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PHASE 1A STUDY REPORT

VOYAGER SPACECRAFT

VOLUME 2
1971 VOYAGER SPACECRAFT

30 July 1965

Prepared for
California Institute of Technology
Jet Propulsion Laboratory
Pasadena, California

Under Contract Number 951113

NAS-7-100

TRW SYSTEMS GROUP

Redondo Beach, California

CONTENTS

CONTENTS (Continued)

ILLUSTRATIONS

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

TABLES

TABLES (Continued)

TABLES (Continued)

INTRODUCTION

This volume presents the preferred design for the 1971 Voyager flight spacecraft and hardware subsystems and has been prepared in keeping with the Phase IA Work Statement to form a part of the final study report. In addition to serving as final documentation it has been evolved and published in draft form during the study in parallel with the technical design work to serve as a coordinating document for the study. Although essentially an integrated document, the volume has been broken down into individual design documents for flexibility in the generation and scheduling of the documentation.

The volume is organized as an hierarchy of technical data starting with the primary guidelines for the Voyager program and evolving into a presentation of design solutions for the various spacecraft hardware elements. In order to be as self-contained as possible, the requirements and guidelines applicable to the spacecraft as given by the "Preliminary Voyager 1971 Mission Specification" and the "Voyager 1971 Mission Guidelines" have been included. These requirements have been codified and interpreted as appropriate to achieve coordinated and consistent in-house implementation.

In addition to presenting a framework of requirements, the volume describes the design and operation of the spacecraft. It presents a summary of the basic engineering data associated with the selected design.

In keeping with the Phase IA Work Statement, the volume contains four sections as follows:

Section I: Mission Objectives and Criteria. Provides over-all ground rules, guidelines, and a definition of the general Voyager spacecraft system design philosophy. It codifies and interprets direction from JPL and augments this information as necessary to achieve coordinated and consistent in-house system implementation. The primary system and mission requirements and the criteria for establishing an order of priority among competing factors are covered.

Section II: Design Characteristics and Restraints. Explains in general how the objectives and criteria of Section I will be met in terms of a breakdown into characteristics and restraints applicable to the spacecraft and its various functional elements. It also describes the preferred design and its operation at the spacecraft system level.

Section III: System-Level Functional Descriptions. To allow better continuity of presentation and more flexibility in preparation, various system-level areas requiring individual treatment are presented as separate functional descriptions. These are established by reference in Section II and incorporated into Section III of this volume.

Section IV: Subsystem Functional Descriptions. Functional descriptions are presented in this section for each identified subsystem area. This documentation is developed to be as self-contained as possible although cross referencing is utilized to minimize duplication. The design criteria and philosophies applicable to the individual subsystems are included and technical descriptions are given covering what the subsystems are intended to do. Functions performance and interactions with other subsystems are included along with a description of physical characteristics.

In keeping with the Phase IA Work Statement, a single spacecraft design is presented here in Volume 2, without the underlying technical justification. That analysis is presented in Volumes 4 and 5, and those two volumes together constitute the basis for the design given here. As is made clear there, many design decisions are tentative, depending on mission details which are still not final. Consequently the design described here, must also be viewed as tentative in many respects.

MISSION OBJECTIVES AND DESIGN CRITERIA

VS-1-110

CONTENTS

CONTENTS (continued)

1. SCOPE

This document presents over-all ground rules, guidelines, and a definition of the general Voyager spacecraft system design philosophy. As the top-level document for presenting the spacecraft design it identifies and references the next level of underlying documentation as appropriate.

The primary system and mission requirements and the criteria for establishing an order of priority among competing factors are covered. These mission and program considerations have many implications on spacecraft design. In the present context, therefore, they are to be interpreted in terms of indirect design criteria or requirements on the spacecraft system. To a large extent the material presented here is of this type and has been excerpted from Reference 1.

2. PROGRAM OBJECTIVES

The primary objective of the Voyager program is to perform experiments on the surface of and in orbit about the planet Mars during 1971, 1973, and subsequent opportunities to obtain information about the existence and nature of extraterrestrial life; the Martian atmosphere, surface, and body characteristics; and the interplanetary environment near Mars.

The primary objective requires an orderly program of continually improving knowledge in science and technology for efficient and timely achievement. Such a program includes:

a) Scientific and engineering observations and experiments directed toward extension of the capability of Voyager systems to operate near the planet and on the planet surface, and the efficient development of this capability during the life of the program.

b) Scientific and engineering observations and experiments directed toward extension of the capability of the scientific instruments to operate near the planet and on the planet surface; more specific definition of future experiments concerning exobiology and planetology, and efficient development of this capability during the life of the program.

c) Scientific observations and experiments concerning possible biology and biochemistry of Mars.

d) Scientific observations and experiments concerning the physics and chemistry of the Martian lithosphere and atmosphere directed toward obtaining information essential to advancement of planetology.

A secondary objective is to perform certain field and particle measurements in interplanetary space between the orbits of earth and Mars.

3. PROGRAM PLAN AND IMPLEMENTATION

3.1 General

The Voyager program will continue and extend the unmanned scientific exploration of the near planets begun by the Mariner, Ranger, and Pioneer programs. It is intended that the Voyager planetary vehicle will conduct both orbital and lander missions to the planet Mars during the 1970's. Inherent in the Voyager design will be the capability of carrying significant scientific payloads to the planet, telemetering large amounts of data back to earth, and having long useful life in orbit about the planet or on its surface.

3.2 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 systems for each opportunity will be designed to be adaptable to subsequent missions; to accommodate a variety of spacecraft science payloads, capsule science payloads, mission profiles, and trajectories; and to accept improvement in technology with minimum redesign.

3.3 Mission Risk

The risk for each mission will be evaluated in terms of reliability, extent of development required, and resources required; mission goals will be selected so that the probability of achieving most of the objectives each time will be high. Particular emphasis will be given to simple and conservative design, redundancy wherever possible, and a complete and integrated program of component, subsystem, and system testing to ensure mission success.

3.4 Gross Missions Plan and Weight Allocation

Two operational space vehicles will be launched on the Saturn IB-Centaur during each Mars opportunity. The mission for each opportunity will vary according to the energy requirement for the particular opportunity as it 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 risk philosophy stated in the preceding paragraph. Tentative gross mission plans and weight allocations for each opportunity are given below. These plans will be adjusted, if necessary, as the design of the 1971 mission and systems progresses and as the program evolves from each Mars opportunity to the next.

Time Period	Mission	Weight Allocations (lb)			Orbiter Lifetime	Lander Lifetime
		Spacecraft Bus and Payload	Spacecraft Propulsion	Flight Capsule		
1971	Orbiter and Lander	2000	3500	2300	6 months	2 days
1973	Orbiter and Lander	2000	3500	2300	6 months	1 month
1975	Flyby and Lander	2000	500	4500	-	6 months
1977	Flyby and Lander	2000	500	4500	-	6 months

3.5 Flight Test

Appropriate deep space flight tests of elements of the flight spacecraft are planned for the 1969 Mars opportunity to permit evaluation of Voyager components and subsystems during long life in space and to demonstrate operations procedures.

4. PROJECT ELEMENTS

4.1 Systems

The Voyager project is organized into the systems described below, each of which contains all flight hardware, developmental hardware, models and spares, operational support equipment, test equipment, software, and associated manpower necessary to accomplish the assigned mission.

4.1.1 Launch Vehicle System

The launch vehicle system includes the three stages of the launch vehicle, nose fairing over the spacecraft, associated supporting ground equipment, software, and associated manpower.

4.1.2 Spacecraft System

The spacecraft system includes the flight spacecraft, its spares, developmental models, associated operational support equipment and software, and the management and engineering teams.

4.1.3 Capsule System

The capsule system consists of the flight capsule, its spares, developmental and sterilization assay models, associated software, operational support equipment, and the management and engineering teams.

4.1.4 Mission Operations System (MOS)

The MOS is that portion of the project which plans, directs, controls, and executes (with support provided by the Deep Space Network) the space flight operation after injection of the spacecraft on its trajectory; the mission-dependent equipment required at the network; and the operational teams.

4.1.5 Deep Space Network (DSN) System

The DSN system is comprised of the Deep Space Instrumentation Facility (DSIF), the Space Flight Operations Facility (SFOF), the DSN Ground Communications System (GCS) connecting these two facilities, and the personnel who operate these facilities.

4.1.6 Launch Operations System (LOS)

The LOS includes those elements of the project responsible for planning and executing the preflight and launch-to-injection phases of the mission.

4.2 Voyager Mission

A Voyager mission includes all preflight and flight activity directly associated with a particular Mars opportunity. The mission starts for each system at the completion of a mission acceptance review and after all associated tests (except prelaunch operations) and training operations have been completed. Mission acceptance of flight hardware is given concurrently with the decision to ship to the Air Force Eastern Test Range (AFETR). The mission is complete when all scientific and engineering data have been returned to earth, reduced, and delivered to the cognizant organizations.

4.3 Voyager Mission System

The Voyager mission system is the over-all system directly associated with a particular Voyager mission as defined in 4.2. It is composed of two Voyager space vehicles and an associated Voyager mission support system (see Figure 1).

4.3.1 Voyager Space Vehicle (VSV)

The Voyager space vehicle consists of the Voyager launch vehicle and the Voyager planetary vehicle.

a. Voyager Launch Vehicle (VLV)

The VLV provides the capability to lift off and inject the Voyager planetary vehicle on a heliocentric trajectory. It is composed of the Saturn IB as a basic launch vehicle, a Centaur stage, and a nose fairing.

b. Voyager Planetary Vehicle (VPV)

The VPV consists of the Voyager flight spacecraft and the Voyager flight capsule.

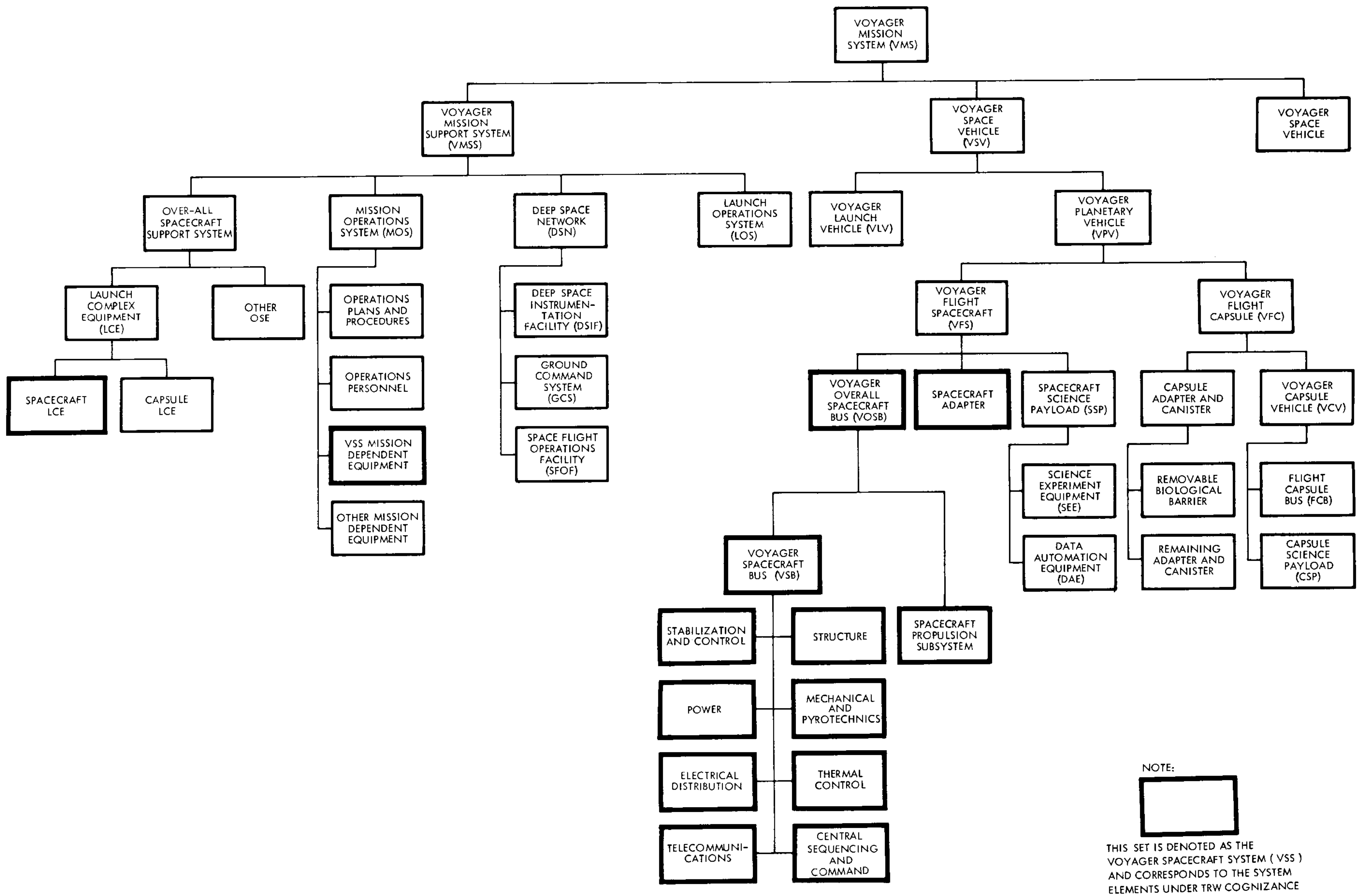


Figure 1. Breakdown for Voyager Mission System

4.3.2 Voyager Mission Support System (VMSS)

The VMSS includes all elements of the Voyager mission system other than the Voyager space vehicle. It includes facilities, supporting ground equipment, spares, software, and associated manpower. The VMSS is broken down into four elements as follows:

a. Launch Operations System (LOS)

The LOS corresponds to the support system for the launch vehicle, including range functions associated with launch vehicle flight.

b. Planetary Vehicle Support System

The planetary vehicle support system is the support system associated with the Voyager planetary vehicle. It consists of the launch complex equipment (LCE) and other OSE. The LCE is used to support and command the Voyager planetary vehicle and to monitor and record its functions (both by hardline and by telemetry) during prelaunch checkout. It includes all such equipment in the spacecraft assembly building, on the launch pad, and in the blockhouse. It is also used to support operations in the explosive safe facility (ESF). The LCE includes both mechanical operational support equipment (MOSE) and electrical operational support equipment (EOSE).

Capsule Launch Complex Equipment. The capsule LCE consists of the support equipment within the LCE that is directly associated with the Voyager flight capsule.

Spacecraft Launch Complex Equipment. The spacecraft LCE consists of the remainder of the LCE after the capsule LCE has been excluded.

c. Deep Space Network (DSN)

The DSN system includes the Deep Space Instrumentation Facility and Space Flight Operations Facility, communications network connecting these facilities, and personnel who operate the mission-dependent equipment of these facilities. The DSN system is utilized to acquire, track, and communicate with one or both Voyager planetary vehicles after injection and until separation of the associated Voyager capsule vehicle. It functions in a similar capacity during the planetary phase of operations with regard to Voyager flight spacecraft in Mars orbit(s) and a lander on the surface of Mars. Planned and existing DSN capability applicable to the Voyager program is given in Reference 2.

Deep Space Instrumentation Facility (DSIF). The DSIF consists of earth-based precision tracking stations with capabilities which vary in time as described in Reference 2.

Space Flight Operations Facility (SFOF). The SFOF is utilized to coordinate and direct all flight operations of the mission, and houses the operating and analysis teams functioning during this phase. It is located at JPL, Pasadena. It operates in conjunction with associated mission dependent equipment (MDE) to fulfill particular requirements of the Voyager mission.

d. Mission Operations System (MOS)

The MOS associated with a particular Voyager mission consists of operations plans and procedures, operations personnel who plan, direct, and control the space flight operations after injection of the spacecraft on its trajectory, and mission dependent equipment (MDE).

The MDE consists of any item required by the DSIF or SFOF to complete the functional requirements of the Voyager mission and which is not required on any other program. The item may be categorized as either software or hardware. MDE for a Voyager mission consists of MDE which is under the cognizance of the Voyager spacecraft system (VSS) contractor and the remainder which is designated as other MDE. The VSS mission-dependent equipment would normally perform various specialized functions and be located with the DSIF. In the past these functions have included such things as the demodulation and decommutation of the data subcarrier.

4.4 Voyager Flight Capsule (VFC)

The VFC contains all elements of hardware required to separate from the Voyager flight spacecraft, survive separated transit, enter and decelerate in the planetary atmosphere, land on the surface, support the operation of a capsule science payload in the atmosphere and on the surface as required, and to telemeter the resulting data back to earth or to a spacecraft relay. Also included as parts of the VFC are the biological barrier required to maintain sterility and the necessary equipment for adjusting the trajectory to permit planetary impact.

THE VFC consists of a capsule adapter and canister and a Voyager capsule vehicle (VCV). After removal of the biological barrier the VCV, consisting of the flight capsule bus (FCB) and the capsule science payload (CSP), separates and proceeds independently to achieve Mars entry and landing.

4.5 Voyager Flight Spacecraft (VFS)

The VFS consists of the Voyager over-all spacecraft bus, the spacecraft adapter, and the spacecraft science payload.

4.5.1 Spacecraft Science Payload (SSP)

The SSP consists of science experiment equipment and data automation equipment.

a. Science Experiment Equipment (SEE)

The SEE consists of instrumentation and other equipment for performing scientific experiments or observations in transit to Mars and in the vicinity of the planet.

b. Data Automation Equipment (DAE)

The DAE consists of control and data handling equipment carried aboard the Voyager flight spacecraft that is associated with the SEE.

4.5.2 Voyager Over-all Spacecraft Bus (VOSB)

The VOSB is a self-contained spacecraft supplying its own power from solar energy or from internal sources, providing propulsion functions as required, controlling its attitude, communicating with earth, and providing its own thermal control. It is capable of monitoring and telemetering its own operation, carrying out on-board sequencing and logic and providing for ground command capability. It serves as a carrying vehicle for the flight capsule and the spacecraft science payload and provides services to these elements as required.

a. Spacecraft Propulsion Subsystem

The spacecraft propulsion subsystem performs propulsion functions corresponding to midcourse corrections, trajectory adjustments near Mars as required, and for retrothrust into Mars orbit.

b. Voyager Spacecraft Bus (VSB)

The VSB consists of those hardware elements which form, when combined with the spacecraft propulsion subsystem, an integrated self-contained vehicle system. It consists of the following subsystems: structure, mechanical and pyrotechnics, thermal control, stabilization and control, power, electrical integration, command and sequencing, and telecommunications.

4.5.3 Spacecraft Adapter

The spacecraft adapter serves as the structural connection between the planetary vehicle and the launch vehicle if such is required.

5. 1971 MISSION OBJECTIVES

5.1 Primary Objective

The primary objective of the 1971 mission is to institute the basic capability to (1) place significant payloads at Mars, (2) conduct observations of Martian phenomena over extended time periods, and (3) transmit the results of these observations to earth.

5.2 Secondary Objective

A secondary objective is to provide experience with both flight and ground systems required to deliver and operate the SSP, to ferry and separate the capsule, and to deliver and operate the CSP.

5.3 Tertiary Objective

A tertiary objective is to obtain scientific and engineering observations in interplanetary space during the transit flight from earth to Mars and to transmit the resulting data back to earth.

5.4 Quaternary Objective

A quaternary objective is to provide flight and ground specific designs and equipment elements compatible with subsequent Voyager missions to Mars.

6. MISSION REQUIREMENTS

The Voyager spacecraft system (VSS) is designed in general accordance with Reference 1.

6.1 General

6.1.1 Launch and Hold Criteria

Launch and hold criteria are established for all system elements and the implications of these criteria are recognized in their system design.

6.1.2 Launch Period and Interval

The spacecraft is compatible with a total launch period of 45 to 60 days and with a minimum daily firing window of 2 hours. It is also compatible with a minimum interval between launches of 2 days.

6.1.3 Arrival Date Separation

The mission calls for a minimum separation of 10 days between the arrival dates at Mars for any two spacecraft.

6.1.4 Launch Facilities

AFETR facilities at Cape Kennedy, Florida, are used for launch and prelaunch operations. Prelaunch assembly and checkout is conducted at the spacecraft checkout facility for the planetary vehicle. An explosive safe facility is used for flight capsule sterilization, propellant and gas loading, final spacecraft alignment, installation of other hazardous components, and spacecraft encapsulation in the launch vehicle nose fairing. Two launch pads are to be utilized for each Voyager mission.

6.1.5 Range Tracking and Telemetry

AFETR launch-through-injection tracking and telemetry reception is required for launch vehicle and spacecraft instrumentation and DSIF acquisition.

6.1.6 Earth-Based Tracking

The spacecraft telecommunications system is designed to be compatible with existing and planned DSN capability as defined in Reference 2. Spacecraft DSN design verficiation tests at Goldstone will be executed prior to design freeze of the spacecraft telecommunications system. The design of the spacecraft is constrained so that the flight project operations requirements to command the spacecraft immediately after initial acquisition by the DSN are compatible with the ability of the DSN to complete initial acquisition.

6.1.7 RF Relay Link

The flight spacecraft design provides a VHF relay receiver and a fixed low-gain antenna. This equipment is capable of receiving post-separation data from the capsule up to and including impact at a rate of 10 bits/sec. This data will be handled and retransmitted to earth by the spacecraft communications system. Relay communications after impact are also considered.

6.1.8 Period of Operation

A useful lifetime of 6 months in Mars orbit is desired for orbiter missions.

6.2 Planetary Quarantine

The probability that Mars is contaminated before the year 2021 as a result of any single launch is not to be greater than 10^{-4}. Consideration is given to the implication of this requirement on the spacecraft and all emissions, ejecta, etc.

6.3 Probability of Success

6.3.1 System Design

All Voyager systems and project elements will incorporate design, test, and operational procedures designed to maximize mission probability of success. These efforts include the following:

a) Failure-mode analysis and design for partial mission success in the event of failure

b) Establishment of adequate design margins

c) Application of functional and parallel redundancy techniques where practical.

6.3.2 Design Philosophy

Emphasis is placed on simplifying the interfaces between the systems of the project to simplify flight operations and ground test and checkout operations and to increase the mission reliability. Particular emphasis is placed on simple and conservative design and a complete and integrated program of component, subsystem, and system testing to increase reliability and assure mission success. Wherever possible, the Voyager design takes advantage of equipment and techniques developed and experience gained in the Ranger, Mariner-R, and Mariner-C designs.

6.3.3 Lifetime

Since a Mars-mission spacecraft must survive and function during earth-to-Mars transit and subsequent orbiter 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.

6.3.4 Numerical Reliability Requirements

The 1971 Voyager mission objective is ordered in the following way, with estimates of desired cumulative probabilities of success for each flight stated for each subordinate objective.

a) Perform a successful launch and injection of the planetary vehicle into a prescribed transfer orbit: 90 per cent probability of success.

b) Perform a successful spacecraft-capsule separation maneuver at a preselected time and location: 80 per cent probability of success.

c) Place an operating science payload in a selected orbit about Mars and perform the functions necessary to begin orbital operations: 65 per cent probability of success.

d) Perform necessary orbital operations to obtain data from the orbital science payload and return the data to earth, for a specified time of 1 month and as long thereafter as possible: 45 per cent of probability of success.

*e) Place the flight capsule on a selected impact trajectory to Mars: 75 per cent probability of success.

*f) Enter the Martian atmosphere and obtain data on the lower Mars atmosphere from the capsule science payload: 65 per cent probability of success.

*g) Land the flight capsule, establish communications with earth; return entry, landing, and system status data to earth: 45 per cent probability of success.

*h) Perform necessary lander operations to obtain data with the capsule science payload over at least one Martian diurnal cycle and return the data to earth: 35 per cent probability of success.

6.4 Schedule Criteria

The 1971, 1973, and 1975 Mars opportunities place absolute constraints on the project schedule. All design and development for systems, subsystems, and components is compatible with the project milestones.

*These objectives involve the spacecraft bus only in regard to relay link support to the capsule vehicle.

6.5 Competing Characteristics

Where there are conflicting technical requirements, the following order of priority relative to acceptable risks governs:

a) Meeting the requirement for planetary quarantine

b) Proper operation of telemetry and communication equipment on downlink

c) Continuous, proper sun-line attitude orientation of spacecraft

d) Continuous, proper temperature control of spacecraft

e) Proper functioning of power equipment on spacecraft

f) Continuous, proper temperature control of capsule

g) Proper functioning of power equipment on capsule

h) Proper operation of communications and command equipment (uplink)

i) Proper roll control of spacecraft

j) Proper execution of midcourse maneuvers

k) Proper spacecraft-capsule separation

l) Proper execution of the maneuver placing the spacecraft in a useful Mars orbit

m) Proper operation of spacecraft instrumentation at Mars

n) Proper execution of the maneuver placing the capsule on a useful Mars landing trajectory

o) Successful capsule entry and landing

p) Proper operation of the capsule instrumentation at Mars

q) Proper operation of the cruise instrumentation

r) Design value to the 1973 and subsequent missions

s) Equipment applicability to the 1973 flight hardware

t) Development of system contractor capability for 1973.

6.6 Trajectories

6.6.1 Ascent Mode

The parking orbit ascent mode will be utilized for the Mars 1971 mission. An arbitrary limit of a 25-minute parking orbit now exists for the 1971 mission; therefore, all vehicle equipment and expendables will be sized for this duration, and all performance calculations will be based upon this limitation. The minimum parking orbit coast time will be 2 minutes or less. Later Voyager missions will require coast times in excess of 25 minutes.

6.6.2 Transfer Trajectories

Type I transfer trajectories will be utilized for the 1971 mission. A maximum C_3 of 18 km^2/sec^2 is assumed, which is compatible with providing an adequate mission weight margin for a separated planetary vehicle weight of 7800 pounds. The hyperbolic excess velocity at Mars will not exceed a maximum of 5 km/sec. In order to improve orbit redetermination geometries, the declination of the departure asymptote (DLA) will be greater than 5 degrees and the inclination of the heliocentric transfer plane to the ecliptic plane will be greater than 0.1 degree. Limiting launch azimuth boundaries require that $|DLA|$ be less than or equal to 50 degrees; for preliminary launch sector planning purposes, $|DLA|$ is assumed less than or equal to 33 degrees for the 1971 mission.

6.6.3 Arrival Over Goldstone

Orbit insertion and capsule entry, descent, and landing will occur in view of the DSIF at Goldstone, California. This requirement may be modified if it significantly restricts the achievement of certain mission objectives, such as the selection of particular Mars surface features for scientific observation, etc.

6.6.4 Satellite Orbit Selection

The geometry selected for a satellite orbit about Mars is to be favorable for obtaining the desired orbiter science data. The following constraints will be satisfied:

a) The planetary quarantine constraint will be satisfied.

b) Occultation of the sun by Mars is to be avoided during at least the first month of orbital operations. After that time, sun occultation is permitted if other orbit characteristics are improved; but sun occultation will not last for more than 10 to 15 per cent of each orbit period.

c) No orbit will be selected which causes permanent loss of the spacecraft roll reference.

d) The orbit insertion maneuver will be performed in view of earth. Sufficient engineering data will be transmitted to earth to permit verification of the orbit-insertion attitude in time to inhibit the propulsion.

6.7 Accuracy

6.7.1 Injection Accuracy

A maximum 1-sigma midcourse velocity increment of 15 meters/sec applied 2 days after injection is sufficient to remove the injection errors resulting from guidance and launch-time dispersions.

6.7.2 Target Accuracy

For the 1971 mission, a target accuracy goal of 500 km (3-sigma, impact-parameter space) is specified. Reduced accuracy may be acceptable in the event that a significant improvement in reliability or simplification of design can be realized.

6.8 Weight Allocation

Weights for the planetary vehicle for the 1971 mission $C_3 \leq 18$ $(km/sec)^2$ are allocated as follows:

Item	JPL Specification Weight (lb)	
Flight capsule vehicle separated weight	1950	
Flight capsule adapter and canister (a maximum of 150 pounds may remain with spacecraft)	350	
Flight spacecraft minus spacecraft adapter weight remaining with Centaur (includes 250 pounds of science)	5500	
Separated planetary vehicle weight		7800
Spacecraft adapter remaining with Centaur (includes spacecraft support above field joint)	250	
Planetary vehicle weight		8050
Spacecraft support below field joint	250	
Total		8300

6.9 Safety

6.9.1 General

Factors that present a real or potential hazard to equipment or personnel typically include pyrotechnic devices, solid propellants, liquid propellants, high-pressure gas storage bottles, high voltages, and possibly toxic materials. Radioactive materials may also be encountered.

Safety factors for pressure vessels, etc., will be considered in the design and use of the equipment, and the rationale for determining the tradeoffs between vehicle performance and personnel safety will be documented; at the same time, the hazards to the flight equipment itself will be considered. Consideration will be given to safety techniques to be employed while testing and verifying flight hardware during times when pressure vessels must be partially or fully charged, squibs installed, radioactive sources used, etc. The design of hoisting, handling, and testing fixtures will give special attention to minimizing hazard to both personnel and equipment. The procedures utilized to fill pressure bottles, install and connect squibs, and load rocket propellants will also consider safety aspects as of paramount importance.

6.9.2 Pyronetworks

Pyrotechnic devices incorporate a switch (or switches) to maintain the equipment in a safe condition until such time as activation of the pyrotechnic will not cause damage. Any unlatching device is to incorporate a safety device to protect against spurious signals. Electro-explosive devices, associated wiring, and firing circuitry conform to Reference 3.

6.9.3 Pressure Vessel Constraints

Pressure vessels for flight hardware fall into two categories: hazardous to personnel and nonhazardous to personnel. Hazardous vessels are designed with a safety factor of 2.2 (burst pressure/ operating pressure). Nonhazardous vessels carry the same safety factor as the associated structure. Rocket motor cases are designed to a factor of 1.15, based on yield strength. Special attention is given to the method of mounting pressure vessels to avoid undue restraint that could induce high-stress concentrations during pressurization. Designs with minimum welding (integral ports and integral mounting bosses) are preferred. Vessel wall thickness-to-diameter ratios are at least 1/1000.

7. RELIABILITY

7.1 General

The design approach for Voyager takes into account the interaction of reliability and performance. A basic policy will be implemented whereby a combined performance-reliability yield is considered for all individual mission phases. Such combined yields are defined consistent with the cumulative probabilities of success specified in 6.3.4 and take into account the precedence in competing characteristics given by 6.5. Mission risk is considered in keeping with 3.3. General design philosophy emphasizes reliability as covered in 6.3.2.

7.2 System Simplicity

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 minimum performance requirements.

7.3 Redundancy

Redundancy is added to the basic system of 7.2 where practical, both functional redundancy (alternate modes) and equipment redundancy (parallel operation and sequential corrections). The former is given special emphasis. The selection of a particular redundancy application is based on a system-level evaluation and a comparison with other such applications on a competitive basis. Redundancy techniques are not used in lieu of high inherent (nonredundant) reliability and product quality.

7.4 Risk Models and Reliability Apportionment

Probabilistic models are constructed to provide a numerical interpretation of design reliability requirements based upon the cumulative mission reliability requirements of 6.3.4. The apportionment of reliability design requirements to system elements is based on such a model. These models also serve to manage the utilization of redundancy techniques as discussed in 7.3.

7.5 Materiel Policy

In order to assure an optimum utilization of available (proven reliability) materiel for all spacecraft systems, specific constraints are established for equipment, parts, and materials selection and application. This requires the extent of any new development to be identified and categorized with regard to reliability risk. Detailed constraints are given in VS-3-120.

7.6 Test Considerations

An integrated test plan forms the basis for reliability development and measurement as covered in Volume 3, Section IV.

Assessed reliability numerics form the basis for statistical test hypotheses where meaningful. Test planning is based upon a joint consideration of:

a) Anticipated failure modes

b) Reliability risk models and redundancy

c) Use of proven materiel

d) Testing time

e) Costs.

7.7 Reliability Documentation

Areas of reliability documentation are provided to delineate detailed requirements and their associated restraints. These are:

a) Reliability Design Objectives. Specific models and mission interpretation with backup data are presented in VS-3-120 (Reference 5).

b) Reliability Program Plan. The definition and scheduling of a comprehensive reliability program plan (in accordance with Reference 6) is provided as part of the program implementation plan.

REFERENCES

1. JPL Project Document No. 45, V-MA-004-001-14-03, "Preliminary Voyager 1971 Mission Specification," 1 May 1965.

2. JPL Engineering Planning Document 283, "The Deep Space Network, A Planning Capability for the 1965 - 1980 Period," W. H. Bayley, July 1965.

3. AFETRP 80-2, General Range Safety Plan, Volume I, and Appendix A.

4. TRW Voyager Spacecraft Design Document VS-2-110, "Design Characteristics and Restraints."

5. TRW Voyager Spacecraft Design Document VS-3-120 "Reliability Design Objectives."

6. NASA Reliability Publication NPC 250-1, "Reliability Program Provisions for Space System Contractors," July 1963 Edition.

DESIGN CHARACTERISTICS AND RESTRAINTS

VS-2-110

CONTENTS

1. SCOPE

This section explains in general how the objectives and criteria of Section I will be met in terms of a breakdown into characteristics and restraints applicable to the spacecraft and to various functional areas.

Design restraints at the spacecraft level presented here are essentially a summary of material from Reference 1.

2. GENERAL

2.1 Description

The Voyager flight spacecraft is a fully attitude-stabilized device utilizing celestial references. It is self-contained in that it supplies its own power, is capable of providing velocity increments for midcourse trajectory correction and for attaining a Mars orbit, is capable of radio communication with the earth, is thermally integrated and controlled, and monitors and telemeters its own operation. In conjunction with the spacecraft science payload, it monitors various scientific phenomena during transit and near Mars and telemeters this information back to earth. On-board sequencing and logic and ground command capability are provided. Services are provided to the flight capsule and spacecraft science payload such as power, timing and sequencing, telemetry, and command. 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.

2.2 Development Freeze

All design concepts, materials, and components that are considered for the Voyager 1971 mission will have a development freeze date of July 1966. Only those design concepts that have demonstrated feasibility and have been fully developed by that date will be considered for inclusion in the Voyager 1971 mission.

2.3 Biological Contamination

The probability of Mars contamination by the spacecraft is compatible with the mission requirements of 6.2 in VS-1-110. Provisions are to be made to avoid contamination of the flight capsule by spacecraft materials or actions after opening the capsule biological barrier in preparation for capsule-spacecraft separation at Mars approach.

2.3.1 Spacecraft Heat Sterilization

The flight spacecraft does not require heat sterilization.

2.3.2 Spacecraft Chemical Sterilization

All spacecraft components are capable of withstanding exposure to a gas mixture of 12 per cent ethylene oxide and 88 per cent freon gas for 10 hours at a relative humidity between 30 and 50 per cent.

2.3.3 Ejecta

All spacecraft ejecta, such as attitude control jet gas, will be biologically sterile.

2.3.4 Orbital Decay

The Mars orbit for orbiter missions will be such that an orbiting body will not decay into the Martian atmosphere prior to the year 2021 within the probability as given in 6.2 of VS-1-110.

2.4 Absence of Capsule

The spacecraft can perform its mission if the capsule is not present.

3. PERFORMANCE CHARACTERISTICS

3.1 Mission Profile

The nominal mission for each space vehicle is composed of the following phases:

a)	Prelaunch	All final assembly, checkout, and test procedures and activities resulting in a commitment to launch.

b)	Launch and Injection	Final space vehicle countdown, launch, parking orbit execution, insertion into the transit trajectory, and separation of the planetary vehicle from the launch vehicle.
c)	Acquisition	Acquisition by the planetary vehicle of external attitude references and execution of all sequences leading to cruise status.
d)	Interplanetary Cruise	All events and sequences during the transit flight to Mars when the planetary vehicle is on external references.
e)	Interplanetary Trajectory Corrections	All events and sequences used to alter the transit trajectory of the planetary vehicle until the return to cruise status.
f)	Spacecraft-Capsule Separation	All events and sequences used to separate the flight capsule from the planetary vehicle, including return of the flight spacecraft to cruise status.
g)	Flight Capsule Trajectory Deflection	All events and sequences used to place the flight capsule on a selected Mars impact trajectory.
h)	Flight Spacecraft Cruise	All flight spacecraft events and sequences between spacecraft-capsule separation and the orbital insertion phase.
i)	Flight Capsule Cruise	All capsule events and sequences between spacecraft-capsule separation and capsule vehicle entry.
j)	Flight Capsule Entry	All capsule events and sequences from the time the capsule vehicle reaches a distance of about 3620 km from the center of Mars until the beginning of terminal velocity descent.
k)	Flight Capsule Descent and Landing	All capsule events and sequences from the beginning of terminal velocity descent.

l)	Landed Operations	All capsule system operations from the time the lander comes to rest until the last time the communications signal is lost or the capsule mission is declared to have ended.
m)	Flight Spacecraft Orbit Insertion	All events and sequences used to insert the flight spacecraft into orbit about Mars.
n)	Orbital Operations	All flight spacecraft system operations from the time external references are reacquired after orbit insertion until the communications signal is finally lost or the orbiter mission is declared to have ended.

3.2 Trajectories

Trajectory considerations in addition to those specified in 6.6 of VS-1-110 are discussed in Reference 2.

3.3 Weights and Mass Properties

3.3.1 Weight Limit

The weight of the flight spacecraft less the flight spacecraft adapter remaining with the Centaur will not exceed 5500 pounds, including 250 pounds of flight spacecraft science payload.

