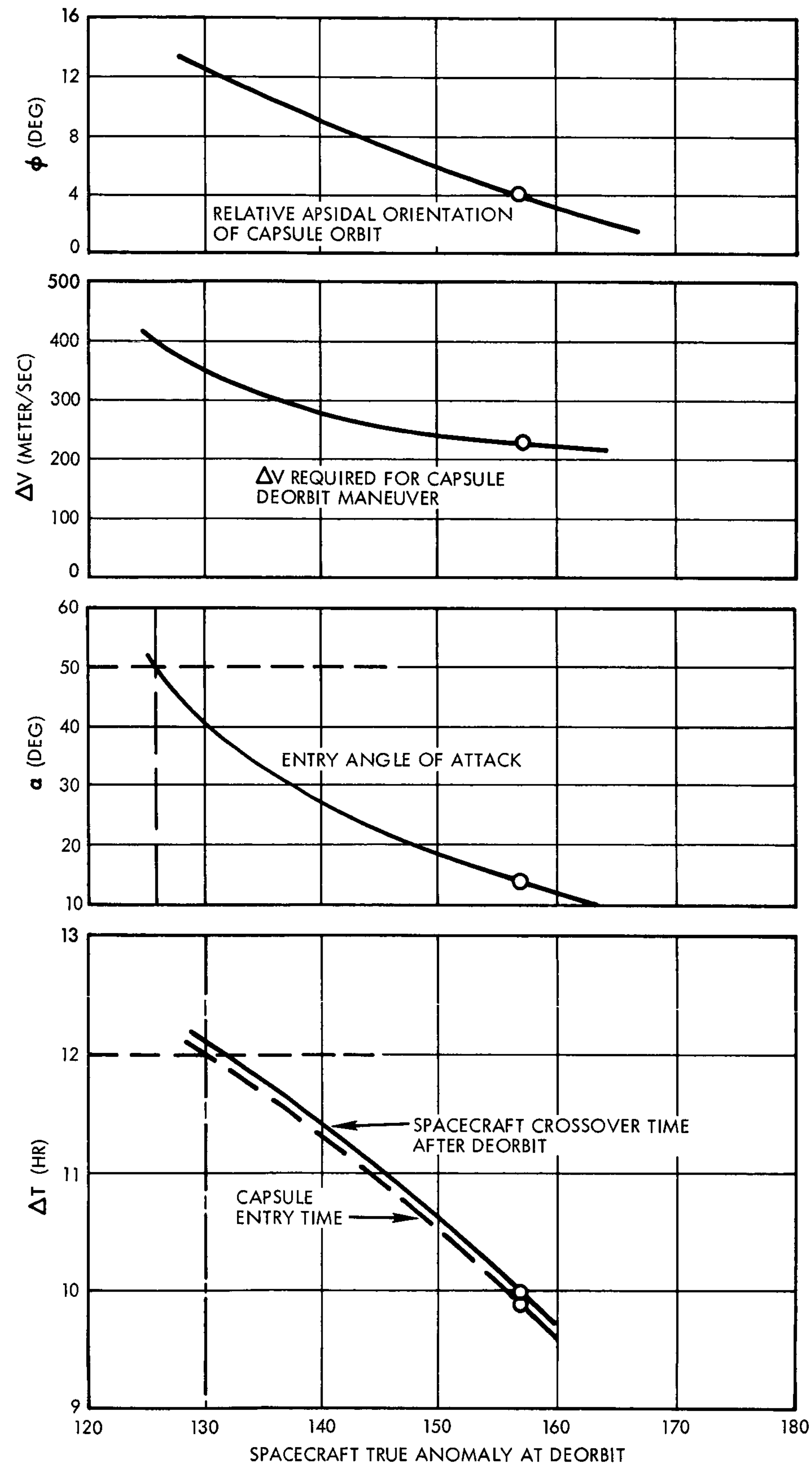

Figure B-3. Capsule Descent Characteristics for Class II Trajectories Versus True Anomaly of Deorbit

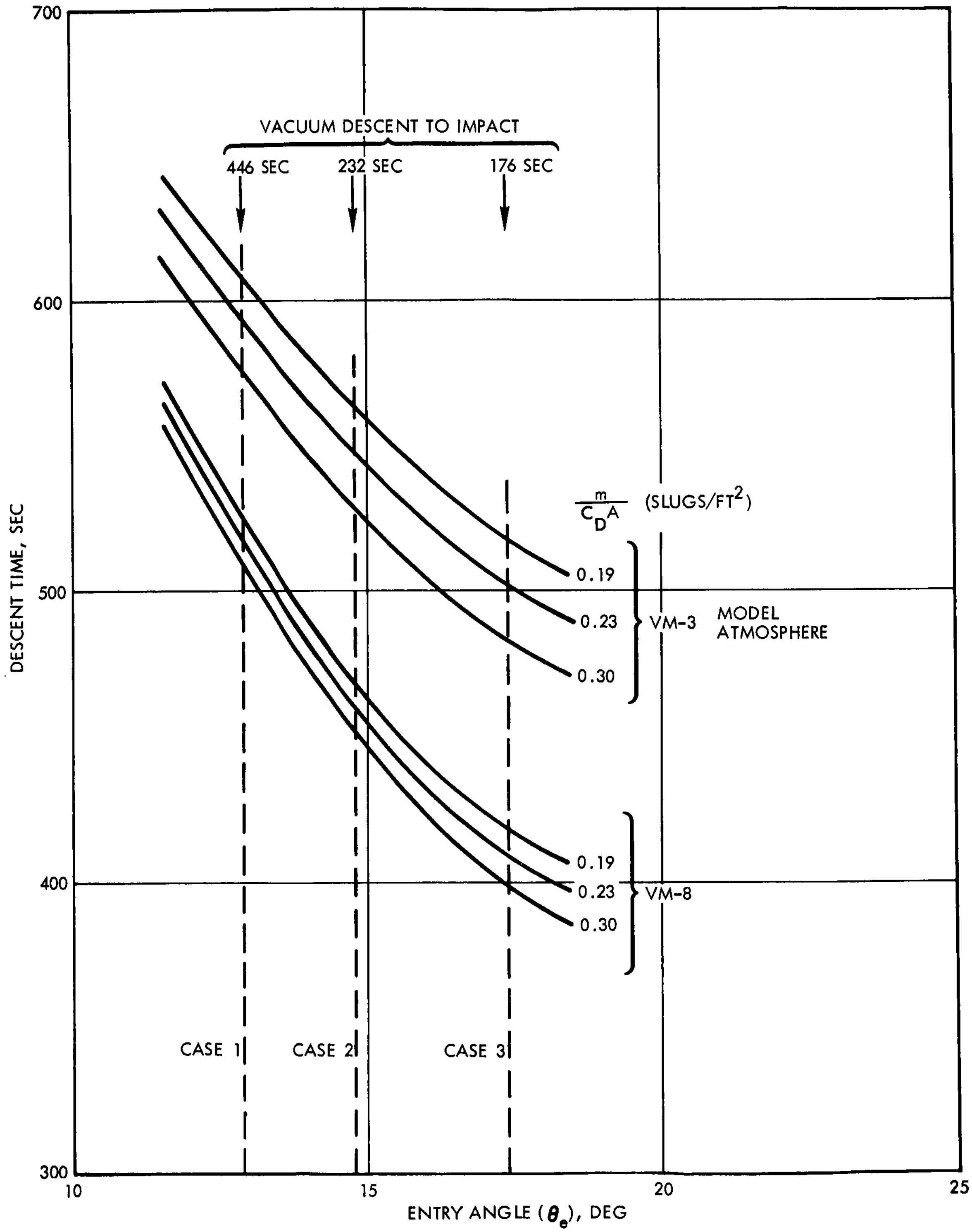

Figure B-4. Capsule Atmospheric Descent Time From Entry at 200 km (With Parachute)

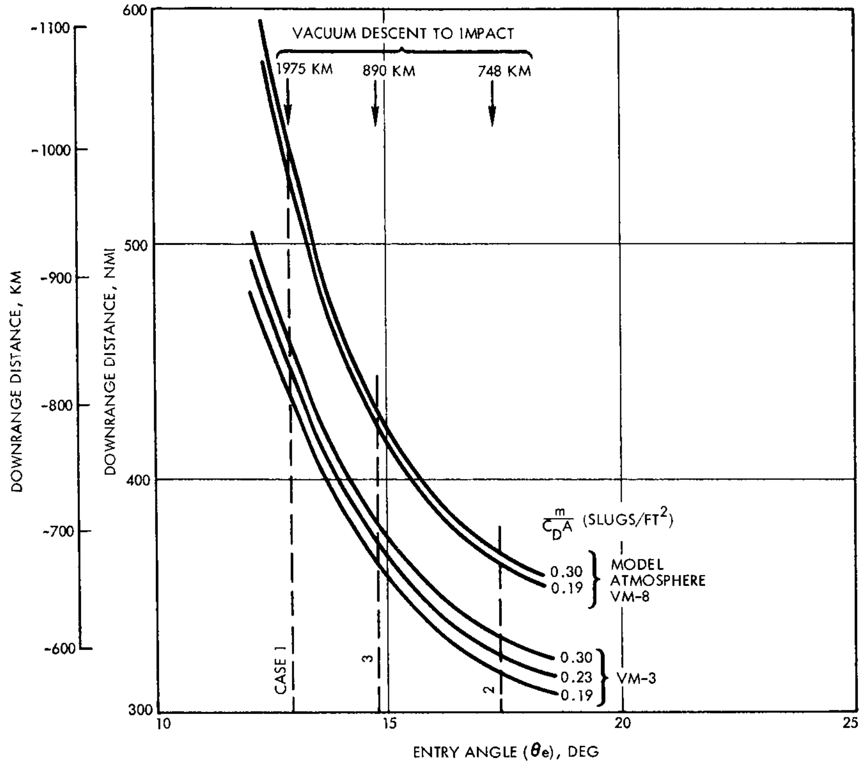

Figure B-5 Horizontal Distance Traveled During Atmospheric Descent From Entry at 200 km Altitude (With Parachute)

Table B-3. Comparison of Terminal Conditions for Capsule Atmospheric Descent and Vacuum Trajectory

Assumed Conditions:
Entry angle 17.4 deg
Entry velocity 14,330 ft/sec
Drag coefficient $C_D = 1.6$ (65 to 70 deg sphere-cone)
Ballistic coefficient $m/C_DA = 0.19$ slugs/ft^2

Deorbit at spacecraft true anomaly of -120 deg (Class I)
Model atmosphere VM-3

	Atmospheric Descent		
	With Parachute	Without Parachute	Vacuum Trajectory
Descent time from entry (sec)	519	351	176
Downrange distance on surface (km)	590	590	748
Terminal velocity (ft/sec)	71	527	14,400
Terminal flight path angle (degrees)	90	85	11
Terminal communication range (km)	2,200	2,140	2,050
Terminal look angle* (deg)	31	22	-4
Look angle at entry* (deg)	1	1	1
Look angle variation during descent (deg)	30	21	-5

* Relative to spacecraft line of apsides

5 to 10 minutes before impact as obtained for atmospheric and vacuum descent trajectories does not yield substantially different design requirements for the spacecraft relay link antenna. Approximation by computation of vacuum trajectories appears satisfactory for purposes of this study. However, a more comprehensive analysis is required before firming up the antenna design.

3. CONE AND CLOCK ANGLE REQUIREMENTS FOR SPACECRAFT RELAY LINK ANTENNA

The preceding discussion of relative motion was simplified by projecting capsule and spacecraft orbits into the Mars ecliptic plane. For antenna design purposes the three-dimensional geometry of the line-of-sight variation during capsule descent must be considered. Figure B-6 shows loci of view angles for several orbital orientations relative to spacecraft coordinates where the spacecraft is assumed to be in a nominal Sun-Canopus-oriented cruise attitude. The coordinates of this diagram

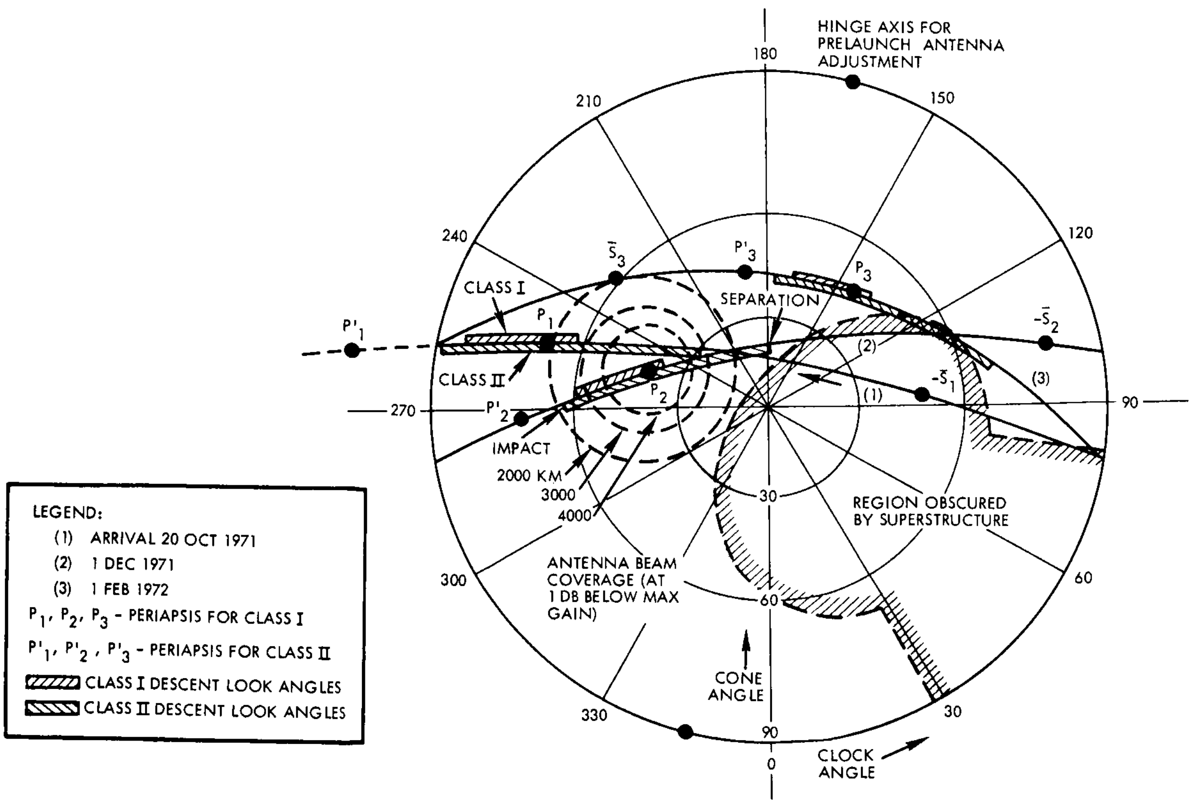

Figure B-6. Coverage of Class I and II Capsule Descent Trajectories by a Fixed Relay Antenna are cone and clock angles projected over the spacecraft rear hemisphere* i.e., for cone angles ranging from 90 to 180 degrees.

The three orbits designated (1), (2), and (3) correspond to arrival dates at Mars on 20 October 1971, 1 December 1971 and 1 February 1972, respectively.

Variations of ZAP and ETS angles with arrival date are reflected by different orbit orientations relative to the spacecraft coordinates. A 45 degree orbit inclination against the Mars equator is assumed. Note that successive suborbit points run from right to left in the projection used in the diagram.** Periapsis positions are indicated on the three sample orbit tracks by points P_1, P_2, P_3. During the capsule descent the line-of-sight vector is confined to the orbit plane, hence it must follow the orbital track. However, the time phasing of the capsule line-of-sight angles. This is illustrated in the orbit diagram, Figure B-7, by a sequence of line-of-sight angles for several capsule descent orbits which intercept the planet's surface at different points. Circular antenna beam contours of fixed relay antennas sized for operation at maximum ranges of 2000, 3000, and 4000 km are shown superimposed on the projected capsule line-of-sight angles in Figure B-6. The required maximum antenna gains are 6.5, 10, and 12.5 db, respectively. The contours correspond to antenna beam widths at 1 db below maximum antenna gain. The plot shows the feasibility of full capsule descent coverage for typical orbit phasing conditions corresponding to Class I capsule descent trajectories. It is seen that for the chosen direction of the antenna beam center line, at a cone angle of 140 degrees and clock angle of 255 degrees, adequate coverage is obtained by the fixed antenna, for early and intermediate arrival dates.

For Class II orbits the same beam orientation also achieves the desired capsule descent coverage if the periapsis positions are shifted counterclockwise along the orbit track by approximately 35 degrees (see Section 1), indicated by points P_1', P_2', P_3', i.e., to the left of the respective positions P_1, P_2, P_3.

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*A stereographic projection is used for convenience.

**Assuming a posigrade orbital rotation.

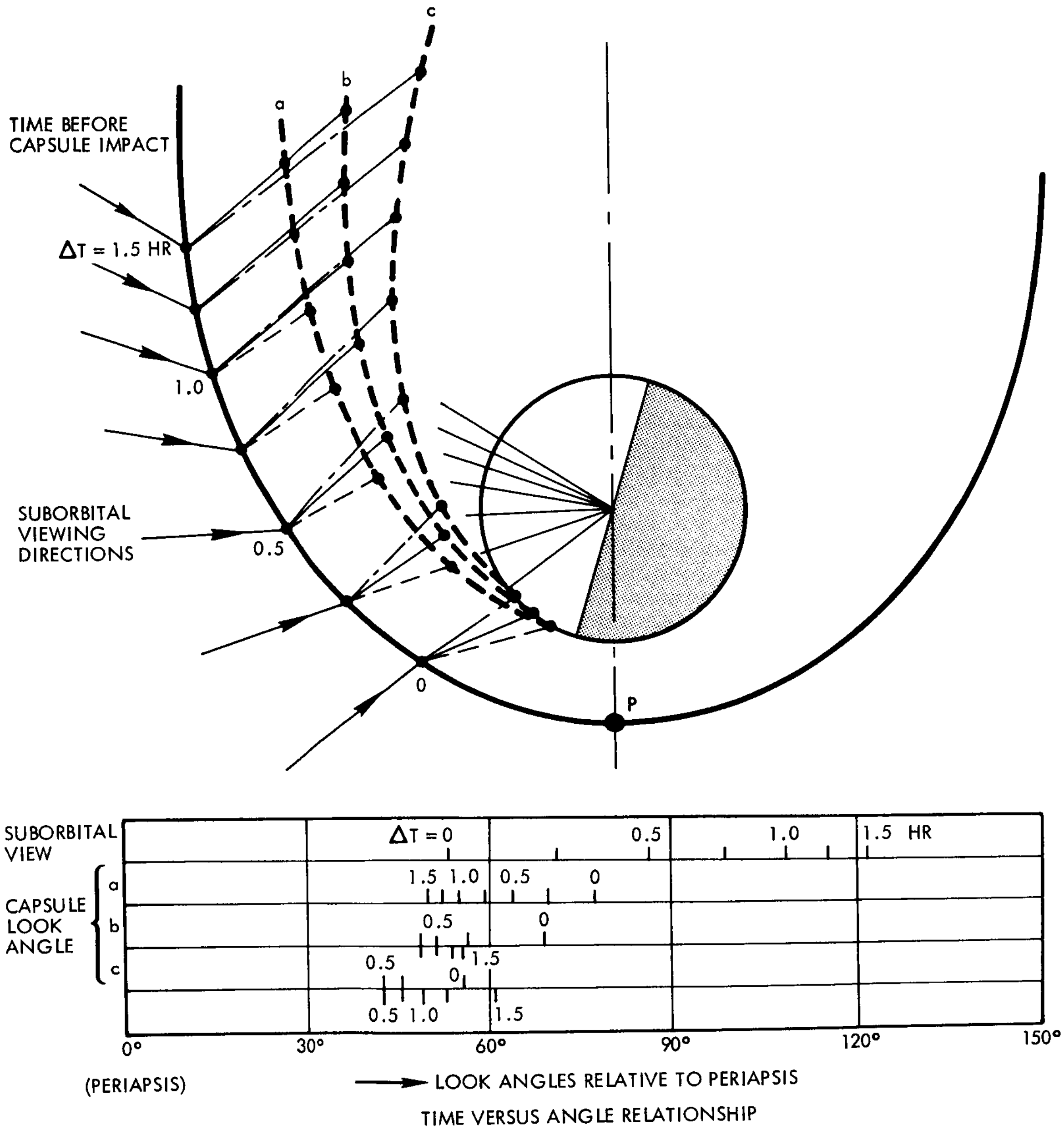

Figure B-7 Capsule Look Angles Versus Time Before Impact (Vacuum Trajectories)

Capsule descents for the case of late arrival represented by orbit track (3) (1 February 1972) cannot be covered by the fixed antenna beam even with a more substantial counterclockwise shift of P_3. However, late arrivals of this type are unlikely to be chosen for the 1971 mission since they do not significantly extend the launch window and require a greater extended interplanetary cruise period (typically, 240 days). Furthermore, such trajectories would fall close to certain mission constraint limits on the declination of launch asymptote and on minimum out-of-plane angles of the transfer trajectory.* The early and intermediate arrivals, cases (1) and (2), originate from a large unrestricted region of the 1971 launch window, and hence are of greater practical interest.