3.3.2 Margins

Weight margins will be carried in all weights during system design, and they will be identified in all weight statements. Margins will be consistent with weight estimation confidence levels.

3.3.3 Reporting

Weights are reported separately by functions, by subsystems, and by the equipment list. The source of each weight estimate is given. Equipment-related weights are reported in VS-3-111, and other weights data are reported in VS-3-112. Supporting data is given in Volume 4.

3.4 Maneuver Requirements

Data associated with spacecraft maneuvers for trajectory correction, retrothrust into Mars orbit, and Mars orbit trim are presented in VS-3-102.

3.5 Terminal Guidance Sensing

A design objective is that the spacecraft provide a capability for utilizing the science payload photographic sensor(s) to measure the direction to the center of Mars (or other known point) for terminal guidance computations.

It is desired that at a range of 5×10^5 km the celestial direction of Mars is to be determinable with an accuracy of 0.75 mr, 3σ. This can be an on-board measurement or on-board sensing plus earth-based data processing.

4. OPERABILITY

4.1 Reliability

In order to achieve a reliability level consistent with mission and spacecraft constraints, reliability design objectives are provided for each subsystem in VS-3-120.

4.2 Environment

Preliminary design environmental conditions are given in II.F of Reference 1. An exception has been taken to the specified radiation levels near Mars. This is discussed in Volume 5, Appendix B.

5. SYSTEM DESIGN STANDARDS

5.1 Selection and Control of Parts, Materials, and Processes

5.1.1 General

On the Voyager project parts, materials, and processes are standardized and controlled. All aspects, including selection, procurement, and utilization of parts and materials and selection and utilization of processes, will be controlled throughout final design and fabrication.

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 demonstrably necessary to meet minimum system performance requirements.

All parts and materials will be selected on the basis of suitability for the intended application and reliable performance during the pre-launch, launch, earth-to-Mars transit, and Martian-orbit phases of the mission. TRW will establish a parts, materials and processes standardization and control program for implementing JPL requirements.

5.1.2 Magnetic Requirements

a. Materials

Nonmagnetic materials will be used wherever possible. All materials used on the spacecraft, including bulk and raw materials will be magnetically evaluated.

b. Interference Magnitudes

A design objective is that all spacecraft assemblies will have total field magnitudes of less than 1 gamma at three times their average dimension measured along their natural rectilinear axes and that small assemblies and stray current loop fields will cause a field of less than 1 gamma at 2 feet. For assemblies containing permanent magnets compensation will be investigated.

c. Magnetic Testing

Magnetic testing considerations are discussed in Volume 3, Section IV., 3.3.

5.1.3 Electronic and Electromechanical Parts

All electronic and electromechanical parts will be selected from a parts list of be established by TRW and approved by JPL.

5.1.4 Materials

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

5.1.5 Processes

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

5.2 Electrical Interface and Grounding

All electrical interfaces will be defined to encompass the interconnecting circuitry between interfacing subsystems and interfacing equipment packages. The electrical interface circuit and grounding criteria follows:

a) Characteristics of transmitting subsystems, receiving subsystems, and interconnecting wiring will be compatible with signal or power being transferred.

b) Circuit radiation and pickup will be below level of interference.

c) Subsystem interaction via common electrical mains and wiring circuits will be below level of interference.

d) Circuits will be referenced to spacecraft structure either directly or via protective networks.

e) Unnecessarily elaborate circuit or wiring features will be avoided.

5.3 Circuit Design Practice

The practices indicated below will be employed in selecting or designing circuits used in the spacecraft.

5.3.1 Reliability

a. Parts

All part applications will be guided by part derating factors established by TRW with JPL approval.

b. Circuits

The following guidelines will be followed:

a) Proven circuits will be used, i.e., circuits that have demonstrated reliability in previous programs or in extensive life and operational testing.

b) A maximum number of common circuits will be used. Circuitry standardization will be optimized throughout the entire spacecraft design. TRW will establish a plan to ensure that standardized circuitry is applied in appropriate design areas.

c) Digital logic will be standardized. TRW will establish a plan to assure that standardized digital logic be employed in appropriate design areas.

d) Transient and DC worst-case analysis will be applied to all circuits.

e) Circuits and subsystems will be designed to facilitate in-process testing at the module and subassembly level in order to detect workmanship errors, miswiring, etc., which were not revealed by other methods and to detect "infant mortality" early in the production process.

f) Circuits performing critical operations will be made independent of any single part failure; this implies redundancy features. Redundancy schemes which require no sensing and switching in case of failure are preferred.

g) Circuits will be reproducible and/or interchangeable. "Part tailoring" and "critical adjustments" will be minimized.

h) All circuits will be amenable to standard analytical techniques to determine performance margins.

5.3.2 Performance

The circuits will operate within the specified tolerances under all of the following conditions:

a) Lifetimes - mission life and field testing

b) Environmental conditions specified for flight acceptance and type approval testing

c) Specified power limitations

d) Specified power supply regulation, impedance and ripple

e) Electrical noise

- The circuits themselves will produce no more than specified radiation noise

- The circuits will produce no greater than a specified variation in impedance on the common power supply

f) The OSE will not require the operation of the spacecraft to verify OSE readiness for subsystem or system testing.

5.3.3 Maintainability

Circuits will operate continuously without adjustments.

5.3.4 Test Circuitry

Test circuits are defined to include all electrical stimulus, monitor, and control interfaces with the spacecraft which are used only during ground operations. In addition, test circuits include circuit elements in spacecraft test cabling and OSE.

a. Test Points

Critical test points and waveforms are brought out to a test plug for trouble-shooting and isolated for protection of spacecraft hardware against inadvertent damage during trouble-shooting operations.

b. Test Circuit Isolation

All test circuits provide appropriate isolation between spacecraft and environment, i.e., OSE and ambient or associated electrical fields. In addition, the spacecraft is protected from damage arising from faults and errors assignable to OSE equipment and personnel.

To satisfy the foregoing, the following design requirements will be met:

a) All test circuits will be capable of being shorted to the common spacecraft electrical ground without deleterious effect on circuit operation and performance.

b) All umbilical functions which are serviced by the same umbilical connector will be capable of being shorted together in any combination without deleterious effects on circuit operation and performance after the short is removed.

c) All umbilical functions will be capable of withstanding a 75-volt pulse 5 microseconds in duration, between each function and spacecraft electrical ground.

d) All test circuits will be required to accomplished the desired test functions without modifying spacecraft performance or behavior.

5.4 Packaging

The Voyager electronic packaging design meets the requirements herein. The packaging design is compatible with the applicable electronic requirements specified. The design meets the spacecraft launch and flight environmental requirements. Bench handling, testing, adjustment, repair, shipping, and modification considerations are included. The packaging design employed meets the requirements for long-life operation and survival in the space environment. Access to the various portions of the equipment that may require adjustment, repair, or modification is provided.

5.4.1 System

a. General

Spacecraft electronic equipment is designed to contain removable and replaceable electronic assemblies. These assemblies are of standardized size and shape to provide for system design flexibility.

b. Arrangement Considerations

The following considerations are applied in the arrangement of subsystems within the electronic assemblies:

a) Where possible a subsystem is confined to a single assembly for ease of handling, checkout, and qualification testing.

b) Subsystems with a large number of interconnections between them are in close proximity to reduce cable lengths.

c) Subsystems with high heat dissipation are uniformly distributed to aid in achieving spacecraft temperature control within specified limits.

d) Location of subsystems aids in achieving spacecraft center of mass within the specified limits.

e) Special equipment and sensor location requirements are met.

5.4.2 Assembly

Subassembly structures are rigidly fastened to the assembly structures and assembly structures are fastened to the spacecraft structure. Details are given in VS-4-550.

a. Geometrical

Assemblies allow standardized subassemblies to be interchanged in any electronic assembly.

b. Cabling and Connectors

Cabling within an assembly is secured to the chassis and does not allow flexing of the wires during installation and testing. Plug-in connectors are used and mounted such that straight and free engagement of contacts is assured. Indexing is provided to prevent equipment damage due to incorrect connection.

c. Thermal

Consistent with special equipment requirements, electronic assemblies are fastened to provide conductive heat paths to the spacecraft. Any subassembly within an assembly is capable of serving as a heat path or sink for heat loads generated within other subassemblies. Adjacent subassemblies utilize radiative and conductive heat transfer between themselves to the maximum extent consistent with other requirements. The assembly chassis provides a surface suitable for application of required temperature control finishes. Surface flatness and the number of fasteners used is compatible with temperature control requirements.

5.4.3 Subassembly

Subassembly profile is standardized to facilitate location flexibility within the assembly unless restricted by electronic design. To optimize design and to reduce interconnections, more than one profile standard is utilized. Subassembly thickness is varied as required to suit particular equipment design requirements and the packaging technique used.

a. Techniques

Packaging techniques are standardized utilizing as few techniques as necessary to design spacecraft electronic equipment. Only qualified processes and techniques are used.

b. Thermal

The subassembly packaging design permits the component parts to operate within the temperature limits for which they have been qualified. The design goal is to provide a minimum thermal stress to the component parts.

c. Structural

The subassembly structures are of adequate stiffness to insure that component fragility levels are not exceeded and that components, modules, and interconnecting devices are not damaged from deflections caused by shock and vibration. The structures are capable of carrying the spacecraft loads to which they may be subjected.

d. High-Voltage Protection

All equipment containing voltages in excess of 250 volts has special provisions for protection from corona or arcing.

5.5 Pyrotechnics

Pyrotechnics consist of electroexplosive devices and associated firing circuitry.

5.5.1 Range Safety

Electroexplosive devices (EED), associated system wiring and firing circuitry, conform to AFETRP 80-2, General Range Safety Plan, Volume I, and associated Appendix A.

5.5.2 Standardization

Consideration will be given to the use of standard squib envelope and match head configurations for all EED operations.

5.5.3 Electroexplosive Devices Design Criteria

Design criteria for electroexplosive devices follows:

a) All squibs contain redundant (i. e., two) bridgewires.

b) EED's utilize connector-type squibs (i. e., not utilize pigtails).

c) Squib bodies are of one-piece construction (i. e., the connector receptacle must be an integral part of the squib body).

d) All EED's utilize 1-amp/1-watt no-fire squibs (i. e., 1 amp/1watt applied to each of the two bridge circuits simultaneously).

e) Squibs are designed to provide continuous circumferential shielding between cable and device and to ensure that the shield circuit is completed before contact is made with the bridge pins.

f) Exploding bridgewire devices are not utilized.

g) Devices are nonventing and nonrupturing and provide positive gas containment.

h) EED's are nondetonating (e. g., the use of materials such as RDX is avoided).

i) All squibs are capable of withstanding static discharge of 25 kv from a 500-picofarad capacitor applied between pins, or between pins and case, at all pressures.

5.5.4 Firing Circuitry Design Criteria

Design criteria for firing circuitry follows:

a) Redundant firing circuitry is utilized.

b) Solid-state switches are used. No electromechanical relays are used.

c) The number of electrical components is minimized.

d) Capacitor discharge circuitry utilizing spacecraft primary power is preferred over other power sources.

e) The design considers as probable the instantaneous and permanent electrical shorting of each and every squib upon firing.

6. INTERFACES

6.1 Launch Vehicle Interface

The interface with the launch vehicle system is described in VS-3-130.

6.2 Launch Operations Interface

The total interface between the spacecraft and the launch operations system (LOS) is comprised of those characteristics of the spacecraft that affect the LOS and vice versa. Prelaunch operations are to be as described in the launch operations plan of Volume 3.

6.2.3 Hardware Interfaces

The hardware interfaces between the space vehicle and the launch operations system are to be specified at a later date. The usual mechanical, electrical and RF interfaces will exist, including propellant and purge gas transfer lines, air conditioning ducts, work platforms, umbilical and test cables, and telemetry checkout links (hardline or RF).

The capability for retropropellant loading or unloading while mated to the launch vehicle is not required for 1971, although such a requirement may exist for later missions.

6.3 Capsule Interface

The interface with the flight capsule is described in VS-3-140.

6.4 Deep Space Network Interface

The spacecraft telecommunications system is designed to be compatible with the existing and planned DSN capability as discussed in Reference 3. Additional interface characteristics are given in VS-4-310.

6.5 Science Payload Interface

The interface with the spacecraft science payload is described in VS-3-180 with additional information given in VS-4-210.

REFERENCES

1. JPL Project Document No. 45, V-MA-004-001-14-03, "Preliminary Voyager 1971 Mission Specification," 1 May 1965

2. JPL Project Document No. 46, V-MA-004-002-14-03, "Voyager 1971 Mission Guidelines," 1 May 1965

3. JPL Engineering Planning Document 283, "Deep Space Network, A Planning Capability for the 1965 - 1980 Period," W.H. Bayley, July 1965

4. TRW Voyager Spacecraft Design Document VS-1-110, "Mission Objectives and Design Criteria"

5. TRW Voyager Spacecraft Design Document VS-3-111, "Spacecraft Components Design Parameters"

6. TRW Voyager Spacecraft Design Document VS-3-112, "Weights and Mass Properties"

7. TRW Voyager Spacecraft Design Document VS-3-120, "Reliability Design Objectives".

8. TRW Voyager Spacecraft Design Document VS-3-140, "Spacecraft - Capsule Interface"

9. TRW Voyager Spacecraft Design Document VS-3-180, "Spacecraft Science Subsystem Integration"

10. TRW Voyager Spacecraft Design Document VS-4-210, "Spacecraft Science Payload"

11. TRW Voyager Spacecraft Design Document VS-4-310, "Communications System"

12. TRW Voyager Spacecraft Design Document VS-4-550, "Electronic Equipment Packaging"

13. AFETRP 80-2, "General Range Safety Plan," Volume I and Associated Appendix A

MANEUVER AND ACCURACY DATA

VS-3-102

CONTENTS

1. SCOPE

This document presents the functional, propulsive, and accuracy requirements on the spacecraft associated with the midcourse trajectory corrections, the capsule vehicle separation, the deboost into Mars orbit, and an orbital trim maneuver if required.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

JPL

V-MA-004-001-14-03	Preliminary Voyager 1971 Mission Specification
V-MA-004-002-14-03	Voyager 1971 Mission Guidelines

TRW 1971 Voyager Spacecraft Design Documents

VS-1-110	Mission Objectives and Design Criteria
VS-2-110	Design Characteristics and Restraints
VS-3-104	Flight Sequence

3. MIDCOURSE TRAJECTORY CORRECTIONS

3.1 Number

The spacecraft has the capability of providing at least four trajectory correction maneuvers with the midcourse propulsion subsystem. The number of planned correction maneuvers during the interplanetary trajectory will be either two or three.

3.2 Times

The first midcourse maneuver will be performed between 1 and 10 days after injection.

If the planned number of interplanetary corrections is two, the second maneuver will normally occur between the midpoint and the three-quarter point (in time) of the interplanetary cruise phase.

If the planned number of interplanetary corrections is three, the second maneuver will normally occur during the first half (in time) of the interplanetary cruise phase, and the third maneuver during the last one-third.

The fourth midcourse trajectory correction maneuver is available as a backup of the two or three which are planned. It also may be used for the orbit trim maneuver, if such a maneuver is required.

3.3 Propulsive Requirements

A sufficient total midcourse velocity allocation is provided to correct trajectory dispersions and perform any required trajectory biasing (to satisfy the planetary quarantine constraint) with a probability of 0.99. Since the midcourse propellant supply is separate from the propellant for the orbital deboost maneuver, this has been interpreted as a requirement that midcourse propellant be available for velocity increments totalling 75 meters per second. (This refers to velocity increments of the planetary vehicle, with approximate gross weight of 7800 pounds.)

It is estimated that a 40 meter per second velocity increment is adequate to carry out all required transit trajectory corrections (including correction for the random errors arising from Centaur injection guidance dispersions) with a probability of 0.99. The additional capability provides for possible nonrandom injection errors or, in their absence, for a Mars orbit trim correction.

3.4 Velocity Increment Accuracy

The timing of the midcourse correction is to be within 10 minutes (3σ) of the time selected by the mission operations system.

The pointing accuracy for the velocity increment achieved in a midcourse correction is to be 30 milliradians (3σ) about each of two defining axes. The magnitude of the velocity increment has two error sources, a proportional error and a nonproportional shutoff error. The proportional error is to be no greater than 3 per cent (3σ). The nonproportional error is to be no greater than 0.01 meter/second (3σ).

3.5 Sequence of Operations

The detailed sequence of operations for a midcourse trajectory correction is given in VS-3-104. In general, the sequence accomodates the following features:

a) Transmission of quantitative commands from the ground to the spacecraft and storage in appropriate spacecraft registers.

b) Re-pointing one of the spacecraft communications antennas to an orientation such that when the spacecraft is placed in the maneuver attitude, that antenna will be able to communicate with the earth.

c) Transmission from the spacecraft to the earth (by means of a second antenna) to verify the gimbal angles of the antenna.

d) Transmission from the earth to the spacecraft to enable the maneuver sequence.

e) Orientation of the planetary vehicle into the maneuver attitude.

f) Verification of achieving the maneuver attitude (transmitted to earth by the first antenna).

g) Transmission of the command from earth to spacecraft to enable the propulsive maneuver.

h) Propulsion start by on-board command at selected time.

i) Propulsion end by on-board command at selected time.

j) Reorientation of the spacecraft to cruise attitude.

4. CAPSULE VEHICLE SEPARATION

4.1 General

The separation of the capsule vehicle from the flight spacecraft will take place between 1 and 5 days before the spacecraft encounters Mars. The exact time at which this maneuver is to take place and the exact attitude which the spacecraft must hold when the separation is initiated will depend on late guidance up to several days before the maneuver is to take place, and therefore will be available for transmission to the spacecraft only several days in advance of the maneuver. However, it is possible that approximate times and pointing directions will be available with sufficient validity that they can be stored for an on-board commanded backup mode prior to launch.

4.2 Accuracy

The timing of the capsule vehicle separation event is to take place within 10 minutes (3σ) of the prescribed time.

The flight spacecraft achieves the prescribed attitude at the time the capsule vehicle is separated to within 0.75 degree (3σ) about each of two defining axes.

4.3 Sequence of Operations

Details are given in VS-3-104. In general, the sequence accommodates the following features:

a) Transmission of quantitative commands from the ground to the spacecraft, and storage in appropriate spacecraft registers.

b) Transmission of the command from the spacecraft to the capsule to jettison the biological barrier as required to allow separation of the capsule vehicle.

c) Transmission to earth of verification of biological barrier jettison.

d) Re-pointing one of the spacecraft communications antennas to an orientation such that when the spacecraft is placed in the maneuver attitude, that antenna will be able to communicate with the earth.

e) Transmission from the spacecraft to the earth (by means of a second antenna) to verify the gimbal angles of the antenna.

f) Transmission from the earth to the spacecraft to enable the maneuver sequence.

g) Orientation of the planetary vehicle into the capsule separation attitude.

h) Verification of achieving capsule separation attitude (transmitted to earth by the first antenna).

i) Transmission of a command from the earth to the flight spacecraft to enable the separation.

j) Transmitting the command for separation from the spacecraft to the flight capsule.

k) Detection and transmission to earth to verify that capsule vehicle separation has occurred.

l) Execution of a spacecraft evasive maneuver, by means of a small cold gas supply provided for this purpose. This evasive maneuver is transverse to the roll axis, and removes the spacecraft from the flight path to be taken by the capsule vehicle.

m) Reorientation of the flight spacecraft to cruise attitude.

5. DEBOOST INTO ORBIT

5.1 General

The use of the flight spacecraft retropropulsion system for inserting the spacecraft into an orbit about Mars can be programmed at the time of launch with sufficient accuracy to be valid for the mission. However, it is desirable that the actual point of closest approach to Mars and the time at which this event takes place as estimated from tracking form the basis for the maneuver. Therefore, provision is made for up-dating the desired attitude the spacecraft is to hold during the retropropulsion maneuver and the time of execution of the maneuver. Because a fixed-impulse solid engine is employed no provision is made for controlling the magnitude of the velocity increment. As the burning time is less than 2 minutes, a constant (celestial) flight spacecraft attitude is maintained during the firing of the engine.

5.2 Accuracy

The time of execution of the orbital insertion maneuver is to be within 15 seconds (3σ) of the time selected by the Mission Operations System.

The attitude of the spacecraft during the firing of the retropropulsion motor is to be held so that the thrust vector remains within 4.5 degrees (3σ) of the desired direction.

5.3 Sequence of Operations

The sequence of operations is given in detail in VS-3-104. In general, it is similar to the sequence given in 3.5 for midcourse trajectory corrections.

6. ORBIT TRIM

6.1 General

The capability of conducting an orbital trim maneuver using the midcourse propulsion system is provided.

For random Centaur injection errors at a 99 per cent probability, less than 40 meters per second midcourse velocity increment will be required for trajectory correction.

As the system is sized for a total capability of 75 meters per second, 35 meters per second (equivalent to 100 meters per second after capsule vehicle separation) is available at a 99 per cent probability for an orbit trim maneuver.

6.2 Accuracy

The timing of the orbit trim maneuver is to be accurate to within 15 seconds (3σ) of the time selected by the Mission Operations System.

The accuracy requirements for the orbit trim maneuver in regard to pointing and the magnitude of the velocity correction are the same as for the midcourse trajectory correction given in 3.4.

6.3 Sequence of Operations

The sequence of operations for the orbital trim maneuver is the same as for midcourse trajectory corrections given in 3.5.

FLIGHT SEQUENCE

VS-3-104

CONTENTS

1. SCOPE

This document covers the sequence of operations performed by the spacecraft from the period immediately preceding launch until completion of the mission.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-2-110 Design Characteristics and Constraints

3. GENERAL REQUIREMENT

The sequence of operations is compatible with the general mission profile in VS-2-110.

4. SEQUENCE OF EVENTS

The tabulation of the flight sequence of events from the Voyager 1971 mission is organized in Table 1 in the following manner.

Column 1 - Events denotes the events which take place and in some cases subevents as known or assumed.

Column 2 - Time defines, to the detail possible, the time at which these events occur. Various times are designated by abbreviation, such as T = time of liftoff, E = Mars encounter, etc. These are all relatable to time from liftoff (T) where necessary.

Column 3 - Source denotes the source of the signal which initiates the particular event.

Column 5 - Backup denotes the source of backup event signals where they have been assumed to be required.

Table 1. 1971 Voyager Reference Configuration—Flight Sequence

Event	Time	Source	Destination	Backup	Remarks
1. Set launch configuration	T -	LOS	Spacecraft		LOS = Launch Operations System Establishes all spacecraft systems in launch configuration.
2. 1st stage ignition and liftoff	T	LOS	Launch vehicle		T is time of liftoff
3. 1st stage cutoff	T + 153 sec	1st stage	1st stage engine		
4. 2nd stage ignition	T + 158 sec	2nd stage	2nd stage engine		
5. Shroud jettison	T + 196 sec	2nd stage	Shroud separation system		
6. 2nd stage cutoff	T + 608 sec	2nd stage	2nd stage engine		
7. 3rd stage ignition	T + 617 sec	3rd stage	3rd stage engine		
8. 3rd stage cutoff: enter coast period					
9. 3rd stage reignition	T + 1176 sec	3rd stage	3rd stage engine		
10. 3rd stage cutoff	T + 1516 sec	3rd stage	3rd stage engine		
11. Spacecraft separation	S (T + 1550 sec)	3rd stage	Spacecraft separation ordnance		S is time of separation initiation by 3rd stage sequencer CS&C = central sequencing and command subsystem
a) Arm spacecraft ordnance bus					a) Armed through separation switches
b) Enable CS&C					b) Enabled through separation switches
c) Start appendage deployment time					c) Deployment delay timed by CS&C
d) Start stabilization and control subsystem					d) Enabled through separation switches. Stabilization and control starts in gyro mode.
12. Appendage deployment	S + 15 sec	CS&C	Deploy initiators	Ground command	
a) High gain antenna					
b) Medium gain antenna					
c) Science appendage(s)					
13. Turn cruise science payload on	S + 20 sec	CS&C	Science DAE	Ground command	Cruise science assumed off during boost and deployment
14. Enter data mode 3	S + 20 sec	CS&C	Communication data handling subsystem	Ground command	Data mode 3 contains cruise science, spacecraft engineering and transit capsule data.
15. Begin sun orientation		CS&C	SCS	Ground command	SCS = Stabilization and control subsystem
a) Sun acquisition complete	S + 50 min	Precise sun sensor	CS&C		
16. Apply spin roll rate bias	S + 50 min	CS&C	SCS	Ground command	If magnetometer calibration is required
a) Apply search roll rate bias	S + 10 hr	CS&C	SCS	Ground command	
b) Turn on Canopus sensor	S + 10 hr	CS&C	SCS	Ground command	
c) Canopus acquisition complete	S + 10 hr, 20 min	Canopus sensor			c) Completion of event automatically places SCS in cruise control mode.
d) Turn off SCS gyros	S + 10 hr, 20 min	CS&C	SCS		
17. Orient high gain antenna	S + 12 hr	CS&C	Antenna servos		
18. Switch down link communications to high gain antenna	S + 12.5 hr	CS&C	Communication system	Ground command	
19. Turn on 20-watt transmitter; turn off 1 watt transmitter	S + 15 hr	CS&C	Communication system	Ground command	
20. Begin midcourse maneuver preparations	M - 2 hr				M is maneuver start time
a) Turn on SCS gyros		Ground command	SCS		
b) Set telemetry mode 1		Ground command	Data handling system		
c) Enter roll turn magnitude and polarity data		Ground command	SCS		Quantitative ground command
d) Enter pitch turn magnitude polarity data		Ground command	SCS		Quantitative ground command
e) Enter engine start and stop data		Ground command	CS&C		Quantitative ground command
f) Enter high gain antenna pointing data		Ground command	Antenna electronics		Quantitative ground command
g) Verify command data reception		CS&C	Communication system (telemetry)		Ground verification from telemetry data
h) Switch down link communications to low gain					
i) Point high gain antenna		CS&C	Antenna servos		
j) Verify high gain antenna angles		Antenna pickoffs	Telemetry		Ground verification from telemetry data

Table 1. 1971 Voyager Reference Configuration— Flight Sequence (Continued)

Event	Time	Source	Destination	Backup	Remarks
21. Begin maneuver sequence	T + 1 to 10 days (M)	Ground command	CS&C		
a) Start roll turn		CS&C	SCS		
b) Stop roll turn		CS&C	SCS		
c) Start pitch turn		CS&C	SCS		
d) Stop pitch turn		CS&C	SCS		d) Under certain conditions a second roll turn can be required after completing this event
e) Switch down-link communications to high gain antenna		CS&C	Communications	Ground command	
f) Verify spacecraft attitude		Telemetry	Ground station		f) Spacecraft attitude verified by gyro telemetry data and signal strength from high gain antenna
g) Remove engine inhibit		Ground command	CS&C		g) Ground controlled enable signal for midcourse engine ignition
h) Start midcourse engine		CS&C	Engine controls	Ground command	
i) Stop midcourse engine		CS&C	Engine controls	Ground command	
22. Post maneuver sequence	M + 5 min				
a) Reset spacecraft to cruise mode		CS&C	SCS	Ground command	
b) Reorient high gain antenna		CS&C	Antenna servos		
c) Turn off SCS gyros		CS&C	SCS		
23. Additional midcourse corrections	As required				Repeat of event sequences 20, 21, and 22. Requirements for additional corrections dependent upon accuracy of previous correction. Spacecraft has capability of 3 additional midcourse corrections.
24. Command science payload	As required	CS&C or ground command			Requirement determined by science payload.
25. Command Canopus angle	As required	CS&C	Canopus sensor	Ground command	
26. Command medium and high gain antenna angles	As required	CS&C	Antenna servos	Ground command	
27. Terminal guidance sensing of Mars	E -				Requirements not firm for on-board terminal guidance sensing. See VS-S-110, 3.5 E denotes Mars encounter time.
28. Final midcourse correction	E -				
29. Capsule system checkout	CVS - 4 hours	CS&C	Flight capsule	Ground command	Capsule vehicle systems checkout prior to sterilization container cover jettison. CVS = event time for capsule vehicle separation.
30. Separate capsule canister cover	CVS - 2.5 hr	Ground command	CS&C		
31. Begin capsule vehicle separation preparations	CVS - 2 hr				
a) Enter spacecraft attitude data		Ground command	CS&C		
b) Enter capsule vehicle separation time		Ground command	CS&C		
c) Enter high gain antenna pointing data		Ground command	CS&C		
d) Verify commands a), b) and c)					d) See event 20g
e) Switch down-link communications to medium gain antenna		CS&C	Communication system	Ground command	
f) Point high gain antenna		CS&C	Antenna servos		
g) Verify high gain antenna angles					g) See event 20j
32. Begin separation sequence	CVS	Ground command	CS&C		Ground command enable command
a) Orient spacecraft to separation attitude					a) Repeat of applicable portion of event sequence 21
b) Switch down-link communications to high gain antenna, enter receiver mode 2					
c) Verify spacecraft attitude					c) See event 21f
d) Remove separation inhibit		Ground command	CS&C		
e) Turn on VHF receivers		CS&C	Communication system	Ground command	
f) Switch to telemetry mode 1		CS&C	Data handling system	Ground command	
g) Separate and inject lander from spacecraft		CS&C	Flight capsule	Ground command	
33. Execute spacecraft evasive maneuver	CVS +	CS&C	Evasive Maneuver Propulsion	Ground command	See explanatory notes for discussion
34. Lander propulsion start	CVS +	Capsule vehicle			On-board capsule vehicle function. No further communications with capsule vehicle after CVS

Table 1. 1971 Voyager Reference Configuration—Flight Sequence (Continued)

Event	Time	Source	Destination	Backup	Remarks
35. Reorient spacecraft to cruise attitude		CS&C		SCS	Repeat of applicable portions of event sequences 15 and 16
a) Reorient high gain antenna		CS&C	Antenna servos		a) Event 31 can precede event 30 under some conditions
b) Switch down-link communications to high antenna		CS&C	Communication system	Ground command	
36. Receive and relay capsule data		VHF link	TLM		
37. Jettison remaining portions of capsule adapter and canister		Ground command	Separation ordnance		May occur prior to this time.
38. Prepare for orbit injection	E - 3 hr				E denotes time of Mars encounter on point of closest approach
a) Enter deboost attitude data		Ground command	CS&C		Quantitative ground command
b) Enter deboost motor ignition time		Ground command	CS&C		Quantitative ground command
c) Enter high gain antenna pointing data		Ground command	CS&C		
d) Verify commands a), b) and c)		CS&C	Telemetry		
e) Switch down-link communications to medium gain antenna		CS&C	Communication system	Ground command	
f) Point high gain antenna		CS&C	Antenna servos		
g) Verify high gain antenna pointing angles		Antenna pick-offs	Telemetry		
h) Start SCS gyros		CS&C	SCS		
39. Lander encounters Mars surface	E - 95 min				
a) Turn off VHF receivers					
40. Orient spacecraft to deboost attitude	E - 75 min	CS&C	SCS	Ground command	
a) Switch down-link communication to high gain antenna		CS&C	Communication system	Ground command	
b) Verify spacecraft attitude					See event 21f
c) Time allowance for ground data analysis	40b + 30 min				
d) Remove deboost motor inhibit	E - 20 min	Ground command	CS&C		
41. Ignite deboost motor	E - 45 sec	CS&C	Deboost motor igniter	Ground command	
42. Encounter	E (T + 177 days)				
43. Retropropulsion thrust termination	E + 45 sec				
44. Commence reacquisition of sun/Canopus references	E + 20 min	CS&C	SCS	Ground command	Repeat of events 15 and 16
a) Return to cruise mode control					
b) Reorient high gain antenna					
c) Switch down-link communications to high gain antenna		CS&C	Communication system	Ground command	
45. Switch POP to Mars scanner control		CS&C	POP controls	Ground command	POP devotes planet oriented package
46. Photograph Mars and transmit data to earth					First orbits. POP operating on nominal preset program.
47. Track spacecraft to determine orbit					DSIF tracking as required to establish orbit.
48. Enter POP photograph program data		Ground command	CS&C/Science DAE		Time dependent upon data from event 43 and establishment of POP program.
49. Start POP program mode		Ground command	CS&C/Science DAE		Operating on mode established in event 44.
50. Relay lander data to earth					As required
51. Command Canopus sensor angles	As required	Ground command	Canopus sensor		
52. Command antenna angles	As required	Ground command	Antenna servos		
53. Repeat 47 through 49	As required				

EXPLANATORY NOTES

Event 1 - Set Launch Configuration. Some time before liftoff it will be necessary to establish the over-all spacecraft in the powered flight equipment operating configuration. Just prior to liftoff, the spacecraft umbilical (with connections via the launch vehicle to the launch complex equipment) will be disconnected by remote control and this disconnection verified.

Events 2 through 10. These events concern the launch vehicle events from liftoff to spacecraft separation and are included for purposes of completeness. The times associated with these events are nominal and subject to variation.

Event 11 - Spacecraft Separation. It is assumed that spacecraft separation is initiated by the launch vehicle and will not require backup from the spacecraft.

The assumed implementation also includes:

1) Arming the spacecraft ordnance bus by separation

2) The requirement to deploy appendages after separation but to delay deployment until the spacecraft has cleared the third stage to prevent mechanical interference during deployment.

3) No requirement for sequencing functions from the CS&C prior to separation; hence the enabling of the CS&C by separation.

4) As it will be some time from separation to sun visibility, the stabilization and control system (SCS) will be enabled by separation, start in the gyro control mode and immediately begin nulling the spacecraft tumbling rates caused by the separation transient.

Event 12 - Appendage Deployment. It has been assumed that the medium and high gain antennas and the science appendages will be deployed at this time. Although some of the appendages will not be required until later in the mission, all are assumed to be deployed here for increased reliability of deployment.

<u>Event 13 - Turn Cruise Science Payload On</u>. This assumes, of course, that there is no science information required prior to this time. Science payload details will be the determining factor.

<u>Event 21 - Begin Maneuver Sequence.</u> The sub-events in Event 21 involve a number of assumptions:

1) The high gain antenna will be pointed to the angles that will assure its looking at earth in the midcourse correction attitude prior to the assumption of that attitude and the angles verified. This assumes cruise communications (down link) on either the low or the medium gain antenna during midcourse preparations.

2) A ground command enable signal will be issued prior to the advance of the sequence, the enable condition being based upon the verification of the high gain antenna angles and the maneuver data.

3) The desired midcourse correction impulse will be achieved by controlling the start and stop time of the engine firing as determined by ground calculation.

4) The attitude of the spacecraft will be verified prior to the ignition of the engine, and an enable must be issued from the ground prior to engine ignition.

5) Upon termination of engine thrust, the stabilization and control subsystem will automatically (under CS&C command) reassume the cruise attitude control mode.

<u>Event 23 - Additional Midcourse Corrections.</u> It is probable that additional midcourse corrections will be required to assure, in particular, proper injection of the capsule vehicle. In view of the fact that the need for additional corrections is predicted upon the results of previous corrections, it is not possible to make a prior determination of the number of corrections required. The spacecraft will be designed to have the capability for at least four midcourse corrections.

<u>Event 24</u>. It has been assumed that the science payload will require the command of calibration sequences either on-board from the CS&C or by ground command.

Events 25 and 26 - Due to the changing position of the canopus reference for attitude control and the spacecraft/earth direction for communications, the canopus sensor and the steerable spacecraft antennas will require changes in pointing directions with respect to the spacecraft body axes as the flight progresses. It is assumed that these corrections will be programmed on board with backup or corrective capability available by ground command.

Event 27 - Terminal Guidance Sensing of Mars. It is possible that on-board terminal guidance sensing of Mars will be required to assure the accuracy of subsequent mission events. Further study is necessary to determine the requirements.

Event 30 - Separate Capsule Canister Cover. This event breaks the sterilization barrier for the lander. It is assumed that this will be done only after preseparation capsule system checkout initiated by the spacecraft, and that it will be accomplished by ground command.

Event 33 - Execute Spacecraft Evasion Maneuver. To avoid the assumption, at this time, that the lander will perform an 180-degree turn after separation and prior to capsule vehicle propulsion start, it has been assumed that the spacecraft will place the capsule vehicle in the proper thrusting attitude. This being the case, the spacecraft, unless maneuvered further, will lie in the direction of lander thrust and a possible collision can result. To avoid this collision, it is assumed that an evasive maneuver will be made by the spacecraft, in this case a simple translation.

Event 35. Among several possibilities, it is assumed here that the spacecraft will reassume cruise attitude as soon as possible after capsule vehicle separation and will remain in this attitude, relaying capsule vehicle data to earth until capsule encounter with the surface of Mars.

Event 38. The timing of the orbit injection events is among the most critical for the entire mission. It is assumed that the preparations for the retropropulsion firing will be made prior to the lander data relay period.

Event 39 - Lander Encounters Mars Surface. The spacecraft will remain in cruise attitude for lander data relay until the lander encounters the surface of Mars. The spacecraft must now orient to retrothrust attitude.

Event 40. As in midcourse corrections during the flight, the spacecraft attitude is verified and a signal is sent from the ground to enable engine ignition. The timing of the event for the ignition of the retropropulsion subsystem is probably the most critical of the mission events, and a time margin for data analysis and adjustment has been allocated.

Events 38, 42 and 43. In the ideal case, the total retrothrust would be applied exactly at the encounter point. This not being possible, the retropropulsion burning time (approximately 90 seconds) has been adjusted to center the burn period at encounter. If orbit parameters other than minimum energy are desired, the burning period and deboost attitude can be adjusted as appropriate.