Antenna beam coverage can be shifted to different regions in the rear hemisphere by a prelaunch reorientation of the antenna beam, if a significant change in launch or arrival dates becomes necessary or if the planned capsule descent orbit is modified. One design approach would provide for a remotely controlled antenna rotation such that the antenna beam can be placed conveniently into the desired new orbital region. Figure B-6 shows a favorable orientation of the antenna hinge axis at a cone angle of 90 degrees and a clock angle of 170 degrees.

A change in the inclination of the spacecraft orbits relative to the Mars equator would be reflected by a rotation of the orbital tracks shown in Figure B-6 around the arrival velocity vector designated $\overline{S}_1$, $\overline{S}_2$, $\overline{S}_3$. It is seen that capsule descent coverage by a fixed antenna is strongly dependent on the orbit inclination. A major change in the intended inclination angle would necessitate a reorientation of the antenna beam prior to launch. Conversely, an adjustment of orbit inclination angle can be used to provide adequate coverage by the fixed-beam antenna over an extended regime of arrival conditions.

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*Voyager Spacecraft Phase IA Study Report, Volume 4, Appendix D, "Nominal 1971 Trajectory and Orbit," TRW Systems Document No. 5410-0004-RU-001, Figure D-1, p. 339, 30 July 1965.

Effects of shadowing and beam pattern interference by the spacecraft superstructure and the overhanging capsule canister bottom must be considered in a more comprehensive relay link analysis. In this preliminary study only the geometrical shadowing effect has been investigated and is depicted in Figure B-6 by the shaded keyhole-shaped region having a center line of 55 degrees clock angle.* The center of the flat array antenna is mounted recessed 10 inches from the rim of the solar panel structure. The capsule canister overhang is assumed to extend to a radius of 120 inches at a distance of 85 inches above the array center.

It is seen that for the proposed antenna location visibility of the capsule is not assured under arrival condition (3) due to the partial obscuration of the upper right hand quadrant of the hemisphere. However, the question of what portion of the canister remains with the spacecraft after orbit insertion and capsule separation is still open to clarification.

*This corresponds to a placement of the relay antenna at a clock angle of 180 +55 degrees.

APPENDIX C

ALLOWABLE INTERPLANETARY TRAJECTORY CORRECTION ERRORS

1. ACCURACY REQUIREMENTS

Reference 1 requires a flight spacecraft performance capability of interplanetary trajectory correction to accuracies compatible with achieving a 3σ semi-major axis of 100 km for the dispersion ellipse in the $\overline{R}$, $\overline{T}$ plane at Mars, under assumed conditions expressed as follows:

- Launch date: May 24, 1971
- Arrival date: December 18, 1971
- A 3σ launch vehicle injection error correctable with a correction velocity increment of 10 meter/sec
- Execution of a first trajectory correction maneuver 5 days after injection
- Execution of a second trajectory correction maneuver 30 days prior to planetary encounter
- No orbit determination error contributions
- No arrival date separation or biasing between successive aiming points.

Further, as a design goal, the flight spacecraft will achieve the specified accuracies with the execution of the second trajectory correction maneuver 30 days after injection. In addition, the error in planetary vehicle time of arrival at Mars attributable to the spacecraft is not to exceed 1 minute (3σ).

The following analysis explains the relation between those accuracy requirements stated in mission terms and the related spacecraft accuracy requirements stated in terms of achieving a specified vector velocity increment.

1.1 Interplanetary Trajectory Characteristics

An integrated trajectory from earth to Mars was computed using the TRW Interplanetary Search computer program. The spacecraft was injected into a geocentric escape trajectory on 24 May 1971, and the trip time was approximately 207 days. The injection and terminal parameters of the trajectory are shown in Tables C-1 and C-2, respectively. The program also computed midcourse sensitivities matrices at 5, 30, and 177 days after injection. The elements of the time dependent midcourse sensitivities matrices (MS) are the partial derivatives of the terminal parameters ($\overline{B} \cdot \overline{T}$, $\overline{B} \cdot \overline{R}$, flight time) with respect to the position and velocity of the vehicle at time t.

Table C-1. Nominal Trajectory Injection Parameters

Geocentric Radius	6717.3199 km
Right Ascension	184.43000 deg
Declination	7.5200000 deg
Geocentric Velocity	11.249054 km/sec
Flight Path Angle	88.115213 deg
Azimuth	119.85973
Time of Injection	24 May 1971 $22^h\ 5^m\ 46.80^s$ GMT

Table C-2. Terminal Parameters

Time of Arrival	18 Dec 1971 $0^h\ 0^m\ 39^s$ GMT
Flight Time	$207^d\ 1^h\ 54^m\ 42.20^s$
$\overline{B} \cdot \overline{T}$	3599.9234 km
$\overline{B} \cdot \overline{R}$	3545.5391 km

The terminal parameters were computed in a $\overline{B} \cdot \overline{T}$, $\overline{B} \cdot \overline{R}$ coordinate system. The impact parameter, $\overline{B}$, is defined as a vector originating at the center of the target and perpendicular to the incoming asymptote of the target-centered approach hyperbola. The impact parameter is resolved into two components which lie in a plane normal to the incoming asymptote, $\overline{V}_{\infty}$. We define a unit vector T lying in a plane parallel to the ecliptic plane according to

$$\vec{T} = \frac{\vec{V}_{\infty} \times \vec{k}}{\vec{V}_{\infty} \times \vec{k}}$$

where $\vec{k}$ is a unit vector normal to the ecliptic plane and pointing towards the north. The $\vec{R}$ axis is defined by

$$\vec{R} = \frac{\vec{V}_{\infty} \times \vec{T}}{\vec{V}_{\infty} \times \vec{T}}$$

The impact parameter, $\overline{B}$, lies in the R-T plane and has components $\overline{B} \bullet \overline{T}$ and $\overline{B} \bullet \overline{R}$.

The state vectors were computed in an X, Y, Z coordinate system. This is a right-handed cartesian geocentric equatorial inertial coordinate system; the X axis points towards the vernal equinox, the XY plane is the equatorial plane, and the Z axis points towards the north along the earth's spin axis. The MS matrix (i.e., midcourse sensitivities matrix) is then

$$MS = \begin{bmatrix} \frac{\partial B \bullet T}{\partial X_m} & \frac{\partial B \bullet T}{\partial Y_m} & \cdots & \frac{\partial B \bullet T}{\partial \dot{Z}_m} \\ \frac{\partial B \bullet R}{\partial X_m} & \frac{\partial B \bullet R}{\partial Y_m} & \cdots & \frac{\partial B \bullet R}{\partial \dot{Z}_m} \\ \frac{\partial T_f}{\partial X_m} & \frac{\partial T_f}{\partial Y_m} & \cdots & \frac{\partial T_f}{\partial \dot{Z}_m} \end{bmatrix}$$

At five days after injection, it was assumed that the 3σ value of each component of velocity uncertainty was 0.01 kilometer per second; the position uncertainty was zero. Thus, the covariance matrix of spacecraft uncertainties in position and velocity could be considered to be a 3 x 3 matrix containing only the velocity uncertainties. Defining the matrix S as the right hand 3 x 3 portion of the MS matrix, the covariance matrix of uncorrected miss at the target is given by

$$\Sigma_{um} = S_5 \, \Sigma_5 \, S_5^T$$

where

$$\Sigma_5 = \begin{bmatrix} (.01)^2 & 0 & 0 \\ 0 & (.01)^2 & 0 \\ 0 & 0 & (.01)^2 \end{bmatrix}$$

The elements of Σ_{um} are

$$\Sigma_{um} = \begin{bmatrix} \sigma_1^2 & \rho_{12}\sigma_1\sigma_2 & \rho_{13}\sigma_1\sigma_3 \\ & \sigma_2^2 & \rho_{23}\sigma_2\sigma_3 \\ \text{symmetric} & & \sigma_3^2 \end{bmatrix}$$

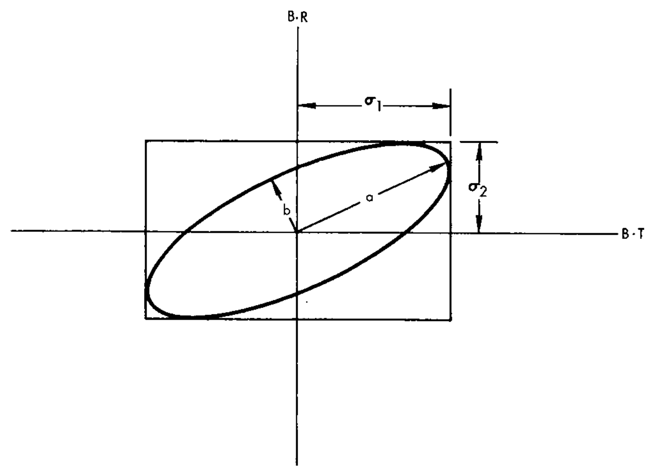
where

$$\sigma_1 = 3\sigma\,(B \bullet T)$$

$$\sigma_2 = 3\sigma\,(B \bullet R)$$

$$\sigma_3 = 3\sigma\,(T_f)$$

The 3σ miss ellipse in the R-T plane will be in a rectangle of size $2\sigma_1$ and $2\sigma_2$ centered at the nominal value of B • T and B • R. The semi-major and semi-minor axes of the miss ellipse are denoted by a and b, respectively.



The 3 x 1 miss matrix, M_{uc}, was assumed to be $\begin{bmatrix}\sigma_1 & \sigma_2 & \sigma_3\end{bmatrix}^T$. This is slightly in error because the semi-major axis, a, is slightly less than $\sqrt{\sigma_1^2 + \sigma_2^2}$. Since we are correcting a slightly larger error than normally exists, the required velocity required to null the miss will be conservative by a slight amount.

The ideal midcourse velocity required to null the miss is given by

$$\Delta V_{I5}\,(S_5) + M_{uc5} = 0$$

Solving for ΔV_{I5}, we get

$$\Delta V_{I5} = -(S_5^T\, S_5)^{-1}\, S_5^T\, M_{uc5}$$

We will now introduce a u, v, w coordinate system such that ideal velocity vector will lie along the v axis

$$\vec{v} = \frac{\vec{\Delta V_I}}{|\vec{\Delta V_I}|}$$

The w axis is parallel to the angular momentum vector such that

$$\vec{w} = \frac{\vec{R} \times \Delta\vec{V}_I}{\left|\vec{R} \times \Delta\vec{V}_I\right|}$$

where $\vec{R}$ is the radius vector of the vehicle at the time of the midcourse correction. The remaining axis, u, is then

$$\vec{u} = \vec{v} \times \vec{w}$$

The rotation matrix, Q, is given by

$$Q = \begin{bmatrix} u_1 & u_2 & u_3 \\ v_1 & v_2 & v_3 \\ w_1 & w_2 & w_3 \end{bmatrix}$$

and

$$\begin{bmatrix} 0 \\ \Delta V_I \\ 0 \end{bmatrix} = Q \begin{bmatrix} \dot{X}_I \\ \dot{Y}_I \\ \dot{Z}_I \end{bmatrix}$$

The actual midcourse correction velocity vector may be considered to lie along the v" axis of a u", v", w" coordinate system. The rotation from the u, v, w coordinate system to the u", v", w" coordinate system is accomplished by first rotating about the v axis through an angle α, which transforms to a u', v', w' coordinate system. The next rotation is about the u' axis through an angle ϵ. Thus

$$\begin{bmatrix} u'' \\ v'' \\ w'' \end{bmatrix} = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\epsilon & \sin\epsilon \\ 0 & -\sin\epsilon & \cos\epsilon \end{bmatrix} \begin{bmatrix} \cos\alpha & 0 & -\sin\alpha \\ 0 & 1 & 0 \\ \sin\alpha & 0 & \cos\alpha \end{bmatrix} \begin{bmatrix} u \\ v \\ w \end{bmatrix}$$

or

$$\begin{bmatrix} u'' \end{bmatrix} = \begin{bmatrix} \epsilon \end{bmatrix} \begin{bmatrix} \alpha \end{bmatrix} \begin{bmatrix} u \end{bmatrix}$$

The value of α is arbitrary, and 45 degrees was selected. The angle ϵ is the pointing error. There may also exist proportional errors and cutoff errors along the v" axis. Thus, the actual midcourse velocity vector, rotated from the u", v", w" coordinate system to the X, Y, Z coordinate system is given by

$$\begin{bmatrix} \dot{X}_A \\ \dot{Y}_A \\ \dot{Z}_A \end{bmatrix} = \begin{bmatrix} Q \end{bmatrix}^T \begin{bmatrix} \alpha \end{bmatrix}^T \begin{bmatrix} \epsilon \end{bmatrix}^T \left\{ (1 + K) \begin{bmatrix} 0 \\ \Delta V_I \\ \end{bmatrix} + \begin{bmatrix} 0 \\ V_{co} \\ 0 \end{bmatrix} \right\}$$

where K is the proportional error and V_{co} is the cutoff error.

The midcourse velocity execution error is the difference between the actual and ideal velocity vectors

$$\Delta V_E = \Delta V_I - \Delta V_A$$

The covariance matrix of miss at the target due to execution errors at the first midcourse is then

$$\Sigma_{Ec5} = E \left[S_5 \, (\Delta V_{E5}) \, (\Delta V_{E5})^T \, S_5^T \right]$$

The second midcourse will attempt to null the miss due to execution errors made during the first midcourse correction. The second midcourse correction will contain execution errors that will cause a miss at the target. Thus, the entire process just described was repeated using the midcourse sensitivities matrix corresponding to the time of the second correction.

1.1.1 Results

The equations above were programmed for the TRW matrix abstraction computer program. First, the magnitude of the cutoff velocity vector required to keep the miss within the desired tolerance was computed by varying V_{co} (using the same value for both corrections) and keeping ϵ and k null. The maximum allowable value of V_{co} was found to be 0.0645 meter per second when the second correction was made at 177 days after injection. Then, with V_{co} constant, ϵ and k were perturbed, with one parameter at zero and varying the other, in order to determine the largest value of ϵ and k required to keep the miss within the desired tolerance. The results are shown in Figure C-1. The two curves are the locus of maximum pointing and proportional errors such that the resultant miss will be within the desired tolerances. Thus, when V_{co} = 0.05 meter per second, the maximum allowable proportional and pointing errors will be bounded by an ellipse with a semi-major axis of 0.59 percent and a semi-minor axis of 0.55 degree.

It should be noticed that the results shown are for a second midcourse correction made at 177 days. The maximum allowable errors for a second correction at 30 days after injection were found to be too small to be practical because of the high sensitivities at the time of the correction.

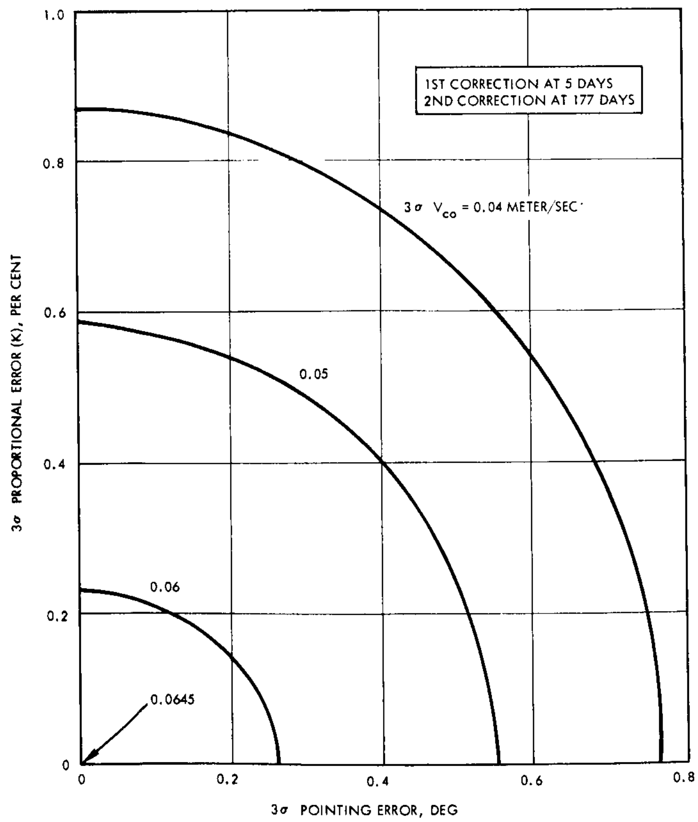


Figure C-1. Maximum Allowable Midcourse Velocity Execution Errors

APPENDIX D

AUTOMATIC SEQUENCING OF SCIENCE ORBITAL OPERATIONS USING TERMINATOR AND LIMB CROSSING TIMES

1. INTRODUCTION

This appendix discusses the scientific data gathering performance achieved with a simple automatic sequence based on data taking signals generated by suitably delayed terminator crossing and limb crossing signals. It is concluded that this automatic sequencing will provide adequate photographic coverage, including reasonable altitudes and lighting angles, throughout 6 months of orbital operations.

The analysis is based on the nominal orbit of Task A (see TRW Phase IA Study Report, Volume 4, Appendix D). As discussed in that report, this orbit provides the basic features pertinent to good photographic coverage of Mars; namely, a suitable latitude-lighting time history at arrival and over a 6-month mission period. As this latitude-lighting time history serves as the basis for the present analysis, the results derived will be valid for all orbits selected to achieve these same basic conditions for favorable photographic coverage. Further, since these conditions are independent of orbit altitude, the same basic sequence derived here can be used for various altitudes within the class of orbits of interest.

2. TERMINATOR AND LIMB CROSSING SIGNALS

2.1 Definitions (see Figure D-1)

a) Lighting Angle, Ω, is the angle between the line from the spacecraft to the sun and the line from the spacecraft to the center of Mars ($0 \leq \Omega \leq 180$ degrees).

b) Terminator Crossing is the time in an orbit at which $\Omega = 90$ degrees. There are generally two terminator crossings.

c) Dark-Light (DL) Terminator Crossing is the time T_{DL} at which $\Omega = 90$ degrees and Ω is increasing (the suborbital point is crossing from dark to light).

d) Light-Dark (LD) Terminator Crossing is the time T_{LD} at which Ω = 90 degrees and Ω is decreasing (the suborbital point is crossing from light to dark).

e) Limb Crossing is the time in an orbit at which the plane which is perpendicular to the line from the spacecraft to the sun and which contains the spacecraft is tangent to the sunlit part of Mars. Limb crossings do not occur for certain orbits (e. g. , when the orbit is perpendicular to the sun line). However, for orbits of interest to Voyager (with TV experiments), there will be two limb crossings.

First Limb Crossing is the time T_1 for the first limb crossing after T_{DL} (Ω is increasing).

Second Limb Crossing is the time T_2 for the second limb crossing after T_{DL} (Ω is decreasing).

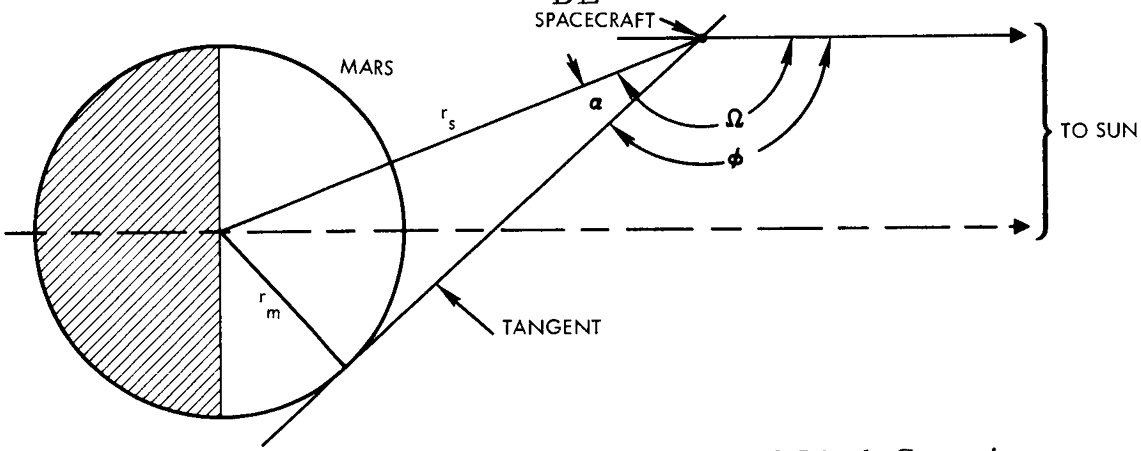


Figure D-1 Geometry for Terminator and Limb Crossings

2.2 Lighting Angles

The curves of Figure D-2 have been plotted from the data of Figures D-5, D-6, and D-7 in Appendix D, Volume 4, of the Study Report for Phase IA, Task A. Shown in Figure D-2 is the lighting angle, Ω, as a function of time from periapsis for the sample orbit at arrival, arrival plus 3 months, and arrival plus 6 months. Since there appears to be little interest in Ω < 90 degrees, those values have been omitted.