Event 44. After retropropulsion thrust has terminated, the spacecraft will reacquire the cruise attitude, i.e., sun/Canopus orientation.

Event 45, 46 and 47. The exact photographic schedule for orbiting photography cannot be established until the orbit has been determined accurately, which will take some time. During the period that orbit determination is being made, photography is assumed to occur under a stored program and with the platform under Mars scanner control. Although the operations are not as accurate as they will be later, this is preferable to waiting for accurate information before beginning photography.

Events 48 and 49. Data for an accurate photographic program can now be entered into the spacecraft by ground command. Once this has been accomplished the POP and photographic equipment are under the modified program control.

Event 50 - Relay Lander Data to Earth. As required, lander data can be received aboard the spacecraft and relayed to earth along with the photographic and other data. It will be necessary to turn on the VHF receivers which were turned off in event 39a).

Events 50 and 51. As the space geometry changes with time, it is necessary to change the canopus sensor and antenna angles to compensate for these changes.

Event 53. When it is desired to adjust the photographic program, it is possible by repeating events 48 and 49.

CONFIGURATION

VS-3-110

CONTENTS

1. SCOPE

This document provides details of the configuration for the 1971 Voyager flight spacecraft.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-3-112	Weights and Mass Properties
VS-3-130	Voyager Spacecraft - Launch Vehicle Interface
VS-3-140	Voyager Spacecraft - Capsule Interface

3. CONFIGURATION GUIDELINES AND SALIENT FEATURES

Guidelines and design principles embodied in the spacecraft configuration are summarized below by functional area, along with some of the salient design features.

3.1 Structure and Mechanical

a) Direct load paths are used to minimize induced loads and structural weight.

b) Structure is designed for multiple purposes to a maximum extent. For example, equipment mounting panels are used for thermal radiation and meteoroid protection. Also, the launch vehicle and capsule mechanical interfaces with the spacecraft bus serve both as field joints and as flight separation joints.

c) Masses are located close to the thrust axis and the center of mass to minimize induced moments and the moment of inertia of the spacecraft.

d) Structural geometry and materials are chosen to yield maximum strength to weight ratios.

e) The number and size of joints are minimized.

f) The structural surface area is minimized to reduce weight.

g) The number of deployed components and deployment extensions are minimized. Symmetrical arrangements are utilized for deployed items when such items are required.

h) Equipment is located close to structural elements to reduce the weight of bracketry and induced moments.

i) Deployable and articulated components are stowed during thrusting to minimize structural requirements.

j) The orientation of the solid retropropulsion motor toward the capsule requires that the remaining capsule adapter and canister must be jettisoned prior to firing.

3.2 Propulsion

a) The spacecraft center of mass is to remain nominally on the thrust axis during consumption of propellant.

b) Engines are located so as to provide point of thrust application for thrust vector control at a maximum axial distance from the spacecraft center of mass.

c) Propulsion systems and associated support structure are modularized and independent of the rest of the spacecraft for ease of installation and test.

d) Multiple tankage for liquid propellant components and pressurants are symmetrically located, with a single tank on the thrust axis as the preferred arrangement.

e) Positive expulsion is required for liquids.

f) The propulsion modules are located on the space vehicle centerline during booster powered flight.

g) The solid retropropulsion motor is oriented so that the plume radiates to the back side of the solar array.

h) The thrust chamber gimballing is not required for thrust vector control. Jet vanes and liquid injection are used.

i) The thrust levels are constrained to minimize acceleration body force (g force), consistent with other propulsion design considerations.

j) High density and high specific impulse propellants are used consistent with other spacecraft requirements.

k) Spherical tankage is utilized and pressure levels selected for minimum weight.

3.3 Thermal Control

a) Adequate thermal radiating area is provided with a maximum view of space.

b) All nonradiating areas are insulated. This controls heat flow into and out of the spacecraft so that adequate temperature control can be achieved by the louvers.

c) Total external surface area is minimized to reduce insulation weight and heat leaks.

d) The radiating louver panels are shaded from the sun or other heat sources as much as practical.

e) Maximum thermal interaction between internal components is provided. This is enhanced by a single equipment compartment arrangement.

f) Heat producing components are uniformly distributed about the spacecraft to a maximum extent consistent with cabling and packaging considerations.

g) Tankage is centrally located for isolation from the sun. Support structure design and internal surface thermal radiation characteristics are utilized to achieve thermal coupling of tankage to the spacecraft.

h) Low-conductance structural attachment fittings are utilized to limit heat input by conduction from propulsion units as a result of firings. High-temperature insulation is utilized on spacecraft surfaces having a large viewing angle to the solid motor exhaust plume.

i) Thermal radiation characteristics of appendages are controlled to minimize solar torque unbalance.

3.4 Stabilization and Control

a) Independent high pressure gaseous nitrogen systems are utilized for generating control torques during cruise. Redundancy is obtained by provision of two separate systems, each with a capability for carrying out the mission.

b) Length of all lines between valves and nozzles is equalized and nozzles are located as far as possible from center of mass.

c) Gas impingement of reaction control jets on spacecraft is minimized.

d) Sensors are located for unobstructed view angles and protected from exhaust contamination.

e) Pitch and yaw moments of inertia are equalized to the maximum extent, as well as pitch and yaw control torques.

f) Position control axes are approximately along spacecraft principal inertia axes to minimize cross coupling.

3.5 Electrical Power

a) Solar power is utilized with a nondeployable, fixed solar array.

b) The solar array is modularized into 6 identical structural panels with 2 electrical sections each.

c) The solar array, with the spacecraft in its cruise orientation, is normal to the sun line.

d) The solar array is located at the opposite end of the spacecraft and facing away from the solid rocket motor plume to minimize contamination and thermal effects. A low absorbtivity coating relative to plume radiation is utilized on the back side of the solar array.

e) The solar array is located and configured within the allowable spacecraft envelope so as to maximize the active area.

f) The solar array structure is utilized to support the planet oriented package, interplanetary science equipment antennas, and stabilization and control elements.

3.6 Telecommunications

a) A high gain parabolic antenna is utilized that is gimballed in two axes. It has an elliptical aperture to maximize the interception area within the allowable spacecraft envelope.

b) A 3-foot diameter single gimbal medium gain parabolic antenna is utilized as a backup for the high gain antenna.

c) A VHF antenna is utilized to provide the relay link capability for capsule-spacecraft-earth communications.

d) A low-gain wide-beam antenna is utilized for transmission during prelaunch and boost flight operation and to allow coverage when the spacecraft does not have its accurate attitude reference.

3.7 Science Support Equipment

a) No magnetic materials are to be utilized.

b) The body-fixed science experiment equipment is located for unobstructed viewing angles.

c) A planet oriented package with two-axis gimballing, provider controlled pointing for installed science instruments.

d) Support to science equipment is required for thermal control and electrical distribution. Shock, vibration, and exhaust contamination are minimized.

e) Protective covers for instruments are provided as required and direct or reflected sunlight is kept from sensor apertures as appropriate.

f) An extendable boom is utilized to position a magnetometer during the earth-Mars transit phase. A second non-deployable boom is also provided to mount a magnetometer for measurements during orbital operations around Mars.

4. PACKAGING CONSIDERATIONS

Some considerations relative to packaging as they affect the spacecraft layout and configuration are given below:

a) Excess thermal radiation panel mounting surface and compartment volume is provided. This represents a growth potential to accommodate 300 pounds of additional equipment.

b) Space is provided for cable routing. Highly interconnected components are located together. Power amplifiers are located close to the associated antennas.

c) Easy access to equipment is provided for installation, test and maintenance.

d) The connections to articulated or deployed items is minimized. Slip rings or gimbals with continuous rotation in one direction are avoided.

5. SPACECRAFT GEOMETRY

5.1 Spacecraft Envelope

The available spacecraft dynamic envelope is shown in Figure 1.

5.2 Spacecraft Coordinates

5.2.1 Station

A spacecraft station coordinate is defined in the axial direction so as to be measured positively in the direction away from the launch vehicle, with the spacecraft in the installed position. The zero station reference plane is at the spacecraft-launch vehicle field joint plane at Saturn station 2048. The spacecraft forward and aft directions correspond to increasing station and decreasing station, respectively.

5.2.2 Line of Direction

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

a) 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.

b) A line of direction parallel to the plane through the spacecraft centerline and the canopus sensor's nominal line of sight has zero clock angle. For any other 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 clock angle.

5.3 Spacecraft Orientation Axes

Right-hand orientation reference axes U, $\vec{V}$, $\vec{Z}$, in the spacecraft are defined as follows (see Figure 2):

a) The vector $\vec{Z}$ in along the spacecraft centerline 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)

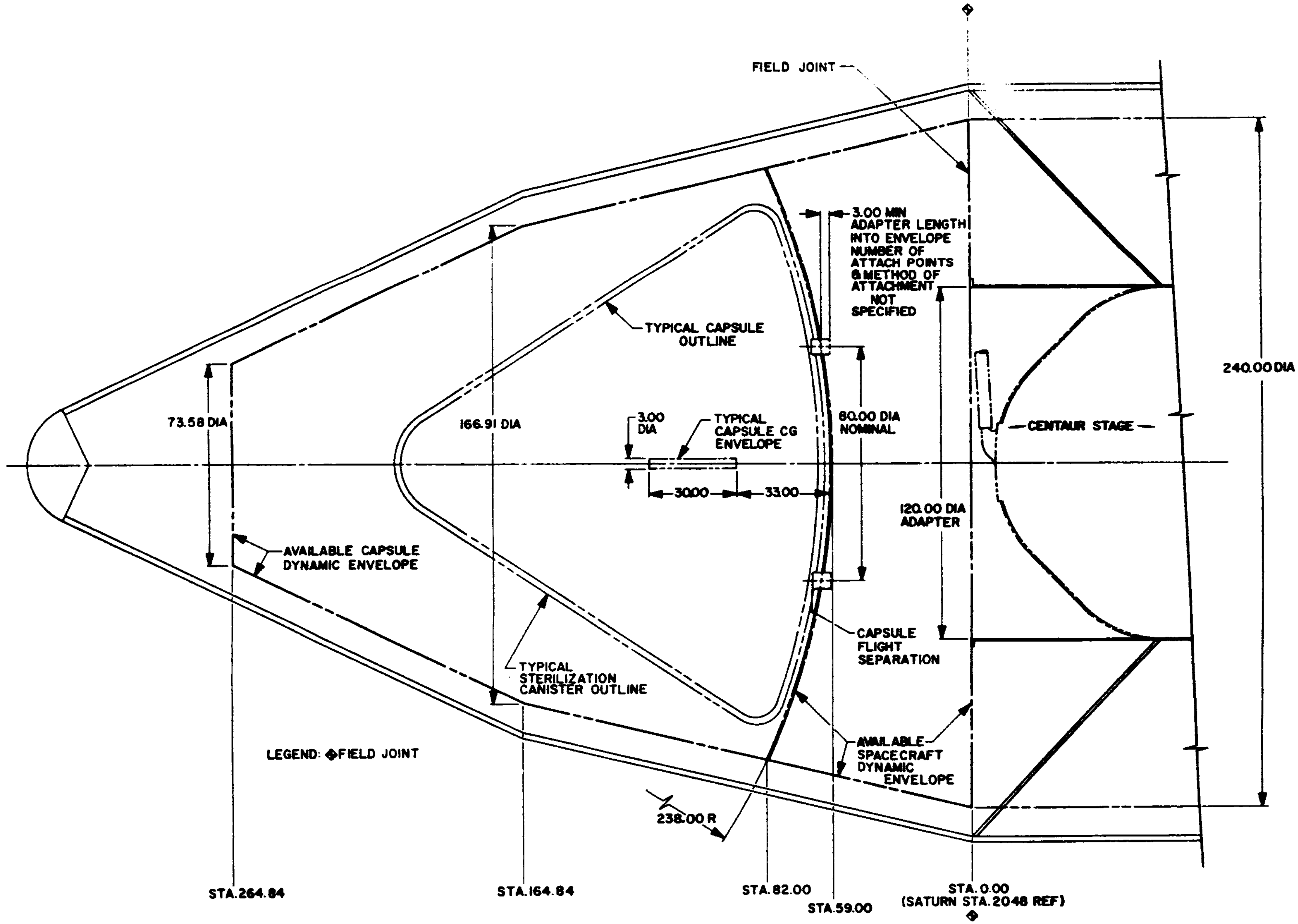
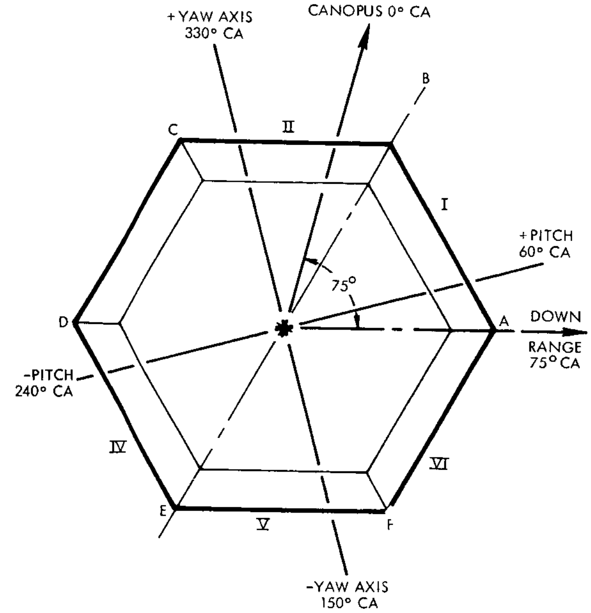


Figure 1. Planetary Vehicle Allowable Envelope



*CA DENOTES CLOCK ANGLE

NOTE:
BUS PLAN VIEW LOOKING AFT (Z AXIS OUT OF THE PAPER AND AWAY FROM THE SUN)

Figure 2. Spacecraft Reference Axes

5.4 Spacecraft Stabilization and Control Axes

Right-hand control axes $\vec{X}$, $\vec{Y}$, $\vec{Z}$ are defined with $\vec{Z}$ as in 5.3 and $\vec{X}$, $\vec{Y}$ corresponding to the pitch and yaw axes respectively as shown in Figure 2.

6. SPACECRAFT DRAWINGS

6.1 Configuration Drawings

Inboard and outboard profiles are given in Figure 3 and 4 respectively. An isometric view is given in Figure 5 and an exploded representation is shown in Figure 6.

6.2 Sensor Location

Spacecraft sensors and other configuration-sensitive components are located and oriented as shown in Figure 7. The spacecraft layout and configuration allows for an unobstructed field of view for each sensor as shown in the figure.

7. CENTER OF MASS AND INERTIAL PROPERTIES

Center of mass data and inertial properties for the spacecraft are presented in VS-3-112. Some associated considerations affecting spacecraft layout and configuration are given below:

a) The spacecraft is configured to achieve symmetry and to have its center of mass nominally on the center line.

b) Consumables are located on or in a symmetrical manner near the spacecraft centerline, which corresponds to the line of thrust for propulsive elements.

c) Allowance is made for a longitudinal and lateral center of mass shift when the capsule vehicle separates and when the remaining capsule adapter and canister is jettisoned.

8. INTERFACES

8.1 Launch Vehicle Interface

8.1.1 Mechanical Attachment

Mechanical attachment to the launch vehicle is described in 3.4 in VS-3-130.

8.1.2 Umbilical Connections

Umbilical connections to the launch vehicle are described in 3.5 of VS-3-130.

8.2 Capsule Interface

8.2.1 Mechanical Attachment

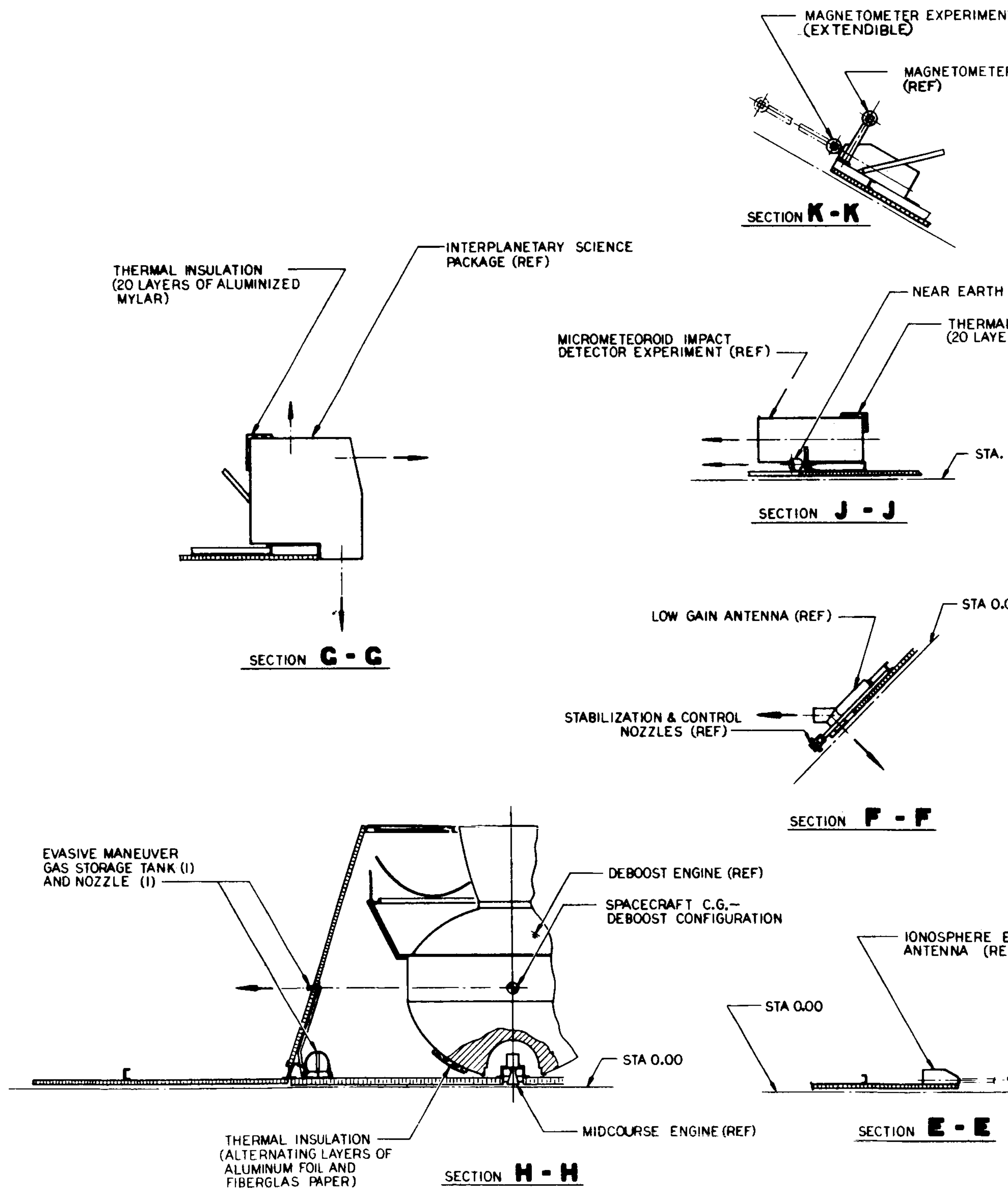
Mechanical attachment to the flight capsule is described in 3.3 of VS-3-140

8.2.2 Umbilical Connection

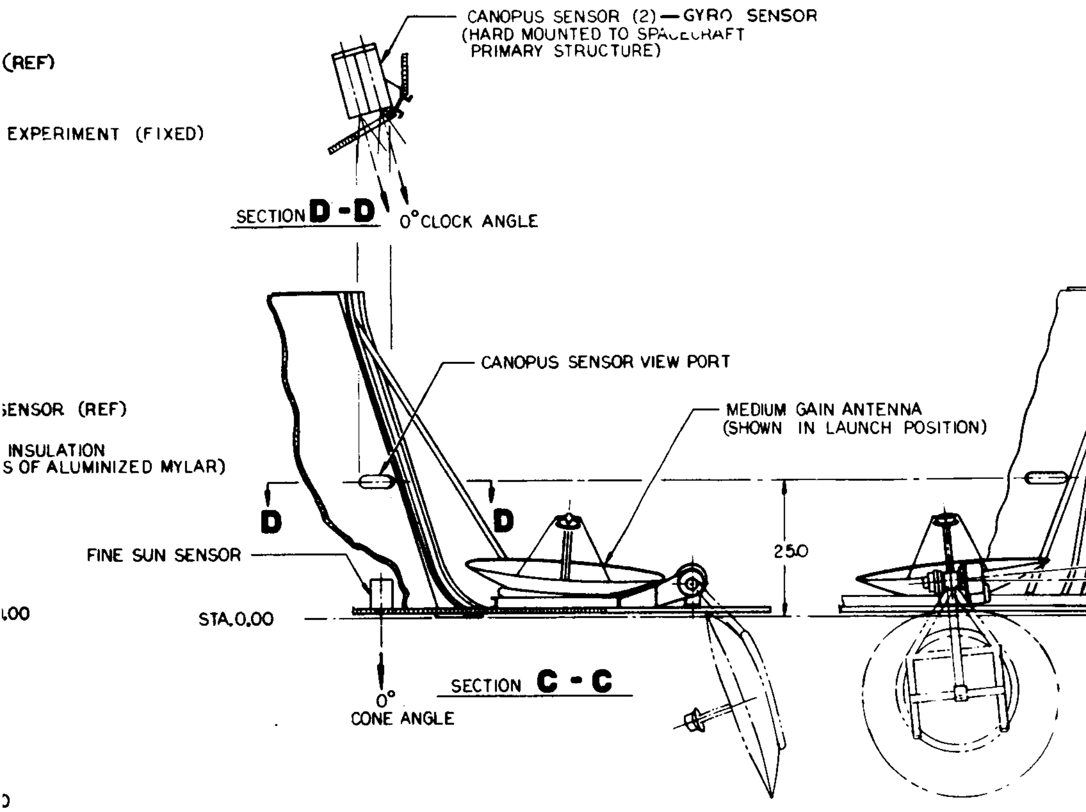
A spacecraft capsule umbilical connection will be provided as described in 3.4.1 of VS-3-140.

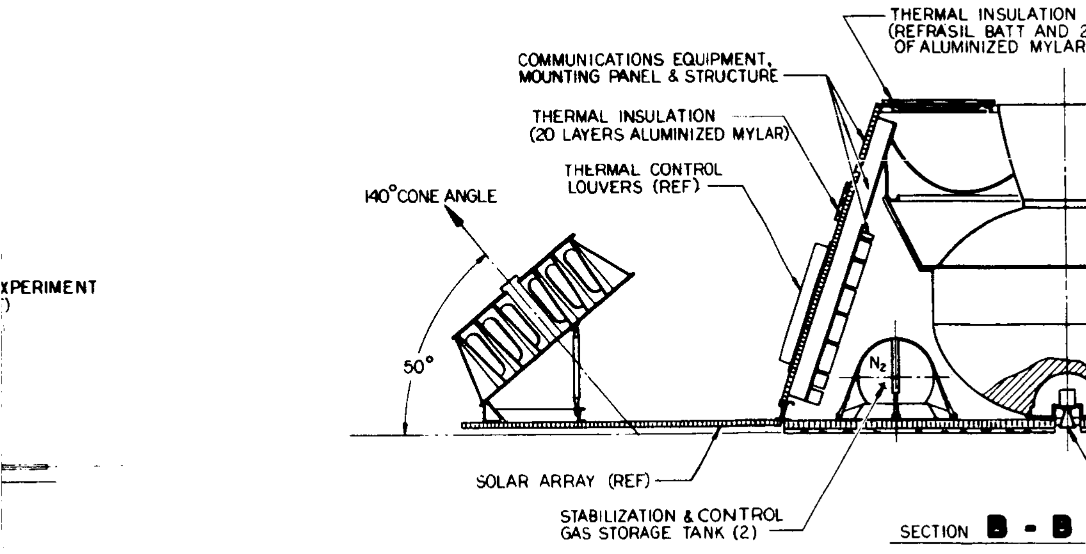
9. ALIGNMENT

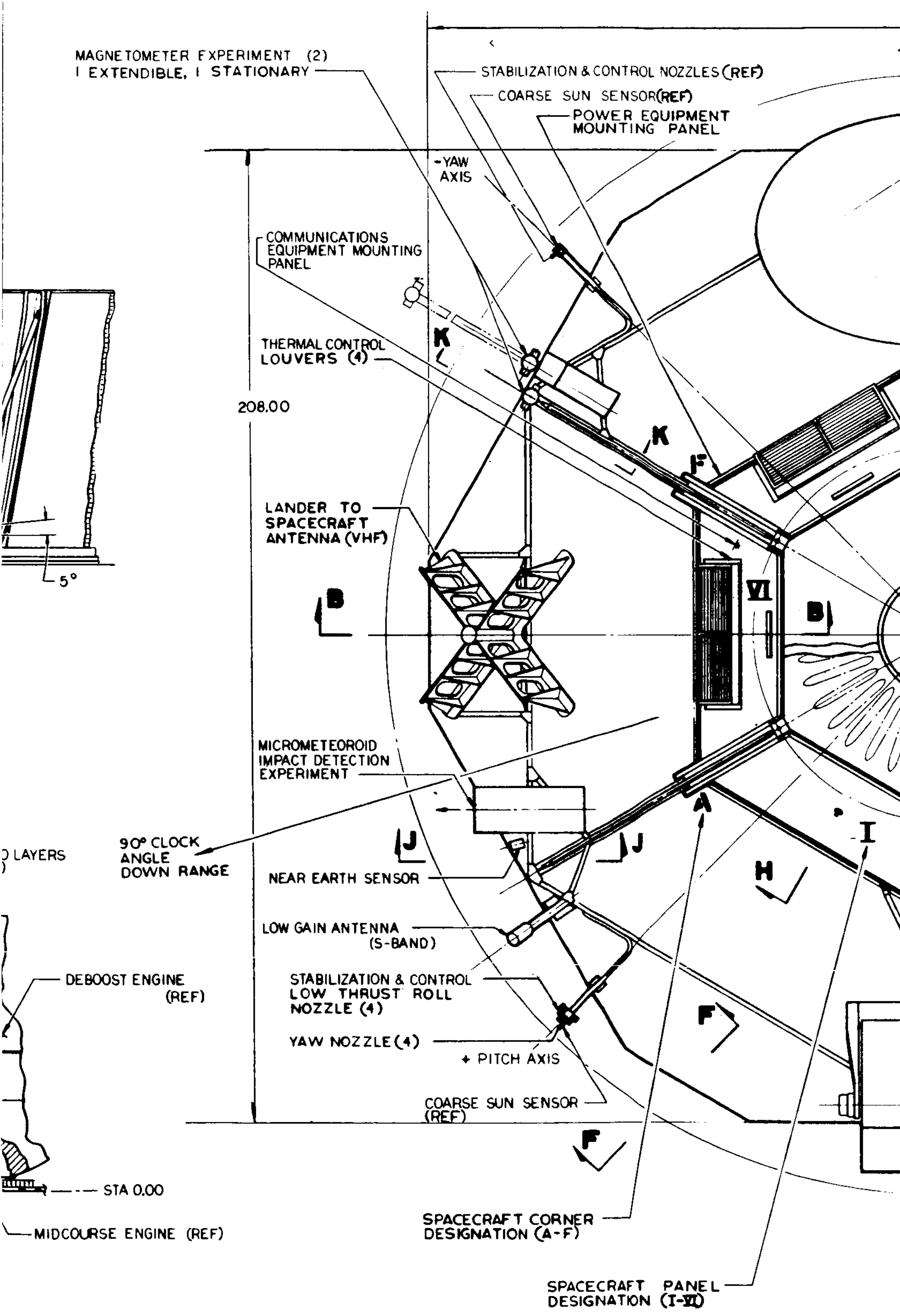
Spacecraft alignment considerations are presented in Volume 6, Appendix D.



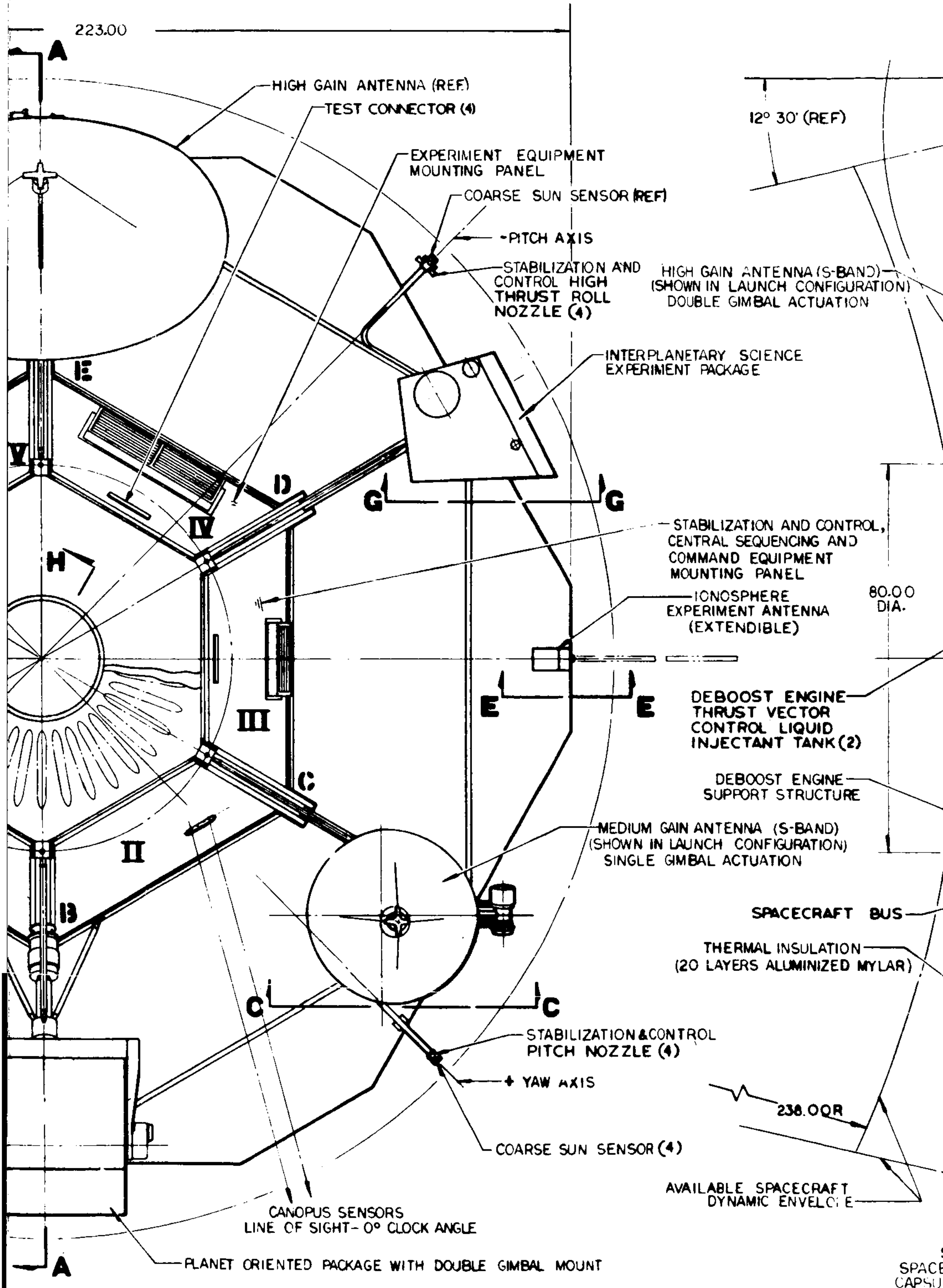
MAGNETOMETER EXPERIMEN
(EXTENDIBLE)
MAGNETOMETE
(REF)
SECTION K - K
INTERPLANETARY SCIENCE
PACKAGE (REF)
THERMAL INSULATION
(20 LAYERS OF ALUMINIZED
MYLAR)
SECTION G - G
NEAR EARTH
THERMA
(20 LAYE
MICROMETEOROID IMPACT
DETECTOR EXPERIMENT (REF)
STA.
SECTION J - J
LOW GAIN ANTENNA (REF)
STA 0.0
STABILIZATION & CONTROL
NOZZLES (REF)
SECTION F - F
EVASIVE MANEUVER
GAS STORAGE TANK (I)
AND NOZZLE (I)
DEBOOST ENGINE (REF)
SPACECRAFT C.G.-
DEBOOST CONFIGURATION
STA 0.00
MIDCOURSE ENGINE (REF)
THERMAL INSULATION
(ALTERNATING LAYERS OF
ALUMINUM FOIL AND
FIBERGLAS PAPER)
SECTION H - H
IONOSPHERE E
ANTENNA (RE
STA 0.00
SECTION E - E


CANOPUS SENSOR (2)—GYRO SENSOR
(HARD MOUNTED TO SPACECRAFT
PRIMARY STRUCTURE)
(REF)
EXPERIMENT (FIXED)
SECTION D-D
0° CLOCK ANGLE
CANOPUS SENSOR VIEW PORT
MEDIUM GAIN ANTENNA
(SHOWN IN LAUNCH POSITION)
SENSOR (REF)
INSULATION
S OF ALUMINIZED MYLAR)
D
D
FINE SUN SENSOR
250
.00
STA.0.00
0°
CONE ANGLE
SECTION C-C


THERMAL INSULATION
(REFRASIL BATT AND 2
OF ALUMINIZED MYLAR
COMMUNICATIONS EQUIPMENT,
MOUNTING PANEL & STRUCTURE
THERMAL INSULATION
(20 LAYERS ALUMINIZED MYLAR)
THERMAL CONTROL
LOUVERS (REF)
140° CONE ANGLE
XPERIMENT
50°
N_2
SOLAR ARRAY (REF)
STABILIZATION & CONTROL
GAS STORAGE TANK (2)
SECTION B-B

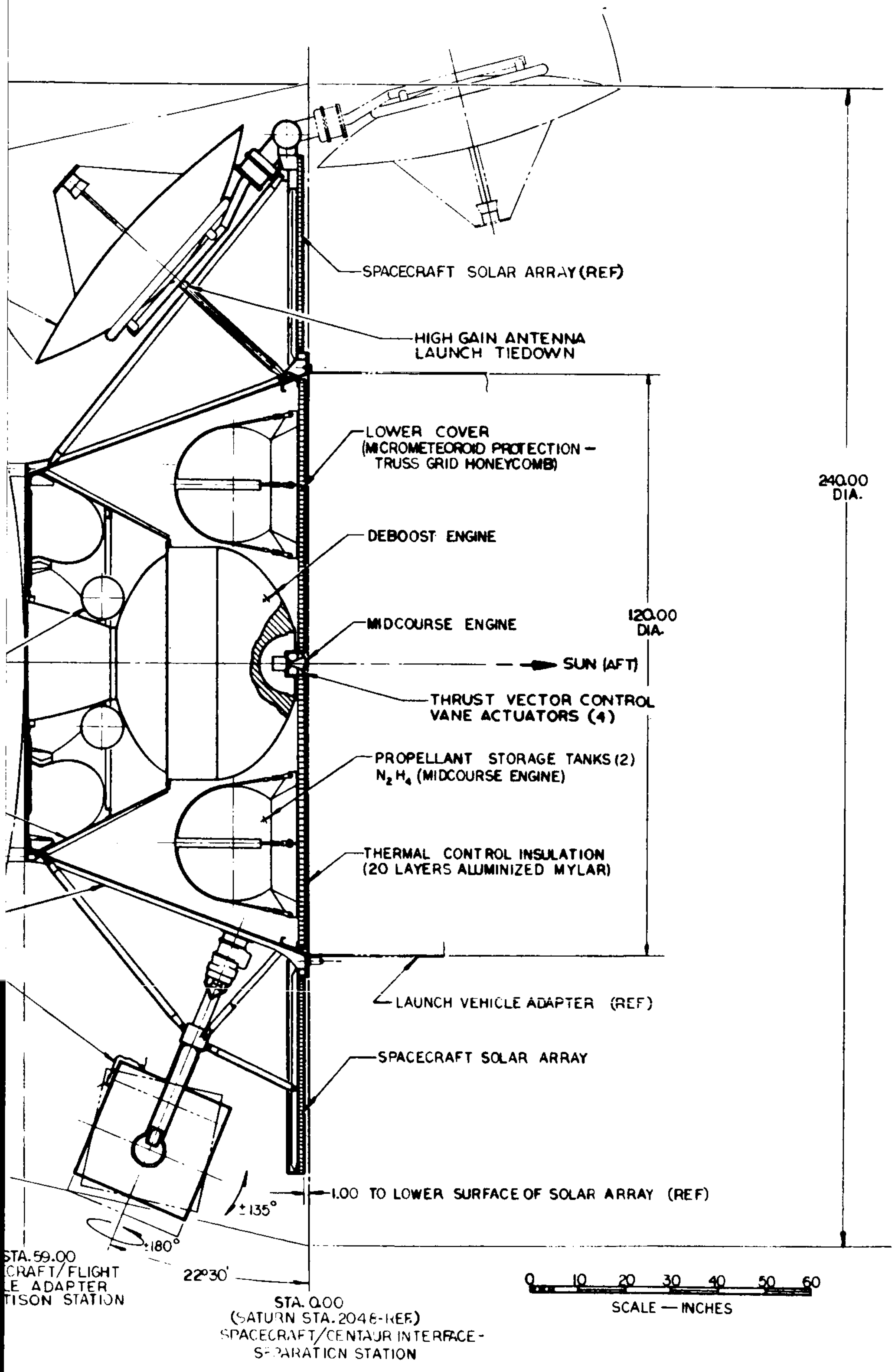

MAGNETOMETER EXPERIMENT (2)
1 EXTENDIBLE, 1 STATIONARY
STABILIZATION & CONTROL NOZZLES (REF)
COARSE SUN SENSOR (REF)
POWER EQUIPMENT MOUNTING PANEL
-YAW AXIS
COMMUNICATIONS EQUIPMENT MOUNTING PANEL
THERMAL CONTROL LOUVERS (4)
208.00
K
F
LANDER TO SPACECRAFT ANTENNA (VHF)
VI
B
5°
MICROMETEOROID IMPACT DETECTION EXPERIMENT
90° CLOCK ANGLE DOWN RANGE
A
I
J
H
LAYERS
NEAR EARTH SENSOR
LOW GAIN ANTENNA (S-BAND)
DEBOOST ENGINE (REF)
STABILIZATION & CONTROL LOW THRUST ROLL NOZZLE (4)
YAW NOZZLE (4)
+ PITCH AXIS
COARSE SUN SENSOR (REF)
STA 0.00
MIDCOURSE ENGINE (REF)
SPACECRAFT CORNER DESIGNATION (A-F)
SPACECRAFT PANEL DESIGNATION (I-VI)

3



223.00
A
HIGH GAIN ANTENNA (REF)
TEST CONNECTOR (4)
EXPERIMENT EQUIPMENT MOUNTING PANEL
COARSE SUN SENSOR (REF)
-PITCH AXIS
STABILIZATION AND CONTROL HIGH THRUST ROLL NOZZLE (4)
HIGH GAIN ANTENNA (S-BAND) (SHOWN IN LAUNCH CONFIGURATION) DOUBLE GIMBAL ACTUATION
12° 30' (REF)
INTERPLANETARY SCIENCE EXPERIMENT PACKAGE
STABILIZATION AND CONTROL, CENTRAL SEQUENCING AND COMMAND EQUIPMENT MOUNTING PANEL
IONOSPHERE EXPERIMENT ANTENNA (EXTENDIBLE)
80.00 DIA.
DEBOOST ENGINE THRUST VECTOR CONTROL LIQUID INJECTANT TANK (2)
DEBOOST ENGINE SUPPORT STRUCTURE
MEDIUM GAIN ANTENNA (S-BAND) (SHOWN IN LAUNCH CONFIGURATION) SINGLE GIMBAL ACTUATION
SPACECRAFT BUS
THERMAL INSULATION (20 LAYERS ALUMINIZED MYLAR)
STABILIZATION & CONTROL PITCH NOZZLE (4)
+ YAW AXIS
238.00R
COARSE SUN SENSOR (4)
AVAILABLE SPACECRAFT DYNAMIC ENVELOPE
CANOPUS SENSORS LINE OF SIGHT - 0° CLOCK ANGLE
PLANET ORIENTED PACKAGE WITH DOUBLE GIMBAL MOUNT
E
V
IV
D
H
III
C
II
B
G
G
E
E
C
C
A

4



SECTION **A A**

Figure 3. 1971 Voyager Spacecraft Inboard Profile

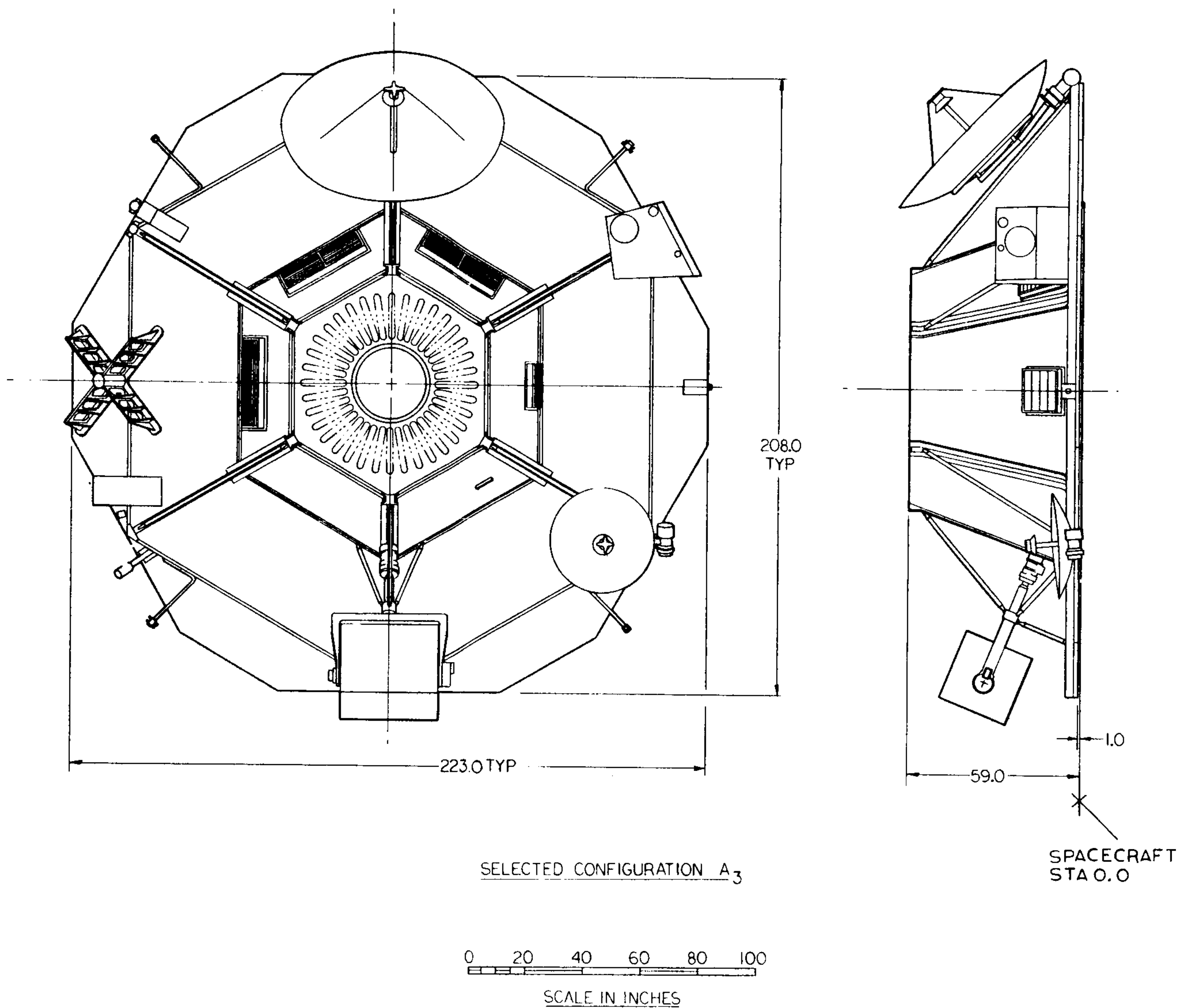
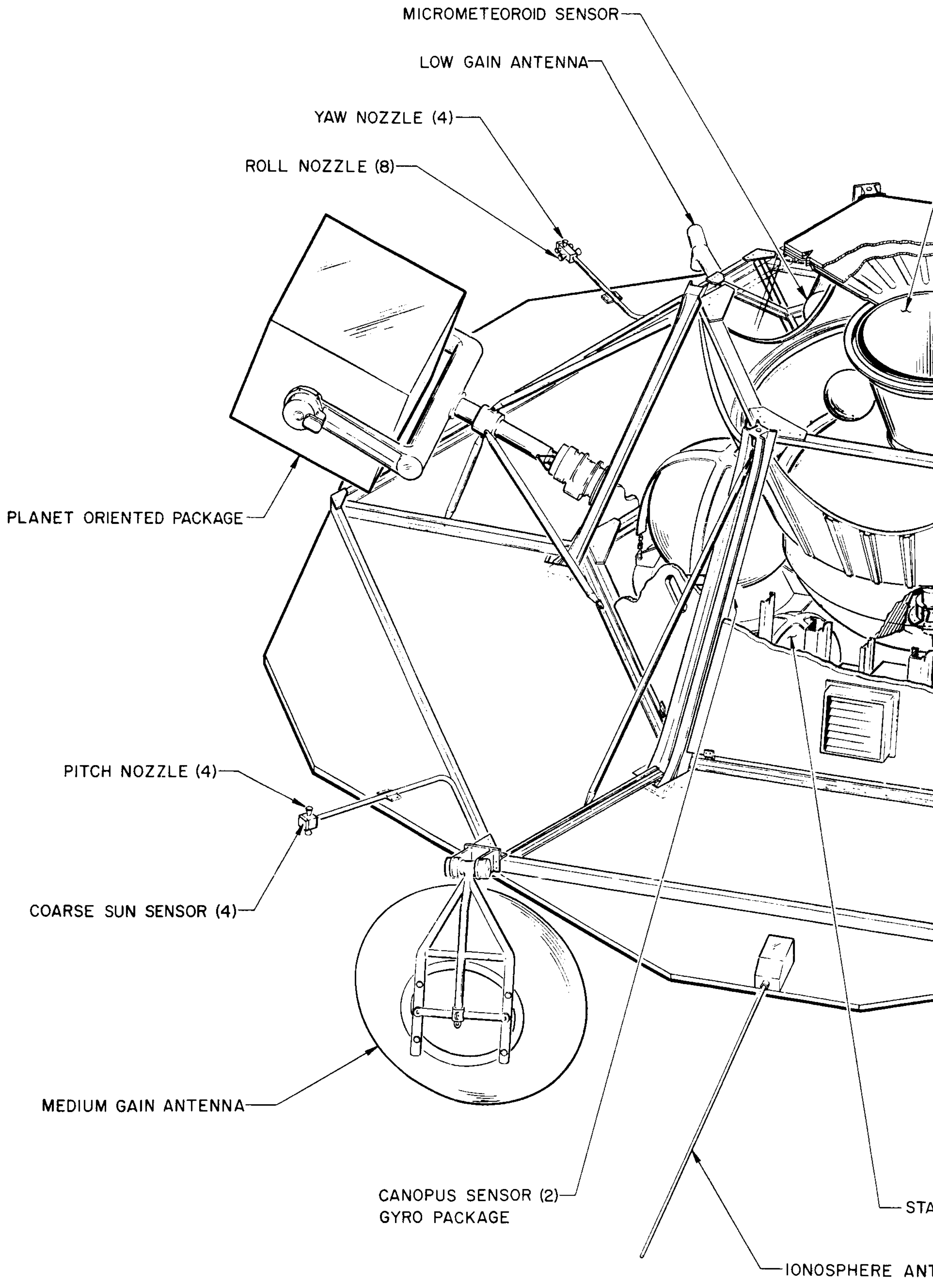


Figure 4. 1971 Voyager Spacecraft Outboard Profile



MICROMETEOROID SENSOR
LOW GAIN ANTENNA
YAW NOZZLE (4)
ROLL NOZZLE (8)
PLANET ORIENTED PACKAGE
PITCH NOZZLE (4)
COARSE SUN SENSOR (4)
MEDIUM GAIN ANTENNA
CANOPUS SENSOR (2)
GYRO PACKAGE
IONOSPHERE ANT
STA

2

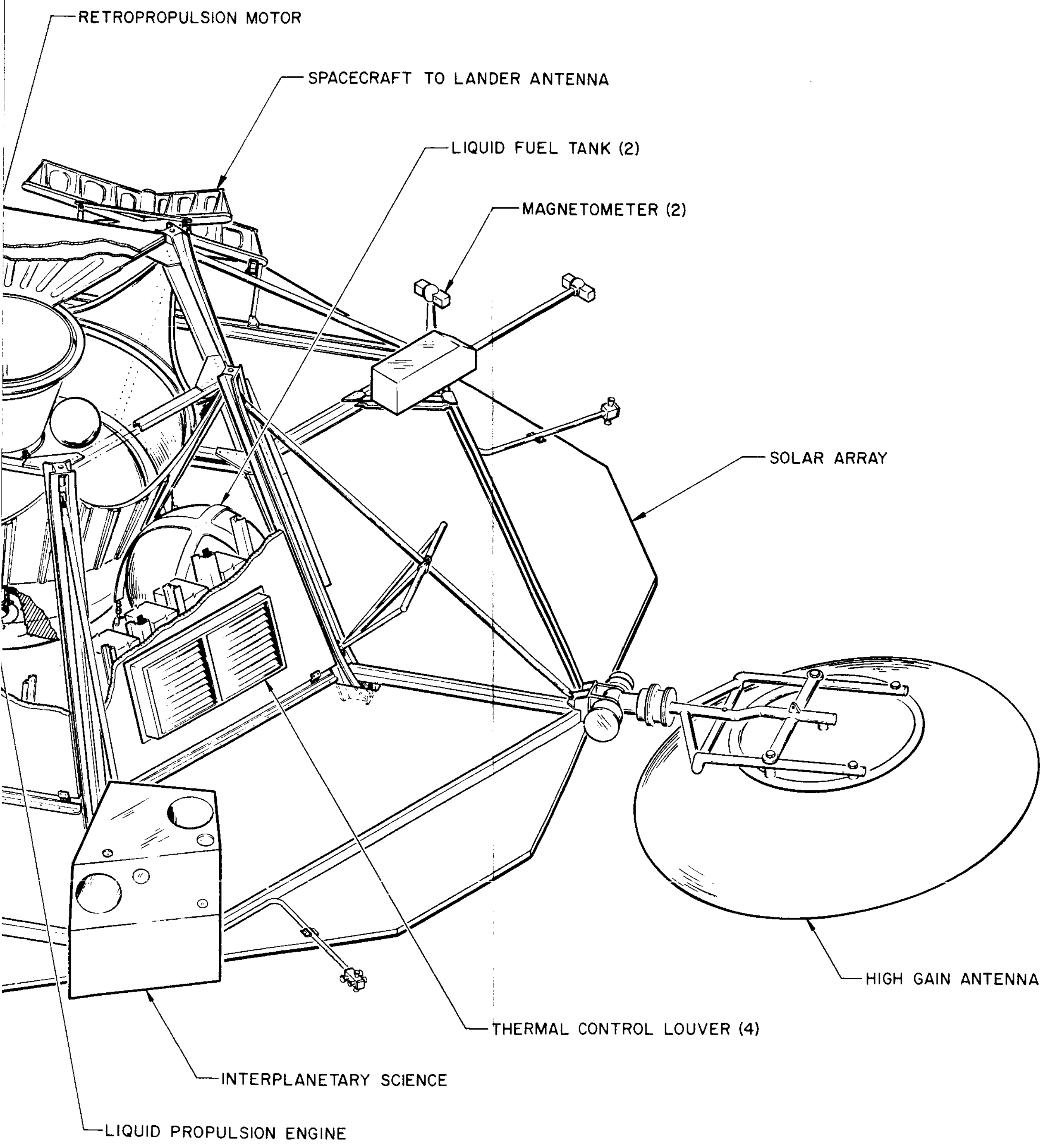


Figure 5. 1971 Voyager Spacecraft Isometric View

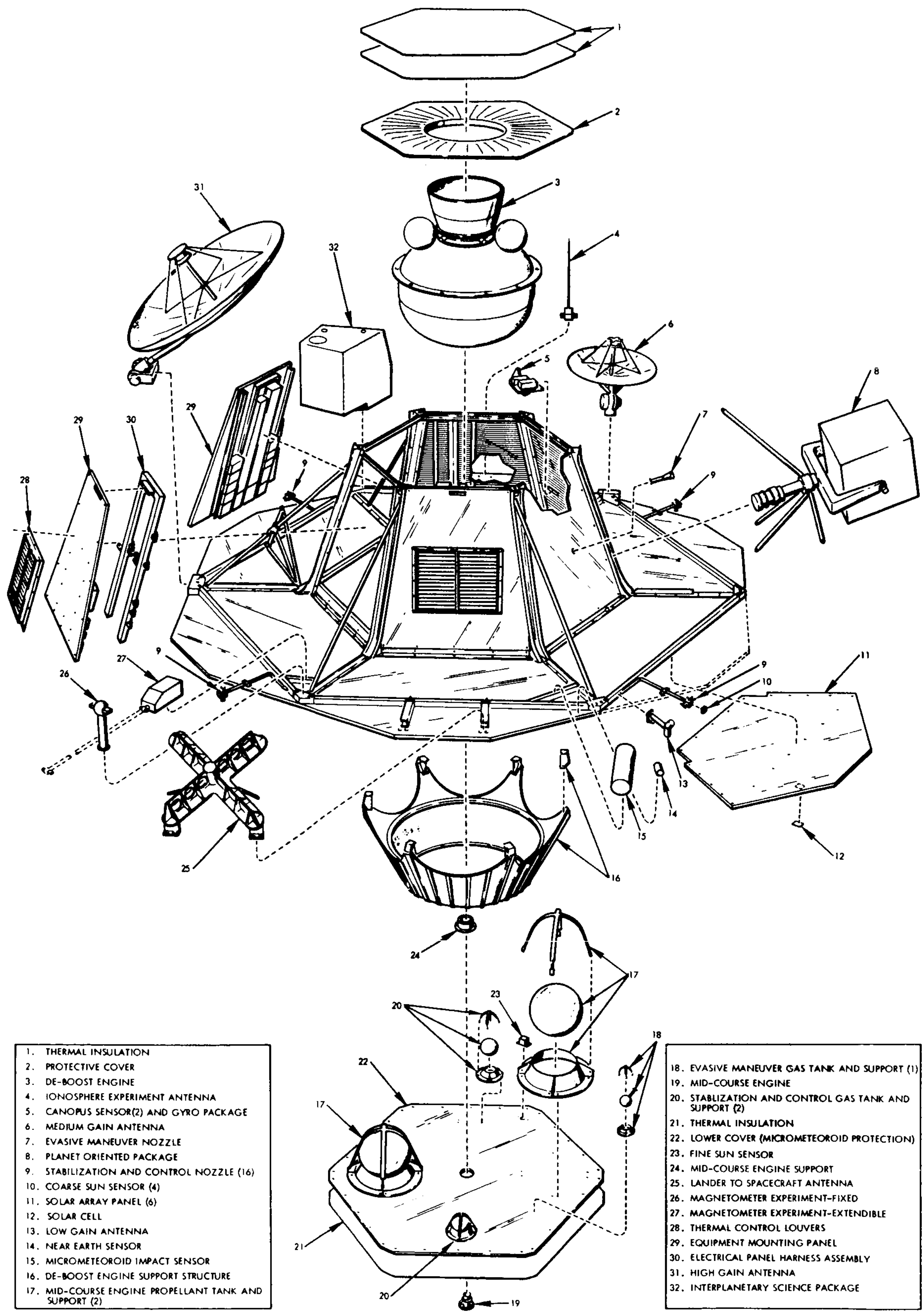
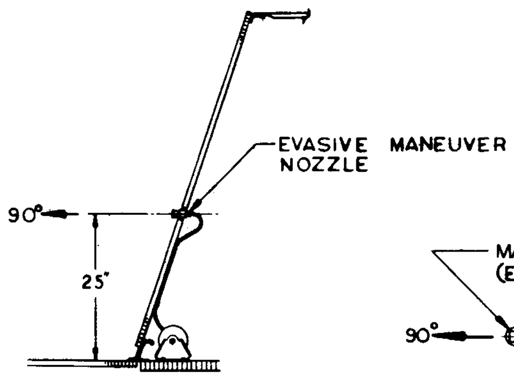
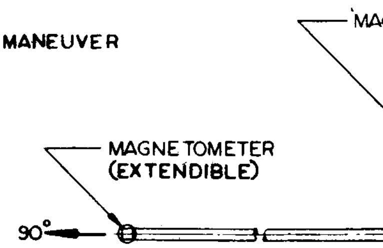
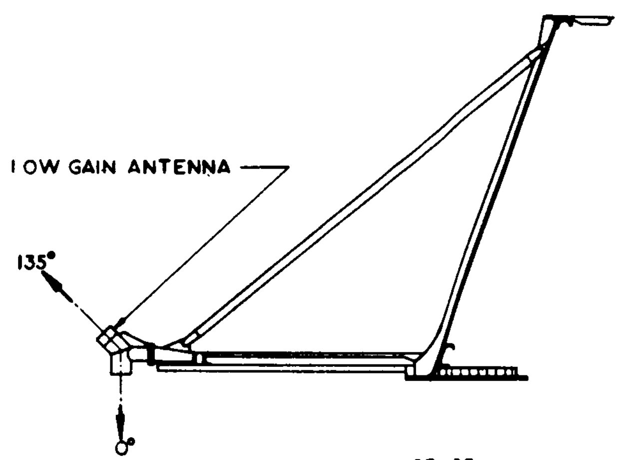


Figure 6. 1971 Voyager Spacecraft Exploded View

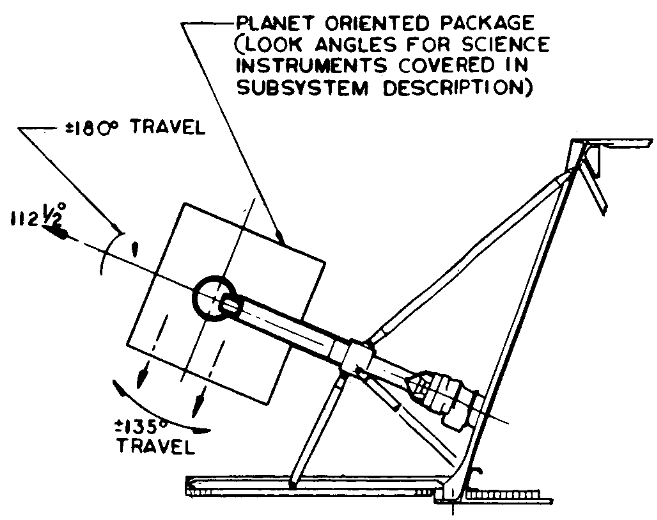


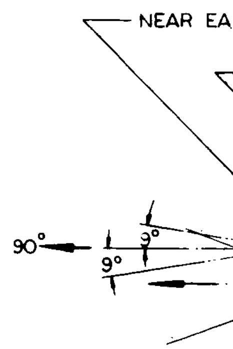


SECTION L-L



SECTION K-K



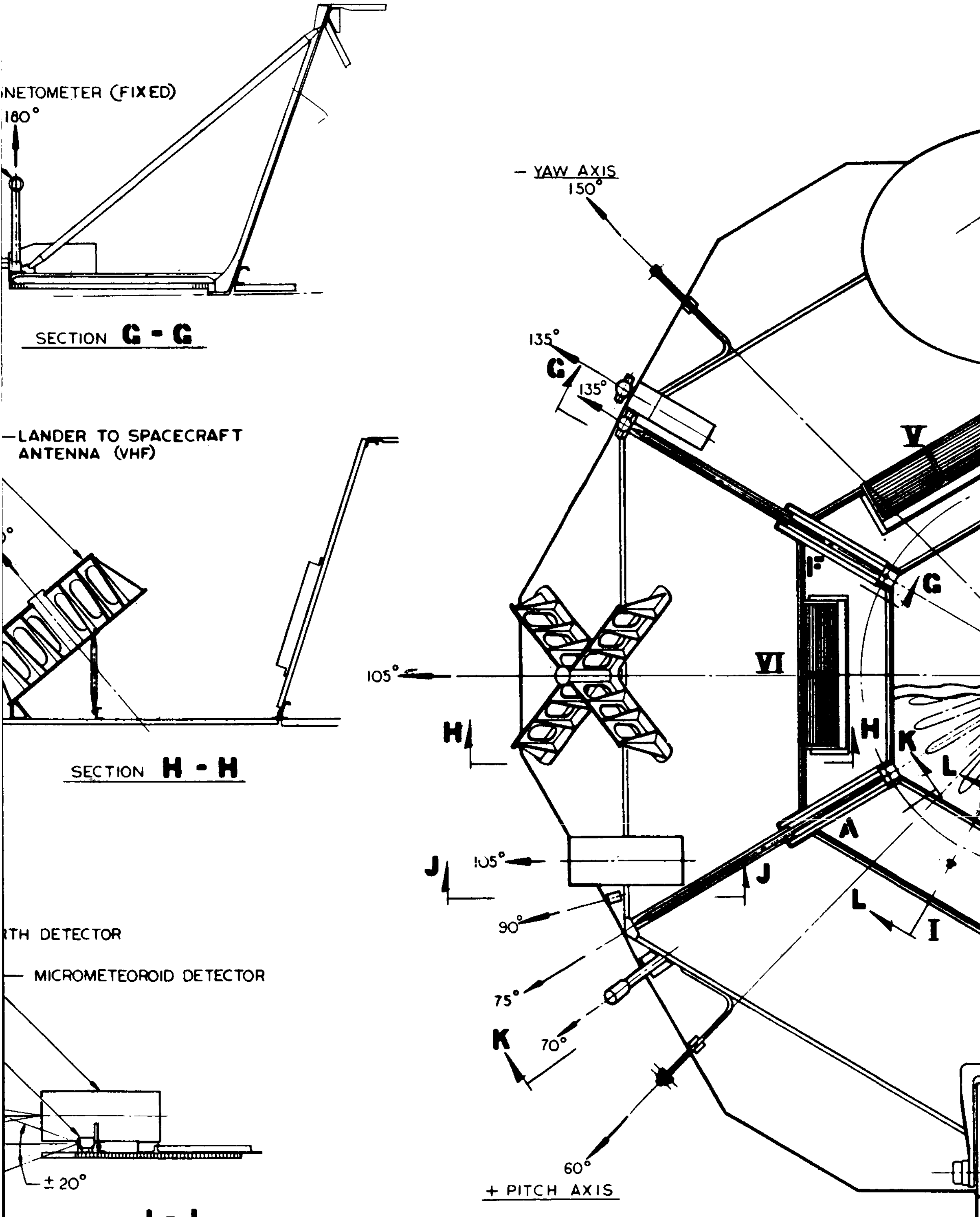


SECTION M-M

NOTES

1) ALL ANGLES SHOWN IN SECTIONS **A-A** DEFINE THE CONE ANGLE AXIS FOR EA
2) 0° CONE ANGLE IS DEFINED AS A LINE SPACECRAFT CENTERLINE POINTING AF
3) SECTIONS SHOWN HAVE BEEN ROTATE

1

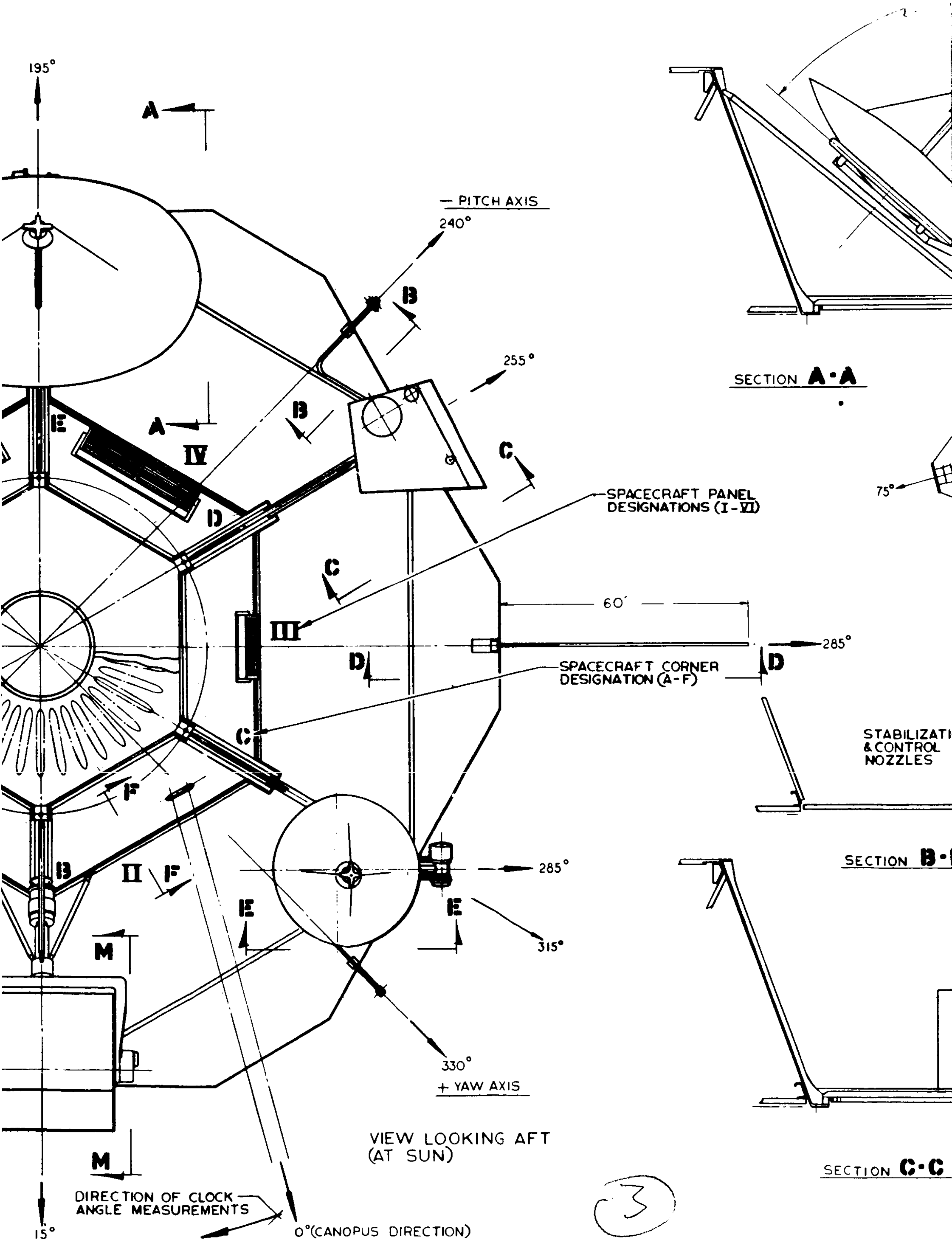


THRU **M-M**
CH SENSOR.
THRU THE
T – AT THE SUN.
D FOR CLARITY.

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~~ LINE OF SIGHT OF THE CANOPUS SENSOR.

(2)


195°
PITCH AXIS
240°
255°
SPACECRAFT PANEL DESIGNATIONS (I-VI)
SPACECRAFT CORNER DESIGNATION (A-F)
60´
285°
285°
315°
330°
+ YAW AXIS
VIEW LOOKING AFT (AT SUN)
DIRECTION OF CLOCK ANGLE MEASUREMENTS
15°
0°(CANOPUS DIRECTION)
SECTION A-A
75°
STABILIZATI
& CONTROL
NOZZLES
SECTION B-
SECTION C-C
3

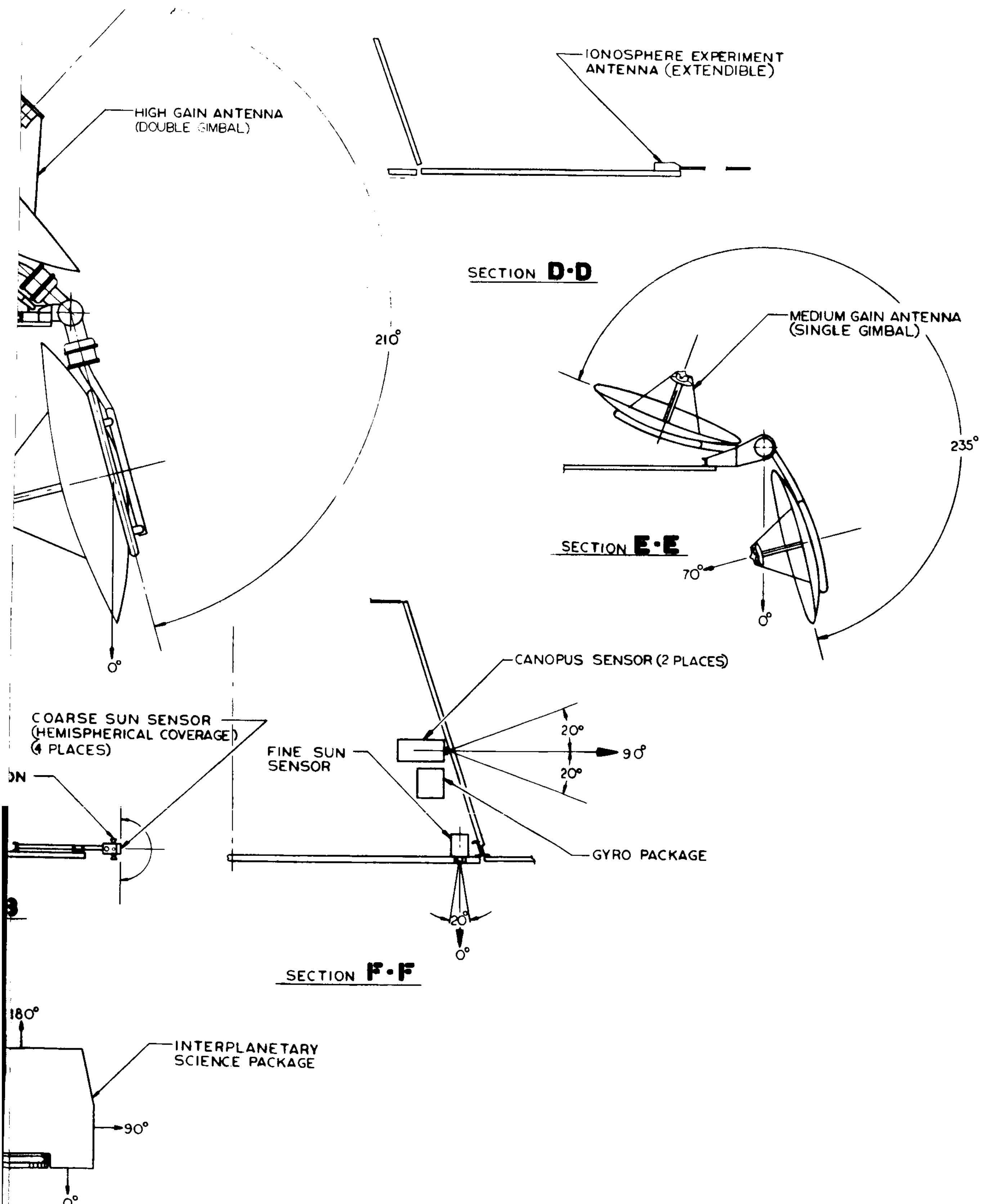


Figure 7. 1971 Voyager Spacecraft Sensor Geometry

4

COMPONENT DESIGN PARAMETERS

VS-3-111

CONTENTS

1. SCOPE

This document provides a listing of spacecraft equipment items and a tabulation of the corresponding design parameters. It encompasses weight, volume, electrical power and sources, and temperature limits.

2. COMPONENT DESIGN PARAMETERS TABLE

The equipment listing and design parameters are given in Table 1.