2.3 Limb Crossings

Figure D-1 shows that conditions for limb crossing are met at the times for which $\phi \equiv \Omega - \alpha = 90$ degrees, with α the first quadrant angle such that $\alpha = \sin^{-1} r_m / r_s$.

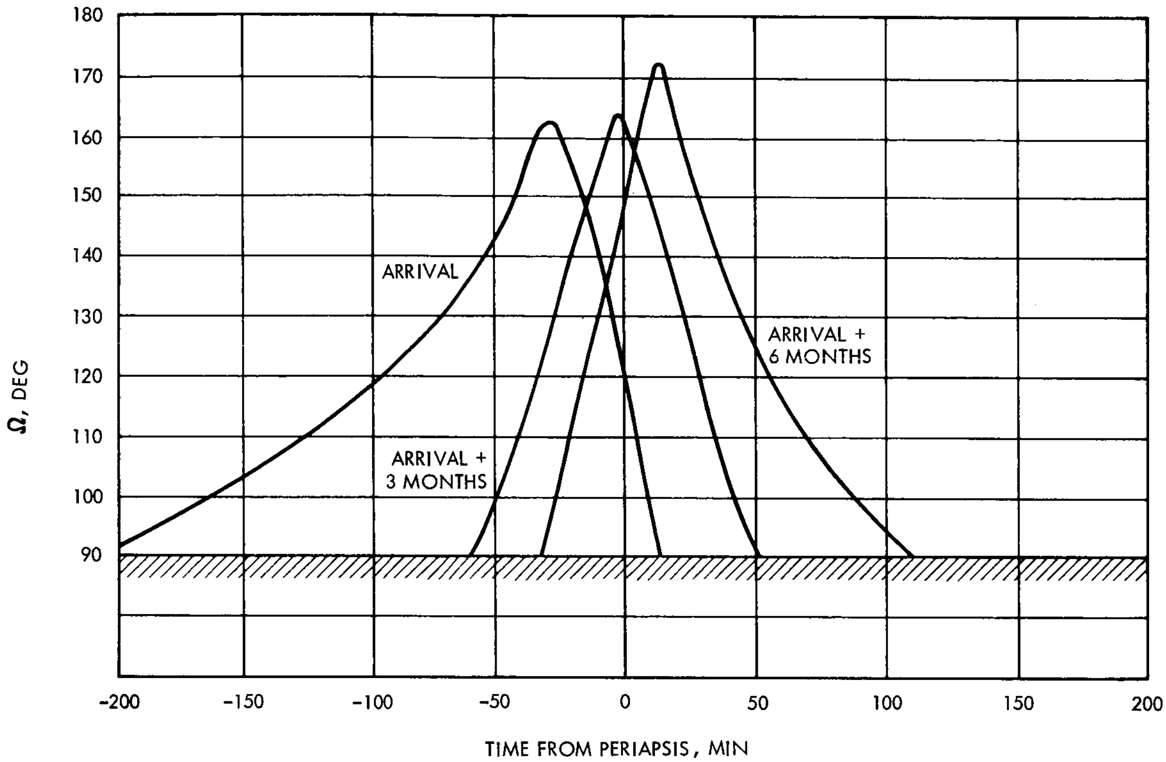


Figure D-2. Lighting Angle Versus Time From Periapsis for Sample Orbit

The curves of Figure D-3 give $\phi = \Omega - \alpha$ versus time from periapsis for arrival, arrival plus 3 months, and arrival plus 6 months. The function α versus time is calculated from the Task A data referenced in 2.2. Subtracting α versus time from the curves of Figure D-2 one obtains the curves of Figure D-3. The time from periapsis for the two limb crossings can then be read off the curves.

2.4 Sequencing

The curves of Figure D-4 show the times from periapsis for terminator and limb crossings and also for various values of lighting angle Ω plotted as functions of time in Mars orbit, for the sample orbit.

Assuming that the photoimaging experiment includes a mapping camera which is set to achieve 1 km resolution at 3330 km altitude (350 km on a side) and which takes strips of 12 pictures (filling the 10^8 bit tape recorder) with 10 per cent overlap, the Mars central angle θ_M between the centers of adjacent pictures is given by

$$\theta_M = \frac{4.8\ h}{3330}\ \text{degrees}$$

where h is the altitude in km.

Applying this assumption to the sample orbit, the lighting angle regions mapped are also shown on Figure D-4 by cross hatched strips for sequences initiated directly by limb crossings. Simplifying the controlled overlap sequence to eliminate the requirement to space the pictures by an amount proportional to altitude would result in equal mapping times, but would leave gaps at low altitude or a large amount of overlap at high altitudes. By delaying the beginning of strip mapping from the first limb crossing, any combination of lighting can be achieved. It appears that sequencing from this limb crossing using a delay which varies linearly with time after arrival from 50 minutes at arrival to 0 minutes after 6 months would produce maximum resolution maps and would cover a reasonable range of lighting angles.

Sequencing from the second limb crossing can be used without manipulation for experiments involving crossing the light-dark terminator. The time between T_2 and T_{LD} varies from about 18 minutes at arrival to about 45 minutes after 6 months.

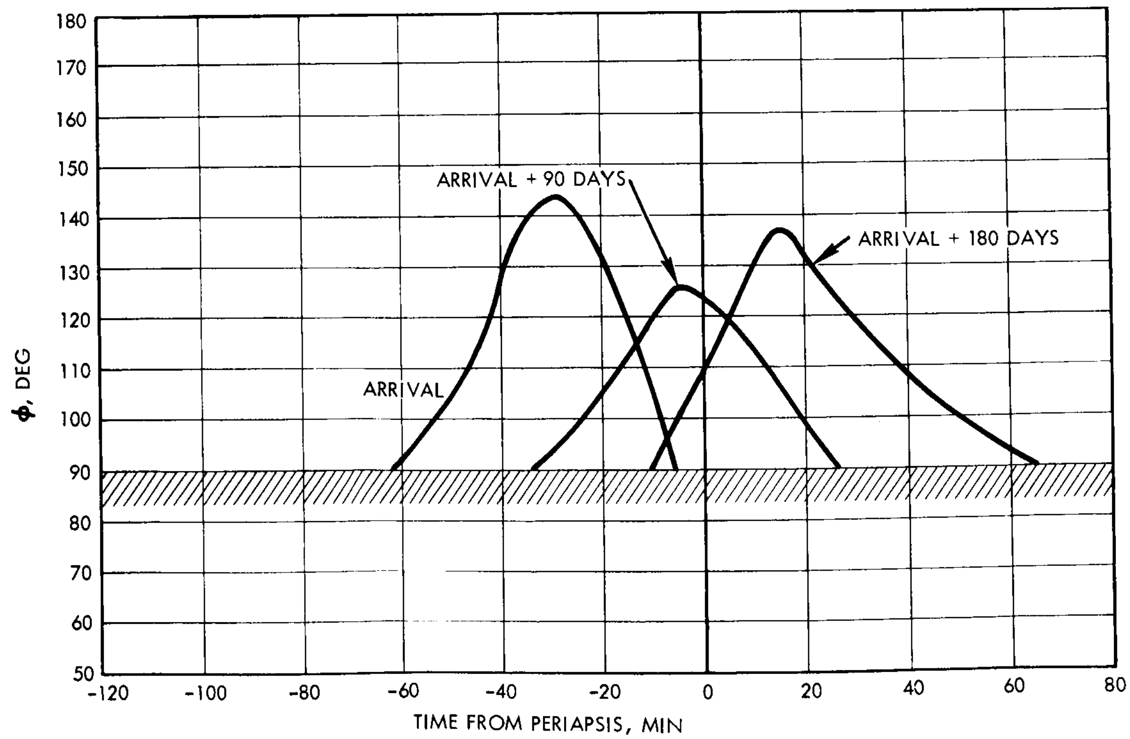


Figure D-3. Angle φ Versus Time From Periapsis for Sample Orbit

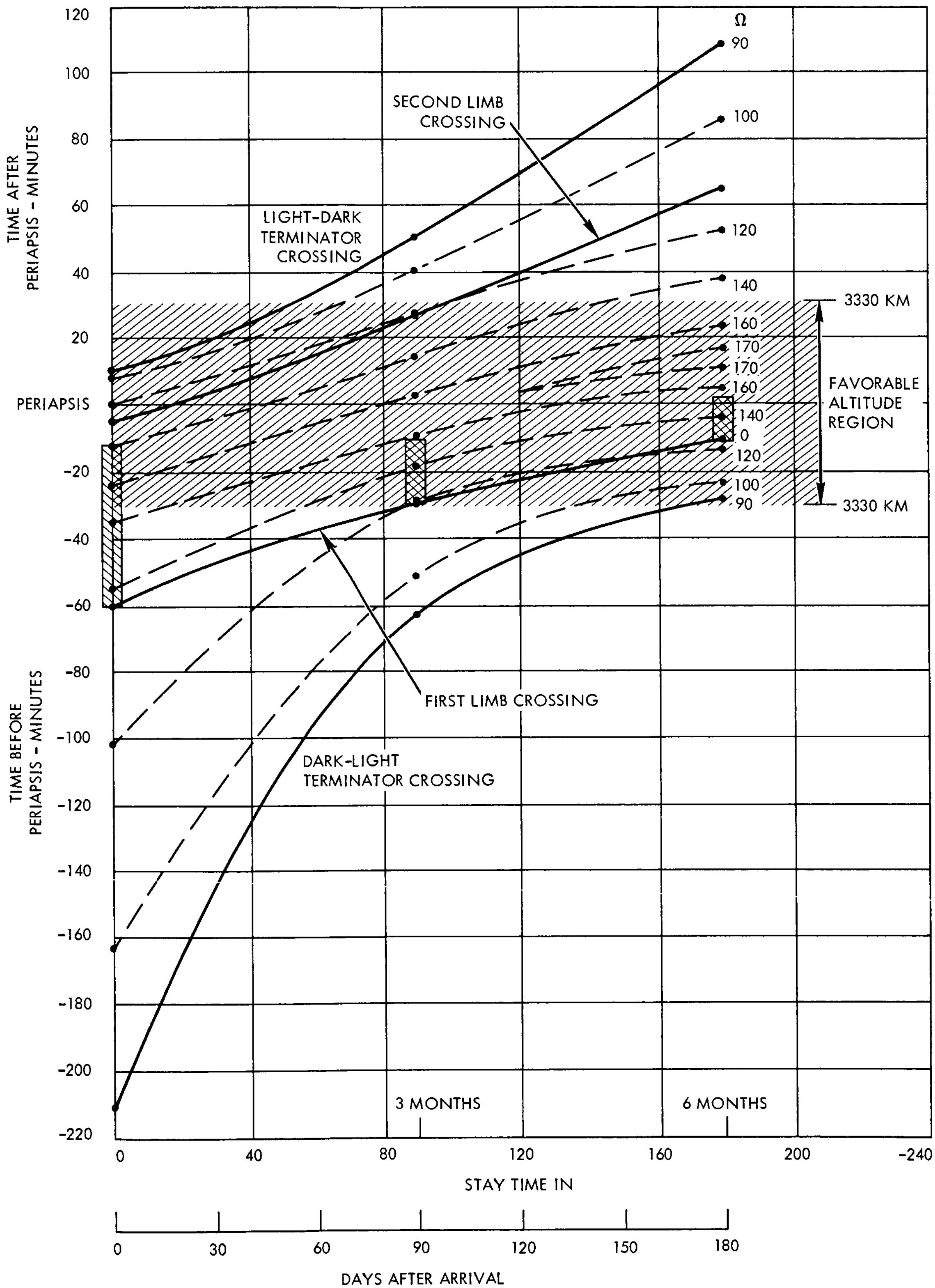


Figure D-4. Time From Periapsis for Terminator and Limb Crossings and for Various Values of Ω Plotted Versus Stay Time in Mars Orbit (Sample Orbit)

3. APPLICATION TO OTHER ORBITS

The Mariner data on Mars' atmosphere indicates that it may be practical to enter an orbit with a considerably lower periapsis than the sample orbit. From the standpoint of photoimaging this would be very advantageous. Although the suborbital speed increases with decreasing periapsis, the focal length of the lens (for a given ground resolution) decreases, an effect which decreases the required exposure much more rapidly than the suborbital speed increase and results in reduced smear. Ground resolution due to the optics and the transducers are both reduced with lower altitudes.

Preliminary analysis indicates that reducing periapsis altitude to 600 km would improve the resolution attainable with the 150 mm diameter lens analyzed in Task A by about a factor of three.

Although detailed orbit studies for a lower altitude orbit have not been completed, the following general observations can be made:

a) Limb crossing times will move closer to periapsis because of the increasing apparent size of Mars (see 2.3 above).

b) The increased suborbital speed would require a shorter time between mapping photographs, and consequently require a higher data rate input to the tape recorder.

c) The available time (non-data taking time) for transmitting picture data would be reduced for orbits with reduced apoapsis, but would not change much with changes in periapsis.

Table D-1 shows the over-all effects of some alternative orbits. It appears that lower periapsis provides significant advantages. Time intervals between terminator and limb crossings become somewhat shorter, so the sequencing delays will be shorter for lower periapsis. Consequently, automatic data sequencing based on relatively short delays from terminator and limb crossings will be satisfactory for all missions.

Table D-1. Alternative Orbit Effects

Periapsis Altitude (km)	Apoapsis Altitude (km)	Suborbital Speed (km/sec)	Orbit Period	Record Data Rate Ratio	Image Quality Improvement
600	10,000	3.5	6.8	1.4:1	3:1
600	20,000	3.6	13.7		
1200	10,000	2.8	7.3	1.2:1	1.5:1
1200	20,000	2.9	14.3		
2000	10,000	2.5	7.7	1:1	1:1
2000	20,000	2.5	14.7		

APPENDIX E

BIOLOGICAL CONTAMINATION

1. INTRODUCTION AND SUMMARY

The purpose of this appendix is to identify the areas of responsibility of the Voyager spacecraft system contractor associated with the observance of the Mars quarantine constraint. It also identifies and assesses various means by which the constraint might be violated, with the objective of calling out design requirements arising from the quarantine constraint.

Throughout this analysis a mission mode is assumed which is based on the Saturn V launch vehicle, with two planetary vehicles injected toward Mars from each launch, and with the capsule descent initiated from orbit about Mars. In addition, the Task B preferred spacecraft system design, based on the LEM descent propulsion stage, is used as the model.

The following general categories of responsibility are identified:

- The spacecraft contractor's responsibility for handling the flight capsule during that portion of the prelaunch sequence devoted to mating the capsule to the spacecraft to form the planetary vehicle, testing the planetary vehicle, and encapsulating it.
- The responsibility for the observance of constraints on the trajectory to prevent the spacecraft from entering Mars.
- Responsibility for minimizing the probability of contamination of Mars by any material emanating from the spacecraft.

The first of these responsibilities does not affect the spacecraft design directly, and will be covered in this appendix only by a brief outline. The second responsibility is a joint responsibility—the design responsibility of the spacecraft contractor to see that adequate capability and reliability characterize the spacecraft design so that trajectory constraints may be executed; and the operational responsibility assigned to the MOS for determining and executing the proper trajectory constraints. These trajectory constraints are treated briefly in outline form.

This appendix considers in detail the third responsibility, that of minimizing the likelihood that contaminating efflux from the spacecraft will impinge on Mars. It is recognized that many of the elements of the analysis of this subject are tenuous in nature, and at best give only order of magnitude results. Furthermore, many of the uncertainties do not appear to be capable of resolution by reasonable experimental procedures.

These analyses are not submitted to verify the satisfaction of the quarantine constraint. They are submitted to identify the mechanisms of contamination which are potentially critical, to gain an insight into the controlling factors for these mechanisms, and to suggest which processes should be investigated further or controlled more tightly to satisfy the constraint.

The mechanisms of possible contamination of Mars which have been identified as potentially jeopardizing the quarantine constraint are given in Table E-1.

The sources of contaminated material which appear in these mechanisms are as follows:

a) Microorganisms residing on the spacecraft surface or within surface or subsurface layers

b) Microorganisms in the spacecraft attitude control gas supply

c) Microorganisms generally residing on the spacecraft structure (in the case of pressure vessel rupture).

Means which are recommended to reduce the number of microorganisms of these sources are given below:

a) Minimize recontamination of planetary vehicle surfaces within the shroud after the ethylene oxide-freon 12 surface sterilization treatment has been employed. This probably requires that only filtered and sterilized air may be introduced into the encapsulated space.

Table E-1. Summary of Critical Mechanisms of Contaminating Mars by Efflux from the Spacecraft

Section	Route	Source	Probability of Live Entry to Mars (upper limit)	Principal Means of Lowering Probability
3.1.4	Direct contamination from transit	Surface and subsurface particles, spontaneously emitted or shaken off	10^{-3}	Minimize recontamination within shroud after surface sterilization
3.2.4	Direct contamination from orbit	Surface and subsurface particles, spontaneously emitted or shaken off	10^{-2}	Minimize recontamination within shroud after surface sterilization
3.2.5	Direct contamination from orbit	Surface and subsurface material ejected by micrometeoroids	$2.5 \cdot 10^{-3}$	Minimize recontamination within shroud after surface sterilization
				Heat sterilize multilayered thermal insulation before applying
				Heat sterilize solar panels before assembly
3.2.6	Direct contamination from orbit	Rupture of a pressurized vessel by meteoroid, and subsequent disintegration	10^{-3} per month	Reduce propellant tank pressure at completion of orbit insertion
3.3.1	Indirect contamination via exposed lander	Attitude control gas	1.5×10^{-1}	Sterilize attitude control gas supply
				Design canister to shield exposed lander
				Program orientation changes during lander descent
3.3.4	Indirect contamination via exposed lander	Surface and subsurface particles, spontaneously emitted or shaken off	7×10^{-3}	Minimize recontamination within shroud after surface sterilization
				Design canister to shield exposed lander
				Program orientation changes during lander descent

b) Maintain clean room conditions generally during spacecraft assembly; keep all external structural surfaces clean and free of any oil or grease which might trap and protect microorganisms.

c) Subject all external multilayered thermal insulation material to heat sterilization before it is applied.

d) Subject solar array panel assemblies to heat sterilization before they are assembled on the spacecraft. A cycle of 53 hours at 125°C has been tentatively approved for this procedure.

e) Heat sterilize the spacecraft attitude control system pneumatic components. Heat sterilize attitude control gas before filling.

f) Use thermal control coatings on exterior surfaces that are self-sterilizing.

Other design and operational procedures which substantially lessen the probability of contaminating Mars are as follows:

a) Design the capsule canister so that when the outer half has been jettisoned, the remaining half of the canister provides maximum line-of-sight shielding between the spacecraft and the capsule vehicle

b) If possible, program a sequence of controlled orientation maneuvers for the descending lander to execute, so as to expose all external surfaces to sunlight

c) Provide a means of reducing spacecraft propellant tank pressure to about 100 psi at (or soon after) the end of the orbit insertion maneuver.

In some areas identified in this analysis, meaningful experimental studies can be undertaken to resolve some of the more uncertain assumptions. Examples of these are:

a) Studies of the probability that meteoroid impacts can eject live microorganisms from a surface or from subsurface layers

b) Studies of the survival rates of microorganisms in a space environment, when they are attached to particles of various sizes. The environment should simulate the high vacuum of space, with and without ultraviolet radiation.