Table 1. Component Design Parameters

Component	No. of Items	Total Weight (lb)	Volume Each (in^3)	Electrical Power and Sources: Average	Electrical Power and Sources: Peak	Electrical Power and Sources: Primary Power Source	Allowable Operating Temperature (°F) Min.	Allowable Operating Temperature (°F) Max.	Allowable Nonoperating Temperature (°F) Min.	Allowable Nonoperating Temperature (°F) Max.
MECHANICAL AND PYROTECHNICS		(37.2)								
Launch Vehicle Separation		5.7					-65	165	-300	165
Capsule Jettison		7.6					-300	165	-300	165
Solar Array Support										
Tubes and Fittings		18.2					-250	240	-250	240
Frames		3.8					-250	240	-250	240
Attachment and Miscellaneous		1.9								
SPACECRAFT STRUCTURE		(489.0)								
Meteoroid Protection Panels		274.8					-250	240	-250	240
Framework		76.2					-250	240	-250	240
Equipment Mounting Provisions		92.0					-250	240	-250	240
High Gain Antenna Supports		2.7					-250	240	-250	240
Medium Gain Antenna Supports		1.1					-250	240	-250	240
Low Gain Antenna Supports		0.8					-250	240	-250	240
VHF Antenna Supports		0.6					-250	240	-250	240
POP Supports		3.2					-250	240	-250	240
Interplanetary Science Sensor Supports		2.6					-250	240	-250	240
Stabilization and Control Supports		0.8					-250	240		240
Attachment and Miscellaneous		34.2					-250	240	-250	240
THERMAL CONTROL		(49.7)								
Insulation										
Aluminized Mylar		17.4					-300	300	-300	300
Refrasil batt		8.1					-300	2000	-300	2000
Louvers		7.4					-100	250	-100	250
Heaters and Thermostats		2.0		2.5	5.0	50 VDC	40	90	NA	NA
Solid Motor Insulation		14.8					-300	800	-300	800
TELECOMMUNICATIONS		(159.6)								
Mod-Exciter	2	6.0	90	2.0	2.0	4.1 KC	30	110	-25	175
4-Port Hybrid Ring	1	0.6	15				30	110	-25	175
Low Power Transmitter	1	3.5	105	10.0	10.0	4.1 KC	30	110	-25	175
Power Amplifiers (20w)										
Tube	2	4.0	84				30	185	-25	250
Power Supply	2	11.0	135	90.0	90.0	50 VDC	30	110	-25	175
Transmitter Selector	1	0.8	15	0.8	0.8	4.1 KC	30	110	-25	175
S-Band Receiver	3	15.0	150	7.5	7.5	4.1 KC	30	110	-25	175
Receiver Selector	1	0.8	15	0.8	0.8	4.1 KC	30	110	-25	175
Command Demodulator	2	4.0	30	1.5	1.5	4.1 KC	30	110	-25	175
VHF Receiver	2	4.0	30	1.0	1.0	4.1 KC	30	110	-25	175
Capsule Demodulator	2	0.8	15	0.6	0.6	4.1 KC	30	110	-25	175
VHF Preamp	1	0.4	15				30	110	-25	175
3-Port Circulator Switch	4	7.3	23				30	110	-25	175
Diplexer	3	2.4	46				30	110	-25	175
DTU	2	6.0	75	4.0	4.0	4.1 KC	30	110	-25	175
Signal Conditioner	1	1.0	20	1.0	1.0	4.1 KC	30	110	-25	175
Data Storage	1	4.0	100	1.5	1.5	4.1 KC	30	110	-25	175
Bulk Storage	2	24.0	350	5.0	15.0	4.1 KC	30	110	-25	175
Low Gain Antenna	1	1.0								
Medium Gain Antenna	1	15.1		.8/.2	4.0/7.0	4.1 KC/410 cps	-350	360	-350	360
High Gain Antenna	1	43.4		1.6/1.0	4.0/33.0	4.1 KC/410 cps	-350	360	-350	360
VHF Antenna	1	4.5					-350	360	-350	360
ELECTRICAL POWER		(314.1)								
Solar Array*	1	190.0					-184	248	-184	248
Batteries	2	80.4	720				50	90	50	90
Inverter 250 watt 4.1 KC	2	7.0	72				-20	120	-50	200
Inverter 30 watt 820 cycle	2	4.0	36				-20	120	-50	200
Inverter 50 watt 410 cycle	2	6.0	48				-20	120	-50	200
Battery Regulator	2	10.4	192				-20	120	-50	200
Power Control Unit	1	6.3	180				-20	120	-50	200
Shunt Element Assembly	1	10.0	216				-20	150	-50	200
ELECTRICAL DISTRIBUTION		(142.0)								
Cabling and Connectors		120.0								
J-Boxes	4	20.0	216							
Umbilicals	2	2.0								
CENTRAL SEQUENCING AND COMMAND		(26.6)								
Input Decoder	2	2.0	20	3.6	3.6	4.1 KC	-31	167	-31	167
Command Decoder	2	2.0	20	1.0	1.0	4.1 KC	-31	167	-31	167
Sequencer	2	15.0	200	9.3	9.3	4.1 KC	-31	167	-31	167
Power Supply	2	7.6	90	4.1	4.1	4.1 KC	-20	120	-50	200
STABILIZATION AND CONTROL		(100.0)								
Control Electronics Assembly	1	13.0	216	9.0	45.0	4.1 KC	30	130		
Gyros and Electronics	1	10.0	180	6/1.5/10	27/1.5/10	820 CY/4.1 KC/DC	30	130		
Coarse Sun Sensor	4	2.0	2				30	140	-20	160
Fine Sun Sensor	1	0.6	32				30	130	-20	160
Canopus Sensor + Electronics	2	11.0	220	3.0	3.0	4.1 KC	-30	100	-30	100
Gas Vessel + Transducers	2	28.0	1150				40	140	0	200
N_2 Gas		20.0					40	140	0	200
Pressure Regulator + Transducers	2	3.0	21				40	140	0	200
Valves + Plumbing Set	2	12.1			48.0	50 VDC				
Earth Detector	1	0.3	7	0.2	0.2	4.1 KC	30	130	-20	160

*190 ft^2 array

Table 1. Component Design Parameters (Continued)

Component	No. of Items	Total Weight (lb)	Volume Each (in³)	Electrical Power and Sources: Average	Electrical Power and Sources: Peak	Electrical Power and Sources: Primary Power Source	Allowable Operating Temperature (°F) Min.	Allowable Operating Temperature (°F) Max.	Allowable Nonoperating Temperature (°F) Min.	Allowable Nonoperating Temperature (°F) Max.
SCIENCE SUPPORT		(114.1)								
POP Science Package Structure		41.9					-250	240	-250	240
POP Support Shaft		6.2					-250		-250	
POP Support Fork		11.6					-250	240	-250	240
POP Thermal Control		2.5		12.5	25.0	50 VDC	-300	300	-300	300
Fork Bearing Housing	2	4.0								
Drives	2	10.4		1.0	14.0	410 CPS				
Two Position Pickoffs	2	5.0								
Magnetometer Booms	2	5.3					-250		-250	240
Magnetometer Drive Mechanism		3.5					20	120	20	150
Magnetometer Thermal Control		0.1					-300	300	-300	300
Science Package Thermal Control		1.4		3.8	7.5	50 VDC	-300	300	-300	300
Science Cabling and Connectors and Wrapups		16.5								
Attachments and Miscellaneous		5.7								
PROPULSION SYSTEM		(483.5)								
Retropropulsion										
Solid Motor Inert Weight										
Case		110.0					40	90	30	100
Insulation **		90.0					40	90	30	100
Nozzle**		90.0					40	90	30	100
Igniter		5.0					40	90	30	100
Solid Motor Support										
Frames		4.9								
Main Struts		2.2								
Stringers		2.0								
Skin		9.2								
End Fittings		1.9								
Attachments and Miscellaneous		1.2								
LITVC Injectors		8.0		0.8	0.8	50 VDC	40	90	30	100
LITVC Fluid + Tankage ***		82.0					40	90	30	100
Midcourse Propulsion										
Containers		34.1								
Pressurization										
Hand Valves		0.8					40	90	35	125
Lines		0.3					40	90	35	125
Fittings and Clips		0.2					40	90	35	125
Propellant System										
Hand Valve		1.0					40	90	35	125
Lines		1.9					40	90	35	125
Fittings and Clips		0.5					40	90	35	125
Bladder System		4.8					40	90	35	125
Thrust Chamber and Valves		5.0					40	90	35	125
Propulsion Module Structure										
Container Supports		12.8					40	90	35	125
Thrust Structure		2.3					40	90	35	125
Attachment and Miscellaneous		0.9					40	90	35	125
Thermal Control		1.0					40	90	35	125
Pressurant		1.2					40	90	35	125
Unused Propellants		4.0					40	90	35	125
Jet Vane Assembly		0.3					40	90	35	125
Jet Vane Actuators		4.0		6.0	12.0	50 VDC	40	90	35	125
Evasive Maneuver Propulsion		2.0								
SPACECRAFT SCIENCE PAYLOAD		(266.6)								
POP Mounted										
TV Experiment	1	36.0	3456		7.0	4.1 KC	0	140	-25	175
UV Spectrometer	1	18.0	2000		2.0	4.1 KC	0	140	-25	175
Scan Radiometer	1	10.0	1800		2.0	4.1 KC	0	140	-25	175
IR Spectrometer	1	20.0	1296		4.0	4.1 KC	0	140	-25	175
Meteoroid Flash	1	5.0	192		1.0	4.1 KC	0	140	-25	175
Mars Sensor	1	12.0								
Bus Mounted Sensors										
Meteoroid Impact	4	6.0	472.5		0.8	4.1 KC	0	140	-25	175
Magnetomter	2	2.6	42		0.5	4.1 KC	0	140	-25	175
Plasma	2	4.0	96		0.6	4.1 KC	0	140	-25	175
Cosmic Ray	4	3.0	12.8				0	140	-25	175
Trapped Radiation	3	9.0	210		0.6	4.1 KC	0	140	-25	175
Ionosphere Experiment	1	3.0					0	140	-25	175
Bus Mounted Remote Hardware										
TV Experiment	1	16.0	1152		15.0	4.1 KC	0	140	-25	175
UV Spectrometer	1	7.0	576		11.0	4.1 KC	0	140	-25	175
Scan Radiometer	1	2.0	144		5.0	4.1 KC	0	140	-25	175
IR Spectrometer	1	2.0	144		4.0	4.1 KC	0	140	-25	175
Meteoroid Flash	1	5.0	192		4.0	4.1 KC	0	140	-25	175
Meteoroid Impact	4	10.0	36		1.2	4.1 KC	0	140	-25	175
Magnetometer	2	10.0	96		1.5	4.1 KC	0	140	-25	175
Plasma	2	5.5	80		5.4	4.1 KC	0	140	-25	175
Cosmic Ray	4	5.0	96		5.0	4.1 KC	0	140	-25	175
Trapped Radiation	1	12.5	1560		4.4	4.1 KC	0	140	-25	175
Ionosphere Experiment	1	6.0	252		1.5	4.1 KC	0	140	-25	175
Data Automation Equipment		57.0	1320		33.0	4.1 KC	0	140	-25	175

**Approx. 70 lb expended

***Approx. 35 lb expended

WEIGHTS AND MASS PROPERTIES

VS-3-112

CONTENTS

1. SCOPE

This document includes weight justification and method of mass properties determination for the Voyager flight spacecraft, 1971 mission. General requirements and constraints considered in the determination of spacecraft mass properties are discussed and a weight summary by subsystem, weight and mass properties history, center of mass envelope, and moments of inertia are included. Functional requirements for mass properties control during design are listed.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

<u>JPL Documents</u>

V-MA-004-14-03	Preliminary Voyager 1971 Mission Specification

<u>TRW 1971 Voyager Spacecraft Design Documents</u>

VS-3-110	Configuration
VS 3-111	Components Design Parameters
VS 4-550	Electronic Packaging

3. GENERAL

The Voyager flight spacecraft mass property estimates included in this document satisfy the requirements specified in JPL Project Document No. 45, "Preliminary Voyager 1971 Mission Specification."

4. WEIGHT AND MASS PROPERTIES DETERMINATION

4.1 <u>Description</u>

Accurate knowledge of the spacecraft weight and mass properties is essential during all phases of design. The weights data listed in Table 1 are estimated based on the selected configuration utilized in the preliminary design and fulfill the requirements imposed upon the spacecraft center of mass during midcourse velocity correction and retropropulsion operations. The radial center of mass envelope is shown in Figure 1, and Table 2 lists the longitudinal center of mass travel during the spacecraft flight sequence after separation from the launch vehicle.

Table 1. Flight Spacecraft Weight Summary - 1971 Mission

Item	Propulsion Weight	Capsule Weight	Bus Weight	Total Weight
Spacecraft Bus				
Mechanical and pyrotechnics			37	37
Spacecraft structure			489	489
Thermal control			50	50
Telecommunications			160	160
Electrical power			314	314
Electrical distribution			142	142
Central sequencing and command			27	27
Stabilization and control			100	100
Science support			114	114
Margin			187	187
Contingency			113	113
Spacecraft Propulsion System				
Retropropulsion				
Inert weight	315			315
Module structure	21			21
Midcourse propulsion				
Inert weight	75			75
Midcourse propellant unused	215			215
Evasive maneuver propulsion	2			2
Contingency	29			29
Spacecraft Science Payload			267	267
Spacecraft Weight in Orbit				2657
Propulsion				
Retropropellant for deboost	2733			2733
Inerts expended	70			70
Spacecraft Weight After Capsule Separation				5460
Flight Capsule				
Remaining capsule components (Ejected after capsule separation)		150		150
Capsule vehicle		1950		1950
Jettisoned canister		200		200
Spacecraft Weight Before Capsule Separation				7760
Propulsion				
Median midcourse propellant used	40			40
Separated Planetary Vehicle				7800
Total	3500	2300	2000	7800
Adapter Allocated Weight Above Field Joint				
Adapter weight above field joint remaining with Centaur				12
Adapter allocated weight not used				238
Total Planetary Vehicle Weight				8050

4.2 Weight

4.2.1 Weight Allocations

The weight allocations for the Voyager flight spacecraft for the 1971 mission specified by JPL are as follows:

Item	Weight, pounds	
Spacecraft bus (including 250 pounds of science)	2000	
Spacecraft propulsion system	3500	
Flight capsule vehicle jettisoned weight	1950	
Flight capsule adapter and sterilization canister (a maximum of 150 pounds may remain with spacecraft)	350	
Separated Voyager flight spacecraft weight		7800
Spacecraft adapter above field joint	250	
Voyager flight spacecraft weight		8050

4.2.2 Weight Margin

A weight margin is included in the weight summary, defined as the difference between the spacecraft bus allocated weight and the spacecraft design weight. This margin may be used for additional redundancy for greater spacecraft reliability, additional science experiments for increased scientific observations, or additional propellants for greater mission capability.

4.2.3 Weight Contingency

A contingency of 6 per cent has been added to the spacecraft bus and propulsion system nominal weights. This contingency reflects the over-all level of confidence of the weight estimates and is consistent with the current level of design. The contingency allows for uncertainties in the weight estimation techniques, slight modifications of the design, and for balance weights to maintain the desired center of mass location. It also includes an allowance for weight growth during design completion and the development phase of the spacecraft.

4.2.4 Weight Justification

Spacecraft bus and propulsion structure weights are calculated based on gages and materials determined by stress analyses. Other spacecraft component weights are estimates based on the current level of design, and the resulting accuracy is consistent with the contingency discussed in 4.2.3. All weight estimates listed in Table 1 are nominal weights.

4.3 Center of Mass

4.3.1 Center of Mass Determination

The spacecraft center of mass has been calculated using the component weights which are listed in detail in Table 1, VS-3-111. Component locations are shown in the inboard profile of VS-3-110 and in the panel installation arrangements of VS-4-550.

4.3.2 Center of Mass Envelope

a. Radial Center of Mass

The spacecraft radial center of mass envelope is shown in Figure 1. This envelope is based on a single point maximum center of mass offset of ± 0.1 inch at retropropulsion burnout.

b. Longitudinal Center of Mass

The spacecraft nominal longitudinal center of mass travel is listed in Table 2. The tolerance about the nominal center of mass is less than ± 0.2 inch. Variations in the center of mass envelope brought about by variations in spacecraft geometry have been determined computationally.

4.3.3 Actual Center of Mass Determination and Control

The spacecraft bus (less appendages) center of mass is to be determined experimentally. The total spacecraft center of mass will be determined by incorporating the calculated center of mass for each of the appendages with the experimentally determined center of mass for the bus.

The spacecraft center of mass is to be maintained within the envelope specified in 4.3.2 by close control of component and subsystem location and, if necessary, by the addition of balance weights.

4.4 Centroidal Moments of Inertia

Centroidal moments of inertia have been determined computationally for the complete flight sequence from spacecraft separation from the launch vehicle to spacecraft retropropulsion burnout. Table 2 lists the moments of inertia about the spacecraft stabilization and control axes as defined in VS-3-110.

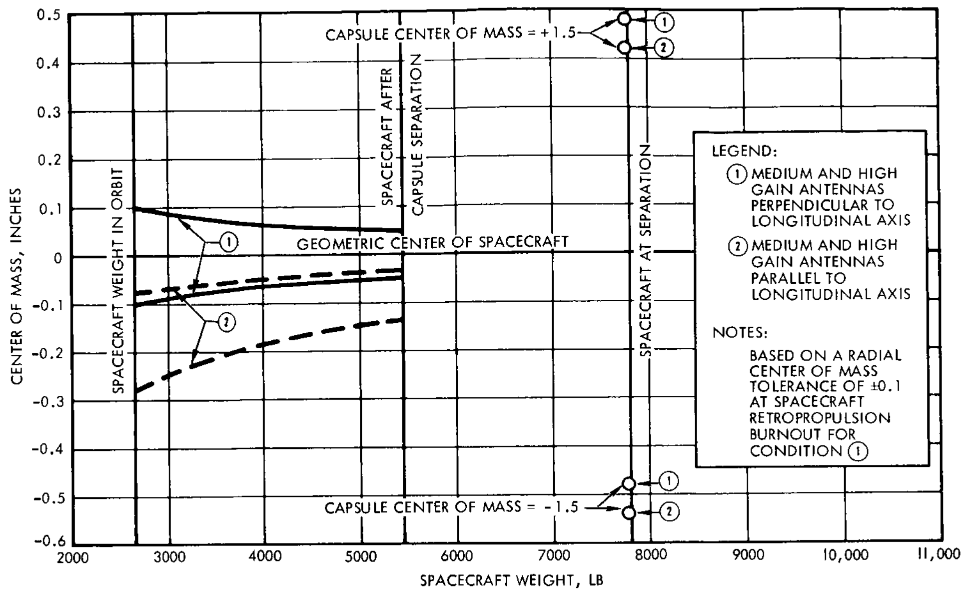


Figure 1. Voyager Radial Center of Mass Envelope

Table 2. Mass Properties History

Condition	Weight, Lb	Center of Mass, Inches*	Moments of Inertia, Slug-Ft^2		
		Spacecraft Station	I_{pitch}	I_{yaw}	I_{roll}
Separated Spacecraft Weight	7800	47.3	4778	5772	4058
Spacecraft Weight Before Capsule Separation	7760	47.4	4769	5751	4046
Spacecraft Weight After Capsule Separation	5460	22.3	886	1880	2389
Spacecraft Weight in Orbit	2657	22.7	829	1824	2241

*Measured from field joint (Station 0)

RELIABILITY DESIGN OBJECTIVES

VS-3-120

CONTENTS

RELIABILITY DESIGN OBJECTIVES

VS-3-120

1. SCOPE

This document provides a consolidated statement of Voyager spacecraft reliability goals for each subsystem as needed to achieve the maximum mission reliability consistent with established mission and spacecraft constraints. It also includes pertinent underlying data.

1.1 Technical Ground Rules and Models

The technical ground rules for allocating reliability requirements to subsystems and their major elements are given prior to the presentation of numerical requirements in tabular form. Essential subsystem reliability models pertinent to the Voyager spacecraft preliminary design are also given to prescribe the degrees of equipment redundancy employed. Alternate operating modes which are known to provide functional redundancy are identified by subsystem.

1.2 Mission Probabilities

Compliance with the subsystem reliability requirements is translated into cumulative mission success probabilities as established by Section II-A of JPL Preliminary Voyager 1971 Mission Specification.

1.3 Data for Reliability Test Hypotheses

A summary of component mean time between failure (MTBF) objectives is given to serve as a basis for reliability test hypotheses when combined with appropriate statistical decision criteria.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

JPL

Project Document No. 45 V-MA-004-001-14-03	Preliminary Voyager 1971 Mission Specification

TRW 1971 Voyager Spacecraft Design Documents

VS-3-104 Flight Sequences

3. RELIABILITY GROUND RULES

3.1 Apportionment and Assessment

Two basic reliability analysis techniques are applicable to the derivation of reliability design objectives for Voyager spacecraft subsystems. These techniques are reliability apportionment and reliability assessment.

3.1.1 Apportionments

Apportionments are normally provided at the outset of a system design process and may utilize gross estimates of design methods, in conjunction with mission needs, to assign specific first-cut reliability objectives to each significant system element. As apportionments seek to become more refined and reflect a more intimate knowledge of specific design features and environments, they depend explicitly upon reliability estimates for each contributing subsystem element.

3.1.2 Assessments

When the contributing elements can be resolved to the parts level, such reliability estimates are termed assessments. For the Voyager study, the reliability apportionments derived are based to a large extent upon detailed reliability assessments, as given in Appendix B of Volume 4.

3.1.3 Validity of Assessments

In utilizing reliability assessment numerics, care was exercised to maintain a consistent level of conservatism so that tradeoff decisions would not be unrealistically biased by assessment errors. Also, a standard of design conservatism was prescribed consistent with the 1971 Voyager mission objectives. The apportioned reliability levels appear to be realistically achievable under the constraints specified.

3.2 Mission Model

For the purpose of reliability analysis a representative Voyager mission was established to permit comparative analysis. Reliability goals are established for spacecraft subsystems in accordance with operating time intervals corresponding to representative mission phases.

3.2.1 Mission Phases

The selected phase definitions are

Mission Phase 1: (0.3 hour)	For the period from liftoff through boost and spacecraft injection.
Mission Phase 2: (4280 hours)	For the period after spacecraft injection through cruise (including midcourse corrections) and capsule vehicle separation.
Mission Phase 3: (50 hours)	For the period after capsule vehicle separation including Mars orbit injection and achievement of orbital operation conditions. This period is taken to include the acquisition of the relayed capsule entry data.
Mission Phase 4: (720 hours)	For the period after successful spacecraft orbit attainment and extending for one month thereafter.
Mission Phase 5: (3600 hours)	For the period after a one month successful spacecraft orbital operation and extending for five months thereafter.

3.2.2 Analysis

In general the reliability associated with any given mission phase will be the probability of success at the phase termination divided by the probability of success at the starting time of that phase. Within some phases discrete events occur which contribute directly to the probability of mission-phase failure. Some Voyager subsystems employ an uncertain pattern of actuation events for which assumed average rates are meaningful when the total number of occurrences throughout that Voyager mission phase is effectively large. Reliability analysis of these factors is taken into account in the failure models employed for reliability assessment.

3.2.3 System Tradeoffs

Failure models are related directly to the specific mission phases defined in 3.2.1. System design tradeoff analyses are conducted using reliability assessments for that set of cumulative mission phases which constrains each Voyager subsystem (see Volume 4, Section IV).

3.3 Competing Characteristic Criteria

In the evolution of a preliminary design for the 1971 Voyager spacecraft, instances of conflict arise between reliability considerations and performance requirements. Guidelines used to resolve these conflicts are based upon the specific "Competing Characteristics" given on Page 21 in the Preliminary Voyager 1971 Mission Specification.

3.3.1 Order of Priority

For the spacecraft subsystem tradeoff analyses the following descending order of priority applies to constrain design decisions for reliability. (Corresponding mission phases and Preliminary Voyager 1971 Mission Specification paragraph numbers are referenced.)

a) Mars contamination constraints (Specification, Section II D 5 (1) and II D 2).

b) Probability of valid checkout of spacecraft by OSE without jeopardy of launch opportunity.

c) Probability of spacecraft boost and injection survival through mission phase 1.

d) Probability of achieving communications telemetry downlink (Specification Section II D 5 (2) during mission phases 2 and 3.

e) Probability of continuous operation of spacecraft sun-line attitude control, thermal control, and spacecraft power subsystem functions during mission phases 2 and 3. (Specification, Sections II D 5 (3), (4), and (5).

f) Probability of spacecraft support of capsule temperature and power requirements. (Specification paragraphs II D 5 (6) and (7) during mission phase 2.

g) Probability of achieving communications and commands uplink to the spacecraft (Specification Section II D 5 (8)) during mission phases 2, 3, and 4.

h) Probability of performing spacecraft roll and midcourse maneuvers (Specification Sections II D 5 (9) and (10) during mission phases 2 and 3.

i) Probability of successful capsule separation from properly oriented spacecraft (Specification Sections II D 5 (11) and (14) during mission phase 3.

j) Probability of successfully achieving a spacecraft orbit at Mars (Specification Section II D 5 (12) during mission phase 3.

k) Probability of operating spacecraft instruments at Mars (Specification Section II D 5 (13) during mission phase 4.

l) Probability of successful operation of spacecraft cruise instrumentation (Specification Section II D 5 (17) during mission phase 2.

3.3.2 Subsystem Ramifications

The ramifications of these ordered criteria are considered for each subsystem in Section IV of Volume 4. In general, the spacecraft subsystems contributing to the risk of completing each mission phase are identified and given reliability objectives commensurate with their priority in the competing characteristic criteria as illustrated in Figure 1. Criteria b and c are ordered above d only because they are effectively conditional

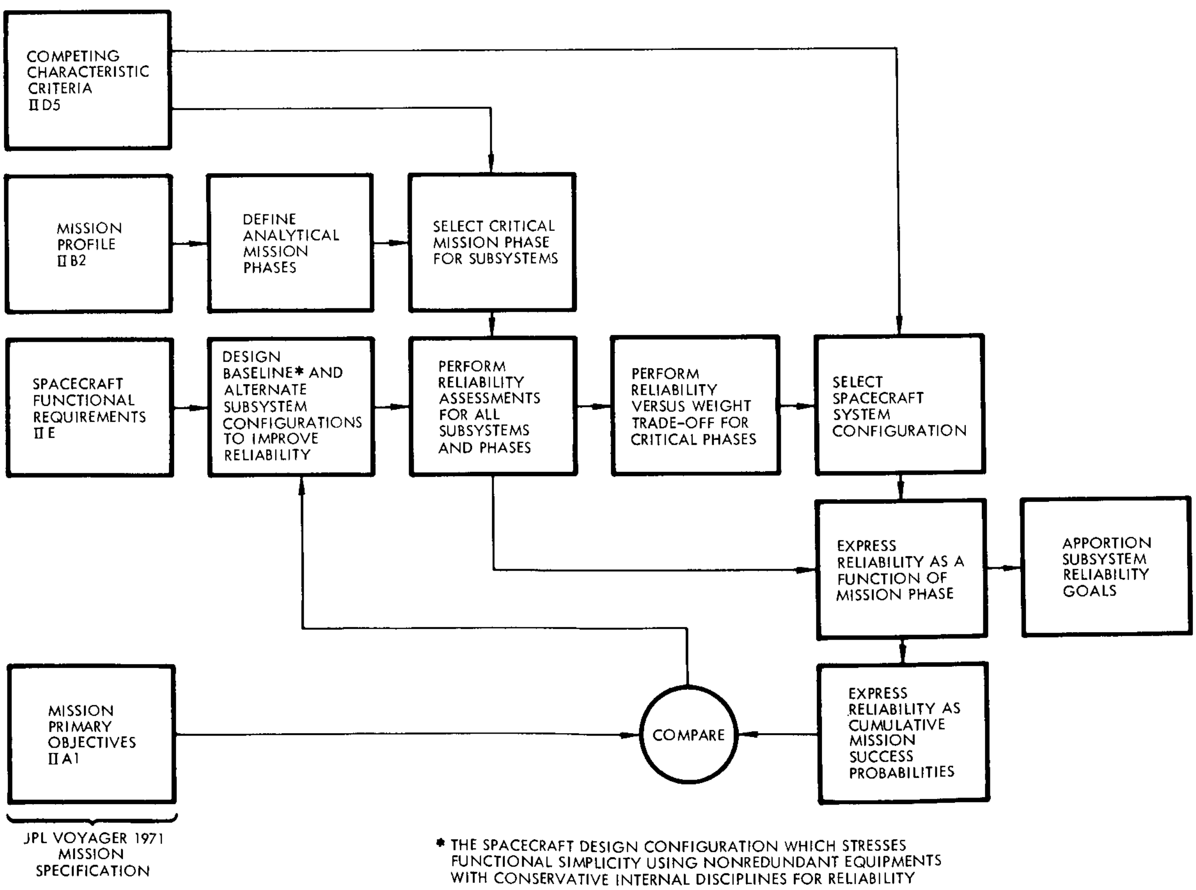


Figure 1. Reliability Apportionment and Analyses

events for d. The specification of reliability design objectives for the OSE, consistent with the priority of criterion b, is presented in Section III of Volume 6.

3.4 Mission Phase Design Objectives

3.4.1 Boost and Injection

The requirements established for successful boost and injection survival (for phase 1 and criterion b) are based upon specific events scheduled to occur within this phase (i. e., ordnance functions, etc.) and the survival of the complex of spacecraft equipment found necessary to accomplish subsequent-phase functions. Probabilistic models depicting these risks are discussed below in 3. 6.

3.4.2 Critical Phase

The specification of reliability design objectives for most spacecraft subsystems is based upon reliability versus weight tradeoff analyses using the probability of successfully completing phase 2. Phase 2 is used as the criterion because of its dominant influence upon the successful completion of mission functions of high priority. The reliability of subsystem elements pertinent to mission phase 1 are apportioned reliability requirements commensurate with their status as conditional factors for the success of phase 2 mission operation.

3.4.3 Mars Contamination

In general, it is foreseen that the criticality of the Mars contamination constraints does not interface significantly with the reliability objectives established for the Voyager spacecraft and pertain predominantly to the capsule design and handling techniques (see Volume 3, Section V).

3.5 Failure Modes and Criticality

3.5.1 Failure Mode Potential

The established Voyager flight sequence of events as given in VS-3-104 constitutes an event-by-event identification of failure mode potentials for the flight spacecraft. The criticality of these events for the spacecraft is dependent on their relation to the mission phases defined above and the corresponding competing characteristic criteria.

3.5.2 Mission Criticality

In general, each major subsystem, with the exception of the retro-propulsion system, is equally critical to the cumulative probability of success for phases 1 and 2. This leads to the selection of these phase reliability levels as design-critical tradeoff criteria for most subsystems. Failure mode recognition within individual subsystems is documented in terms of the reliability models constructed and discussed below. In non-redundant 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.

3.5.3 Effect on Redundancy

In the Voyager spacecraft preliminary design, significant levels of equipment redundancy are employed. 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 fore-seen 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.

3.6 Reliability Models and Distributions

3.6.1 Exponential Representation

Reliability models for Voyager system and subsystem analyses were based upon an assumed exponential representation for all nonredundant electronic equipment called upon to operate over extended periods. In these instances the electrical stress levels were found to be capable of design control so that electronic equipment failure rates were found to be essentially the same in their unenergized state as in their energized state. Thus, the projected probabilities of survival for the equipment is 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 redundance 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.

3.6.2 Redundancy Representations

Reliability functions of a nonexponential form were used to characterize redundant configurations at equipment level. These R(t) functions were derived (except for phase 1 and other one-shot events) from the equivalent parallel redundant models using exponential model constituents. In all instances where equipment redundancy entailed the use of sensing and switching functions with significant risk magnitudes, model adjustments were made to include series-risk increases.

3.6.3 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, was achieved by dividing the R(t) probability for the time of phase termination by the R(t) probability for completing the preceding phase of operation.

3.6.4 Phase 1 Failure Rate Modification

The reliability models used to depict probability of success for phase 1 are based upon a modified interpretation of the exponential R(t) function. Recourse is taken in the estimation of a significantly increased environmental ambient for all equipment for a short phase 1 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.

3.6.5 Time-Independent Probabilities

Reliability models pertinent to selected elements of the Voyager system have been based upon the binomial distribution. Analysis of one-shot devices and other actuation (rather than time dependent) probabilities has been treated in this manner. Selected use was also made of reliability assessment models where survival is a function of cumulative events rather than time. These probability numerics are related to the established mission phases in accordance with their time of expected occurrence throughout the flight sequence for the Voyager mission.

3.6.6 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 the solar array such modeling is necessary. In this case the power-reserve characteristic of mission phase 2 is not limiting but the long cruise condition is a significant factor for possible cell failures. In contrast, mission phase 5 places more power demand on the solar array, thus lowering the threshold to failure and significantly reducing the probability of surviving a given period of orbital operation. For identified wearout modes, 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.

3.7 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 are discussed as part of the iteration of system design for reliability given as Section IV of Volume 4. The part that this process plays in the determination of subsystem reliability objectives is illustrated in Figure 1.

3.8 Command, Switching and Redundancy

One area of system planning for reliability pertains to the selection of alternate spacecraft functional modes and alternate equipment operation (from a redundant set) for a given functional mode. Particular attention has been given to the problem of recognizing functional modes which are conditional to the reliability of high priority modes. Thus, it was foreseen that while downlink telemetry is given a high priority among the competing characteristic criteria, the spacecraft subsystem configuration necessary to assure this function will entail a level of redundancy which must be managed (functionally sensed, commanded and switched) or simultaneously energized. This in turn places a conditional-probability emphasis upon either on-board redundancy management or 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 telecommunications receiver and the spacecraft central sequencing and command subsystem. As indicated above, the reliability risks entailed in the management of redundancy have been incorporated in the reliability models established for spacecraft subsystems.

3.9 Environments and Derating Policies

3.9.1 Operating Temperatures

The thermal environment conditions assumed are based upon estimated capabilities of the thermal control subsystem using a conservative upper limit. The nominal operating temperature assumed for all electronic equipments is $50^{\circ}C$. This is an estimate for the maximum sustained temperature for which there is an effectively constant failure rate over the mission environmental profile.

3.9.2 Electronic Equipment Derating

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, viz. Mars radiation levels.

3.9.3 Structures

Mechanical structures are analyzed for reliability potential using a conservative margin of safety commensurate with the specific preliminary designs developed for the spacecraft frame.

3.9.4 Mechanical and Pneumatics

Mechanical and pneumatic systems analyses are based upon estimated degradations as a function of mission time so that critical factors such as reserve strength or gas supply are accounted for as probability parameters pertinent to all critical mission phases for the elements involved.

3.10 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 reliability ramifications of this policy are productive in three significant areas.

a) Each integrated circuit function incorporated is judged to be more amenable to the circuit tolerance controls characteristic of integrated circuit technology. See Appendix C of Volume 4.

b) Weight reductions inherent in a transition to integrated circuit packaging have allowed equipment level redundancies within the constraints of subsystem weights for equivalent nonredundant discrete part assemblies.

c) Repetitive integrated circuit packages have afforded a means for planning statistical test verification under stress conditions directly representative of circuit usage and with cumulative sample sizes enhanced through integrated circuit standardization to approximately eight basic types.

4. SUBSYSTEM RELIABILITY APPORTIONMENTS

Using the ground rules outlined in Section 3 each mission-critical subsystem of the spacecraft is assigned a reliability objective. Also, each major element of the subsystems has been apportioned a reliability goal based upon the subsystem reliability models discussed in Section 5 below and the other factors shown in Figure 1. Table 1 lists the reliability objective summary for each major subsystem, and Table 2 provides a more detailed reliability objective listing for the principal elements of all subsystems. Both tables give reliability numerics as appropriate in the four categories described below.

4.1 Probability of Launch Survival

Spacecraft subsystems must be capable of surviving the launch period (defined as mission phase 1) with its excursion into intense environments for a short interval. Liftoff through two Saturn stage ignitions and two Centaur stage ignitions represent the principal environmental influences. Functioning of ordnance devices and their associated circuits constitute additional risk elements within phase 1. Equipment elements must demonstrate the ability to meet these stress requirements.

4.2 Probability of Surviving Cruise

The extended time associated with spacecraft cruise (representative mission of 4280 hours) necessitates a corresponding design objective. The establishment of a success probability objective in conjunction with a specific time interval is necessary in lieu of an MTBF objective to cover the general case when failure distributions are not exponential.

4.3 Probability of Actuation

One-shot devices are allocated reliability objectives in terms of the probability of single-event success. The retropropulsion motor is the one major element in this category.

4.4 Wearout Life

The useful life of all subsystems is limited by the shortest expected life of any element within it. The maximum mission time or accumulated actuations will dictate the lower limit objective for known wearout phenomena. To assure the controlled absence of wearout modes during a composite period of spacecraft testing and flight operation, a 3-sigma (lower limit) time is established for each subsystem design.

Table 1. Apportioned Voyager Spacecraft Reliability Objectives for Major Subsystems

	Subsystem	Probability of Launch Survival	Probability of Surviving 4280 Hours	Probability of Successful Actuation	Assured Wearout Life 3σ Lower Limit
1.0	Mechanical, Ordnance, and Separation	0.99742	0.97698	0.99964	—
2.0	Thermal Control	0.99999	0.99939	—	12,000 Hr
3.0	Telecommunications	0.99824	0.96637	—	12,000 Hr and 75,000 record actuations
4.0	Power	0.99883	0.98991	—	12,000 Hr
5.0	Central Sequencing and Command	0.99997	0.99251	—	12,000 Hr
6.0	Electrical Distribution	0.99849	0.97840	—	12,000 Hr
7.0	Stabilization and Control	0.99134	0.97549	Covered by simulated use for 4280 hours	12,000 Hr
8.0	Propulsion	0.99831	0.99514	0.97416	12,000 Hr
9.0	Science Support	0.99876	0.98238	0.99995	12,000 Hr

Table 2. Apportioned Voyager Spacecraft Reliability Objectives for Subsystem Elements

Subsystem		Equipment	Launch Survival	Probability of Surviving 4280 Hours	Probability of Successful Actuation	Assured Wearout Life 3 σ Lower Limit
1.0 Mechanical, Ordnance and Separation	1.1	Spacecraft Frame	0.99742	0.99998	—	—
	1.2	Launch Vehicle Separation System	—	—	0.99982	—
	1.3	Spacecraft/Capsule Base Jettison	—	—	0.99982	—
	1.4	Meteoroid Protection	—	0.07700	—	—
2.0 Thermal Control	2.1	Louvers	0.999997	0.999536	—	12,000 Hr
	2.2	Heaters and Thermistats	0.999999	0.999851	—	12,000 Hr
3.0 Telecommunications	3.1	S-Band Receiver	0.99963	0.99331	—	
	3.2	Data Handling Unit	0.99999	0.99771	—	12,000 Hr and 75,000 record actuations
	3.3	S-Band Transmitter	0.99887	0.97885	—	12,000 Hr
	3.4	Capsule Receiver	0.99975	0.99618	—	12,000 Hr
	3.5	1-Watt Transmitter	0.99849	0.97845	—	12,000 Hr
4.0 Power	4.1	Solar Array	0.99900	0.99900	—	12,000 Hr
	4.2	Shunt Regulators				
	4.3	Power Control Units	0.99983	0.99090	—	12,000 Hr
	4.4	Batteries				
	4.5	Battery Regulators				
	4.6	Inverters				
5.0 CS&C	5.1	Input Decoders				
	5.2	Command Decoders	0.99997	0.99251	—	12,000 Hr
	5.3	Programmers				
	5.4	Power Converters				
6.0 Electrical Distribution	6.1	Cables, Connectors, Etc.	0.99849	0.97840	—	12,000 Hr
	6.2	Command Distribution Unit				
7.0 Stabilization and Control	7.1	Control Electronics				
	7.2	Gyros and Electronics				
	7.3	Sun Sensors and Electronics	0.99134	0.97549	Covered by Simulated Use for 4280 Hours	12,000 Hr
	7.4	Canopus Sensor				
	7.5	Reaction Controls				
8.0 Propulsion	8.1	Retroproulsion	—	—	0.97421	—
	8.2	Midcourse Propulsion	0.99831	0.99519	—	12,000 Hr
	8.3	Emissive Marouver Propulsion	—	0.99995	0.99995	—
9.0 Science Support	9.1	Planet Oriented Package	0.99876	0.98238	—	12,000 Hr
	9.2	Magmatometer Deployment	—	—	0.99995	—

5. SUBSYSTEM RELIABILITY MODELS

Figures 2 through 13 show the reliability models representing the preliminary spacecraft design. The redundant configurations established as a basis for the selected design are illustrated. Numerical reliability levels are shown for essential subsystem components and the appropriate mathematical reliability function is given for each subsystem (or subsystem element) reliability model. These models are the basis for the reliability apportionments given in Section 4, and the interpretation of subsystem reliability capabilities in terms of the Voyager mission objectives, as given in Section 7.

6. ALTERNATE MODE CAPABILITIES

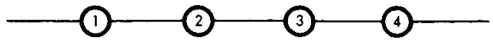
The construction of reliability models required the recognition of alternate spacecraft operational modes. Table 3 summarizes the alternate modes accounted for in the reliability models given in Section 5.

7. RELIABILITY BY MISSION PHASE

Based upon the reliability objectives for subsystem elements, the probability of successfully completing the individual and cumulative mission phases can be determined as shown in Tables 4 and 5, respectively. In the latter table, a comparison is made between the capabilities of the preliminary spacecraft design and the mission reliability objectives as established in Section II A 1 of the JPL Preliminary Voyager 1971 Mission Specification.

8. RELIABILITY DEMONSTRATION REQUIREMENTS

The planning of reliability demonstration tests for individual subsystem components can utilize their MTBF objectives as appropriate statistical design hypothesis. Table 6 provides a summary of such MTBF values as determined directly from their failure rate (λ) characteristics as given in the subsystem reliability models in Section 5.



	COMPONENT (i)	RELIABILITY (i)	CRITICAL PHASE
1	SPACECRAFT FRAME	0.99742	1
2	LAUNCH VEHICLE SEPARATION	0.99982	1
3	SPACECRAFT/CAPSULE BASE JETTISON	0.99982	3
4	METEOROID PROTECTION	0.9770	2

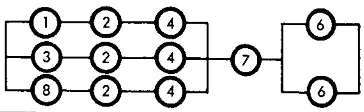
$R(t) = R_1R_2R_3R_4$ for all phases

Figure 2. 1.0 Mechanical Ordnance and Separation

	COMPONENT (i)	RELIABILITY (i)	CRITICAL PHASE
1	LOUVERS	0.999586	2
2	HEATERS AND THERMOSTATS	0.999851	2

$R(t) = R_1R_2$ for all components

Figure 3. 2.0 Thermal Control Subsystem



	COMPONENT (i)	λ_i FAILURE RATE, PARTS PER 10^9 HOURS
1	ANTENNA, 2 GIMBAL	8,100
2	DIPLEXER	250
3	ANTENNA, OMNI	1,170
4	RECEIVER COMPONENT	17,228
6	COMMAND DETECTOR	5,412
7	RECEIVER SELECTOR	1,209
8	ANTENNA, 1 GIMBAL	4,670

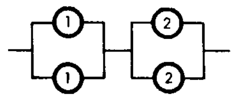
$R(t) = (1 - Q_A Q_B Q_C) R_7 (1 - Q_6^2)$

where $Q_A = 1 - R_1 R_2 R_4$

$Q_B = 1 - R_3 R_2 R_4$ $\quad Q_6 = 1 - R_6$

$Q_C = 1 - R_8 R_2 R_4$ and $R_i = \exp(-\lambda_i t)$

Figure 4. 3.1 S-Band Receiver



	COMPONENT (i)	λ_i FAILURE RATE, PARTS PER 10^9 HOURS
1	DIGITAL TELEMETRY UNIT	6633
2	BULK DATA STORAGE	8110

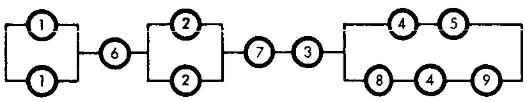
$R(t) = (1 - Q_1^2)(1 - Q_2^2)$

where

$Q_1 = 1 - R_1$ $\quad R_i = \exp(\lambda_i t)$

$Q_2 = 1 - R_2$

Figure 5. 3.2 Data Handling Unit



	COMPONENT (i)	λ_i FAILURE RATE, PARTS PER 10^9 HOURS
1	MODULATOR EXCITER	5,034
2	POWER AMPLIFIER	15,313
3	CIRCULATOR SWITCHES (2)	500
4	DIPLEXER	250
5	ANTENNA, 2 GIMBAL	8,100
6	FOUR-PORT HYBRID	250
7	TRANSMITTER SELECTOR	2,921
8	CIRCULATOR SWITCH (1)	250
9	ANTENNA, 1 GIMBAL	4,670

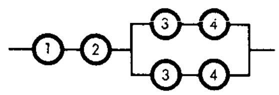
$R(t) = (1 - Q_1^2) R_6 (1 - Q_2^2) R_7R_3 (1 - Q_AQ_B)$

where

$Q_A = 1 - R_4R_5$ $\quad Q_1 = 1 - R_1$ $\quad R_i = \exp(-\lambda_i t)$

$Q_B = 1 - R_8R_4R_9$ $\quad Q_2 = 1 - R_2$

Figure 6. 3.3 S-Band Transmitter



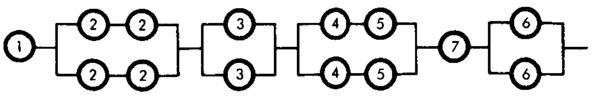
	COMPONENT (i)	λ_i FAILURE RATE, PARTS PER 10^9 HOURS
1	VHF ANTENNA	500
2	PREAMPLIFIER	340
3	RECEIVER COMPONENT	1760
4	DEMODULATOR	1480

$R(t) = R_1R_2 (1 - Q_A^2)$

where

$Q_A = 1 - R_3R_4$ $\quad R_i = \exp(-\lambda_i t)$

Figure 7. 3.4 VHF Capsule Receiver



	COMPONENT (i)	λ_i FAILURE RATE, PARTS PER 10^9 HOURS
1	SOLAR ARRAY	NOT APPLICABLE, RELIABILITY = $R_1(t)$
2	SHUNT REGULATOR	260
3	POWER CONTROL UNIT	13,520
4	BATTERY	7,200
5	BATTERY REGULATOR	4,605
6	INVERTER (SET)	4,048
7	SWITCHING DEVICE	480

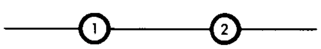
$R(t) = R_1 (1 - Q_A^2)(1 - Q_3^2)(1 - Q_B^2)(1 - Q_6^2)$

where

$Q_A = 1 - R_2^2$ and $R_i = \exp(-\lambda_i t)$

$Q_B = 1 - R_4 R_5$

Figure 8. 4.0 Power Subsystem



	COMPONENT (i)	λ_i FAILURE RATE, PARTS PER 10^9 HOURS
1	CABLES AND CONNECTORS	2,700
2	COMMAND DISTRIBUTION UNIT	2,340

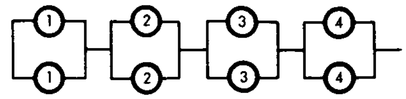
$R(t) = R_1 R_2$ where $R_i = \exp(-\lambda_i t)$

Figure 10. 6.0 Electrical Distribution Subsystem

	COMPONENT (i)	RELIABILITY R_i	CRITICAL PHASE
1	RETROPROPULSION	0.97421	3
2	MIDCOURSE PROPULSION	0.99519	2

$R(t) = R_1 R_2$ for all phases

Figure 12. 8.0 Propulsion Subsystem

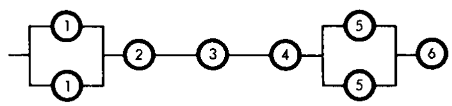


	COMPONENT (i)	λ_i FAILURE RATE, PARTS PER 10^9 HOURS
1	INPUT DECODER	1,622
2	COMMAND DECODER	13,370
3	SEQUENCER	11,760
4	POWER CONVERTER	6,265

$R(t) = (1 - Q_1^2)(1 - Q_2^2)(1 - Q_3^2)(1 - Q_4^2)$

where $Q_i = 1 - R_i$ $R_i = \exp(-\lambda_i t)$

Figure 9. 5.0 Central Sequencing and Command



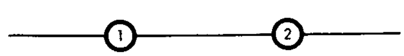
	COMPONENT (i)	λ_i FAILURE RATE, PARTS PER 10^9 HOURS
1	CANOPUS SENSOR	6,494
2	COARSE SUN SENSOR	980
3	FINE SUN SENSOR	790
4	GYROS AND ELECTRONICS	536 FOR 2% DUTY CYCLE
5	REACTION CONTROLS	9,103
6	CONTROL ELECTRONICS	15,368 FUNCTIONALLY REDUNDANT

$R(t) = (1 - Q_1^2) R_2 R_3 R_4 (1 - Q_5^2) R_6$

where $Q_1 = 1 - R_1$

$Q_5 = 1 - R_5$ and $R_i = \exp(-\lambda_i t)$

Figure 11. 7.0 Stabilization and Control



	COMPONENT (i)	RELIABILITY (i)	CRITICAL PHASE
1	PLANET ORIENTED PACKAGE	0.98238	3
2	MAGNETOMETER DEPLOYMENT	0.99995	2

$R(t) = R_1 R_2$ for all phases

Figure 13. 9.0 Science Support Equipment

Table 3. Identified Alternate or Functional Redundancy

Primary Function	Alternate Operating Mode	Reliability Model Effects
S-Band Receiver	Alternate antenna(s) with associated switches provide equal antenna characteristics for critical mission phases.	Shown as equivalent equipment redundancy when reliability differences are given.
S-Band Transmitter	Alternate antenna(s) with associated switches provide equal antenna characteristics for critical mission phases.	Shown as equivalent equipment redundancy when reliability differences are given.
Digital Telemetry Unit	Functional mode alternatives are provided at circuit level. Estimated effect redundancy, three to one.	Estimated effect as reduced equivalent failure rate by one-third. Exponential model assumed as conservative equivalent.
S-Band Transmitter	An auxiliary 1 watt transmitter is provided for Phase 1 and early Phase 2 operation. This is a back up capability for a short part of the total mission.	Reliability models do not show effect of this short period function as an improvement factor. The models are conservative to this extent.
Central Sequencing and Command	Various functional redundancies are recognized but depend upon specific S&C operations.	For reliability models no functional redundancy is shown as a conservative assumption.
Stabilization and Control	Alternate sequences and repeated trial features of some stabilization and control events are recognized. Extended midcourse corrections are effectively redundant control events. Circuit level redundancy is used for electronics.	Reliability models reflect a nominal actuation plan and are conservative in the omission of detailed back-up events. Circuit level redundancy is reflected as reduced failure rates.
Electrical Distribution	Only a fraction of all power circuit and command distribution controls are known to be mission critical. Estimates at 20 percent for connections and 25 percent for command events are based upon a study of "in-line" functions relative to all functions provided.	Reliability models are exponential but modified to reflect reduced equivalent failure rates relative to all "in-line" conditions.
Power Subsystem	Means are provided to evaluate battery charge states. These individual cell assessments will enhance overall reliability. The equipment to measure battery conditions is not included as in-line reliability risks.	Battery failures rates are based upon a per-cell risk modified to reflect its status sensing for charge control.
Thermal Control	Louver failures which represent spacecraft risks must fall into a pattern of adjacent control elements. This provides a protective feature equivalent to functional redundancy.	The estimated louver reliability reflects this consideration.
Propulsion	In the propulsion feed system a method of functional back-up is used where solenoid valves back up squib valves. This redundancy is particularly effective because of the alternate failure modes associated with the two basic part types.	The estimated propulsion system reliability reflects this part level redundancy.
Meteoroid Protection	The main shielding from meteoroid impact is provided by the external spacecraft surfaces. There is, however, a significant protection of mission critical equipment provided by all internal structural elements.	Reliability estimates for meteoroid protection are made to include the effect of secondary (internal) shielding effects.

Table 4. Reliability of Spacecraft by Individual Mission Phases

System Elements: Subsystem	Equipment		Mission Phases: 1 (0.3 Hours)	2 (4280 Hours)	3 (50 Hours)	4 (720 Hours)	5 (3600 Hours)
1.0 Mechanical Ordnance and Separation	1.0	Total	0.99724	0.97698	0.99977	0.999997	0.999995
	1.1	Spacecraft Frame	0.99742	0.99998	0.99995	0.999997	0.999995
	1.2	Launch Vehicle Separation	0.99982	-	-	-	-
	1.3	Spacecraft/Capsule Base Jettison	-	-	0.99982	-	-
	1.4	Meteoroid Protection	-	0.9770	-	-	-
2.0 Thermal Control	2.0	Total	0.999996	0.999387	0.999993	0.999897	0.999485
	2.1	Louvers	0.999997	0.999536	0.999995	0.999922	0.99961
	2.2	Heaters and Thermostats	0.999999	0.999851	0.999998	0.999975	0.999875
3.0 Telecommunications	3.0	Total	0.99824	0.96637	0.99867	0.99238	0.95448
	3.1	S-Band Receiver	0.99963	0.99331	0.99990	0.99843	0.98991
	3.2	Data Handling Unit	0.99999	0.99771	0.99995	0.99921	0.99435
	3.3	S-Band Transmitter	0.99887	0.97885	0.99887	0.99512	0.97345
	3.4	(VHF) Capsule Receiver	0.99975	0.99618	0.99995	0.99961	0.99613
4.0 Power	4.0	Total	0.99883	0.98991	0.99971	0.99222	0.97382
	4.1	Solar Array	0.99900	0.99900	0.99990	0.99500	0.99000
	4.2	Shunt Regulators					
	4.3	Power Control Unit					
	4.4	Batteries	0.99983	0.99090	0.99981	0.99721	0.98366
	4.5	Battery Regulators					
	4.6	Inverters					
5.0 Central Sequencing and Command	5.0	Total	0.99997	0.99251	0.99983	0.99740	0.98417
	5.1	Input Decoder					
	5.2	Command Decoder					
	5.3	Sequencer					
	5.4	Power Converters					
6.0 Electrical Distribution	6.0	Total	0.99849	0.97840	0.99974	0.99628	0.98135
	6.1	Cable, Connectors and Junction Boxes					
	6.2	Command Distribution Unit					
7.0 Stabilization and Control	7.0	Total	0.99134	0.97549	0.99912	0.99797	0.98101
	7.1	Control Electronics					
	7.2	Gyros and Electronics					
	7.3	Sun Sensor and Electronics					
	7.4	Canopus Sensor					
	7.5	Reaction Controls					
8.0 Propulsion	8.0	Total	0.99831	0.99519	0.97421	0.99968	0.99891
	8.1	Retro Propulsion					
	8.2	Midcourse Propulsion					
9.0 Science Support	9.0	Total		0.99995		0.99702	0.98510
	9.1	Planet Oriented Package		-		0.99702	0.98510
	9.2	Magnetometer Deployment		0.99995		-	-
Total System			0.98252	0.88047	0.97113	0.97313	0.86615
Voyager 1970 Specification (derived)		(Possible inclusion of launch vehicle reliability)	0.900	0.888	0.813	0.692	-

Table 5. Cumulative Mission Reliability by Mission Phase

System Elements			Cumulative Mission Phases				
Subsystem		Equipment	Phase 1 only	Phases 1 and 2	Phases 1, 2, 3	Phases 1, 2, 3, 4	Phases 1, 2, 3, 4, 5
1.0 Mechanical Ordnance and Separation	1.0	Total	0.99724	0.97428	0.97406	0.97406	0.97405
	1.1	Spacecraft Frame					
	1.2	Launch Vehicle Separation					
	1.3	Spacecraft/Capsule Base Jettison					
	1.4	Meteoroid Protection					
2.0 Thermal Control	2.0	Total	0.999996	0.999383	0.999376	0.999273	0.998758
	2.1	Louvers	0.999997	0.999533	0.999528	0.999450	0.999060
	2.2	Heaters and Thermostats	0.999999	0.999850	0.999848	0.999823	0.999698
3.0 Telecommunications	3.0	Total	0.99824	0.96468	0.96339	0.95605	0.91252
	3.1	S-Band Receiver	0.99963	0.99295	0.99284	0.99128	0.98128
	3.2	Data Handling Unit	0.99999	0.99770	0.99765	0.99686	0.99122
	3.3	S-Band Transmitter	0.99887	0.97774	0.97664	0.97187	0.94607
	3.4	(VHF) Capsule Receiver	0.99975	0.99593	0.99589	0.99550	0.99165
4.0 Power	4.0	Total	0.99883	0.98875	0.98845	0.98078	0.95510
	4.1	Solar Array	0.99900	0.99800	0.99790	0.99291	0.98298
	4.2	Shunt Regulators					
	4.3	Power Control Unit					
	4.4	Batteries					
	4.5	Battery Regulators	0.99983	0.99073	0.99053	0.98778	0.97164
	4.6	Inverters					
5.0 Central Sequencing and Command	5.0	Total	0.99997	0.99249	0.99233	0.98975	0.97409
	5.1	Input Decoder					
	5.2	Command Decoder					
	5.3	Sequencer					
	5.4	Power Converters					
6.0 Electrical Distribution	6.0	Total	0.99849	0.97692	0.97667	0.97304	0.95489
	6.1	Cable Connectors and Junction Boxes					
	6.2	Command Distribution Unit					
7.0 Stabilization and Control	7.0	Total	0.99134	0.96704	0.96619	0.96423	0.94592
	7.1	Control Electronics					
	7.2	Gyros and Electronics					
	7.3	Sun Sensor and Electronics					
	7.4	Canopus Sensor					
	7.5	Reaction Control					
8.0 Propulsion	8.0	Total	0.99831	0.99351	0.96789	0.96758	0.96653
	8.1	Retro Propulsion					
	8.2	Midcourse Propulsion					
9.0 Science Support	9.0	Total		0.99995	0.99995	0.99697	0.98212
	9.1	Planet Oriented Package					
	9.2	Magnetometer Deployment					
Total System			0.98252	0.86508	0.84010	0.81753	0.70810
Voyager 1971 Mission Specification		(Possible inclusion of launch vehicle reliability)	0.90	0.80	0.65	0.45	

Table 6. Summary of MTBF Values Determined from Failure Rate Characteristics

System Elements: Subsystem	Equipment	MTBF Objective
S-Band Receiver	Command Detector	184,774
	Receiver Selector	827,129
	Receiver Component	58,045
	1 Gimbal Antenna	214,132
	2 Gimbal Antenna	123,456
	Omni Antenna	854,700
S-Band Transmitter	Modulator Emitter	198,649
	Power Amplifier	65,303
	Transmitter Selector	342,348
	1 Watt Transmitter	198,649
VHF Capsule Receiver	Preamplifier	2,941,176
	VHF Receiver Component	568,181
	Capsule Demodulator	675,675
CS&C	Input Decoder	616,522
	Command Decoder	74,794
	Sequencer	85,006
	Power Converter	159,616
Power	Power Control Unit	73,964
	Shunt Regulator	3,846,153
	Battery	138,888
	Battery Regulator	217,155
	Inverters	247,035
Data Handling	Bulk Data Storage Unit	123,289
	Digital Telemetry Unit	150,761
Electrical Distribution	Command Distribution Unit	198,412
Stabilization and Control	Control Electronics	65,070
	Gyros and Electronics	37,285
	Sun Sensor, Coarse	1,020,408
	Sun Sensor, Fine	1,265,822
	Canopus Sensor	153,988
	Reaction Control (less valves)	109,853
Science	Planet Oriented Package (2 gimbal)	242,130

SPACECRAFT-LAUNCH VEHICLE INTERFACE

VS-3-130

CONTENTS

1. SCOPE

This document presents the functional and physical interface between the Voyager flight spacecraft and the Saturn IB-Centaur launch vehicle.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

JPL

V-MA-004-001-14-03	Project Document No. 45, Preliminary Voyager 1971 Mission Specification.

TRW 1971 Voyager Spacecraft Design Documents

VS-3-110	Configuration
VS-4-470	Electrical Distribution
VS-4-570	Launch Vehicle-Spacecraft Separation

3. REQUIREMENTS

3.1 General

The launch vehicle boosts and injects the planetary vehicle onto a proper earth-Mars heliocentric trajectory. It provides air conditioning for the spacecraft on the launch stand and umbilical cabling from the spacecraft to the launch facility. During atmospheric flight the launch vehicle protects the planetary vehicle from the aerodynamic environment by a fairing. The fairing incorporates a window to allow transmission of the spacecraft radio frequency link*. Separation of the spacecraft after injection is initiated by the launch vehicle.

*A window in the fairing to allow spacecraft radio transmission is a nominal solution suggested in the current study. The final interface definition will have to be made by JPL.

3.2 Space Vehicle Configuration and Geometry

3.2.1 Space Vehicle Arrangement

The Voyager planetary spacecraft in the installed position is placed at the forward end of the Saturn IB-Centaur launch vehicle and within the launch vehicle nose fairing. The general Voyager space vehicle arrangement is shown in Figure 1.

3.2.2 Reference Axes

The reference axes for the Voyager space vehicle and the relationship of these to axes for the spacecraft are shown in Figure 2.

3.2.3 Direction of Flight

To establish the direction of flight, the space vehicle pitches so as to fly over fin 1, which is shown in Figure 2.

3.3 Dynamic Envelope

The dynamic envelope to prevent interference with the launch vehicle nose fairing is shown in Figure 1 of VS-3-110.

3.4 Mechanical Attachment

The mechanical attachment between the launch vehicle and the flight spacecraft is a joint between the spacecraft and the Centaur stage mounting structure at station 2048. Nominal geometry and other features are shown schematically in Figure 3.

3.5 Electrical Interface

3.5.1 Spacecraft Umbilical

The spacecraft umbilical is a connector providing hardline connections between the spacecraft and the LCE. The associated cabling from the spacecraft to the launch facility is made through the Centaur umbilical connector as shown in Figure 3. The hardlines provide battery charging, critical function sense lines, etc. As the functions supplied through this connector are not required after liftoff, the umbilical will be disconnected quite late in the terminal countdown, with the disconnect initiated from the ground and verified prior to liftoff.

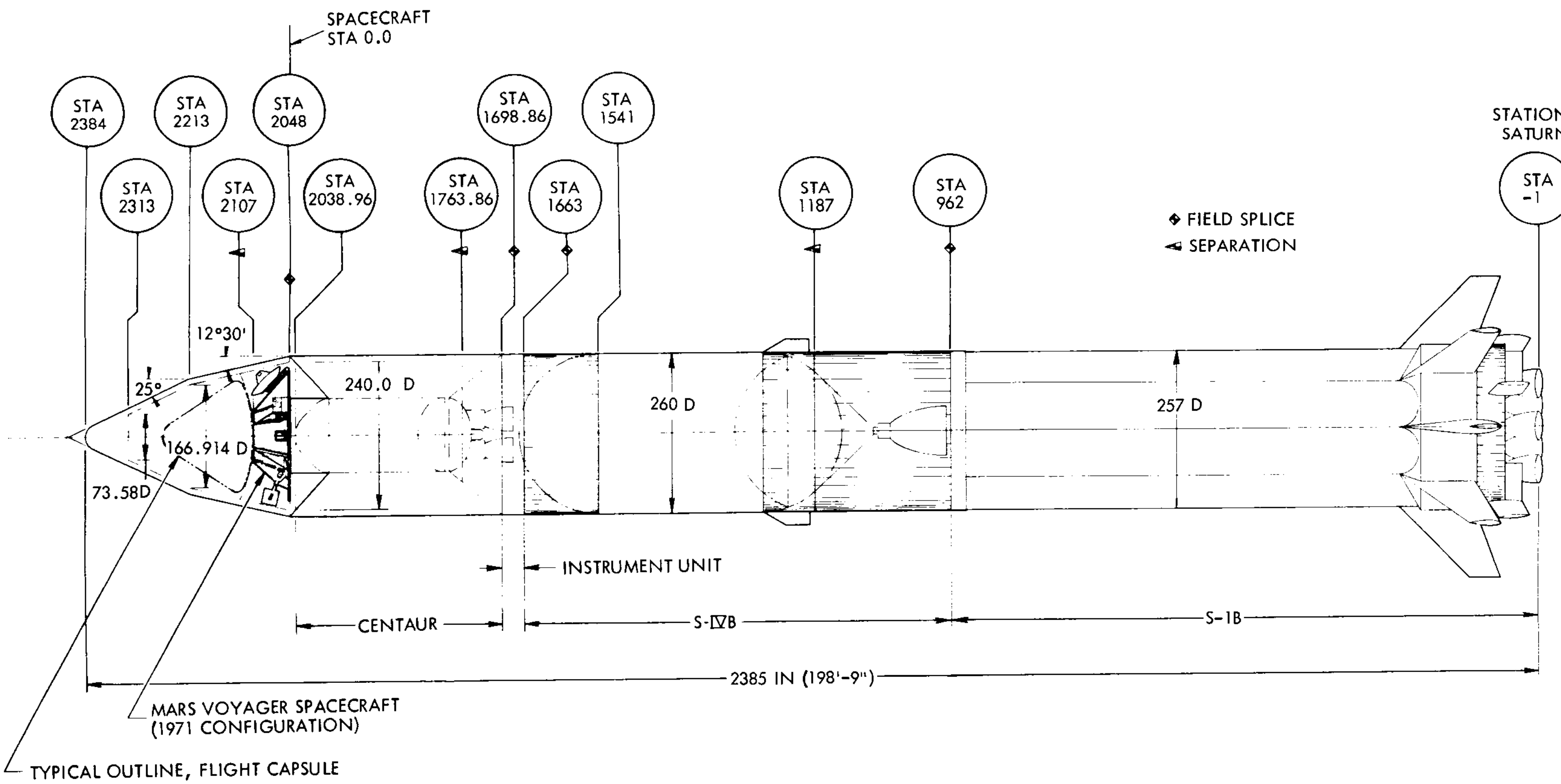


Figure 1. Voyager Space Vehicle Arrangement—Launch Configuration

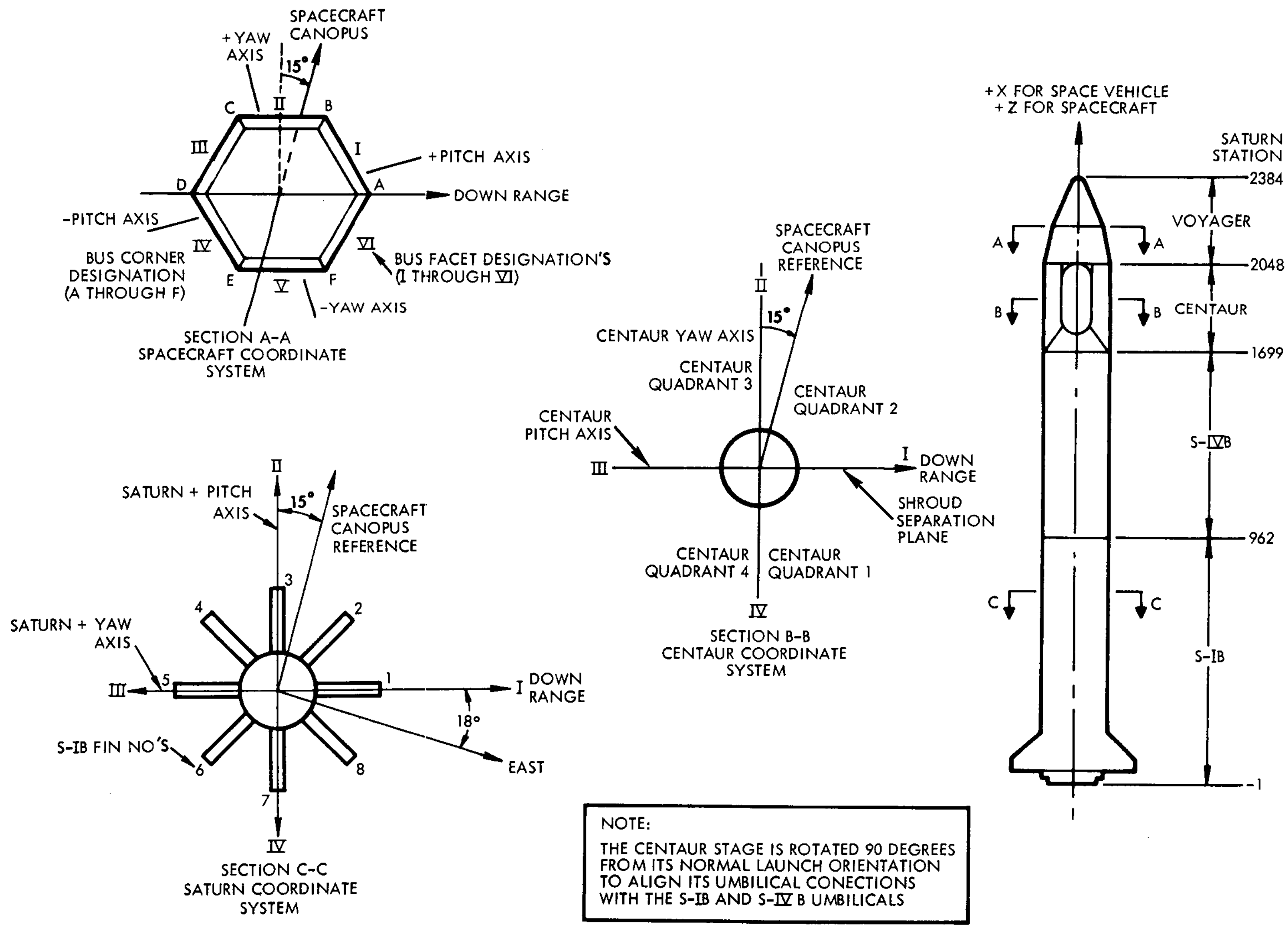


Figure 2. Reference Axes for the Spacecraft and Launch Vehicle

The spacecraft umbilical utilizes an explosively disconnected connector whose application places it in a Class B ordnance category, i. e., one whose inadvertent firing is not injurious to personnel or property. It will be necessary to stow the disconnected harness as shown in Figure 3 during powered flight.

Additional discussion of the electrical interface is given in VS-4-470.

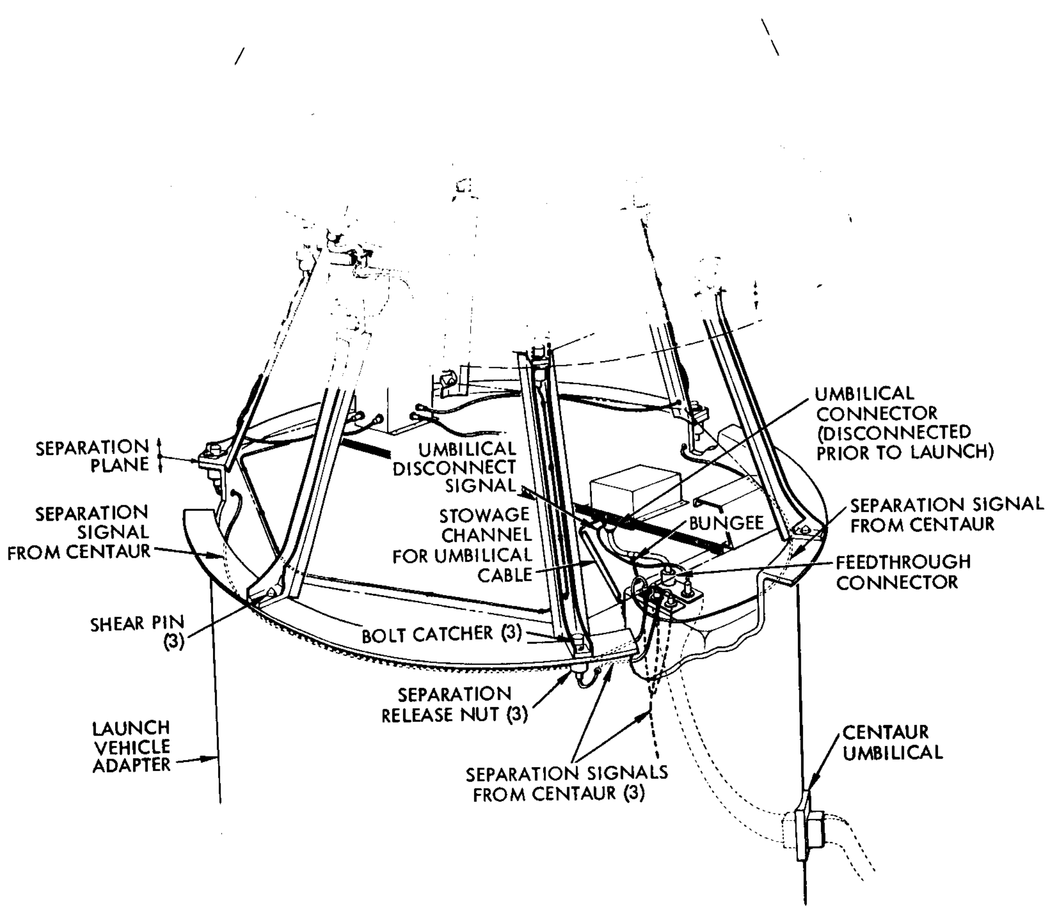


Figure 3. Launch Vehicle-Spacecraft Interface Schematic

3.5.2 Separation Signal Cabling

A separation signal is supplied by the launch vehicle at the appropriate time to fire three Class A explosive devices at the interface plane (1 amp no fire, 5 amp all fire for each device). The cabling and the three connectors for attachment to the explosive bolts, as shown in Figure 3, are to be supplied as spacecraft support equipment by the launch vehicle agency. The firing signal does not cross the interface plane.

3.5.3 RF Window

The presence of the nose fairing requires a window in the fairing or some other device to allow radiation of the spacecraft RF link when the nose fairing is in place.

3.6 Spacecraft Support

Spacecraft support is that equipment which is installed in the launch vehicle below the field joint interface to meet the requirement of a particular spacecraft. Although spacecraft support equipment is not now defined, the following may be required:

a) A launch vehicle adapter with mechanical and electrical fittings

b) Spacecraft destruct system, if required

c) Electrical cabling between the field joint and the launch vehicle.

The weight for spacecraft support is not to exceed 250 pounds.

3.7 Environment

3.7.1 Prelaunch

Prelaunch thermal and humidity control for the planetary vehicle are supplied by the launch vehicle system.

3.7.2 In-Flight Heating

The nose fairing provides protection from the aerodynamic environment and is not removed until suitable flight conditions have been achieved. The maximum temperature time-history of the nose fairing will be specified along with the material and geometry of the nose fairing configuration.

Heat rates to the spacecraft will not exceed those specified in Section II-F. 3. f (page 68), JPL Mission Specification.

3.7.3 Pressure Reduction

Venting during atmospheric flight is provided by the launch vehicle. The nominal ambient pressure versus time is given in Section II-F. 3. g (page 68), JPL Mission Specification.

3.7.4 Dynamic Environment

The dynamic environment to which the spacecraft is subjected during launch vehicle flight is given in Section II-F. 3. a-e (pages 65-67), JPL Mission Specification.

3.8 Cleanliness and Sterilization

The nose fairing and associated launch vehicle hardware must present clean surfaces consistent with the level of cleanliness to be achieved for spacecraft surfaces. The manufacturing processes should assure freedom from burrs or fragments which could be detached by shock or vibration and subsequently transferred to the spacecraft. Just prior to final assembly, all surfaces in contact with the spacecraft cavity are to be cleaned of dust and foreign matter by means of vacuum and wiping with appropriate solvents.

The nose fairing must be designed to prevent the possibility of outgassing or smoking of materials which could deposit on the spacecraft. It must also be assured that during shroud separation no particles or fragments will be set free which could come in contact with and contaminate the spacecraft.

3.9 Spacecraft Center of Mass

The center-of-mass limitation of the planetary vehicle in the boost mode configuration is a cylinder 3 inches in radius, with centerline on the vehicle roll axis, and with the upper end of the cylinder at vehicle station 2170.00. (Refer to Figure 1 for vehicle station locations.) With flight capsule center-of-mass and weight limitations as given in VS-3-140, this is equivalent to a maximum allowable offset of 3.6 inches for the center of mass of the flight spacecraft alone.

3.10 Ascent Mode

The parking orbit ascent mode is to be utilized for the Mars 1971 mission. An arbitrary limit of a 25-minute parking orbit presently exists for the 1971 mission; therefore, all vehicle equipment and expendables will be sized for this duration, and all performance calculations will be based upon this limitation. The minimum parking orbit coast time is to be 2 minutes or less. Later Voyager missions will require coast times in excess of 25 minutes.

3.11 Injection Accuracy

For preliminary design purposes, the miss plus time-of-flight dispersions of the planetary vehicle produced by the launch vehicle are to be correctable with a maximum 1-sigma midcourse velocity increment of 15 meters/sec applied 2 days after injection.

3.12 Separation

3.12.1 General

After achieving the proper velocity for injection into the desired earth-Mars heliocentric trajectory, the Centaur stage shuts down and initiates the spacecraft-launch vehicle separation operation.