Two factors which have not been introduced in the analyses of this appendix, but which may tend to alleviate the results and reduce the resultant probability of contaminating Mars, should be mentioned. The first is the fact that microorganism populations are inherently heterogeneous. Whereas the initial populations are composed of many varieties and strains, the attenuation rates assumed to result from the various environments have been chosen to correspond to those exhibited by the most resistant strains. For exposure to vacuum, high temperature, and ultraviolet radiation, Bacillus subtilus, variety niger is one of the most resistant and it has been the model. However, studies of the composition of populations of microbes anticipated to accrue in the spacecraft manufacturing, assembly, and prelaunch operations would undoubtedly show that only a small percentage of the initial spacecraft microorganism population consists of the most resistant varieties, and that the remainder can be expected to show substantially greater attenuation rates. Such studies should be made.

The other factor has to do with the probability that a microorganism, unattached or attached or embedded in a small particle, would survive entry into the Martian atmosphere. This would apply in those instances where direct contamination of Mars is threatened by microorganisms aboard surface or subsurface flakes or particles which have been spontaneously emitted or ejected from the spacecraft. Atmospheric heating is likely to kill microorganisms on particles of certain sizes, upon entry. The very smallest particles may not be subject to heating intense enough to destroy the microorganism (but these may be sensitive to ultraviolet radiation). Larger particles will certainly be subjected to heating; particles as small as 10^{-6} grams are heated to incandescence when they enter the earth's atmosphere as meteoroids. When the particles are still larger, more heat is generated, but then the probability of survival again increases, because microorganisms embedded in the interior of such bodies may be insulated from the intense surface heat. It is appropriate for a review of these thermal characteristics associated with the entry of particles to be conducted to evaluate the effect on the potential contamination of Mars.

2. HANDLING THE CAPSULE DURING THE PRELAUNCH SEQUENCE

This section considers the operations which must be performed under the responsibility of the spacecraft system contractor as they pertain to the flight capsule. It is assumed that the flight capsule has been received in a sealed canister and has been heat sterilized through a controlled, standard procedure, such that the probability that a single live microorganism exists within the canister is estimated at some number less than 10^{-4}, with satisfactory confidence in the estimate. It is also assumed that the flight capsule mechanical design has been such that it has a reasonable immunity to damage caused by the required prelaunch operations. It is assumed that the 10^{-4} probability covers not only the likelihood of a live microorganism remaining within the flight capsule after the heat sterilization process, but also covers the possibility of undetected rupture of the sealed canister or other compromise of this sterility through the prelaunch sequence and up to injection of the planetary vehicle by the launch vehicle.

Therefore, the operations undertaken during the prelaunch sequence must have as one of their objectives a very low probability of resulting in a rupture of the sealed canister, and in particular, a significantly lower probability, of the order of 1×10^{-5}, of permitting an undetected rupture of the canister. These operations are as follows:

a) Accepting the flight capsule, with the capsule vehicle sterilized and sealed within its canister

b) Mating the flight capsule to the spacecraft

c) Performing combined tests on the planetary vehicle

d) Mating the planetary vehicle to the shroud, and adding diaphragms to encapsulate the planetary vehicle

e) Surface sterilizing the encapsulated planetary vehicle by the use of an ethylene oxide-Freon 12 gaseous mixture within the space enclosed by the shroud and the diaphragms

f) Testing the encapsulated planetary vehicle

g) Mating the encapsulated planetary vehicle assembly to the launch vehicle (the responsibility for this operation is assumed to lie with the launch vehicle system)

h) Subjecting the flight capsule only to the expected environment during launch and injection (limiting the amplification of the launch vehicle induced environment by the spacecraft is the responsibility of the spacecraft contractor)

i) Subjecting the flight capsule to the space flight environment. This environment, under the jurisdiction of the spacecraft contractor, includes the discharge of pyrotechnic devices, the thermal environment of the capsule, as influenced by the spacecraft, and mechanical and thermal environments caused by the operation of spacecraft propulsion.

3. TRAJECTORY CONSTRAINTS ON THE SPACECRAFT

This section outlines various aspects of the design responsibility to make the spacecraft compatible with and capable of observing trajectory constraints to avoid spacecraft entry to Mars, and the various aspects of the operational responsibility for identifying and executing these constraints. The Task A spacecraft system specification (in force prior to the adoption of the mission mode which is based on the Saturn V launch vehicle, and which directs two planetary vehicles to Mars for each launch) allocated a maximum probability of 2×10^{-5} for spacecraft Mars entry. It is necessary to redefine this allocation for the present mission mode. However, assuming that each spacecraft is allocated a maximum probability of the same order of magnitude, the concept of trajectory biasing will be similar to that outlined in the Phase 1A, Task A, report (Volume 4, Section 5.2). This approach must be modified somewhat to account for the requirement that the capsule be taken into orbit about Mars before it is released for planetary entry, but the general principles still apply.

3.1 Design Responsibilities

Design responsibilities may be listed as follows:

a) Velocity increment capability of the propulsion subsystem

b) Resolution and variability of the impulse of the propulsion subsystem

c) Number of engine starts available

d) Ability of the communication subsystem to provide data for spacecraft tracking and to provide verification of maneuvers

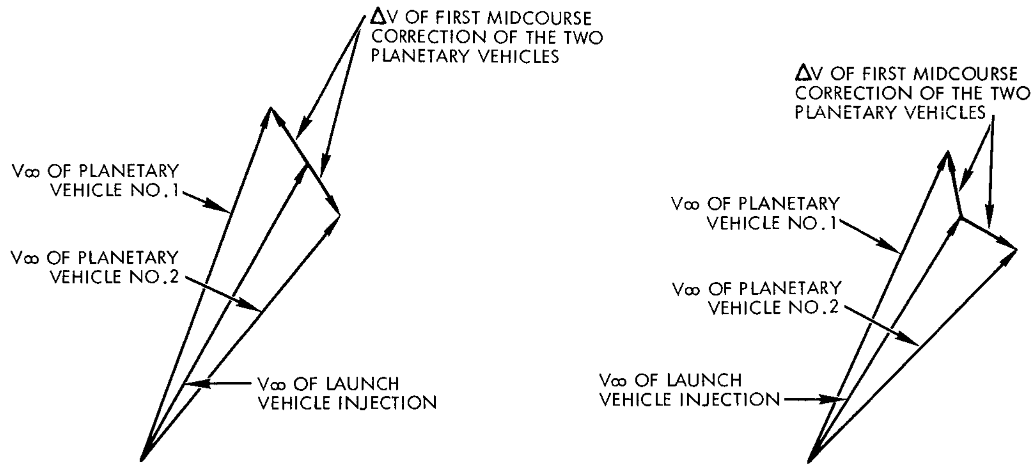
e) Capability of the attitude control system to provide flexibility and accuracy in the performance of maneuvers, and the orientations required by them

f) Probability of successfully performing all of the above functions during the mission life.

3.2 Operational Responsibility for Trajectory Constraints

3.2.1 Launch

The targeting of the injection of the two planetary vehicles by the launch vehicle has a strong influence on the contamination constraint. Since the two planetary vehicles, launched at the same time, are destined to arrive at Mars on two different dates, separated by at least 10 days, they ultimately must follow different Earth-Mars tracks having different V_∞ vectors (initial asymptotic departure velocity relative to the earth). In order to make most efficient use of the propellant supply on board the spacecraft, it is reasonable that the separation from the injection trajectory to the individual Earth-Mars trajectories be accomplished within several days of injection, and specifically, as a part of the first interplanetary trajectory correction. For a specific launch date and specific arrival dates, the propellant necessary for this first midcourse correction of the two planetary vehicles is further minimized if the injection V_∞ is the vector mean of the two required planetary vehicle V_∞'s. However, such an injection procedure results in a trajectory which comes very close to intersecting Mars (the third stage of the launch vehicle and a portion of the shroud which, unless otherwise deflected, remain on this trajectory). However, it is possible to target the initial injection away from the midpoint between the two V_∞'s, as shown in Figure E-1.



(a) INJECTION IS TARGETED TO THE MEAN OF THE TWO PLANETARY VEHICLE V∞'S.

(b) INJECTION IS TARGETED AWAY FROM THE MIDPOINT BETWEEN THE TWO PLANETARY VEHICLE V∞'S.

Figure E-1. Initial Injection from Midpoint

Because of the geometry involved, such a change in the injection targeting results in very little increased propellant utilization by a planetary vehicle in its first midcourse correction maneuver, but is adequate to reduce greatly the probability of impact on Mars of any portion of the injected system which is not deflected by a propulsive maneuver. Specifically, this pertains to the launch vehicle third stage, the shroud section between the two planetary vehicles, any separation hardware which may not be retained and attached to the launch vehicle, and the planetary vehicles themselves in the event of failure of their propulsion systems.

3.2.2 First Interplanetary Trajectory Correction Maneuver

At the time of the first midcourse maneuver the following components are accommodated by the vector velocity increment performed:

a) A correction for the intentional bias of the injection by the launch vehicle. This is a "bias" in the sense that it is not directed toward the aim point of either of the two planetary vehicles.

b) A correction for unintended dispersion by the launch vehicle from the nominal injection conditions.

c) A trajectory bias in the target point of the first midcourse maneuver—that is, it is not the final target point of the interplanetary phase — because the dispersion resulting from this maneuver will be too great to permit aiming as close to Mars as for the final target point.

3.2.3 Second Interplanetary Trajectory Correction Maneuver

This maneuver must account for the following components:

a) Correction for the intentional bias of the first midcourse maneuver.

b) Correction for the unintended dispersion of the first midcourse maneuver.

c) The possibility that this second maneuver may have to observe a new bias from the final targeting point. Whether such bias is necessary at this time depends on several factors:

- The time of execution of the second maneuver. The accuracy of execution of the second maneuver (and to some extent the corrective accuracy of all midcourse maneuvers).

- The magnitude of B, the impact parameter, which is dependent upon the intended orbit, and the angle of apsidal rotation to be effected upon entry.

- The magnitude of the first midcourse maneuver.

Of course, the imposition of a bias in the second maneuver implies that a third maneuver will have to be executed.

3.2.4 Third Interplanetary Trajectory Correction Maneuver

This maneuver, if it is necessary, must accommodate considerations similar to those given above for the second maneuver.

3.2.5 Approach to Mars and Orbit Insertion

During the final stages of the planetary vehicle approach to Mars, the operational responsibility for avoiding impact by the spacecraft involves the following tasks:

a) By tracking the vehicle, the orbit determination must be established and monitored to verify the accuracy of the preceding midcourse correction.

b) The orbit insertion parameters must be adjusted to account for the current estimate of the approach trajectory, and the intended resulting orbit about Mars should be verified for estimated lifetime.

c) The intended orbit insertion procedure must be analyzed for the possibility of failures which might threaten to violate the quarantine constraint. For example, the probability of failure of the orbit insertion engine to shut off at the commanded time, although very small from the viewpoint of mission reliability, may be significant for contamination control if this failure would produce an orbit of unduly low altitude.

3.2.6 Orbital Operations

After insertion into orbit, tracking data is utilized to establish the orbital elements and the accuracy of their estimation. The orbit trim maneuver must be formulated, based on these estimated orbital elements.

3.2.7 Orbit Trim Maneuver

A failure analysis, similar to that described in paragraph 3.2.5, must be employed to assess the risk of contaminating Mars in the event of operational failures connected with the orbit trim maneuver.

3.2.8 Capsule Canister Segments

Associated with the release of the capsule on its descent trajectory to impact Mars, there is the programming of the opening of the canister and the disposition of the segments of the canister. The following considerations apply:

a) In the mission sequence there are the alternatives of jettisoning the upper half of the canister during the approach phase, i.e., before orbit insertion, and of retaining the canister sealed until just prior to the capsule separation maneuver.

b) In the former alternate, there is the disadvantage that the landing capsule stands exposed to contamination for a much longer time—some 10 days as compared to several hours in the second alternate.

c) On the other hand, if the canister lid is jettisoned while in orbit, the orbit lifetime constraint now becomes more critical when applied to the jettisoned canister segment than it is when applied to the flight spacecraft. It is estimated that M/C_DA is only 1/10th as large for the canister half. This may require that the minimum periapsis altitude be double that which is possible based on the spacecraft alone. Of course, after the capsule has separated, and all segments of the canister have been jettisoned, it would then be feasible to lower the spacecraft periapsis with a subsequent orbit trim.

3.2.9 Evaluation of Orbital Decay Rate

It is desirable during the orbital phase of the mission to evaluate the tracking data for minute changes in orbital altitude, in order to estimate the actual decay rate of the orbit. This is a fairly sensitive operation, because in a period of one or several months it is desirable to predict orbital lifetime of 50 years or more. If a shorter lifetime is predicted by the orbital tracking data, the opportunity would exist to perform an orbital trim maneuver which raises the periapsis altitude of the spacecraft to a satisfactory height.

3.2.10 End of Mission

At a nominal end of mission, it may be desirable to raise the periapsis by a last orbit trim maneuver, if this is indicated to be necessary by the evaluation discussed in the preceding paragraph. In addition, it might be desirable to release the pressures in propellant storage tanks, and perform other similar operations which will minimize the possibility of an explosive failure (possibly initiated by meteoroid penetration) at some future time, which could lead to the ejection of nonsterile material on a path toward Mars.

4. CONTAMINATION BY MATERIAL EMANATING FROM THE SPACECRAFT

In determining what is required of the spacecraft design to minimize the probability that material emanating from the spacecraft will contaminate Mars, it is necessary to identify the possible mechanisms by which such contamination might occur. These mechanisms may be classified by the route to Mars from the spacecraft, and by the source of the material on the spacecraft.

The route to Mars is distinguished as follows: Section 4.1 is concerned with material departing from the spacecraft during the interplanetary cruise phase (except for the final approach to Mars) and proceeding directly to impinge on the planet. Section 4.2 analyzes material which leaves the spacecraft at any time from the final approach phase through orbital operations until 50 years has passed, and proceeds directly to the planet. Section 4.3 is concerned with contamination by an indirect route—material leaves the spacecraft and settles on the lander vehicle, which then carries it to Mars. The period of concern here is brief; it starts when the canister seal is broken—possibly shortly before encounter, but possibly only an hour or two before the capsule is separated from the spacecraft—and ends at capsule entry.

In each of these sections (as appropriate) the following sources of contaminated material are considered:

a) Spacecraft attitude control gas

b) Spacecraft propulsion exhaust products

c) Planetary vehicle surface outgassing and sublimation

d) Surface chips and flakes and dust particles spontaneously emitted or shaken off by pyrotechnic operations, deployments, articulation of appendages, and engine firings

e) Surface or subsurface material ejected by micrometeoroid impact

f) Jettisoned capsule canister segments

g) Rupture of a pressurized vessel, initiated by micrometeoroid penetration, and resulting in a violent disintegration of structure.

When the mission mode was based on one planetary vehicle launched by a Saturn IB-Centaur, the corresponding spacecraft specification allocated a total probability of 3×10^{-5} for unsterile impingement onto Mars by any of the above means (possibly excepting those associated with flight capsule hardware). For the present mission mode, in which a Saturn V injects two planetary vehicles, the allocation will have to be redefined.

4.1 Spacecraft Efflux During Interplanetary Cruise

Direct impingement on Mars of efflux departing from the spacecraft during interplanetary cruise, including trajectory corrections is discussed below. The final approach phase is excluded from this category because the departure velocity necessary for a particle to impact Mars during this phase increases to high values, and the analysis is treated separately in Section 4.2.

4.1.1 Attitude Control Gas

There is no threat of contaminating Mars by attitude control gas because (1) the attitude control gas can be sterilized if necessary (see paragraph 4.3.1), and (2) there is a high rate of attenuation of unattached microorganisms in vacuum, exposed to the ultraviolet light in solar radiation.*

4.1.2 Midcourse Propulsion Exhaust

With a liquid engine, exhaust products carry no threat to Mars because (1) the propellants tend to be self-sterilizing, so that very few

*In vacuum, the most resistant bacterial spores show a sensitivity to ultraviolet radiation which corresponds to a half life of 6 seconds or less when exposed to sunlight at 1.4 AU from the sun. (Actual experimental data indicate half lives ranging from 1 to 6 seconds, depending on experimental conditions.) A half life of 6 seconds implies attenuation of one decade in 20 seconds, and 10 decades in 200 seconds. Thus, it is concluded that any process in which all microorganisms are exposed unattached to sunlight for over 200 seconds carries no threat of survival of viable organisms. "Unattached" is used because even very small particles to which the spores are attached offer some protection from the effects of the radiation, and larger ones may more effectively shield them.

live microorganisms remain after several days, (2) the combustion process is a sterilizing one, and (3) any surviving microorganisms in the exhaust would be unattached (as all products are gaseous) and exposed to solar ultraviolet radiation.

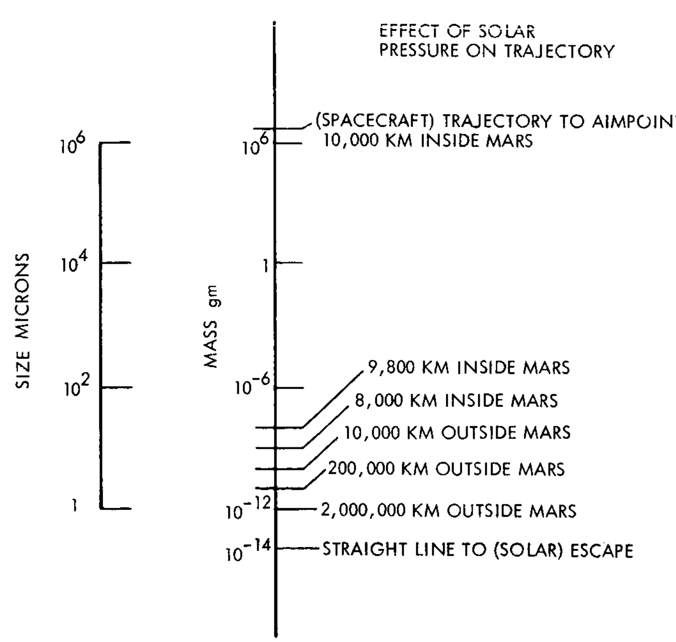
4.1.3 Surface Outgassing and Sublimation

No threat is posed by surface outgassing and sublimation because of exposure to solar ultraviolet radiation. (Any microorganisms emitted in this process are "unattached," as the efflux is gaseous.)

4.1.4 Particles

Surface chips, dust particles, or flakes that are spontaneously emitted, or shaken off by pyrotechnic operation, deployments, articulations, vibrations from engine firings represent particles whose ejection will be at low velocities relative to the spacecraft (0.001 to 1 meter/sec).

For the very smallest particles, subsequent motion will be straighter than the spacecraft orbit around sun—because solar pressure tends to cancel solar gravitation. Roughly, as shown in the sketch below, particles heavier than 10^{-9} gms, for all practical purposes follow the spacecraft inside of Mars (except for effect of lateral—R, T—components of the relative velocity of departure), particles $3 \cdot 10^{-10}$ gms have their heliocentric motion straightened enough to hit Mars, particles 10^{-10} gms



are straightened enough to go 10,000 km outside Mars, and lighter particles miss by greater amounts outside Mars. Those small enough to have appreciable orbit straightening are less than 10^{-9} grams in mass, less than 10 microns in size. This is also so small that a viable microorganism will be unlikely to "hide" from solar ultraviolet radiation. Therefore, we are concerned only with heavier particles with no appreciable orbit straightening.

For the analysis of this appendix we consider the external surface area of the planetary vehicle to be comprised of the following:

Spacecraft		
Solar cells	290 ft^2	
Ablative engine skirt	60	
Antennas	200	
Metallic structure (adapter)	40	
Thermal insulation (layered)	470	
Thermal louvers	20	
Miscellaneous, including science packages, sensors, gimbal assemblies, etc.	60	
Total		1140 ft^2
1971 Capsule		
Canister forward half	160 ft^2	
Canister aft half	100	
Total		260 ft^2
Planetary Vehicle Total		1400 ft^2

Although all the above area is external, that is, visible from outside the spacecraft, some of the surfaces are reentrant. The average cross section of the planetary vehicle is about 350 ft^2.

It is necessary to estimate the number of particles which may be dislodged, or shaken off, and the number of microorganisms which may be borne by these particles. The Spacecraft System Specification (Reference 2) calls for preventing loose or unattached particles greater than 4 mils (100 microns) from being within the encapsulated enclosure at launch time. Assuming this specification is essentially observed, particles of concern are predominantly of smaller size, although some larger particles may later be dislodged, due to possible deterioration of surface materials.

Because of the surface sterilization of the planetary vehicle by gaseous ethylene oxide within the shroud before launch, the possibility of viable microorganisms existing unattached on exterior surfaces, or attached to small particles is vastly reduced. However, in the prelaunch sequence after this surface sterilization, the vehicle spends several days, as a minimum, in test and being mated to the launch vehicle. During this time, it is anticipated that cooled, filtered air must be circulated through the encapsulated space, to maintain sufficiently low temperatures for the operation of electronic subsystems. The sterilization of this cooling air is not simple, although technically feasible. Estimates of the viable microorganism population after this prelaunch sequence range from zero, if cooling air is sterilized, to 10^3 organisms per square foot of exterior surface, if not. These would either be unattached to particles, or attached to very small particles.

The fraction of these dislodged or shaken from the spacecraft during transit is almost impossible to estimate. However, it is noted that, if the injection by the launch vehicle is aimed somewhat away (say 40 meters/sec) from the mean V_∞ vector of the two spacecraft, then neither planetary vehicle nor the spent launch vehicle third stage is aimed on a path which comes near Mars. Only after the end of the first midcourse correction is the planetary vehicle directed to an aim point within 10, 000 km of Mars. This would mean that a particle, in order to threaten to impact Mars, would have to remain aboard the planetary vehicle during separation from the launch vehicle, deployment of antennas, initial acquisition of references, orientation for first midcourse, and midcourse engine firing for a number of minutes. It would then have to be detached at the end of the first midcourse propulsion, or during the 6 to 8 months interplanetary cruise phase. The implication is that nearly all the detachable particles will probably be detached very early in the mission, before the vehicle is aimed close enough to Mars for detached particles to threaten to impact. We therefore assume that the fraction of microorganisms present on the surface which will be dislodged after the first midcourse correction is between 10^{-4} and 10^{-3}.

Next, the probability that a dislodged particle will impact Mars is strongly dependent on the ejection velocity. During most of the interplanetary cruise phase, the spacecraft is on a path which will miss Mars by some 5000 km; however, an ejection velocity of only 0.4 to 1 meter/sec is sufficient to cause impact. Also, remember that particles under 10 microns in size are deflected from Mars impact by solar radiation pressure. For larger particles, if ejected in random directions with normally distributed velocities, the probability of impacting Mars is approximately

$$P \cong 0.15 \exp\left[-\frac{.6^2}{2\sigma^2}\right]$$

where σ is the standard deviation of the ejection velocity, in meters/sec. This gives, for different values of σ:

σ =	.1	.2	.3	meter/sec
P =	$2.4 \cdot 10^{-9}$	$1.5 \cdot 10^{-3}$	$2 \cdot 10^{-2}$	

Mariner and Gemini experience has shown that at least some dislodged dust particles—presumably the larger ones—have very low ejection speeds. A reasonable value of σ is assumed to be 0.2 meter/sec, but P may easily range from 10^{-4} to 10^{-2}.

Because most microorganisms dislodged from the vehicle onto impact trajectories are on small particles (of the order of 10 to 20 microns), the attenuation due to solar ultraviolet radiation must attenuate them substantially. The survival rate is assumed to be 10^{-2} to 10^{-4} between ejection and impact.

Combining the factors discussed above gives these results:

$$\begin{pmatrix}\text{spacecraft}\\\text{external}\\\text{area (ft}^2\text{)}\end{pmatrix}\begin{pmatrix}\text{live micro-}\\\text{organisms}\\\text{per ft}^2\end{pmatrix}\begin{pmatrix}\text{fraction}\\\text{dislodged}\end{pmatrix}\begin{pmatrix}\text{fraction on}\\\text{particles over}\\\text{10 microns}\end{pmatrix}\begin{pmatrix}\text{fraction of}\\\text{these on impact}\\\text{trajectories}\end{pmatrix}\begin{pmatrix}\text{fraction}\\\text{surviving}\\\text{transit}\end{pmatrix} = \begin{pmatrix}\text{probable number}\\\text{of live impacts}\end{pmatrix}$$

$$\begin{pmatrix}1400\\ \\ \end{pmatrix}\begin{pmatrix}10^{3}\\ \\ 0\end{pmatrix}\begin{pmatrix}10^{-3}\\ \\ 10^{-4}\end{pmatrix}\begin{pmatrix}10^{-2}\\ \\ \end{pmatrix}\begin{pmatrix}10^{-2}\\ \\ 10^{-4}\end{pmatrix}\begin{pmatrix}10^{-2}\\ \\ 10^{-4}\end{pmatrix} = \begin{pmatrix}10^{-3}\\ \\ 0\end{pmatrix}$$

With 10^3 live microorganisms per square foot, the resulting probability of a live impact is from 10^{-8} to 10^{-3}, depending on the conservatism of the estimate of the processes by which these microorganisms might be transported to Mars. Unless the upper limits of these estimates can be discounted—a very difficult experimental task—the above analysis indicates the importance to be attached to minimizing recontamination within the shroud after the treatment by gaseous sterilant.

4.1.5 Micrometeoroid Effects

Surface or subsurface material ejected from the spacecraft by micrometeoroids differ from that of the preceding section by the velocity of the departing material. Whereas particles spontaneously emitted or shaken off will depart at velocities in the range 0.001 to 1 meter/sec, micrometeoroid ejecta have departure velocities from 100 to 10,000 meter/sec. During the interplanetary cruise phase, departures at such high relative velocities leads to a very low probability of impact. Later, in the orbiting phases, these higher departure velocities produce a substantial probability of impact. Since the pertinent orbiting time is 50 years, or 100 times the transit time, the micrometeoroid problem is more severe in orbit than in transit, and therefore is not considered here

4.2 Materials Ejected Near Mars

Direct impingement on Mars of material departing from the spacecraft during late approach, orbit insertion, and orbital operations (including orbital trim maneuvers) must be considered for a 50-year period. The possible mechanisms of contaminating Mars during this period are characterized by a requirement of high departure velocities—100 to 5000 meter/sec—but relatively short transit times.

4.2.1 Attitude Control Gas

By the argument of paragraph 4.1.1, there is no threat of contamination.

4.2.2 Exhaust Products

By the argument of paragraph 4.1.2, there is no threat of contamination by orbit insertion and orbit trim propulsive exhaust products.

4.2.3 Surface Outgassing and Sublimation

By the argument of paragraph 4.1.3, there is no threat of contamination.

4.2.4 Surface Particles

Surface chips, flakes, and dust particles, spontaneously emitted or shaken off by pyrotechnic operations, deployments, articulation of appendages, and engine firings are now considered. Because departure velocities due to these phenomena are small—less than 1 meter/sec—such efflux would not initially be on a course which could impact Mars. However, because the detached particles have a very low M/C_DA ratios, the orbital lifetime would be severely curtailed by atmospheric drag. Decay into the Martian atmosphere would take place in a period of months to several years, depending on the spacecraft orbit and the particle size.

An estimate of the factors involved in this mechanism is made, similar to the analysis of paragraph 4.1.4:

$$\begin{pmatrix}\text{spacecraft}\\\text{external}\\\text{area, ft}^2\end{pmatrix}\begin{pmatrix}\text{live micro-}\\\text{organisms}\\\text{per ft}^2\end{pmatrix}\begin{pmatrix}\text{fraction}\\\text{dislodged}\end{pmatrix}\begin{pmatrix}\text{fraction on}\\\text{particles over}\\\text{10 microns}\end{pmatrix}\begin{pmatrix}\text{attenuation}\\\text{during}\\\text{orbit decay}\end{pmatrix}=\begin{pmatrix}\text{probability}\\\text{of a live}\\\text{impact}\end{pmatrix}$$

$$\begin{pmatrix}1100\end{pmatrix}\begin{pmatrix}10^{2}\\\text{to}\\0\end{pmatrix}\begin{pmatrix}10^{-4}\\\text{to}\\10^{-5}\end{pmatrix}\begin{pmatrix}10^{-2}\end{pmatrix}\begin{pmatrix}10^{-1}\end{pmatrix}=\begin{pmatrix}10^{-2}\\\text{to}\\0\end{pmatrix}$$

The number of live microorganisms on the spacecraft surfaces is taken at 10^2 to account for attenuation throughout the transit phase. The fraction dislodged is similarly reduced to recognize that any loose particle is more likely to be dislodged early in the mission than late.

The result of the estimate, similar to that of 4.1.4, again indicates the importance of minimizing recontamination within the shroud after gaseous surface sterilization.

4.2.5 Micrometeoroid Effects

Surface or subsurface material ejected by micrometeoroids are considered here. Estimates exist as to the micrometeoroid flux in interplanetary space, and in the region of Mars, with estimates of mean particle velocity and the distribution of the flux according to particle mass. Other experiments have shown what the penetrating power of meteoroids is, how a surface struck by a meteoroid is cratered or pierced, and how the debris from such impacts is dispersed. Typically, the impact of a particle on a relatively thick metallic sheet results in the forming of a roughly hemispherical crater by the removal of material with some 50 times the mass of the impacting particle. Most of this material is removed in a high velocity spray of fine particles, probably mostly heated to above the melting point, but some larger fragments may be ejected from the periphery of the crater. Where the target sheet is thinner, say, less than 6 radii of the particle, then spalling or ejection of large flakes and fragments from the side opposite the impact is likely. When the sheet is thinner than 3 to 4 radii of the particle, a complete penetration will result.

What is harder to estimate is the probability that, from an impact of a meteoroid of a certain mass, any microorganisms in surface or subsurface layers will be ejected alive onto a path which will impact Mars. The processes and factors which are involved include the number of live microorganisms initially present in the area affected by the impact, the likelihood that the impact will detach a particle large enough (3 microns or more) to harbor a microorganism, the fraction of the affected area from which such particles will be ejected, and the probability that an ejected microorganism will survive. These factors are all a function of the mass of the meteoroid, and all of these probabilities are greater for impacts by meteoroids of larger mass. The total probability of contamination of Mars by this means is then found by integrating over meteoroid particles of variable mass as follows:

$$P = 0.1\, A_s\, T \int \sigma\, A(m)\, F(m)\, P_1(m)\, P_2(m) \left[- dN(m)\right]$$

The terms are defined as follows, and estimated with reference to Figure E-2:

0.1	is the estimate of the fraction of live microorganisms on ejected particles of over 3 microns in size which reach Mars alive. It considers that microorganisms on the smaller particles, whose orbits decay through atmospheric drag to enter Mars' atmosphere sooner, are most subject to attenuation by ultraviolet radiation, whereas those on larger particles take much longer (on the average) to impact Mars. Essentially all ejected particles are assumed to hit Mars eventually, either because high departure velocities and appropriate departure direction cause the initial departure orbit to intersect Mars, or because of atmospheric drag acting on particles of low M/C_DA.
m	is the mass of the impacting micrometeoroid, in grams
σ	is the number of live microorganisms per unit area initially present on spacecraft surface and subsurface layers
A(m)	is the area of the crater resulting from impact. It is taken as the area of a hemisphere which, at 3 gm/cm^3, has a mass of 50 m
F(m)	is the fraction of the affected area which is ejected as solid particles greater than 3 microns in size
$P_1(m)$	is the probability that the impact will detach a particle as large as 3 microns
$P_2(m)$	is the probability that a microorganism on a particle over 3 microns will survive the ejection of over 100 meter/sec

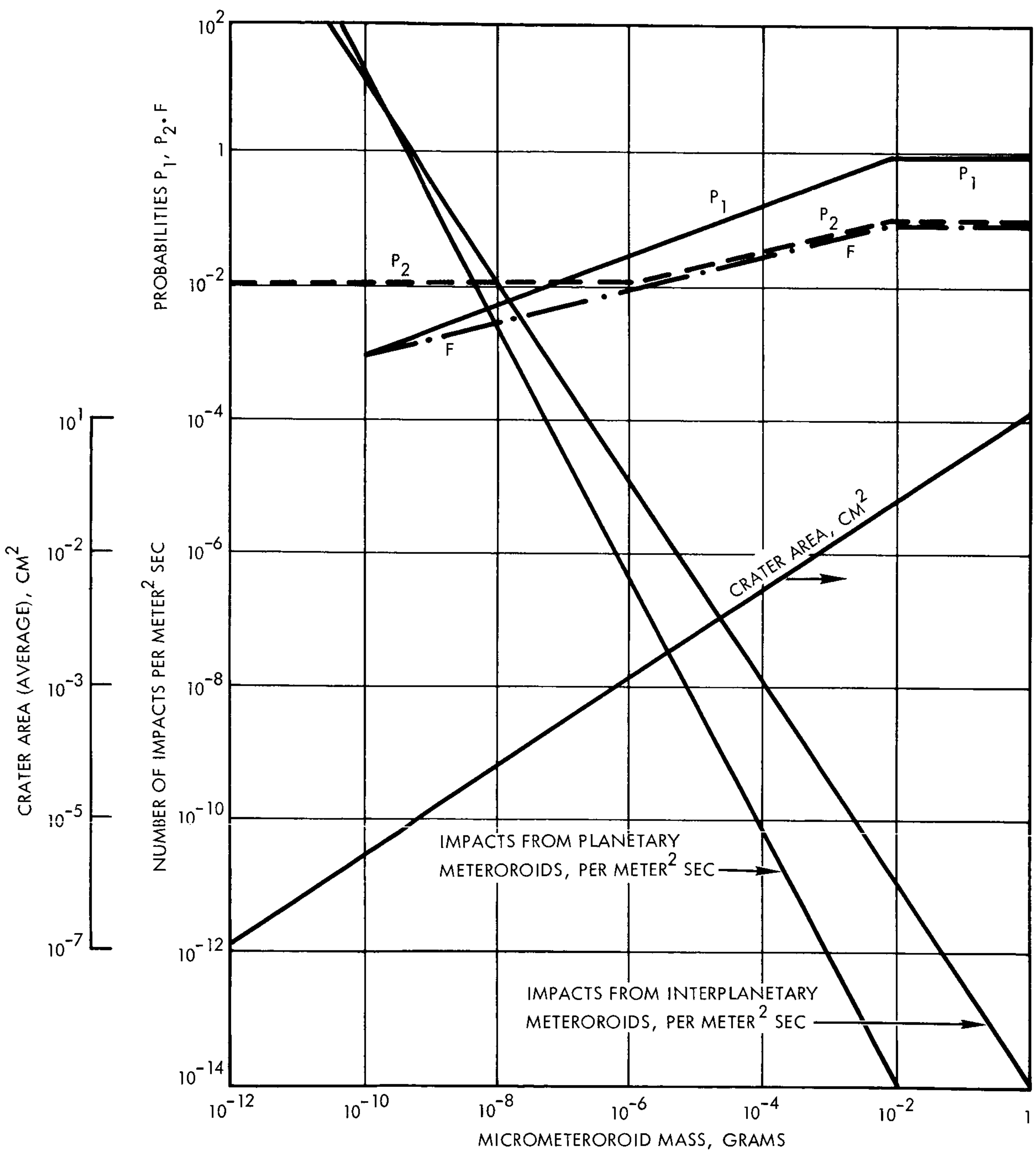


Figure E-2. Variation of Effects with Micrometeoriod Mass

$\left[-dN(m)\right]$ accounts for the distribution of micrometeoroids by mass. Although the environment of concern is near Mars, the interplanetary flux is used, because it is greater than the planetary flux in the range of heavier meteoroids, 10^{-6} to 10^{-2} grams, which have a greater contribution to the probability of contamination. From the JPL Voyager Environmental Predictions Document, we take, for the interplanetary flux 1.5 AU from the sun:

$$N(m) = 2.5 \cdot 10^{-14.48}\, m^{-1.34} \left(\frac{\text{impacts}}{\text{meter}^2\ \text{sec}} \right)$$

whence

$$\left[-dN(m)\right] = 1.11 \cdot 10^{-14} \left(\frac{\text{impacts}}{\text{meter}^2\ \text{sec}} \right) \cdot m^{-2.34} dm$$

The (-) sign arises from the definition of N(m) as the total impacts due to meteoroids of "mass m or larger"

A_s = 70 meters^2 is used. This is approximately double the cross section area

$T = 1.5 \times 10^9$ seconds, the 50-year quarantine period.

Figure E-2 shows N(m) for the interplanetary flux, given by the above equation, and for the flux of planetary micrometeoroids as well. Figure E-2 also shows A(m) and the estimated attenuating fractions F, P_1, and P_2. These fractions all terminate below $m = 10^{-10}$ grams, recognizing the extreme unlikelihood that smaller micrometeoroids could eject a particle large enough to harbor a live microorganism. The fractions rise to their maximum values at $m = 10^{-4}$ and 10^{-2} grams; the probability of dislodging live microorganisms which can impinge on Mars rises to its maximum value only when the impacting micrometeoroid is large enough to not simply crater the surface but to tear out substantial fragments. For example, where the surface is multilayered thermal insulation, micrometeoroids of about 10^{-9} grams will be large enough to penetrate one 75-micron layer; a mass of perhaps 10^{-8} grams would be able to tear out fragments of a layer. In surfaces composed of thicker layers, correspondingly larger meteoroid particles would be required to shatter rather than merely crater.

The result of the indicated integration is that the probability that a microorganism ejected from the orbiting spacecraft by micrometeoroids will impinge alive on Mars in a 50-year period is:

$$P = 2.5 \times 10^{-3} \sigma \text{ cm}^2$$

For

σ = 1 microorganism per cm^2 (about 10^3 per ft^2), we get

$P = 2.5 \times 10^{-3}$ probability of contamination.

For the probability of contamination by this mechanism to be held to a tolerable level, P must be reduced 2 orders. Unless further study indicates that the P and F factors used above are over conservative, the appropriate conclusion is that serious consideration must be given to means of holding the contamination level of surface and subsurface layers to as low as one microorganism per square foot.

4.2.6 Pressure Vessel Rupture

Rupture of a pressurized vessel, initiated by meteoroid penetration, and resulting in a violent disintegration of structure, differs from that considered in the preceding section in that a much greater mass of fragments is expelled than would be ejected by the meteoroid impacting unpressurized structure. The occurrence of such a rupture requires a meteoroid great enough to penetrate a pressurized vessel (helium or nitorgen tank, or propellant tank), and requires that the pressure within the tank exceed a critical level. In the Task B preferred spacecraft design, a brief analysis of wall thicknesses, pressures, and sizes, indicates that the propellant tanks pose a greater risk of meteoroid-induced failure than the other components.

This mechanism is not included in the interplanetary cruise phase, because in the Task B preferred design, the propellant tanks are pressurized to only 100 psi during that phase, and are subjected to the design pressure level of 250 psi only for the orbit insertion and subsequent maneuvers. Only at this higher pressure level can penetration lead to violent structural disintegration of the vessel.

The probability of a meteoroid-induced penetration leading to such a disintegration while in orbit about Mars has been estimated at 0.001 for a 1-month period. Again, the probability of contaminating Mars may be attenuated by factors representing the likelihood that the expelled fragments carry live microorganisms, and the probability that a contaminated fragment will impinge on Mars.

As to the existence of live microorganisms on the fragments, the probability must be essentially 1, resulting from contamination of the exterior surfaces of propellant tanks, unless the surface decontamination achieved by the ethylene oxide treatment during the prelaunch sequence is preserved by sterile isolation of the encapsulated planetary vehicle until launch and injection. However, if this sterile surface condition is maintained, then propellant tank fragments may be sterile. The sterility on their inner surfaces is derived from the self-sterilizing properties of the propellant, that of the outer surface stems from the prelaunch surface decontamination, and the sterile state of the interior of the wall dates back to its original manufacturing process. Even though pressure vessel wall fragments may, in this manner, be free of contamination, expelled fragments of inlet and outlet fittings, spacecraft structure, and insulation carried away by the disintegration are likely to be contaminated.

We estimate the appropriate attenuating factors for this mechanism of contaminating Mars as follows:

Probability that some expelled fragments are contaminated - 1
Probability that some contaminated fragment impacts Mars - 1

Thus the total probability of contaminating Mars is estimated as 10^{-3} for each month in orbit if the propellant tanks retain the high-level pressurization.

Although it has not been incorporated in the Task B preferred design, it would be possible to have a means of depressurizing the propellant tanks to the lower 100 psi level by venting through a relief valve at the end of the orbit insertion maneuver, or soon after, without sacrificing the orbit trim capability. Such a sequence would limit this means of contamination to an acceptably low probability by curtailing the period in which a meteoroid penetration might induce structural disintegration. However, before adopting such a design requirement, the factors leading to the figure of 10^{-3} per month should be studied in more detail.

4.3 Indirect Contamination

Indirect contamination of Mars is possible if efflux from the spacecraft settles on the exposed capsule vehicle. As indicated in 3.2.8, the period of concern depends on the sequence for jettisoning the canister lid. If this is done during the final approach phase (i.e., before encounter) the period may be 5 to 10 days; if it is just before capsule-spacecraft separation, the period is more like 2 or 3 hours of exposure to spacecraft-originated efflux. Another possible sequence avoids almost completely any exposure to spacecraft efflux, although other disadvantages lessen its desirability: in this sequence the capsule (with the canister intact) is separated from the spacecraft, and the canister is split and jettisoned from the capsule only when some distance away from the spacecraft.