3.12.2 Mechanism

The separation mechanism is a part of the spacecraft system and is described in VS-4-570.

3.12.3 Separation Signal

A "separation initiate" signal is to be generated by the launch vehicle as described in 3.5.2.

3.12.4 Angular Rates

The maximum angular rate at separation about any axis attributable to the launch vehicle is estimated to be 3 deg/sec.

3.12.5 Retromaneuver

After being separated from the spacecraft, the Centaur backs away from the spacecraft by employing retrorocket thrust. Retrorocket exhaust gas is not to impinge on the spacecraft. The retromaneuver capability is to have a magnitude and direction to satisfy the planetary quarantine requirement, and is to be performed at an appropriate time to meet the requirement for separation distance from the spacecraft.

SPACECRAFT-FLIGHT CAPSULE INTERFACE DESCRIPTION

VS-3-140

CONTENTS

1. SCOPE

This document presents the functional and physical interface between the Voyager flight spacecraft and the flight capsule.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-3-102	Maneuver and Accuracy Data
VS-3-104	Flight Sequence
VS-3-110	Configuration
VS-4-310	Communications
VS-4-311	Data Handling
VS-4-470	Electrical Distribution
VS-4-573	Spacecraft Separation

3. REQUIREMENTS

3.1 General

The flight capsule is combined with the flight spacecraft to form the planetary vehicle. Integration and support of the planetary vehicle during mission preparation is a part of the Voyager spacecraft system responsibility.

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 also provides the capability to serve as a communications relay for the capsule during entry and during Mars surface operations.

3.2 Geometry and Arrangement

The flight capsule is mounted forward of the flight spacecraft on the launch vehicle. The general arrangement and the capsule envelope is shown in Figure 1 of VS-3-110.

3.3 Mechanical Attachment

The mechanical attachment between the capsule and the spacecraft is a joint at spacecraft station 59.00. It involves six load transmission points with shear pins at three of these and explosive release nuts at the remaining three as shown schematically in Figure 1.

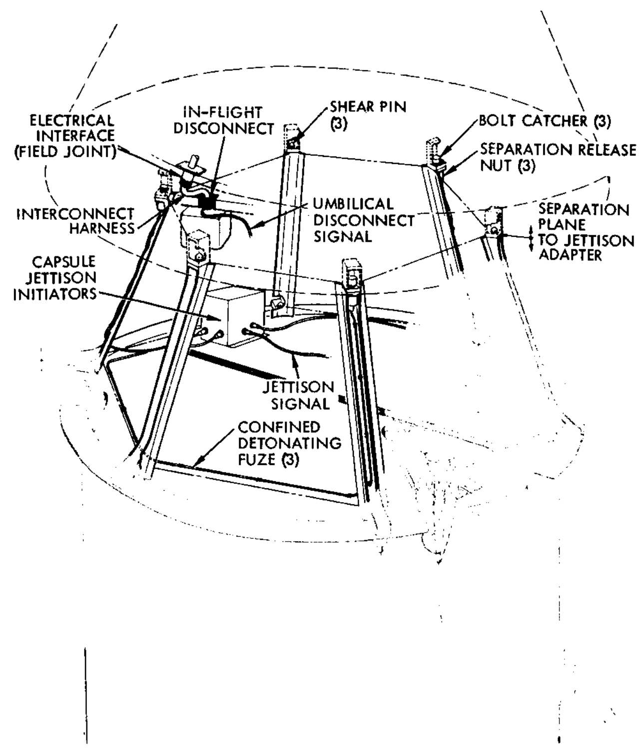


Figure 1. Capsule-Spacecraft Interface Schematic

3.4 Electrical Interface

3.4.1 Electrical Connector

An electrical connector is provided to accommodate umbilical functions between the spacecraft and capsule as shown in Figure 1. Additional information on this connector is given in 4.2 of VS-4-470.

3.4.2 Electrical Interference

Conducted or radiated energy between the flight capsule and the flight spacecraft systems must not interfere with the operation of either system.

3.4.3 RF Relay Link

An RF relay link is provided in accordance with 3.5.3.

3.5 Capsule Support

3.5.1 Data

The spacecraft bus is capable of accepting data at the rate of 10 bits/sec from the capsule. This is provided by hard line connection in a form compatible with the spacecraft telemetry for transmission to earth. Additional definition of this data is given in 4.1.4 of VS-4-311.

3.5.2 Commands

The spacecraft has the capability of transferring at least five commands to the flight capsule before separation. These commands may be spacecraft stored commands or ground commands transmitted through the spacecraft. In addition, a command is sent for separation of the capsule vehicle.

3.5.3 RF Relay Link

The flight spacecraft design provides a VHF relay receiver and a fixed low gain antenna. This equipment is capable of receiving post-separation data from the capsule vehicle up to and including impact at a rate of 10 bits/sec. This data will be handled and retransmitted to earth by the spacecraft communications system. Relay communications at a similar data rate are also possible after the lander is on the surface of Mars. Link design data is presented in VS-4-310.

3.5.4 Power

The spacecraft power system can supply quasi-steady DC power to the flight capsule during cruise at a level of zero to 200 watts at a voltage between 25 and 50 volts.

3.6 Environment

3.6.1 Thermal

The cruise orientation for the planetary vehicle is such that the flight capsule is pointed away from the sun. This orientation is maintained except for short duration trajectory adjustment maneuvers and just before capsule vehicle separation. The flight capsule is to be designed to minimize heat transfer to the spacecraft.

3.6.2 Dynamic Environment

The spacecraft-generated dynamic environment on the flight capsule will be defined when required.

3.7 Sterilization

3.7.1 Chemical Sterilization

All exposed planetary vehicle components must be capable of withstanding exposure to a gas mixture of 12 per cent ethylene oxide, 88 per cent freon gas for 10 hours at a relative humidity between 30 and 50 per cent.

3.7.2 Ejecta

All spacecraft ejecta such as attitude control jet gas must be biologically sterile.

3.8 Weights and Mass Properties

3.8.1 Flight Capsule Weight Allocation

Weight allocation for the 1971 flight capsule follows:

	Pounds
Flight capsule vehicle separated weight	1950
Flight capsule adapter and canister (A maximum of 150 pounds may remain with spacecraft)	350
Total flight capsule weight	2300

3.8.2 Flight Capsule Center of Mass

The flight capsule center of mass is to be within the envelope depicted in Figure 1 of VS-3-110.

3.8.3 Flight Capsule Moments of Inertia

The flight capsule moment of inertia about any of its principal axes and relative to its center of mass has a nominal value of 1600 slug ft^2.

3.9 Separation

The maneuver for separation of the capsule vehicle is discussed in VS-3-102. Additional information on the detailed sequence is given in VS-3-104.

3.9.1 Separation Mechanism

The mechanism for separation of the capsule vehicle from the capsule adapter/canister is within the flight capsule system area of responsibility.

3.9.2 Capsule Separation Disturbance

The angular velocity imparted to the spacecraft as a result of separation of the capsule vehicle is assumed to be less than 3 degrees/sec about each control axis.

3.9.3 Spacecraft Contamination

The capsule vehicle propulsive maneuver must not produce any injurious effects on spacecraft equipment such as contamination of science instruments or solar cell modules.

3.10 Jettison of the Capsule Adapter and Canister

At an appropriate time after separation of the capsule vehicle and after the spacecraft evasive maneuver but before firing of the spacecraft retropropulsion it is necessary to jettison the remaining capsule adapter and canister. The mechanism for accomplishing this is within the spacecraft bus system and is presented in VS-4-573. In the event of a failure such that the capsule vehicle does not separate, the spacecraft capsule adapter and canister jettison subsystem will jettison the complete flight capsule to allow the flight spacecraft to proceed with its orbital mission.

SCIENCE SUBSYSTEM INTEGRATION

VS-3-180

CONTENTS

1. SCOPE

This document presents the TRW-generated functional description and requirements for spacecraft science subsystem integration.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

<u>TRW 1971 Voyager Spacecraft Design Documents</u>

VS-4-210	Science Payload
VS-3-310	Communications
VS-3-311	Data Handling
VS-3-312	Tape Recorder
VS-4-450	Control Sequencing and Command
VS-4-460	Electrical Power
VS-4-470	Electrical Distribution
VS-4-510	Thermal Control
VS-4-550	Electronic Equipment Packaging
VS-4-571	Planet-Oriented Package

3. REQUIREMENTS

3.1 <u>General</u>

The spacecraft science subsystem consists of the hardware and software elements peculiar to achieving the spacecraft scientific objectives of a Voyager mission. It includes:

a) Spacecraft Science Payload (SSP)

b) Science Support Equipment (SSE)

c) Science Mission Dependent Equipment (SMDE)

3.1.1 <u>Spacecraft Science Payload</u>

The spacecraft science payload consists of:

a) Science experiment equipment (SEE)

b) Data automation equipment (DAE)

Science experiment equipment includes sensors and instrument electronics. SEE senses physical phenomena and provides suitable electrical indications to the DAE. SEE may be categorized according to the general objectives of the investigations as planetary observation and planetary and interplanetary environment observation. For the purpose of describing general physical properties of representative instrumentation, two divisions may be made for each instrument according to physical location: (1) primary sensor and directly associated hardware package, and (2) remote hardware. Remote hardware is typically all electronic and may be located in a common volume; the former may be individually located, as dictated by the requirements of the investigation, on a movable scan platform or at fixed locations on the spacecraft bus body.

Data automation equipment performs the functions of sequencing the science instruments, providing temporary storage of data, and sequencing stored information into the bulk data storage equipment of the spacecraft bus or into its telemetry system in a fixed format for transmission to ground receivers.

3.1.2 Science Support Equipment

Science support equipment for spacecraft assembly and checkout is used for conducting tests both before and after mating SEE with the bus. This SSE will be provided by JPL and combined with the bus test equipment in forming the system test complex and launch complex equipment.

3.1.3 Science Mission Dependent Equipment

Hardware and software for performing functions of science data handling and data analysis and for controlling experiments are required in the Deep Space Network for each function peculiar to the Voyager mission. The most likely place for SMDE is in the SFOF.

3.1.4 General Interface Considerations

Interface requirements include mechanical attachment, electrical attachment, adequate fields of view, and adequate isolation from the bus. Isolation includes radiation, vibration, static magnetic fields, static electric fields, etc. and includes various covers which must be removed as part of the flight sequence. These covers are on the bus side of the interface. In addition, the planet-oriented equipment requires support from the bus in the form of planet-sensing and pointing-control functions. Additional scan modes will be assigned to the science payload side of the interface.

3.2 Requirements on the Spacecraft

3.2.1 Mounting

The planetary and interplanetary observation SEE is body-mounted on the main structure or appendages, as shown in VS-3-110. The planetary observation SEE is mounted on an articulated planet-oriented package (POP) as described in VS-4-571. Remote hardware is panel-mounted as shown in VS-4-550.

3.2.2 View Angles

Mounting locations will be provided for sensors which provide unobstructed fields of view up to 1 steradian for sensors mounted on the main structure and approaching 4π steradians for sensors mounted on appendages.

3.2.3 Pointing

a. Planetary Observations

Means will be provided for pointing sensor axes in the selected directions with a precision of 0.5 degree, 3σ, and with an angular rate of less than 10^{-4} rad/sec. This pointing will be provided by the planet-oriented package in accordance with VS-4-571.

b. Planetary and Interplanetary Observations

The sensor axes for body-mounted instruments will be known to a 3σ accuracy of 3 degrees.

3.2.4 Alignment

Means will be provided to align axes of sensors mounted on the POP within 0.05 degree of their nominal directions (relative to spacecraft axes). Alignment remains within 0.25 degree throughout the mission. Detailed alignment data is given in VS-4-210.

3.2.5 Center of Mass

The spacecraft will be capable of compensating for changes in weights and locations of elements of the SEE equivalent to static balance changes of about $\pm$ 50 ft-lb around nominal.

3.2.6 Commands

In normal operation, commands for reading data, calibrating, and sequencing experiments are furnished to the science equipment by the data automation equipment. The spacecraft furnishes backup commands in accordance with VS-4-450. Basic science payload command requirements upon the spacecraft central sequencing and command subsystem (CS C) are given in Table 1.

Table 1. Science Command Sequencing Requirements Upon CS&C

Functions	Primary Requirements				
	Number of Direct Commands	Number of Quantitative Commands	Ground or On-Board	Data Rqmts. (bits)	Remarks
DAE Mode Control	4		Ground		
Sequence Time Update	-	1	Ground	14	
Orbit Period	-	1	Ground	14	
Impulse Commands	70 to 100	-	Ground		Includes Calibration Commands
POP Angles 1 & 2	-	8	Both	12	4 groups of 3 pictures
Camera Shutter Actuation	-	12	Both	-	
Magnification	2	-	Both	-	
Filter Selection	4	-	Both	-	4 positions assumed
Image Motion Compensation	-	8	Both	7	
Exposure Time	3	-	Both	-	3 positions assumed

3.2.7 Power

Power is to be a 50 volt peak to peak square wave at 4096 cps. The detailed characteristics are given in VS-4-460.

3.2.8 Intercabling

Interconnections will be made in accordance with VS-4-470.

3.2.9 Thermal Control

a. Temperature Range

The temperature range will be -25 to 175°F nonoperating and 0 to 140°F operating. Additional data is provided in VS-4-510.

b. Gradients

At sensor locations, temperature gradients will be controlled to 1°F per inch in precise interface regions and 10°F per inch in general.

3.2.10 Interference

a. Electrical Interference

Conducted and radiated interference limits will be specified.

b. Attitude Control and Exhaust Gas

The release of attitude control gases is controlled to minimize interference to instruments. Exhaust gases must not contaminate the scientific instruments with materials which will interfere with subsequent scientific measurements.

c. Static Field

The static electric field limits will be specified. The static magnetic field is given in Table 2.

Table 2. Static Electric Field Limits

Flight Sequence	Magnetometer Requirements		
	Alignment Prediction	Static Field	Field Stability
Cruise	3°, 3σ	1γ	$0.2\ \gamma$
Orbit	1°, 3σ	25γ	$5\ \gamma$

3.2.11 Protection

Adequate protective covers are provided to protect sensors from meteoroid flux or solar input during periods of nonoperation as required.

3.3 Requirements on the Voyager Spacecraft Mission Support System (SMSS)

3.3.1 Operational Support Equipment

EOSE of the SMSS makes provisions for mounting space, power, interconnecting cables, and environmental control for the electrical science support equipment. MOSE of the SMSS furnishes the precision mounting surfaces, fiducial marks, etc., necessary for enabling use of the mechanical science support equipment.

3.3.2 Test Complexes

Test complexes make provision for incorporating SSE.

3.3.3 MDE

Although science MDE will usually be furnished to the SFOF by the procuring agency and the MDE furnished by the spacecraft contractor will usually be furnished to the DSIF sites, the spacecraft MDE will include elements of the science MDE as required.

3.4 Requirements on the Spacecraft Science Subsystem

Equipment of the spacecraft science subsystem is designed to operate in a spacecraft system conforming to the requirements of 3.2.

3.4.1 Flight Hardware Packaging

Sensors and directly associated hardware are packaged as appropriate to the functions performed and to the spacecraft design. Remote hardware and data automation equipment are packaged in accordance with the requirements of VS-4-550.

3.4.2 Environment

Equipment of the spacecraft science subsystem will be designed in accordance with environments to be specified.

3.4.3 Telecommunications

Science data will be read into bulk storage or buffer storage or transmitted in real time in accordance with the requirements of VS-4-311 and VS-4-312.

SCIENCE PAYLOAD

VS-4-210

CONTENTS

1. SCOPE

This document describes the essential characteristics of a hypothetical spacecraft science payload. The instruments included are based on extensions of present space science instruments considered appropriate for the Voyager program. The intent has been to provide enough detail so that essential design integration items are identified within the framework of the selected spacecraft design. At the same time, an attempt has been made to achieve a reasonable balance among the competing desires of various experimenters and the allocated weight, volume, and power.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

JPL

V-MA-004-001-14-03	JPL 1971 Voyager Mission Specification
	JPL 1971 Voyager Mission Guidelines

TRW 1971 Voyager Spacecraft Design Documents

VS-3-180	Spacecraft Science Payload System Integration
VS-4-311	Data Handling Subsystem
VS-4-550	Electronic Packaging

3. FUNCTIONAL DESCRIPTION

The spacecraft science payload (SSP) will perform planetary observations and planetary and interplanetary environment observations for the 1971 Voyager spacecraft.

The nomenclature and characteristics of the SSP are in accordance with VS-3-180.

3.1 Experiment Observations

3.1.1 Orbit Characteristics

Planetary observations will be carried out from the spacecraft in orbit around Mars. For purpose of the present description, a sample orbit is used which has the following parameters:

- Periapsis altitude: 2000 km
- Apoapsis altitude: 20,000 km
- Period: 52,000 sec
- Inclination (Mars equator): 45 degrees
- Arrival condition: ZAP = 119°; $\gamma_p = -3^{\circ}$: $V_{\infty} = 3.25$ km/sec

The spacecraft science payload will be capable of operating for other orbits based on prelaunch selection and adjustment.

3.1.2 Planetary Observations

Planetary observations will accomplish the following:

a) Obtain photographs of the planet surface at a ground resolution of 1 km for mapping and approximately 0.05 km for surface analysis. 150 ft.

b) Make vacuum UV spectral scattering measurements and air glow measurements.

c) Make IR radiometric measurements for measuring temperature/emissivity properties of the planet surface.

d) Make IR spectral measurements for deducing atmospheric and surface properties.

e) Make vacuum UV and visible photometric measurements for aurora and meteor flash observations.

3.1.3 Planetary and Interplanetary Environment Observations

Observations carried out continuously (except during propulsion operations) in cruise and in orbit are the following:

a) Measure three components of the interplanetary magnetic field.

b) Measure three components of the planetary magnetic field.

c) Detect meteoroid impact, velocity, and composition for four directions of arrival.

d) Measure cosmic ray flux rate and energy spectrum for energies from 100 Mev to above 2 Kev for three directions of arrival.

e) Measure moderately energetic electron and proton flux rates and energy spectra for energies of 50 Kev to 5 Mev for electrons and 10 to 200 Mev for protons.

f) Measure the solar plasma flux.

g) Measure the doppler difference effects between 50 and 400 Mc. During earth occultation by Mars, evaluate the density profile of the Mars atmosphere. Consider making the same kind of measurement during earth occultation by the sun.

3.2 Data Automation Equipment (DAE)

3.2.1 General Operation

The DAE performs the function of sequencing the science instruments, provides temporary storage of data, and sequences the stored information into the bulk data storage or into the telemetry system in fixed formats.

3.2.2 Cruise Mode of Operation

Low rate science data is formatted into a single bit stream by the DAE to be interleaved with the spacecraft and capsule engineering data for transmission to earth. Science instrument engineering data may be formatted by the spacecraft data handling system, if desired. The data is stored temporarily within the DAE in small registers before being mixed with the spacecraft data. The formats and readout rates vary with spacecraft modes of operation and telecommunication system capability.

3.2.3 Solar Flare Mode

To obtain a profile of a "flare mode" in which it is important to obtain data from the fields and particle instruments at a higher sampling rate, a special data handling mode is provided. This mode is automatically enabled when a flow is sensed. In addition, a storage system is provided to accommodate the recording of a limited segment of this profile.

3.2.4 Orbital Mode of Operation

During the orbit phase, the DAE will sequence the cameras for photographic operations, control the readout of TV data into the spacecraft bulk storage, and relinquish control of the bulk storage to the data handling subsystem for telemetry readout. The DAE then controls the operations of the other science instruments and supplies low rate science data for interleaving with the high rate data. The data recorded in bulk storage

will include the TV data, frame and format identifiers, high rate scan instrument data, and associated identification data such as POP angles and camera exposure times. It is assumed that blank spaces will be left at the end of each picture to enable transmission of stored low rate scientific and engineering data. Additional information is presented in VS-H-311.
(Recorder may be stopped.

3.2.5 Instrument Calibration

Calibration sequences are initiated by means of periodic commands from the spacecraft bus sequencer or by real-time commands from ground-based transmitters. Upon receipt of a calibration command, the data automation equipment provides the calibration sequence which commands the science instruments through their calibrate mode.

3.2.6 General Physical Properties

The data automation equipment will be designed in accordance with the spacecraft packaging concept. For preliminary design purposes, the DAE is assumed to conform with the following maximum constraints:

- Volume: 1320 cubic inches
- Weight: 57 pounds
- Power consumption: 33 watts

3.3 Planet Observation Equipment

The planetary observation equipment makes observations from which properties of the planetary atmosphere and surface can be deduced. With the exception of the ionosphere experiment, these experiments all operate in the wavelength regions of 0.1 to 20μ. Longer wavelengths are probably not of interest because of the general clarity of the Martian atmosphere. Generally, these instruments cannot look closer to the sun than 45 degrees (except during eclipse).

3.3.1 Photography

The surface of Mars will be photographed with and without color filters to obtain moderate resolution (≈1 km) photographs for mapping and high resolution (≈0.05 km) photographs for investigation of surface features.

a. Mapping

Mapping photographs are taken at altitudes between 2000 and 3000 km, at lighting (sun-spacecraft-Mars) angles between 100 and 140 degrees. Conditions for high definition color photographs are altitudes near 2000 km and lighting angles near 180 degrees.

b. High Resolution Photography

Surface detail photographs will be taken (nested in the mapping photographs) simultaneously with mapping photographs. Image motion compensation prediction signals will be accepted, or image motion may be sensed and used to enhance resolution. As an alternate mode, high resolution photographs may be taken at higher altitudes for additional mapping coverage.

3.3.2 Atmospheric Scattering

The spectral radiance of the Mars atmosphere will be measured under all angles of sun illumination with vertical observations at the terminator and limb measurements most desired. The primary measurement band is 0.11 to 0.34μ. The purpose of the measurements is to determine upper atmosphere constituents, presence of aerosols, and aurora and air glow phenomena if trapped radiation is present.

3.3.3 Radiometry

A multi-channel scanning radiometer will be used to obtain maps of surface temperature, visual features, and H_2O distribution. Primary interest will be the terminator, with secondary interest in other sun illuminations and the dark side. The purpose of the experiment is to develop information about the surface temperatures and temperature gradients. The regions thermally mapped will be correlated or overlapped with the visual data and photographs on TV pictures.

3.3.4 Absorption Spectroscopy

The planet surface, particularly the terminator, will be viewed and the IR spectrum in the range of 0.7 to 20 μ will be measured. The purpose of these measurements is to deduce composition and pressure of the atmosphere and data about the surface reflectance properties. A secondary objective is to make direct absorption measurements viewing the sun through the atmosphere as the spacecraft enters and leaves eclipse.

3.3.5 UV Photometry/Meteor Flash

Band-limited observation in the vacuum UV will be employed to map out UV intensities over the planet atmosphere, including dark side polar regions. It is predicted that such measurements will provide good statistical data on the impact of meteoroids on a large volume of the planetary atmosphere.

3.4 Planetary and Interplanetary Environment Observation Equipment

3.4.1 Cruise

During cruise, the planetary observation equipment (POE) will be off and the planetary and interplanetary observation equipment will be on.

a. Magnetometer

The interplanetary magnetic field can be measured with a threshold of 0.25 gamma. It is expected that the dynamic range of the instrument will be $\pm$ 500 gamma. To achieve the necessary stability in the effects of the spacecraft magnetic field,it will be necessary to reduce the total field to less than 0.25 gamma at the magnetometer. It appears desirable to rotate the spacecraft 360 degrees about two of the three spacecraft axes several times during cruise and in Mars orbit to establish the magnitude of the spacecraft field.

b. Cosmic Ray Telescope

The energy, composition, and angular distribution of the high energy protons will be measured using three sensors. Each sensor consists of a solid state detector and a Cerenkov detector. Proton energies from 100 Mev to above 2 Bev and alphas from 400 Mev to above 8 Bev will be measured.

c. Plasma Probe

Solar plasma density and spatial distribution will be measured for the stream arriving at the spacecraft. Measured currents will be correlated with the magnetometer readings to study the mechanisms involved in the propagation of particles from the sun.

d. Meteoroid Impact (Cosmic Dust)

Although the impact rate is expected to be low, statistical data on the density and velocity distributions of meteoroids will be accumulated. In addition, the magnitude of particle velocity and the composition of the particles will be measured.

e. Trapped Radiation

Although primarily of interest for planetary environment measurements, intermediate energy electrons and protons will be measured in cruise also. Observation of particles of energy up to 200 Mev will be implemented for three directions of travel relative to the spacecraft.

3.4.2 Planetary Environment Observations

a. Magnetometers

Recent information indicates that the magnetic fields in the vicinity of Mars are very low. The magnetometer dynamic range from 0.25 to 500 gamma has been selected to cover these low levels and accommodate any solar flare activity.

The spacecraft field at the boom mounted, deployable magnetometer is expected to be less than 0.2 gamma. A second manetometer fixed at the edge of the spacecraft solar array is included to provide back-up and better orientation predictability of the sensor.

b. Trapped Radiation

Trapped particle radiation, if any, in the Mars orbit will be measured in the proton ranges from 10 through 200 Mev and in the electron ranges from 50 through 5 Kev, using three sensors.

c. Ionosphere Measurement

Two phase locked receivers, one at 50 mcs and the other at 400 mcs, will measure the phase advance and group delay of the low frequency signal compared to that of the higher frequency signal. In this way the integrated electron density and its time variation will be determined.

d. Meteoroid Impact

Cruise measurements of meteoroid impacts will continue in orbit about Mars.

3.5 Orbit Science Data Sequencing

3.5.1 Time History

After injection in orbit, the planetary observation equipment will be turned on. Typical time histories for data collecting and the relation to orbit variables are shown in Figures 1 through 3 for the sample orbit. Planetary and interplanetary environment observations will continue.

3.5.2 Telemetry Rates

The data collection sequences shown will be used when the telemetry rate is 2048 bits/sec. Adjustment to different rates will generally be made by changing the number of photographs per orbit. The nominal data rate is 4096 bits/sec allowing twice the data shown in these illustrations. However, the DAE will sequence the experiments in accordance with ground command to obtain the best coverage possible considering the needs of all experimenters.

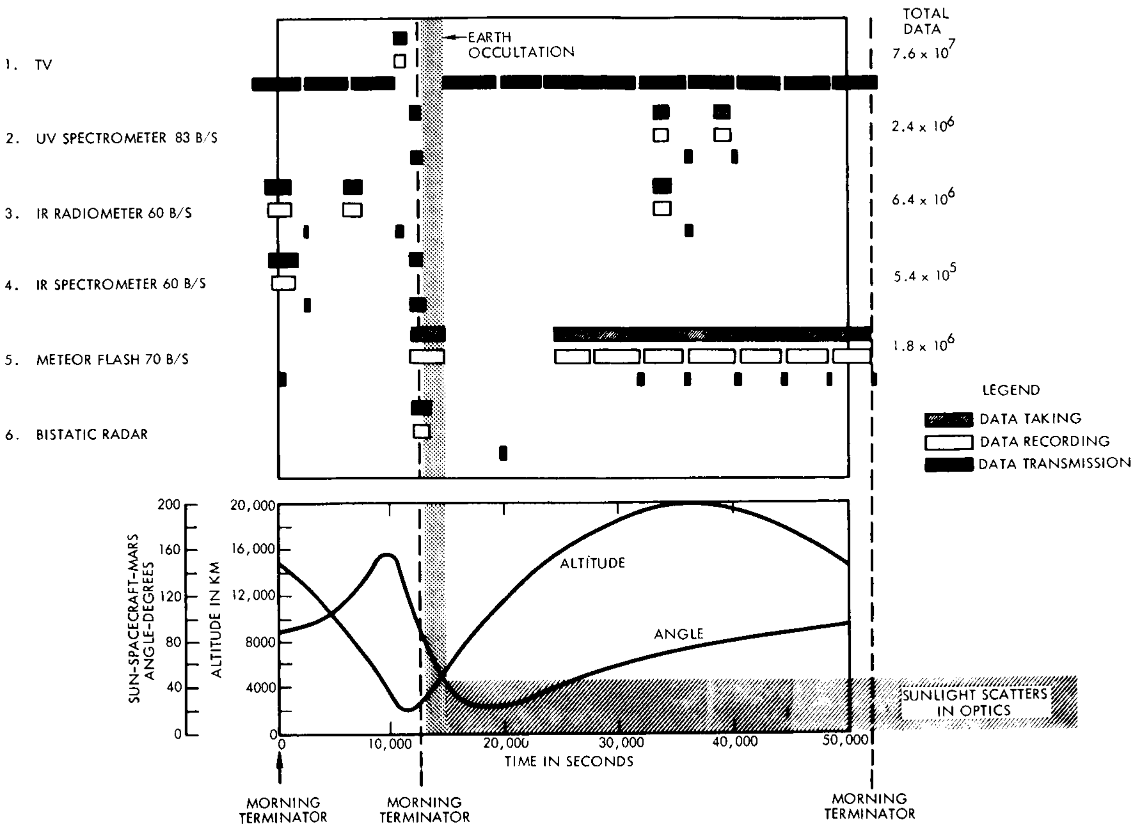


Figure 1. 12 November 1971 Sample Orbit Data Rate 2048 Bits/Sec

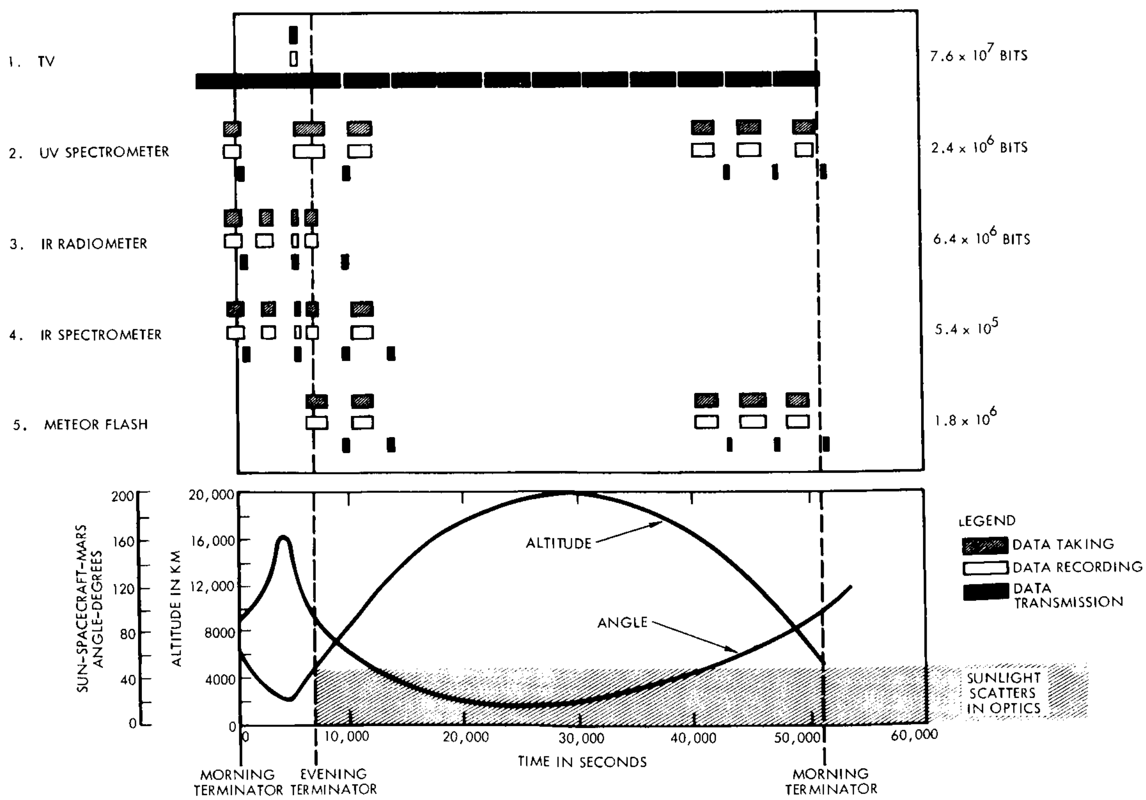


Figure 2. 10 February 1972 Sample Orbit Data Rate 2048 Bits/Sec

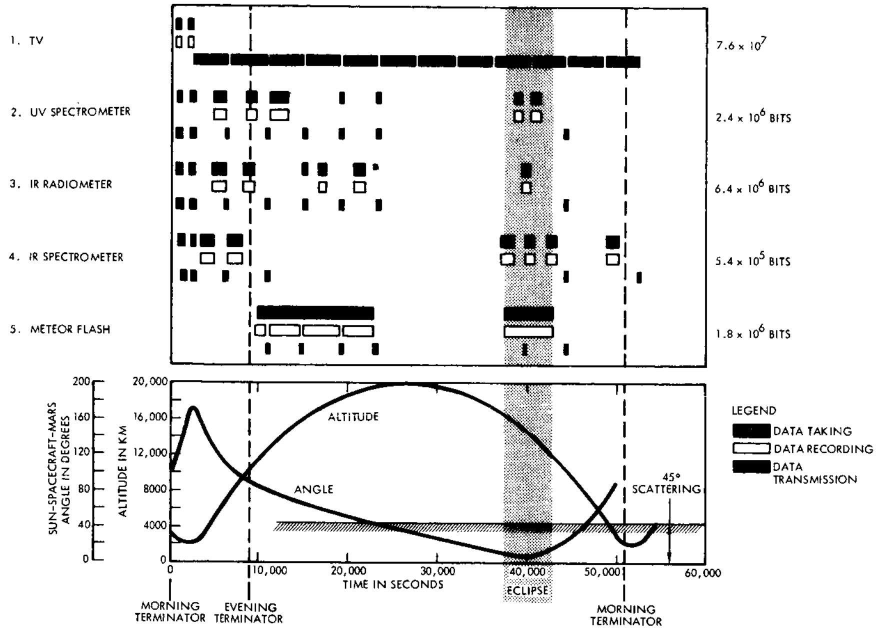


Figure 3. 10 May 1972 Sample Orbit Data Rate 2048 Bits/Sec

4. EXPERIMENT CHARACTERISTICS AND CONSTRAINTS

The science experiment equipment for planet observations will generally conform to the data set forth in Table 1 and the planetary and interplanetary environment observation equipment will generally conform to the data of Table 2. Additional detail regarding the science payload experiments is given in Figures 4 through 14 as follows:

Figure	Experiment
4	Photographic (TV) Experiment
5	UV Spectrometer
6	IR Scanning Radiometer
7	IR Spectrometer
8	Meteor Flash
9	Meteoroid Impact
10	Magnetometer
11	Plasma Probe
12	Cosmic Ray Telescope
13	Trapped Radiation Detector
14	Ionosphere Experiment

Table 1. Science Experiment Equipment Data

Experiment	Sensor FOV	Sensor Align-ment	Sensor No. Leads	Sensor Volume (in)^3	Sensor Weight (lb)	Sensor Power (watt)	Remote Hardware Volume (in)^3	Remote Hardware Weight (lb)	Remote Hardware Power (watt)	* Com-mand	Engr. Meas.	Data Rate (bits/sec)	Data Bits per Orbit
1) Mapping TV	9^o	0.1^o	40	3456	36**	7	4000	16	15	8D	7	163×10^3	1.54×10^8 (12 picture pairs)
High Resolution TV	0.4^o	0.1^o								3Q			
2) UV Spectrometer	$2\text{-}1/2^o$	1^o	12	2000	18	2	576	7	11	5D	4	1×10^3	2.4×10^6
3) IR Scan Radiometer	1 mr	0.1^o	15	1800	10	2	144	2	5	18D 1Q	4	4.3×10^3	6.4×10^6
4) IR Spectrometer	0.05 rad	0.5^o	12	1296	20	4	144	2	4	4D	4	4.8×10^3	4.8×10^6
5) Meteor Flash	30^o	3^o	8	216	5	1	144	2	4	3D	4	0.025×0.14	——

*D - Discrete Commands
Q - Quantitative Commands

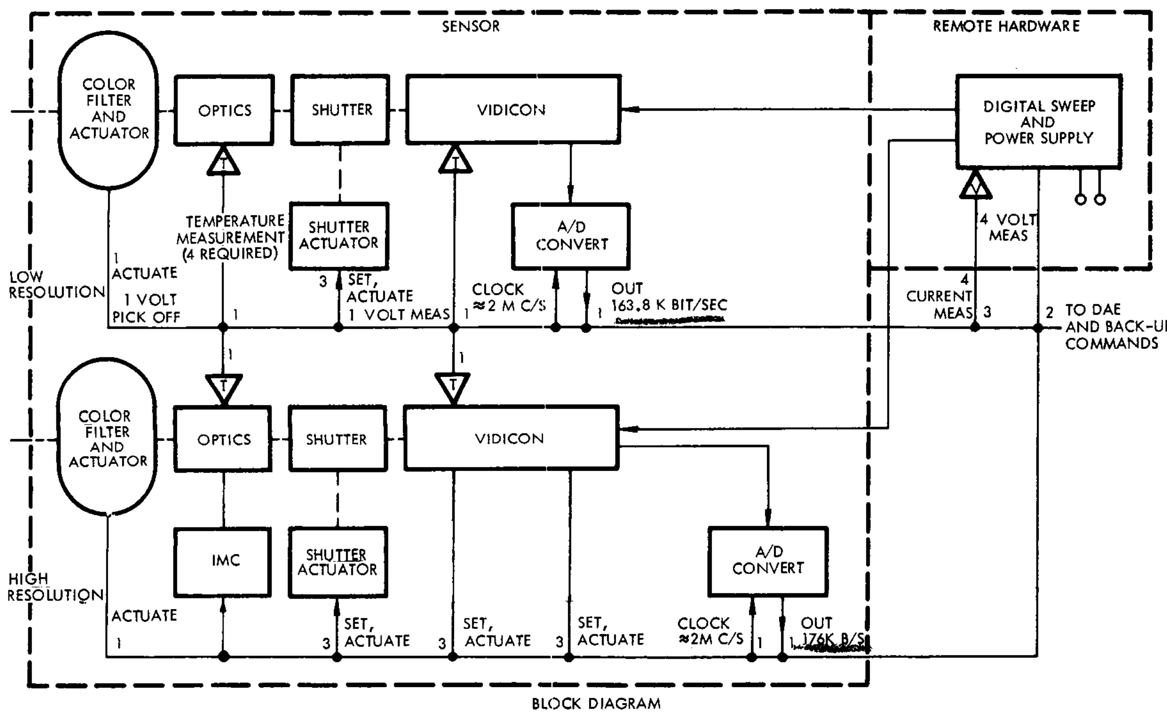
**High resolution optics, approx. 10 lb.