Regardless of which canister program is followed, one procedure can be inserted into the capsule sequence which will reduce the probability of contamination of Mars by spacecraft efflux carried on the surface of the lander. This procedure, requiring a certain flexibility of the capsule attitude control system, makes use of programmed changes of capsule orientation, while it is descending from the spacecraft orbit, so as to expose toward the sun successively each portion of its exterior surface. This exposure to solar ultraviolet radiation would reduce by several orders the probability of survival of microorganisms on the exterior surface of the capsule. Particularly on those surfaces such as a heat shield which have a smooth shape and a vitreous finish, the bacteria would be almost as sensitive to the ultraviolet exposure as they are if "unattached," in which case 20 seconds' exposure attenuates the population by one decade.

4.3.1 Attitude Control Gas

The mechanism by which viable microorganisms might be expelled with the attitude control gas and settle on the exposed capsule vehicle is, of course, sensitive to the decontamination measures the attitude control gas has been subjected to, and is dependent on the geometry of the spacecraft-capsule configuration. In filling the spacecraft attitude control gas vessels during the prelaunch sequence, careful handling and efficient filtering (but not sterilizing) of the air can keep the number of live microorganisms as low as 25 per pound. If the pneumatic components of the attitude control system are heat sterilized and filled with gas which has been sterilized, careful control of the process may result in a probability as low as 10^{-1} or even lower that any viable microorganisms are in the supply. For the 60 pounds aboard, this is $1.5 \cdot 10^{-3}$ per pound, or 4 orders of magnitude less. The amount of attitude control gas used during the period the capsule is exposed is about 1.3 pounds for the long (10-day) period, or about 0.3 pound for the short alternate (2 hours), based on the maneuvers necessary during the two periods. The probable number of microorganisms, then, can range from 5×10^{-4} to 30.

The planetary vehicle geometry influences the fraction of these expelled microorganisms which can impinge on the capsule surface. If the lander is exposed by discarding the canister half most remote from the spacecraft, the remaining canister half may shield the entire capsule vehicle from a direct line of sight from any of the spacecraft attitude control nozzles. Because expelled microorganisms are in a medium consisting of a gas rapidly expanding into vacuum, they follow essentially straight-line paths, and such shielding will reduce the fraction which impinges to an extremely low value—perhaps 10^{-5}. If the shielding afforded by the remaining canister half is less efficient, then the fraction which impinges could be significantly higher; but even with adverse geometry it is probably limited to 10^{-2}.

A further exposure will occur when the capsule vehicle separates from the spacecraft, for it then abandons the protection of the remaining canister half. However, at a departure velocity of 50 meter/min, it rapidly achieves the immunity to contamination afforded by distance—because it subtends continually smaller angles—and the amount of gas expelled from the spacecraft during this time is very small. The use of spacecraft attitude control gas could be discontinued for a short time during this period, if desired, with no adverse effects; but if the attitude control gas supply were sterile, there would be no need for this.

Some attenuation of the probability that live microorganisms are on the lander surface will occur during the capsule orbital descent phase, due to solar radiation. However, if the programmed orientation changes (Section 4.1) are not followed, some surfaces will remain shaded from the sun, and this attenuation will not be great.

Combining the factors discussed above gives these results:

$$\begin{pmatrix}\text{live micro-}\\ \text{organisms per}\\ \text{pound of gas}\end{pmatrix}\begin{pmatrix}\text{pounds of gas}\\ \text{used while}\\ \text{lander exposed}\end{pmatrix}\begin{pmatrix}\text{fraction}\\ \text{impinging on}\\ \text{lander surface}\end{pmatrix}\begin{pmatrix}\text{attenuation}\\ \text{during}\\ \text{descent}\end{pmatrix} = \begin{pmatrix}\text{probability of live}\\ \text{microorganism}\\ \text{entering Mars'}\\ \text{atmosphere}\end{pmatrix}$$

$$\begin{pmatrix}25\\ \text{to}\\ 1.5 \times 10^{-3}\end{pmatrix}\begin{pmatrix}1.3\\ \text{to}\\ 0.3\end{pmatrix}\begin{pmatrix}10^{-2}\\ \text{to}\\ 10^{-5}\end{pmatrix}\begin{pmatrix}0.5\\ \text{to}\\ 10^{-3}\end{pmatrix} = \begin{pmatrix}1.5 \cdot 10^{-1}\\ \text{to}\\ 5 \cdot 10^{-12}\end{pmatrix}$$

Based on this analysis, it may be concluded that either the attitude control gas supply should be sterilized, or both of these measures should be taken, to keep the result under 10^{-5}:

- The capsule design should provide line-of-sight shielding from the spacecraft attitude control nozzles when the outer canister half is jettisoned

- The capsule descent program should include a controlled rotation to expose all external surfaces to the sun.

Whether the canister lid is jettisoned before or after orbit insertion is not a significant factor.

4.3.2 Spacecraft Propulsion Exhaust Products

Impingement of microorganisms from spacecraft propulsion exhaust products on the lander could be a contamination mechanism only if the canister is jettisoned prior to the orbit insertion or planetary vehicle orbit trim maneuvers. However, even in this case, there is no threat of contamination, because (1) the propellants tend to be self-sterilizing, (2) the combustion process is a sterilizing one, (3) the exhaust products are gaseous and directed in the opposite direction from the lander, and (4) the lander is shielded from impingement by the spacecraft.

4.3.3 Surface Outgassing and Sublimation

The number of microorganisms which might be liberated by outgassing and sublimation of the spacecraft surface must be extremely limited, as late in the mission as the capsule descent. Additionally, the same attenuation due to shielding by the unjettisoned canister half will apply as in paragraph 4.3.1. Therefore contamination by this mechanism is considered to have a negligible probability.

4.3.4 Surface Particles

Surface chips, flakes, and dust particles spontaneously emitted, or shaken off by pyrotechnic operations, deployments, articulation of appendages, and engine firings are considered here. The analysis for this mechanism for contaminating Mars follows the line of reasoning of Section 4.1.4. The upper limiting number of live microorganisms per square foot of spacecraft surface is reduced from 10^3 to 10^2 due to the long exposure to the space environment during transit from earth. The fraction dislodged is reduced to 10^{-5} for a 10-day exposure period, and 10^{-7} for a 2-hour exposure period. (Engine-firing, as a cause of dislodging particles during the 10-day period, is discounted, because the steady acceleration caused by the propulsion is in a direction so as to prevent detached particles from approaching the exposed capsule vehicle.) The fraction of detached particles which impinges on the lander is 10^{-5} to 10^{-2}, depending on geometrical shielding (see paragraph 4.3.1). The fraction surviving descent is 10^{-3} to 0.5, the smaller number applying if the programmed rotations described in 4.1 are employed.

The resulting summary is

$$\begin{pmatrix}\text{spacecraft}\\ \text{external}\\ \text{area (ft}^2)\end{pmatrix}\begin{pmatrix}\text{live micro-}\\ \text{organisms}\\ \text{per ft}^2\end{pmatrix}\begin{pmatrix}\text{fraction dis-}\\ \text{lodged (10-day}\\ \text{or 2-hour period}\end{pmatrix}\begin{pmatrix}\text{fraction}\\ \text{impinging}\\ \text{on lander}\end{pmatrix}\begin{pmatrix}\text{attenuation}\\ \text{in}\\ \text{descent}\end{pmatrix} = \begin{pmatrix}\text{probability of live}\\ \text{microorganism}\\ \text{entering Mars'}\\ \text{atmosphere}\end{pmatrix}$$

$$\begin{pmatrix}1400\\ \\ \end{pmatrix}\begin{pmatrix}10^2\\ \\ 0\end{pmatrix}\begin{pmatrix}10^{-5}\\ \\ 10^{-7}\end{pmatrix}\begin{pmatrix}10^{-2}\\ \\ 10^{-5}\end{pmatrix}\begin{pmatrix}0.5\\ \\ 10^{-3}\end{pmatrix} = \begin{pmatrix}7 \times 10^{-3}\\ \\ 0\end{pmatrix}$$

Based on this analysis, it again appears important to minimize recontamination within the shroud after gaseous surface sterilization during the prelaunch sequence. As in paragraph 4.3.1, it is seen that optimum shielding provided by the remaining canister design and a programmed sequence of capsule orientations during descent will markedly reduce the probability of contaminating Mars. In this case, the duration of the period of capsule exposure before separation is significant.

The possibility that particles will be dislodged by the separation of the canister should not be overlooked. Assuming the products of the pyrotechnic device are contained, then any unsterile particles dislodged by the shock from this event would have to come from external surfaces of the spacecraft or canister, and geometric shielding would prevent impingement on the capsule vehicle, as discussed in paragraph 4.1.4. Thus this event is not outside the scope of the preceding analysis.

4.3.5 Micrometeoroid Effects

Surface or subsurface material ejected by micrometeoroid impact are now considered. Because the exposure period is very short compared with the 50-year quarantine period, which applies to the spacecraft in orbit, the probability of contamination by meteoroid ejecta impinging on the lander is very small compared with the probability during the 50-year period in orbit, which is considered in paragraph 4.2.5.

APPENDIX F

MAGNETICS REQUIREMENTS ANALYSIS

1. INTRODUCTION

This appendix contains a preliminary analysis to estimate the magnetic field of the recommended spacecraft design. The results are tied to the assumption that an effective program is instituted to monitor and control the magnetic properties of all elements of the Voyager flight spacecraft. It is concluded that the requirements specified in 9.1.4 of Section III can be met.

Selecting the proper method of achieving the specified field depends on analyzing and evaluating the over-all cost, reliability, development, and schedule risks in the tradeoff between providing a more complex deployment device for the magnetometer and controlling the magnetic properties of the spacecraft. Longer booms involve additional weight, present more difficult alignment problems, and, to some extent, affect reliability. On the other hand, experience with the magnetics control program used on Pioneer indicates that controlling and reducing the magnetic properties of the spacecraft equipment (probably the flight capsule, as well) so as to permit mounting the magnetometer directly on the spacecraft would be very expensive, and would involve high development and schedule risks. It presently appears that a boom length of 10 to 20 feet is a reasonable compromise, and the analysis here is based on boom lengths in this range.

The analysis of the field at the magnetometer due to components and current loops is presented in 2 and 3, and a possible in-flight calibration for alignment is discussed in 4.

2. ANALYSIS: COMPONENT FIELDS

The analysis presented computes the field which would exist at the magnetometer if all contributions from spacecraft elements were to arise from equivalent dipoles oriented to maximize the field at the magnetometer sensor. The extensive testing and analysis on OGO and Pioneer have shown that actual field levels are between 1 per cent and 10 per cent of this maximum value. This experience has also indicated that it will not be difficult to meet the magnetic levels shown in Table F-1 for the various components of interest, providing strict control of the parts, materials, and processes is maintained, with magnetic material permitted only for those items for which no satisfactory substitute can be found.

Table F-1. Typical Levels of Component Magnetic Fields

Item	Equivalent Field (Field in gamma at 1 ft of an equivalent dipole)	Comment
Standard electronic packages - TWT's, motor and gear train, microwave components, batteries, inverters, etc.	500	Operating and Nonoperating
Tape recorders	2000	Operating and Nonoperating
Solenoid valves	100	Nonoperating
	1000	Operating
Experiment sensors	500	Except ionization sensor which is approximately 1 gamma
Experiment remote hardware	500	Operating and Nonoperating
Guidance and Control sensor	500	Operating and Nonoperating
Louvers	500	Operating and Nonoperating
Two-axis drive and gimbal	1000	
DeHaviland antennas	500	Drive motor

The magnetic effects of an equipment item at the magnetometer can be conservatively represented by a suitable equivalent field. Such an equivalent field is taken to be the magnetic field from a magnetic dipole which would produce the same distant magnetic field as that producted by the unit. This equivalent field is then conveniently expressed by the magnetic field, i.e., flux density B_1 expressed in gamma (10^{-5} gauss), which would be measured 1 foot away from the equivalent dipole. As the equivalent dipole field varies inversely as the cube of the distance, the field B_m at a distance R from the unit can be expressed as follows:

$$B_m = \frac{B_1}{R^3} = B_1 A$$

where A is the attenuation factor $1/R^3$.

The spacecraft arrangement for the Task B preferred design is shown schematically in Figure F-1, with three boom lengths indicated. For this analysis, the spacecraft has been subdivided into zones. For equipment items within a zone, the magnetic field attenuation factor relative to the magnetometer location is assumed to be uniform and equal to the minimum at the boundary closest to the magnetometer. A new zone, moving away from the magnetometer, is defined at a radius where the attenuation factor decreases by a factor of 2.

The approximate attenuation factor and resultant maximum field level for three boom lengths are tabulated in Table F-2.

For this preliminary analysis, it has been assumed that the field produced by the structure will be zero. Preliminary information available on the LEM-D stage indicates that some "nonmagnetic" 304 stainless tubing and some steel valves are used. The extent to which these materials may present a problem has not been analyzed. However, it can be seen from Figure F-1 that a considerable number of items having equivalent fields of 500 gamma can be located near the center of the spacecraft without serious effect.

If one uses a rule of thumb that the spacecraft is 10 per cent of the maximum possible field, one concludes that the DC field specification can be met for a boom length of 15 feet. Furthermore, as all equipment items throughout a spacecraft zone are assumed to have a magnetic contribution as if at the near edge, this produces a pessimistic prediction. The over estimation is considered to be about 40 per cent. Thus, the 1 gamma requirement can probably be met with a 10 foot boom, although

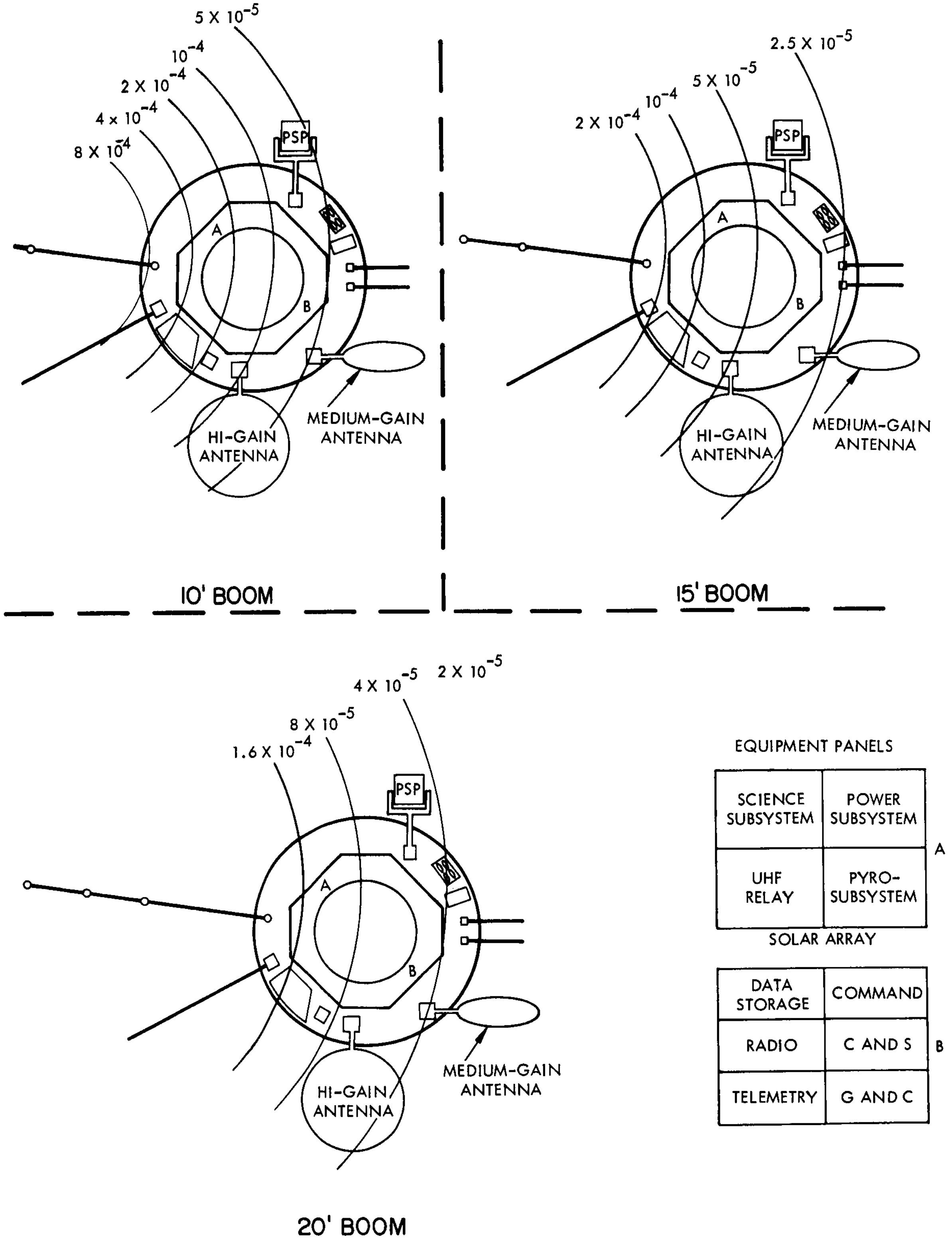


Figure F-1. Magnetic Dipole Attenuation Factors for Different Boom Lengths

Table F-2. Field Contributions at Magnetometer Sensors

Subsystem	Parts	Equivalent Field (gamma at 1-ft)	20-ft Boom Atten.(1)	20-ft Boom Total	15-ft Boom Atten.(1)	15-ft Boom Total	10-ft Boom Atten.(1)	10-ft Boom Total
Structure	Nonmagnetic	0	-	0	-	0	-	0
Command	4 electronic packages	2000	4(-5)	0.08	5(-5)	0.10	1(-4)	0.20
Propulsion	1 motor gear train	500	4(-5)	0.02	1(-4)	0.05	2(-4)	0.10
	4 solenoid valves	400/1400(2)	4(-5)	0.02/0.06	1(-4)	0.04/0.14	2(-4)	0.08/0.28
	2 latch solenoid	200/1200	4(-5)	0.01/0.05	1(-4)	0.02/0.12	2(-4)	0.04/0.24
Cabling	Nonmagnetic	0	-	0	-	0	-	0
Computing & Sequencing	2 electronic packages	1000	4(-5)	0.04	5(-5)	0.05	1(-4)	0.10
S-Band Radio	8 electronic packages	3500	4(-5)	0.14	5(-5)	0.20	1(-4)	0.40
	2 TWT	1000	4(-5)	0.04	5(-5)	0.05	1(-4)	0.10
	8 microwave components	4500	4(-5)	0.18	5(-5)	0.20	1(-4)	0.40
Guidance & Control	3 electronic packages	1500	4(-5)	0.06	5(-5)	0.08	1(-4)	0.15
	1 antenna drive, 2 axis	1000	4(-5)	0.04	1(-4)	0.10	2(-4)	0.20
	1 antenna drive, 1 axis	500	4(-5)	0.02	5(-5)	0.02	1(-4)	0.05
	2 actuators TVC	1000	4(-5)	0.04	1(-4)	0.10	2(14)	0.20
	16 solenoid valves	1600/3600(3)	4(-5)	0.06/0.14	1(-4)	0.16/0.36	2(-4)	0.32/0.72
	5 sensors	2500	4(-5)	0.10	5(-5)	0.12	1(-4)	0.25
	4 sensors	2000	8(-5)	0.08	1(-4)	0.20	2(-4)	0.40
Telemetry	2 electronic packages	1000	4(-5)	0.08	5(-5)	0.05	1(-4)	0.10
Temperature Control	4 louvers	2000	8(-5)	0.16	5(-5)	0.10	1(-4)	0.20
	4 louvers	2000	4(-5)	0.08	2(-4)	0.40	4(-4)	0.80
Power	14 electronic packages	7000	8(-5)	0.56	2(-4)	1.40	4(-4)	2.80
Data Storage	6 tape recorders	12,000	4(-5)	0.48	5(-5)	0.60	1(-4)	1.20
Relay Link	4 electronic packages	500	8(-5)	0.04	1(-4)	0.05	2(-4)	0.10
	1 tape recorder	2000	8(-5)	0.16	1(-4)	0.20	2(-4)	0.40
Pyrotechnics	Nonmagnetic	0	-	0	-	0	-	0
Science	2-axis drive	1000	4(-5)	0.04	5(-5)	0.05	1(-4)	0.10
	5 sensors	2500	4(-5)	0.10	5(-5)	0.12	1(-4)	0.25
	3 sensors	1500	2(-5)	0.03	5(-5)	0.08	5(-5)	0.08
	11 electronic packages	5500	8(-5)	0.44	1(-4)	0.55	2(-4)	1.10
	1 sensor	500	1.6(-4)	0.08	2(-4)	0.10	8(-4)	0.40
	16 sensors	8000	8(-5)	0.64	2(-4)	1.60	4(-4)	3.20
TOTAL				3.83		8.79		13.72
Solenoid valves operating				0.06		0.10		0.20
Solenoid valves operating				0.08		0.20		0.40

[1] Number in parenthesis is power of 10 multiples, i.e., 4 (-5) = 4×10^{-5}.

[2] Change indicates one solenoid operating at the same time.

[3] Change indicates two solenoids operating at the same time.

the risk is higher. However, the specified 0.1 gamma stability requirement appears to require the 20 foot boom. The magnetic field changes shown in Table F-2 are based on assumed values for the fields of solenoid valves and on the assumption that the solenoid valves operate in pairs in the guidance and control subsystem. It may prove practical to enclose the solenoid valves in a magnetic material of sufficient thickness to reduce the fields from the operating solenoids sugnificantly. Also, the operation of such equipment may occur at known time of sufficiently short duration to permit exceeding the 0.1 gamma specification at those times. Under these circumstances, a 10 foot boom might be adequate.

3. ANALYSIS: CURRENT FIELDS

The distance field from a coil of wire in its plane can be approximated by

$$B = 100n \frac{A\,I}{x^3} \text{ gamma}$$

where n is the number of turns

A is the area of the coil in square meters

x is the distance in meters from the center of the coil to the point of measurement in meters ($x^2 >> A$)

I is the current in amperes.

The solar array has about 50 amperes, total, flowing in it. If all this current were flowing across the spacecraft in a single loop 0.1 meter by 6 meters, then the field at the magnetometer would be as follows:

Boom length (ft)	x	B (gamma)
10	6.1	0.80
15	7.62	0.40
20	9.15	0.24

The design of the solar array employs wiring techniques which keep the total area much less than 0.6 m^2. Also, the arrangement of current flow around the array achieves cancellation betweel loops. The conclusion is that reasonable care in arrangement of wires will keep the magnetic fields due to current flow at negligible levels.

4. ALIGNMENT CALIBRATION

The following analysis examines whether or not the orientation of the magnetometer can be checked by generating an accurate magnetic field on the spacecraft. The most obvious method is to mount a triaxial coil assembly at the root of the boom with known orientation and energize the coils from a precision current source.

For a sensor with 0.2-gamma maximum error in each axis, alignment can be verified within 1 degree by generating known fields of 150 gamma successively along each axis and measuring cross-coupling. For a coil 2 meters in diameter with 100 turns, the current required to generate 150 gamma at the magnetometer sensor is plotted in Figure F-2 for various boom lengths.

If the coil were made of number 20 wire, the resistance would be about 20 ohms and the weight of the wire about 6 pounds. For the 10-foot boom, these numbers are reasonable, but for the longer booms the power dissipated in the coil becomes rather large. The power dissipated in the above coil is also plotted in Figure F-2.

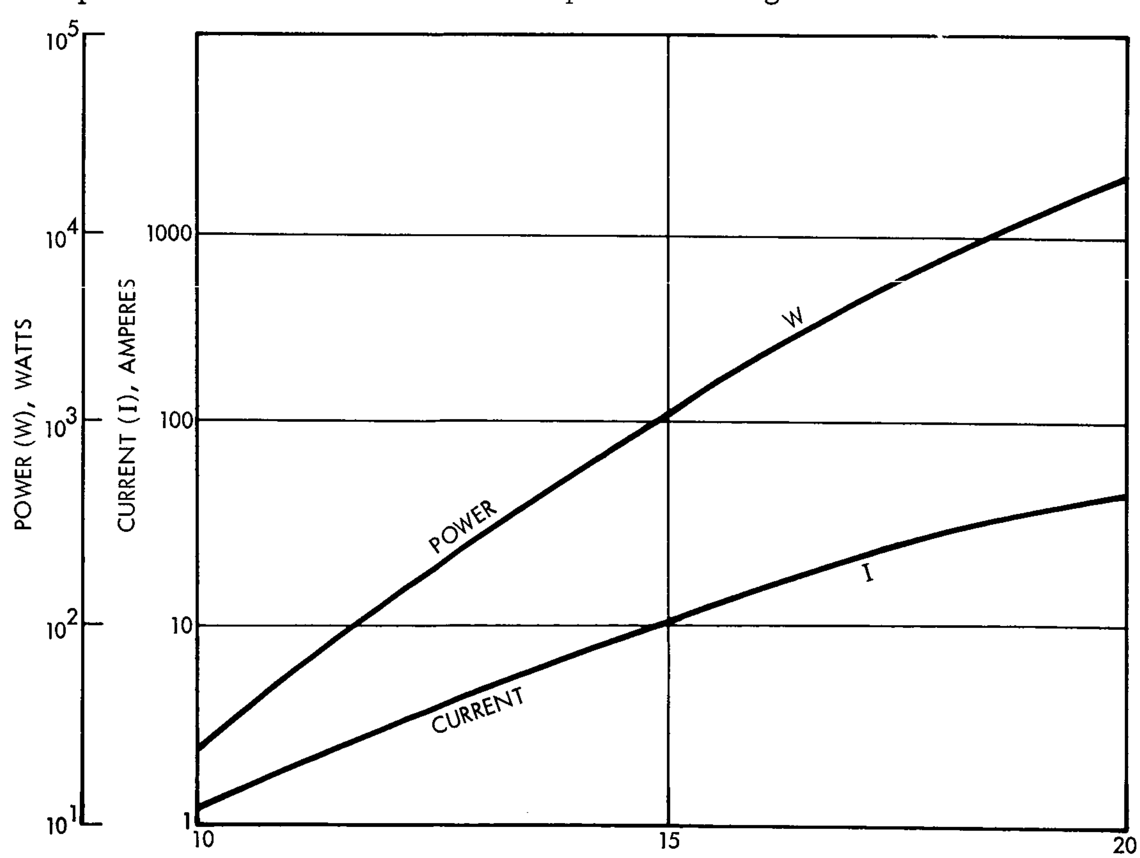


Figure F-2. Boom Length, feet

APPENDIX G

ELECTROMAGNETIC COMPATIBILITY DESIGN CONSIDERATIONS

The goal of the proposed EMC design and control requirements for the Voyager spacecraft system is to ensure that all equipment will operate as required in the integrated system, without malfunction or degradation in performance due to electromagnetic interference (EMI). This is achieved ultimately by identifying and controlling sources of electromagnetic energy, both conducted and radiated, and by providing suitable protection for critically susceptible circuits. This must be done for both the self-generated and the external electromagnetic environments.

This appendix is concerned with a design approach for reducing the interaction of the existing EMI with system circuits. As is true for any complex design discipline, there are several avenues to achieve such a goal, each having certain advantages and disadvantages. However, the basic design approach which is felt to produce the greatest benefit for the least weight, cost, and with the highest reliability is that of electrical bonding, grounding, and shielding to obtain circuit electromagnetic isolation. The fundamental principle involved here is that if the efficiency of interference energy transfer from one circuit to another is greatly diminished by minimizing the number and magnitude of the common impedance elements between circuits, then any significant energy transfer must be accomplished by electromagnetic propagation through a dielectric media where mismatches in impedance and polarization are controlling conditions. This type of energy transfer can then be adequately controlled by proper shielding and cabling techniques, and by control of power and signal waveforms.

The major emphasis in the approach is placed on electrical bonding of the various mechanical elements of the system, the primary reason being that the better the electrical bonding, the more closely does the system mechanical structure approach the ideal equipotential reference level. Secondly, the well-bonded structure and equipment enclosures afford a high degree of shielding efficiency against external electromagnetic fields, e.g., launch site EMI environment. A number of standard techniques are also employed, e.g., shielding, twisting wires, etc. Special fixes, such as decoupling networks may be required, but generalizations about such devices are not profitable at this stage.

The importance of achieving the best reference level possible is illustrated by considering the method of handling many digital signal lines in the presence of both AC and DC power and analog signals. The basic compatibility problem arising out of handling a multiplicity of functional signals is centered around the circuit return current paths. Every circuit must complete a loop from source to load and back to source, regardless of the ground reference system employed. On a circuit-by-circuit analysis, few restrictions are placed upon the nature of the circuit current return path; almost any electrically conductive substance can be made to satisfy the circuit return requirements under a given set of circumstances. One approach to return path control is the "single-point" ground system, which has been devised particularly to avoid "ground loops." The usual result, however, in systems containing numerous transistor devices is that interference persists. This is the case even though all circuit returns and shields have been carefully carried back to the system ground point so that power, signal, and shield grounds will be common at only one point. Such a situation is caused by the nature of the active current elements. For example, when a transistor, which is a three-terminal device, is utilized to switch power, it is nearly impossible to operate this circuit without connecting the power and signal circuit returns together.

This problem can be most easily recognized by analyzing a typical pulse amplifier, shown in Figure G-1, used in many systems as a command pulse amplifier in order to raise the incoming low level command to a usable level. The amplifier is turned on by supplying a voltage, normally a pulse voltage, at terminal 1. This voltage pulse allows current to flow through R1, raising the transistor base potential with respect to signal ground, terminal 2. The transistor will not turn on until the base potential is positive with respect to power ground, terminal 3, since the base-to-emitter junction is actually a reversed-bias diode. The amount of positive potential required is a function of the transistor and the diode from emitter to ground. The diode is added to increase the circuit "noise band" and is typically greater than 4 volts.

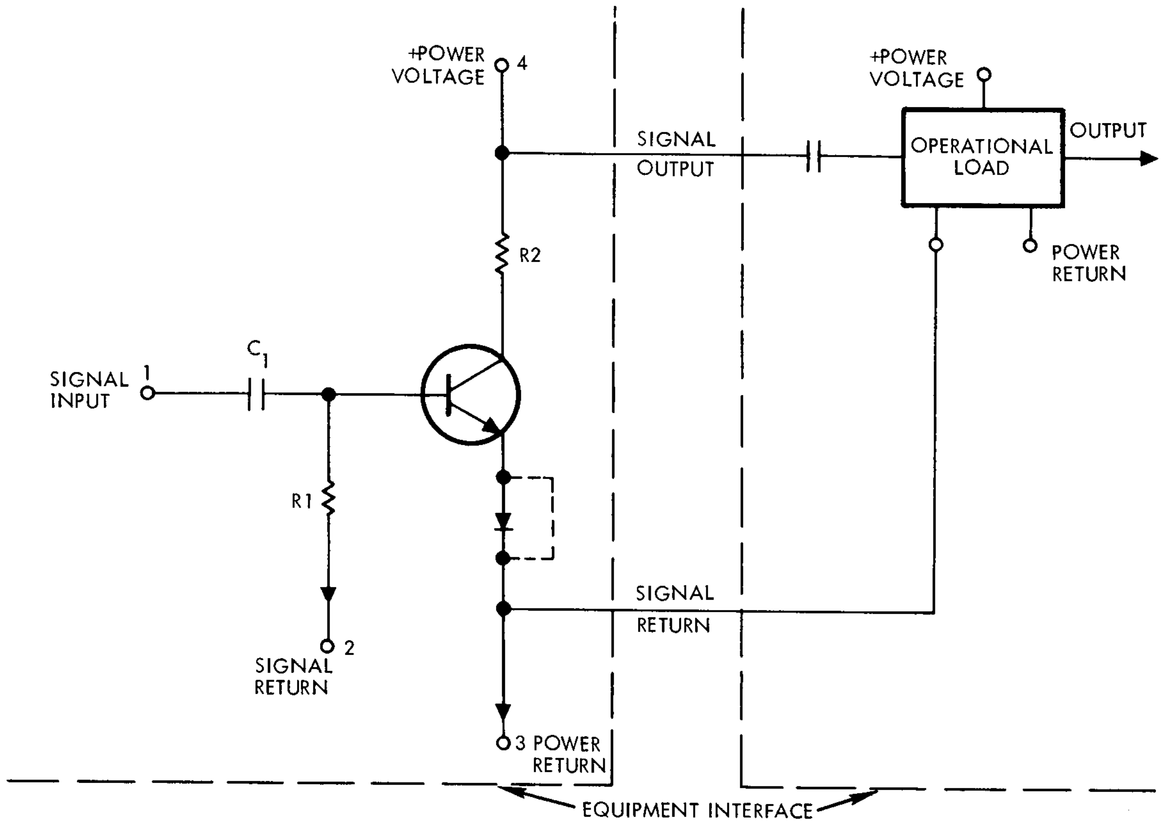


Figure G-1. Typical Pulse Amplifier

Obtaining the potential difference across the base-to-emitter junction necessary to allow enough current to flow to turn the transistor on is the critical area in isolating circuit grounds. There are several ways of realizing this potential difference; among these are adding a capacitor or diode between terminals 2 and 3, changing the base input to accommodate transformer coupling, or connecting terminals 2 and 3 by hard wire. These approaches are illustrated in Figures G-2a, G-2b, and G-2c.

The circuit approaches shown in these figures satisfy the physical requirements for turning on the amplifier. However, in Figure G-2a, where the diode is employed, and in Figure G-2c, where hard wire connection is employed, there is no true isolation of signal and power grounds. It is therefore necessary to employ the circuits of Figure G-2a, with isolating capacitor, or that in Figure G-2b, with the isolating transformer, to achieve signal and power ground isolation.

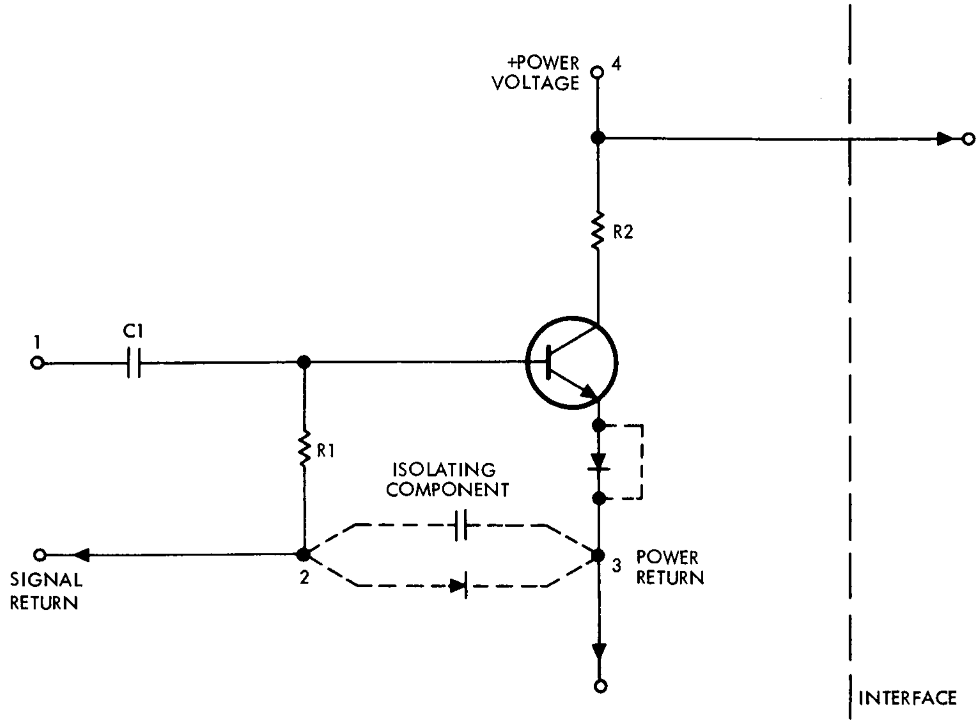


Figure G-2a. Typical Circuit Utilizing A Diode

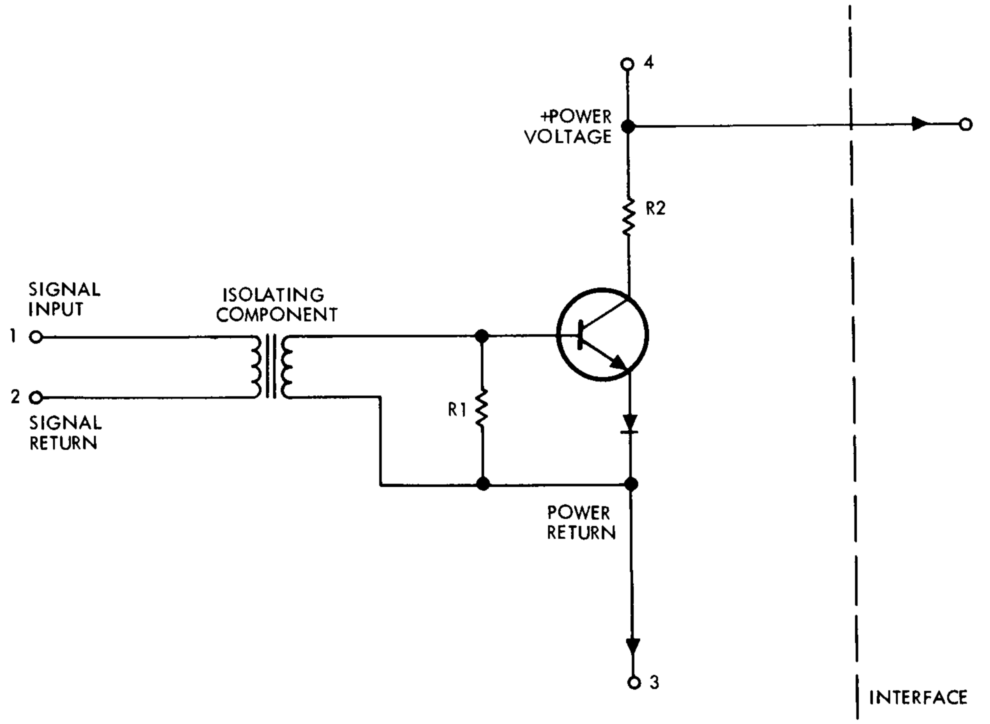


Figure G-2b. Typical Circuit Utilizing an Isolating Transformer

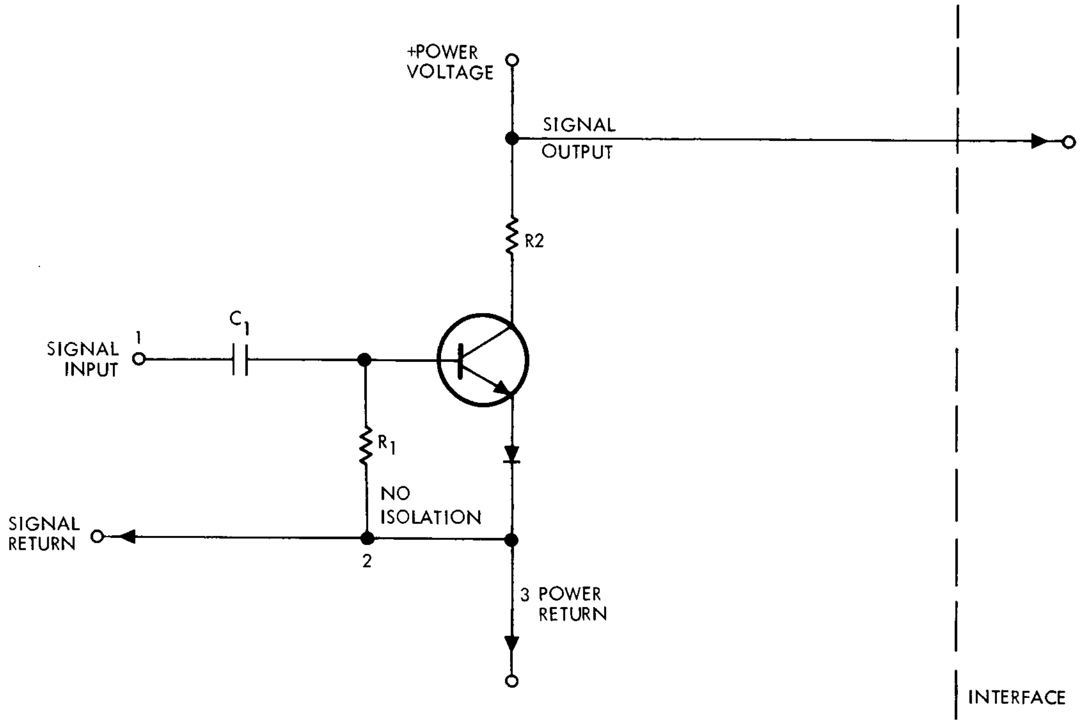


Figure G-2c. Typical Circuit Utilizing a Connecting Wire

If this example is carried one step further, it can be assumed that the signal output voltage across R_2 in Figure G-1 or G-2, is conducted to another item, where it is operated upon by some other circuitry with a separate power supply. A typical situation would be the circuitry normally employed to execute a ground command to a satellite so as to change the position of a coaxial switch or to pulse a solenoid valve. This case is shown in Figure G-3, employing the capacitor isolation of Figure G-2a.

It is readily seen from Figure G-3 that to truly implement the single-point ground philosophy would require AC coupling at every pulse circuit interface with another equipment or circuit which utilizes a different power supply. If, for example, the system had 60 commands and 60 AC coupling networks, 60 pairs of wire cables would be required at the decoder and command and sequencing unit interface. In addition, any pulse signal entering or leaving the command and sequencing unit and the digital telemetry unit must utilize AC coupling and two-wire cabling throughout the system. In general, the system weight would increase due to AC coupling networks and additional isolated return wires; the reliability would decrease as a result.

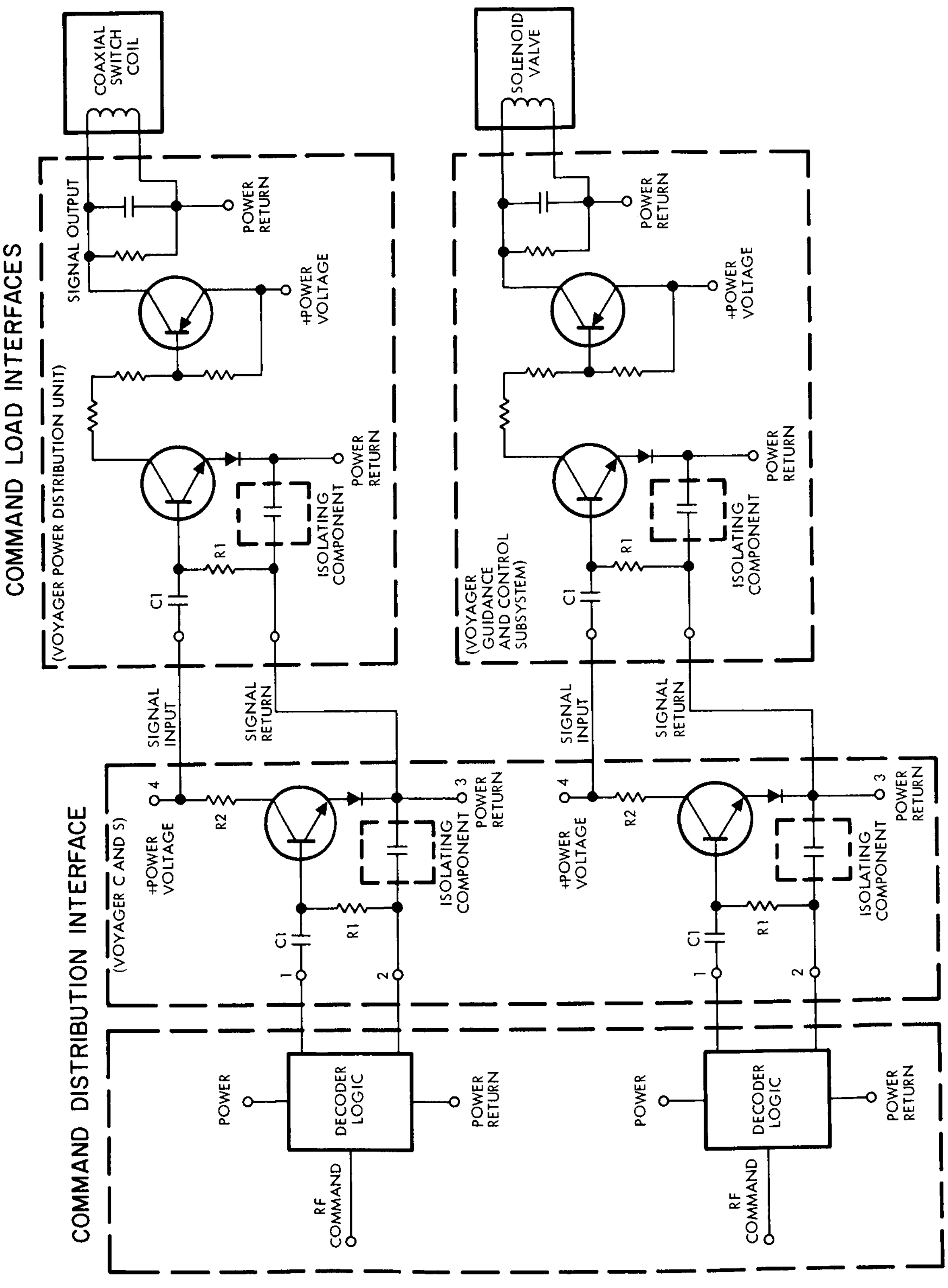


Figure G-3. Typical Ground Command Execution Chain

Furthermore, since normally at equipment interfaces the various grounds cannot be segregated for the reasons stated above, a system weight saving is implemented so as to employ one circuit return for several different circuits. A simplified diagram depicting this situation is shown in Figure G-4. The end result is a maze of interconnecting power, signal, and shield returns which has some large finite impedance (common impedance) at the frequencies which comprise the operational signals. This finite impedance will pass some undefined currents, resulting in every circuit which utilizes the common return being exposed to parallel voltage generators in their return circuit.

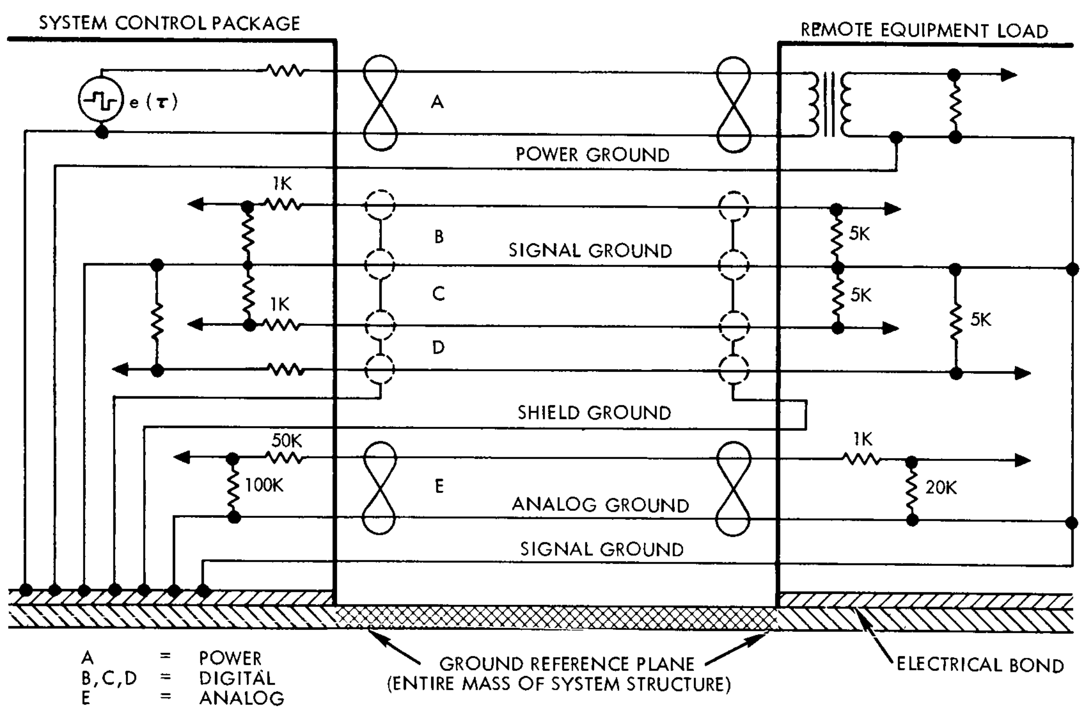


Figure G-4. Typical Implementation of Single-Point Ground Concept

The parallel voltage generation results from the internally interconnected return lines of the various subsystems being subjected to a L(di/dt) voltage drop in the parallel return line impedance. The change in current, di/dt, of a typical digital signal, having a rise or fall time of 0.1 microsecond and a current amplitude of 10 milliamperes, is sufficient to produce a 1-volt spike at the rise and fall of the pulse across a cable run of 3 to 6 feet of 24 AWG cable. The problem is magnified as the cable run lengthens, wire gauge decreases, current increases, and rise/fall time speeds up. The mechanism for this problem area is shown in Figure G-5.

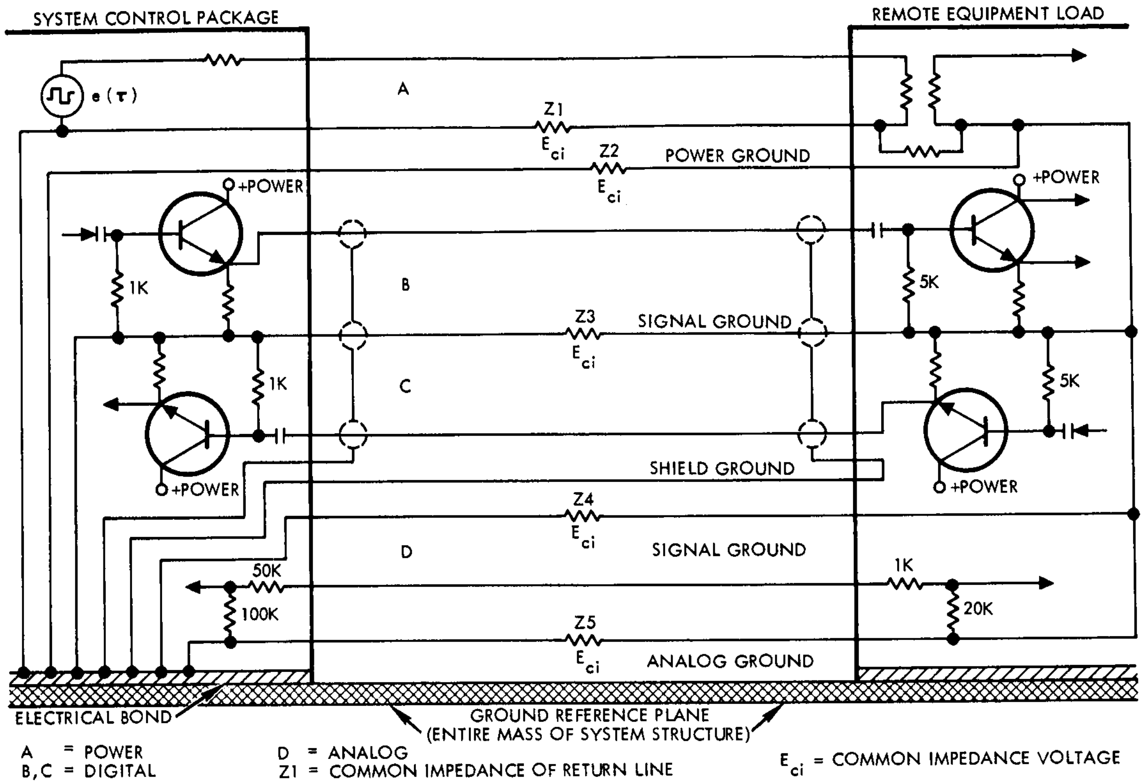


Figure G-5. Typical Interface for Common Impedance Coupling

The most logical method of resolving the common impedance problem is the reduction of the magnitude of the common impedance. For a space system, with weight a major consideration, the least expensive way of minimizing common impedance effects is to lower the common impedance by increasing the dimensions (mass) of the circuit returns. This last statement is the basic justification for the proposed Voyager electrical bonding requirements, since the well-bonded structure is an ideal answer for achieving a lower impedance return than a piece of wire. Since even the large mass of the structure still has some inductance (therefore impedance), it is necessary to use a compromise on the circuit grounding scheme to avoid switching heavy current loads through the structure. To achieve this requirement, the system and subsystem requirements specify that power circuits be connected to the system electrically bonded structure at one point per circuit. Furthermore, all power grounds will be DC isolated from each other. This requirement dictates the use of transformer or DC/DC converter between power circuits which will eliminate the problem of having common power return impedances. In addition, the fact that the power returns are isolated permits the signal returns to be isolated with a minimal amount of AC coupling of signal circuits.

The signal returns themselves are isolated through the use of AC coupling on all digital lines to isolate the digital circuits from the analog and power circuits. A simplified verson of this grounding scheme is shown in Figure G-6. Conflicting philosophies regarding ground networks will be analyzed and subjected to EMC tests during Phase IB.

To carry the discussion one step further, the electrically bonded structure is also utilized to provide a common reference for shield grounding. Considering the fast rise time digital telemetry and command

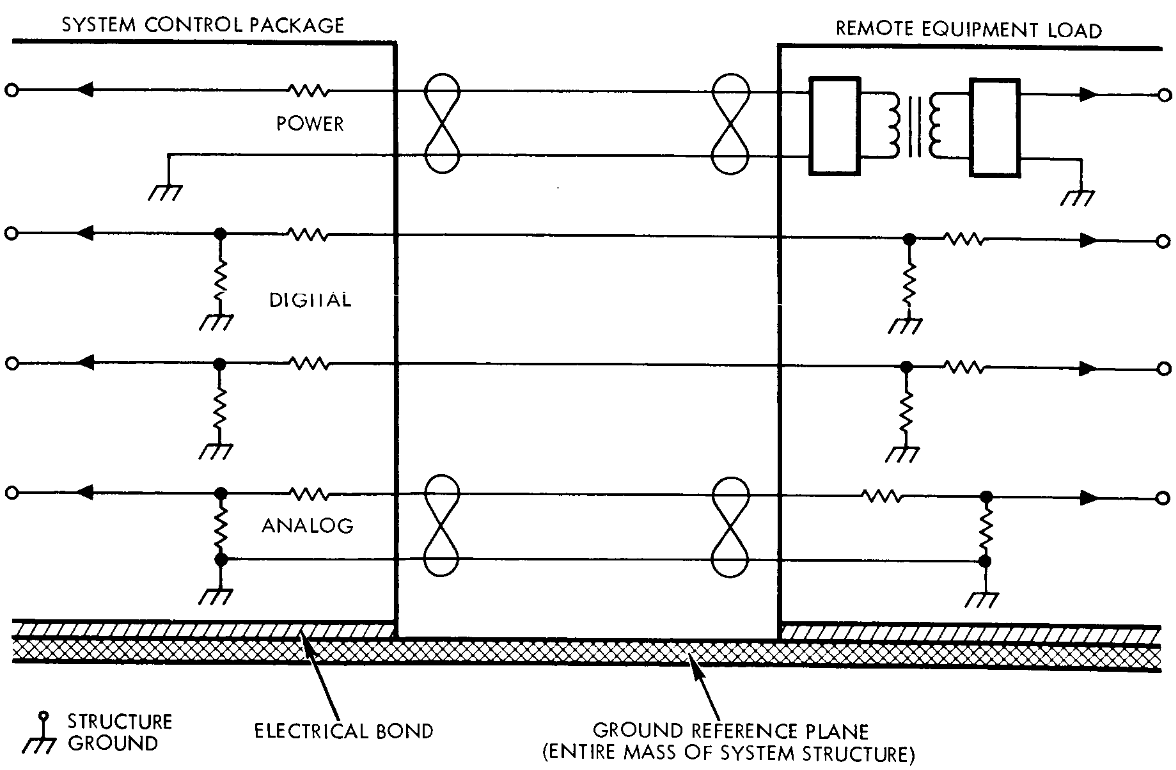


Figure G-6. Typical Structure Return Ground System subsystem, the Fourier frequency components comprising the normal signals extend, in most cases, from DC to beyond 10 mc. Employing conventional analysis techniques for the cable routing in a confined system layout, it is not uncommon to expect as high as 75 percent capacitive signal coupling between lines. Since some rise/fall time is functionally required by the system, the only alternative is to reduce the wiring capacitance between any two adjacent wires. Considerable quantities of both analytical and experimental data from various programs are available which prove that, in most cases, shields should be chassis grounded at both the source and load end of the cable and at any discontinuity to minimize intercable capacitive coupling in systems which have a multiplicity of signal types. These analyses also demonstrate that the optimum shielding technique makes use of coaxial cable. Therefore it is proposed that the Voyager system utilize an RF shield grounding philosophy. This requires shields to be individually terminated to chassis at every shield discontinuity (both source and load) with as short a shield termination as possible, as specified in the design requirements. While this policy will be used for the initial design, alternate shield grounding schemes will be analyzed and tested during hardware development.

A complete sketch of the proposed circuit and shield ground philosophy is shown in Figure G-7.

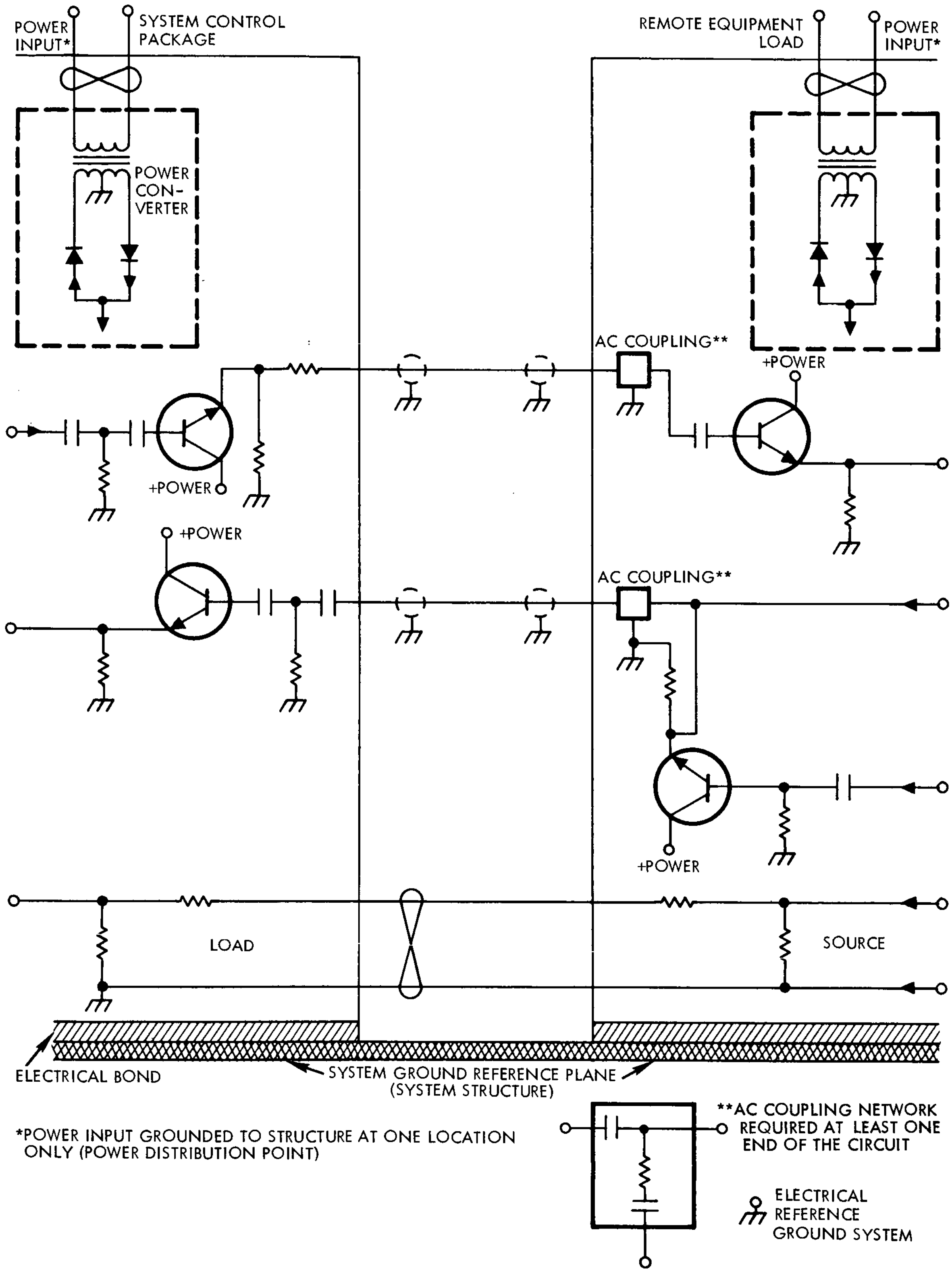


Figure G-7. Voyager Grounding Concept