Table 2. Planetary and Interplanetary Environment Observation Equipment Data

Experiment	Sensor FOV(2)	Sensor Main Axis Clock°	Sensor Main Axis Cone°	Sensor Align-ment	Sensor No. Leads	Sensor In^3	Sensor Lb	Sensor Watts	Remote Hardware In^3	Remote Hardware Lb	Remote Hardware Watts	Com-mand	Engrg Meas.	Data Rates Quiet	Data Rates Flare
Magnetometer(1)															
a. Cruise	4π	0	0	3^o	10	180	1.3	1.8	96	5	5	3D, 1Q	3	6B/S	30B/S
b. Orbit	4π	0	0	1^o	10	180	1.3	1.8	96	5	5	3D, 1Q	3		
Cosmic Ray							3	-	384	5	5	4	8	0.4B/S	4B/S
Sensor A	45^o			5^o	6										
Sensor B					6										
Sensor C					6										
Plasma						192	4	0.6	160	5.5	5.4	4	6	1B/S	10B/S
Probe A	20^o	-	0	3^o	4										
Probe B		-	0	1^o	4										
Meteoroid						7560	6	0.8	144	10	1.2	4	8	10^{-3}B/S up to 0.1B/S	
Sensor A	10^o	-	0	1^o	5										
Sensor B		-	180		5										
Sensor C		90	90		5										
Sensor D		270	90		5										
Trapped Radiation						210	9	0.6	1560	12.5	4.4	4	8	2.5B/S	25 B/S
Sensor A	45^o	-	0	5^o	6										
Sensor B		-	180		6										
Sensor C		90	90		6										
Ionosphere	180^o	270	45	3^o	Coax	-	3	-	252	6.0	1.5	4	2	10B/S	

(1) Only one magnetometer may be provided, data for Helium magnetometer shown. See Figure 4-7b for data on toraxial fluxgate magnetometer.

(2) Total angles



CHARACTERISTICS

- FILTER 4 SECTION, CLEAR, SPECTRAL FILTERS F1, F2, F3
- OPTICS
  - LOW RESOLUTION 25 MM DIA 630 MM FL
  - HIGH RESOLUTION 150 MM DIA 100 MM FL
- SHUTTER SPEED ~ 0.1 SEC AT 400 FT LAMBERTS FOR CLEAR FILTER
- STORAGE VIDICON
  - FRAME READOUT 80 SEC MAX
- FORMAT
  - 1024 x 1024 ELEMENTS AT 6 BITS/ELEMENT
- DATA RATE 163.8 KB/SEC
- IMAGE MOTION COMPENSATION PROVIDED FOR HIGH RESOLUTION CAMERA BASED ON OPEN LOOP SYSTEM ACCURATE TO 5%

ELECTRICAL INTERFACE

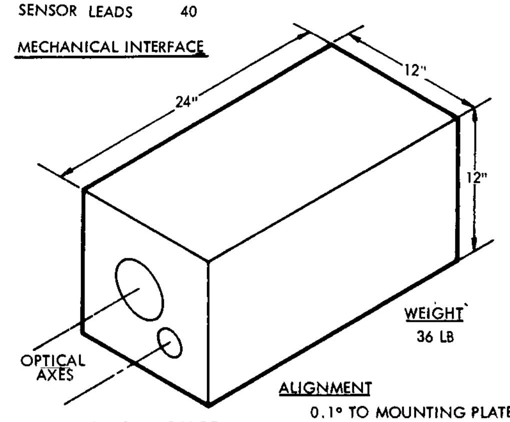
DAE COMMANDS	DISCRETE	QUANTITATIVE
ACTUATE FILTER	2	
SHUTTER		
SETTING		2 AT 3 BITS
ACTUATE	2	
POWER ON, OFF	2	
SWEEP CONTROL	2	
$\dot{x}$ IMC		6 BITS
$\dot{y}$ IMC		6 BITS

DAE SIGNALS	LEADS	
PICTURE SIGNALS	2	AT 6 BIT/WORD, SERIAL 163.8K BIT/SEC
SWEEP SYNC	2	
VERTICAL SYNC	2	
4 TEMPERATURE, ENGINEERING MEASUREMENTS		

POWER 30 WATTS REG,

SENSOR LEADS 40

MECHANICAL INTERFACE



WEIGHT 36 LB

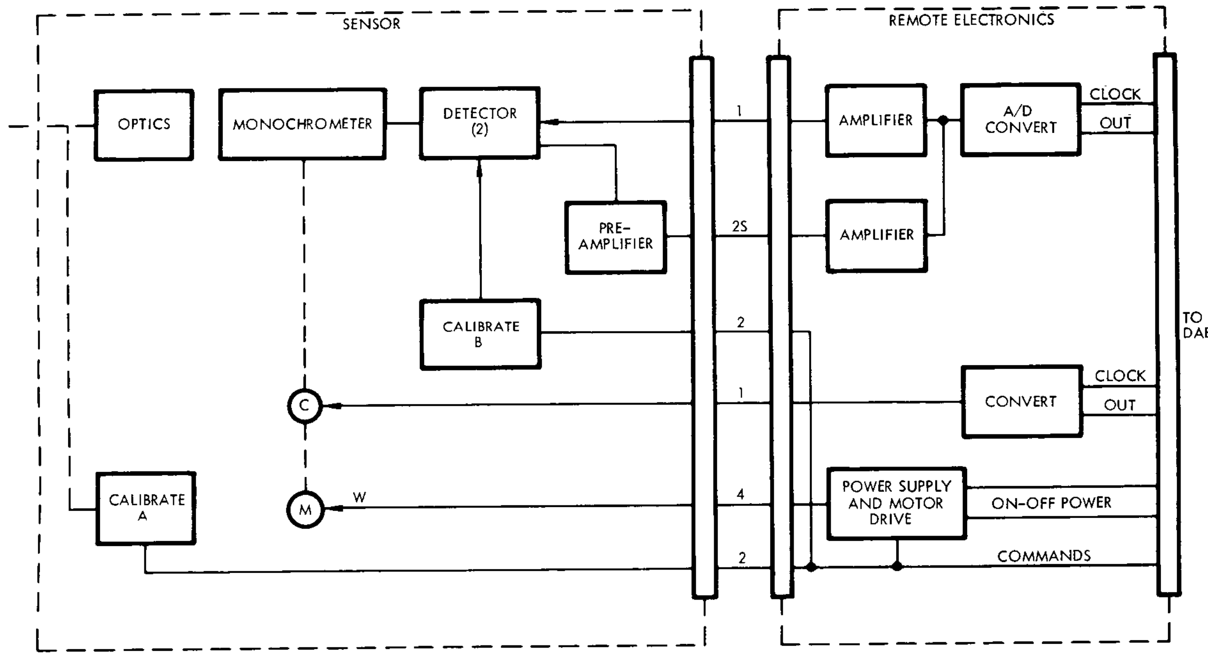
ALIGNMENT
0.1° TO MOUNTING PLATE
0.01° BETWEEN SENSORS

	VOL	WT
REMOTE ELECTRONICS	$4000\ IN^3$	16 LB

THERMAL INTERFACE; CONTROL GRADIENTS, VS-4-510

CONSTRAINTS
a) MAGNETIC
b) 1% LENS FLARE MAX
c) VS-3-180

Figure 4. Photographic Experiment Data Sheet



UV SPECTROMETER-FUNCTIONAL
BLOCK DIAGRAM

CHARACTERISTICS

- OPTICS: 5" x 5" CASSEGRAIN
- MONOCHROMETER: RANGE 0.11-0.34 μ IN 30 SEC
  RESOLUTION $10^{-3}\mu$
- FIELD OF VIEW: 2.5° x 0.025° APPROXIMATE
- TO MEASURE SCATTERING AND AIR GLOW. VIEW TERMINATOR FOR 25 MIN PER ORBIT
- DYNAMIC RANGE 2.5×10^{7} PHOTONS SEC^{-1} CM^{-2} $STERAD^{-1}$ TO 2.5×10^{14} PHOTONS SEC^{-1} CM^{-2} $STERAD^{-1}$
  (APPROXIMATE 300 TO 3×10^{9} RAYLIEGHS)
- SCATTERED LIGHT
  OUTPUT RESPONSE FALLS OFF APPROXIMATELY ONE ORDER OF MAGNITUDE PER 3 DEGREES OUTSIDE FIELD OF VIEW

ELECTRICAL INTERFACE

DAE COMMANDS

POWER ON-OFF	1	
SCAN SELECT	1	
SCAN ON-OFF	1	
CALIBRATE A	1	
CALIBRATE B	1	

DAE SIGNALS

DETECTOR OUTPUT	2	6 BIT WORD AT 1500 B/S TOTAL
WAVELENGTH OUTPUT	1	6 BIT WORD AT 500 B/S

2 TEMPERATURE, 2 VOLTAGE ENGINEERING MEASUREMENTS

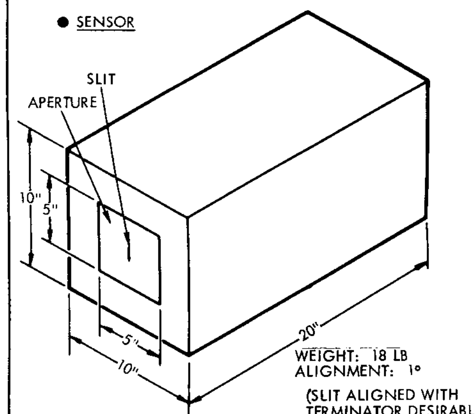
POWER 13 WATTS

2 WATTS IN SENSOR
11 WATTS IN REMOTE

SENSOR LEADS 12

MECHANICAL INTERFACE

- SENSOR



WEIGHT: 18 LB
ALIGNMENT: 1°
(SLIT ALIGNED WITH TERMINATOR DESIRABLE)

- REMOTE HARDWARE

VOL	576 $IN.^{3}$
WT	7 LB

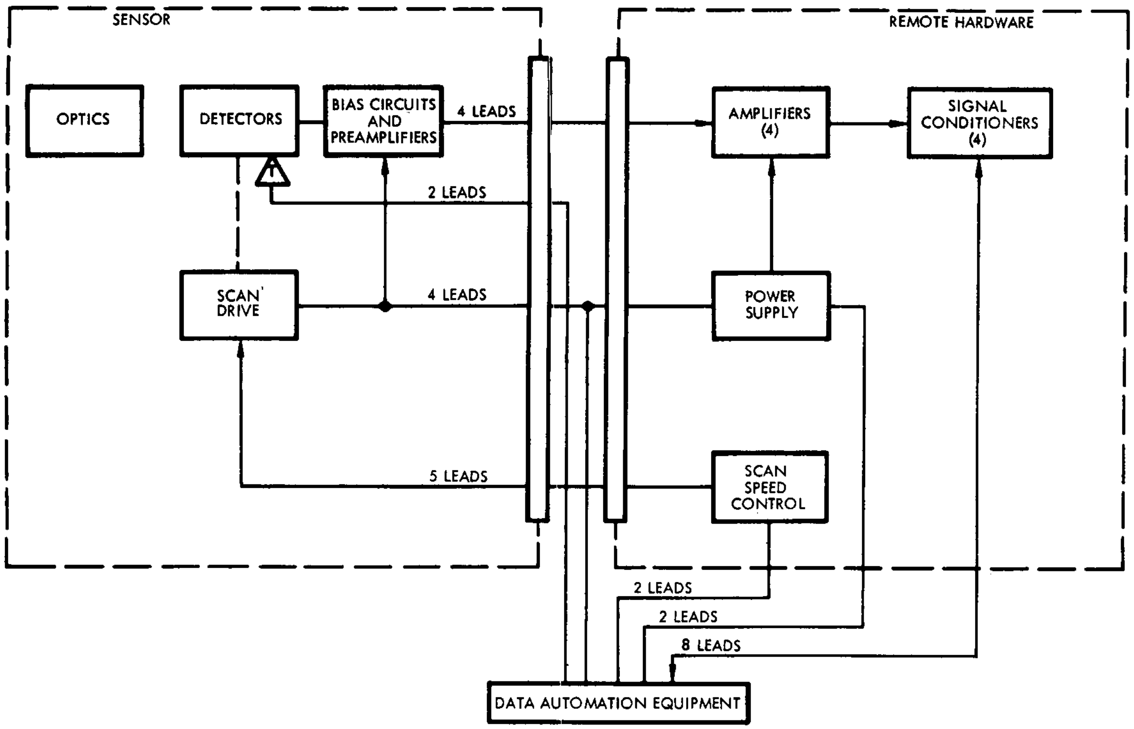
THERMAL INTERFACE:

CONTROL TEMPERATURE, V-4-510

CONSTRAINTS

1971 VOYAGER MISSION SPECIFICATION
VS-3-180

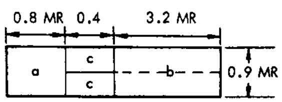
Figure 5. UV Spectrometer Data Sheet



4 CHANNEL
SCANNING RADIOMETER
FUNCTIONAL BLOCK
DIAGRAM

CHARACTERISTICS

SCAN DIRECTION



FIELD OF VIEW
a) 8-12 μ BOLOMETER
b) 2.55-2.65 μ, 2.32-2.45 μ PbS
c) 0.5-1.0 μ Si CELLS

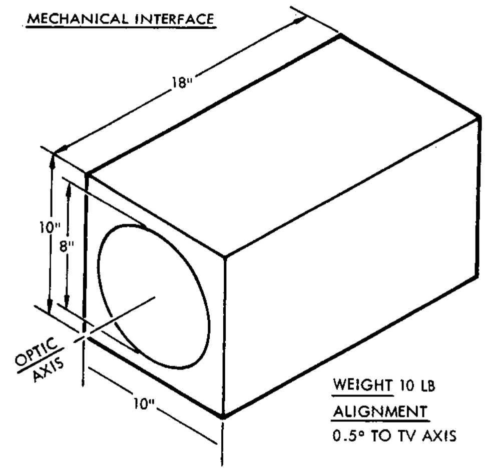
- SCAN 80 MR IN 0.72 SEC
- SLIGHT OVERLAP AT 3.330 KM TRACKING MARS CENTER
- OPTICS 8" CASSEGRAIN
- OUTPUT 4 SIGNALS 6 BIT WORDS
  DATA RATE 3460 BITS/SCAN
  OR 4300 BITS/SECOND
- SCATTERED LIGHT
  1% FLARE IN OPTICS, SUNLIGHT 45° OR MORE AWAY

ELECTRICAL INTERFACE	LEADS	QUANTITY
DAE COMMANDS		
POWER ON, OFF	2	
SCAN DRIVE	2	
CHANNEL CONTROL	4	
DAE SIGNALS		
DETECTOR SIGNALS	8	6 BIT WORDS
SCAN SYNC	2	
2 TEMPERATURE, 2 VOLT, ENGINEERING MEASUREMENTS		

POWER

7 WATTS REG — 2 WATTS SENSOR, 5 WATTS REMOTE HARDWARE

MECHANICAL INTERFACE



WEIGHT 10 LB

ALIGNMENT

0.5° TO TV AXIS

SENSOR

REMOTE HARDWARE 144 IN^3 2 LB

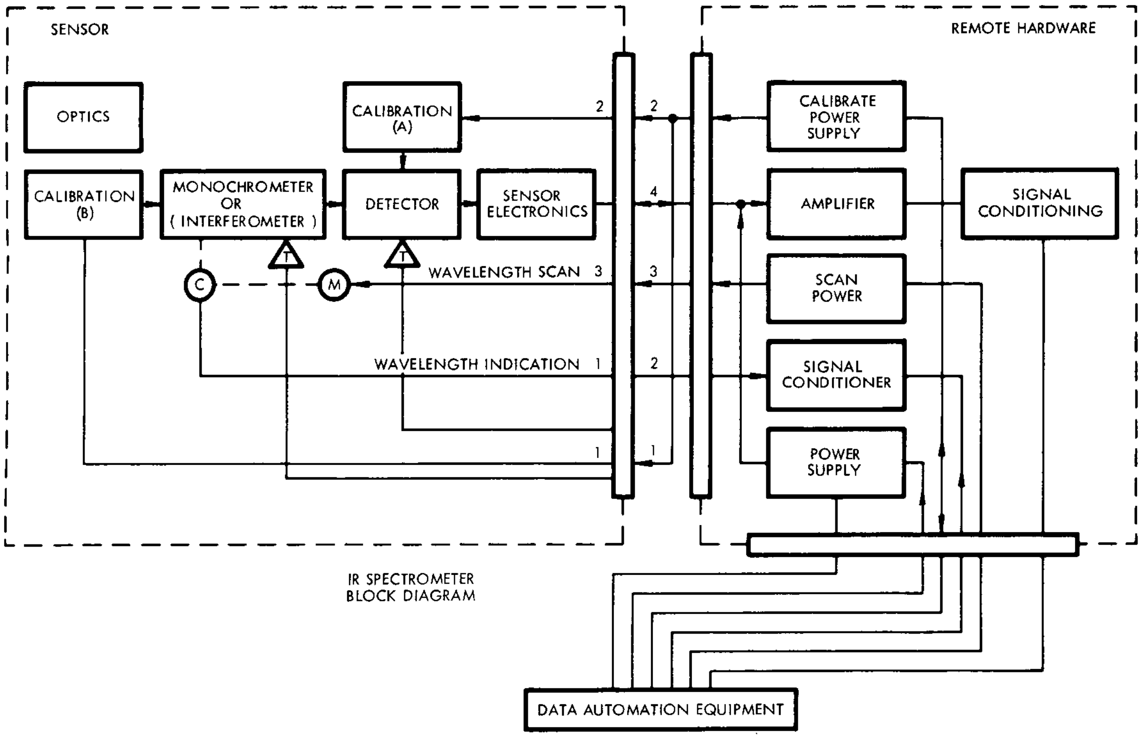
THERMAL INTERFACE VS-4-510

DETECTORS COOLED TO -40°C
(INSTRUMENT CAN RUN COLD)

CONSTRAINTS

VS-3-180
1971 VOYAGER MISSION SPECIFICATIONS

Figure 6. Four-Channel IR Scan Radiometer Data Sheet



IR SPECTROMETER
BLOCK DIAGRAM

CHARACTERISTICS:

- 6" DIA CASSEGRAIN OPTICS + MONOCHROMETER
- WAVELENGTH RANGE 0.7μ TO 20μ
- RESOLUTION: APPROXIMATELY 0.02μ
- FIELD OF VIEW $\simeq$ 1/20 RAD
- OUTPUT
  30 SEC PER SCAN 6000 DATA BITS (SPECTROMETER)
  20 MIN PER ORBIT AT 5000 BITS/SEC (INTERFEROMETER)
- OBSERVE TERMINATOR
  a. DAWN
  b. DUSK
- SECONDARY OBJECTIVE POLAR CAPS
- TERTIARY OBJECTIVE DAY SIDE AND DARK SIDE
- THERMISTOR BOLOMETER DETECTOR

ELECTRICAL INTERFACE

DAE COMMANDS — NO.

POWER ON-OFF	1
WAVELENGTH SCAN	1
CALIBRATE	2

DAE SIGNALS

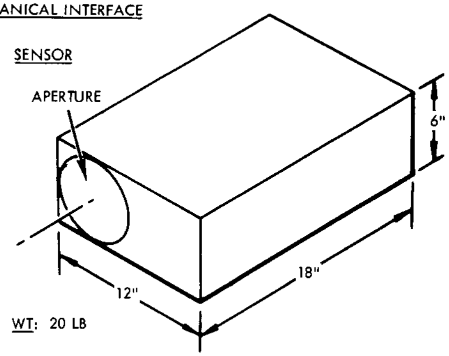
SIGNAL OUTPUT 6 BIT WORDS 200 BITS/SEC (SPECTROMETER)
2 TEMPERATURE, 2 VOLTAGE ENGINEERING MEAS
5000 BITS/SEC (INTERFEROMETER)

POWER 8 WATTS
4 SENSOR
4 REMOTE HARDWARE

SENSOR LEADS 12

MECHANICAL INTERFACE

SENSOR



WT: 20 LB

ALIGNMENT

1° TO PHOTOGRAPHIC

REMOTE HARDWARE

VOL 144 IN^3 WT 2 LB

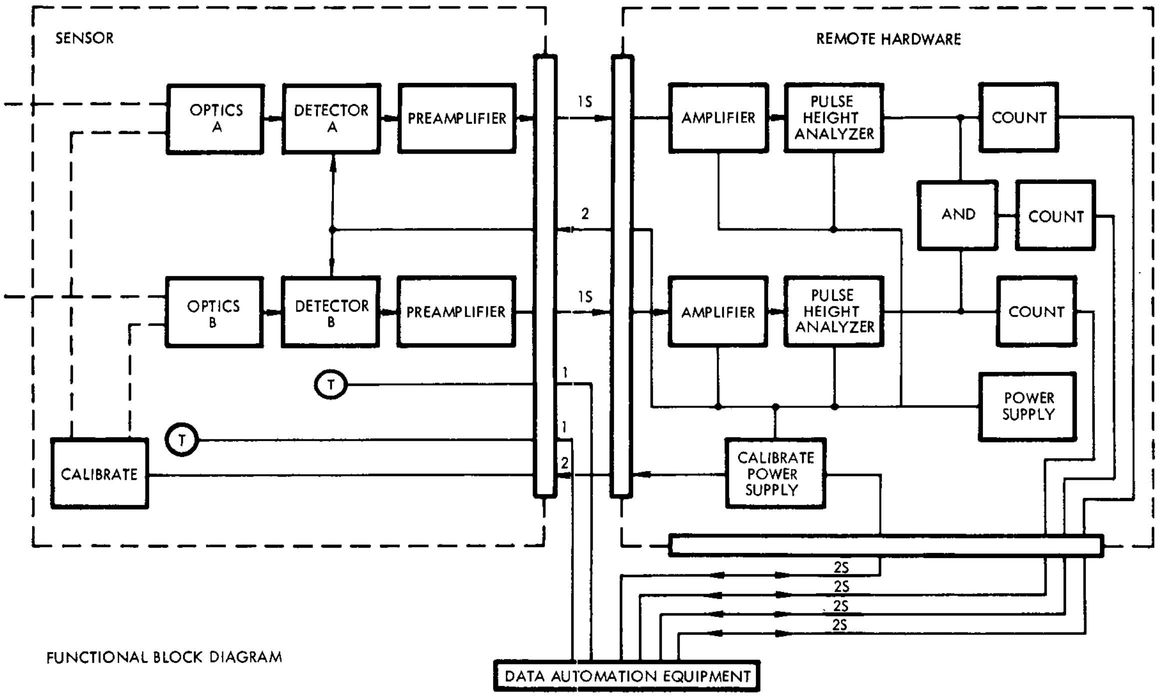
THERMAL INTERFACE VS-4-510

CONTROL TEMPERATURE APPROXIMATE 250°K

CONSTRAINTS

MAGNETIC

Figure 7. IR Spectrometer Data Sheet



FUNCTIONAL BLOCK DIAGRAM

CHARACTERISTICS

- TWO UV PHOTOMETER CHANNELS
  - a) 0.25μ - 0.3μ WAVELENGTH FOV - 30° AND 15°
  - b) 0.3μ - 0.45μ WAVELENGTHS FOV-30° AND 15°
- CALIBRATION MEANS
- OUTPUTS OF
  - METEOR FLASH DETECTION
  - PLANETARY RADIANCE
- YIELDS COUNT, SPECTRUM OF $f_1(V \cdot \cos\theta)$ AND SPECTRUM OF $f_2\left(\frac{m}{\cos^3\theta}\right)$ FOR METEORS

ELECTRICAL INTERFACE

DAE COMMANDS NO.

POWER ON-OFF 1
CALIBRATE ?

DAE SIGNALS

BACKGROUND	2,	6 BIT WORDS	12 BITS/SEC
FLASH TIME HISTORY	2,	6 BIT WORDS	48 BITS/SEC
PULSE-HEIGHT ANALYZE			10 BITS/SEC
			70 BITS/SEC

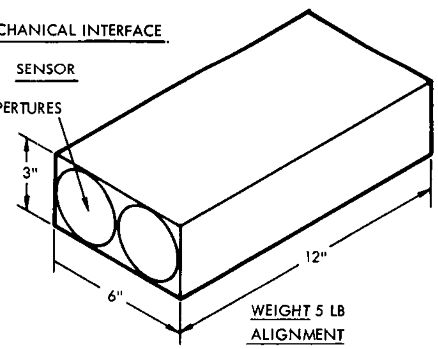
2 TEMPERATURE, 2 VOLTAGE ENGINEERING MEASUREMENTS

POWER: 5 WATTS
1 SENSOR
4 REMOTE

SENSOR LEADS 8

MECHANICAL INTERFACE

SENSOR



WEIGHT 5 LB
ALIGNMENT
3° TO MARS AXIS

REMOTE ELECTRONICS:

VOLUME 144 IN^3 WT 2 LB

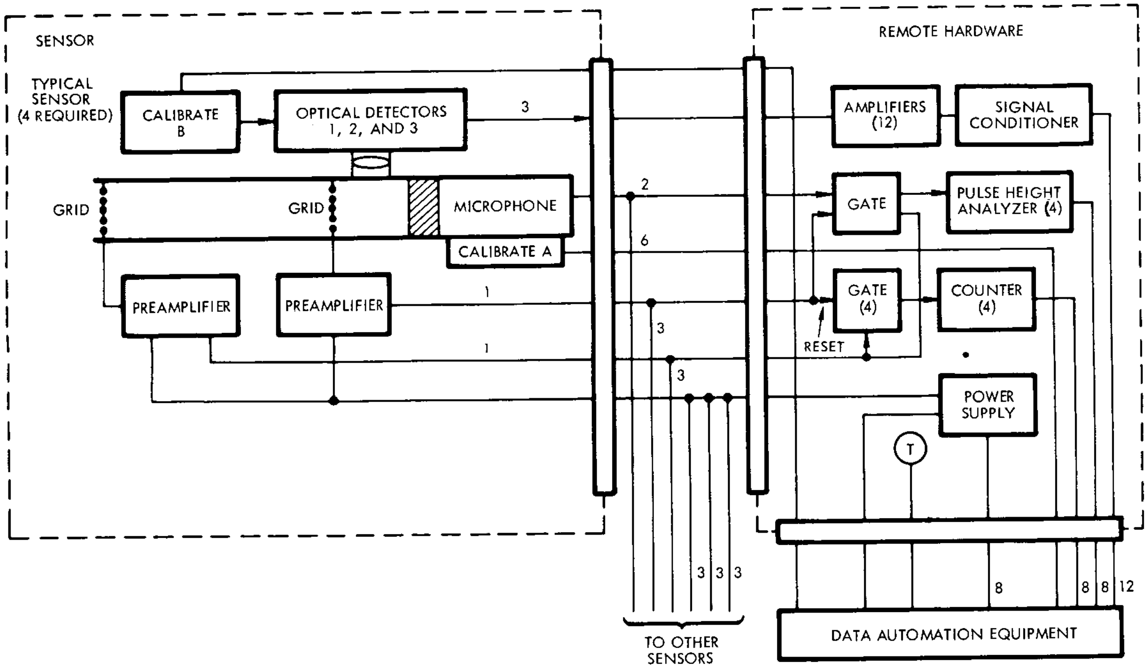
THERMAL INTERFACE VS-4-510

NOMINAL TEMPERATURE CONTROL

CONSTRAINTS

MAGNETIC 1971 VOYAGER MISSION SPECIFICATION

Figure 8. Meteor Flash Data Sheet



METEORITE IMPACT EXPERIMENT FUNCTIONAL BLOCK DIAGRAM

CHARACTERISTICS

- 4 SENSORS:

CONE ANGLE	CLOCK ANGLE
0°	—
90°	270°
90°	0°
180°	—

- COLLECTING AREA 300 IN^2
- ACCEPTANCE ANGLE APPROX 10° TOTAL
- MEASUREMENT

  COLLECT IONS AT GRIDS TO MEASURE METEORITE SPEED AND Q/M OF COLLISION IONS

  MEASURE ACOUSTIC ENERGY

  MEASURE SPECTRAL DISTRIBUTION (IN 3 WAVELENGTH BANDS) OF FLASH

ELECTRICAL INTERFACE

DAE COMMANDS

POWER ON-OFF	1
CALIBRATE	2

DAE SIGNALS — EACH EVENT

TRANSIT TIMES	12 BITS
PULSE/HEIGHT	12 BITS

6 TEMPERATURE 2 VOLT ENGINEERING MEASUREMENTS

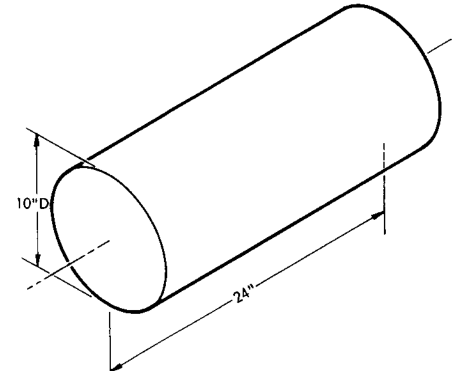
POWER

2 WATTS	
SENSOR, 0.2	0.8
REMOTE HARDWARE	1.2

SENSOR LEADS

EACH SENSOR 10 40

MECHANICAL INTERFACE



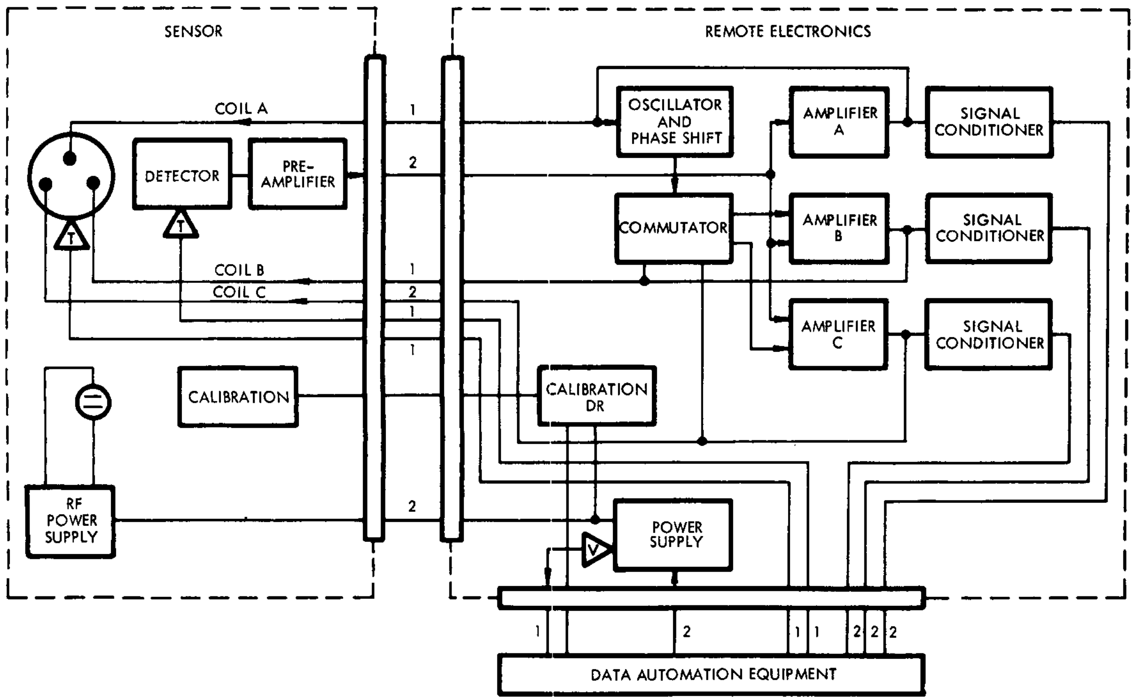
ALIGNMENT	3°
FOV	10° TOTAL
WEIGHT	3 LB

SENSOR ENVELOPE

REMOTE ELECTRONICS: VOLUME 144 IN^3, WT 4 LB

THERMAL INTERFACE	VS-4-510
CONSTRAINTS	VS-3-180

Figure 9. Meteoroid Impact Experiment Data Sheet



ELECTRICAL INTERFACE

DAE COMMANDS

POWER ON-OFF	1	
RANGE CHANGE	1	
CALIBRATE	1	PLUS 3 BIT WORD

DAE SIGNALS

3-AXIS OUTPUTS 6 BIT WORDS AT 6 OR 30 BITS/SEC

2 TEMPERATURE 1 VOLTAGE ENGINEERING MEASUREMENTS

POWER

6.8 WATTS
1.8 IN SENSOR
5 IN REMOTE HARDWARE

SENSOR LEADS 10

MECHANICAL INTERFACE

SENSOR

6" LONG WT: 1.3 LB
3" DIA ALIGNMENT
INTERPLANETARY ±3°
PLANETARY ±1°

REMOTE HARDWARE

VOL 96 IN^3 WT: 5 LB

THERMAL INTERFACE

CONTROL TEMPERATURE, VS-4-510

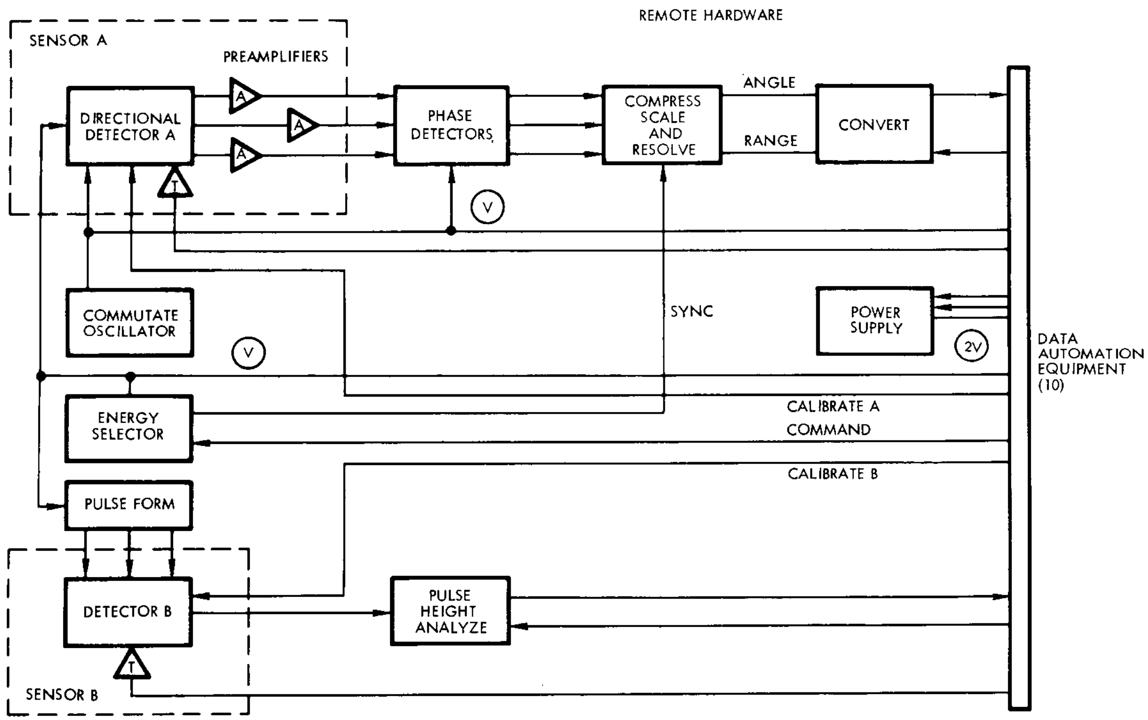
CONSTRAINTS

MAGNETIC FIELD AT BOOM MOUNTED MAGNETOMETER LESS THAN 0.258

CHARACTERISTICS

- MEASURE: 3 ORTHOGONAL COMPONENTS OF MAGNETIC FIELD
- RANGE: -500 TO +500
- RESPONSE 1 CPS
- NULL STABILITY AND RESOLUTION 0.25γ
- LINEARITY: 1% OF THE VALUE + NULL STABILITY
- IN-FLIGHT CALIBRATION FIELDS -80, -40, +40, +80γ

Figure 10. Helium Magnetometer Data Sheet



CHARACTERISTICS:

- DETECTOR A MEASURES A MODULATED CURRENT TO DETERMINE DIRECTION OF ARRIVAL AND ENERGY
- DETECTOR B SWEPT DETECTOR TO RESOLVE ENERGY LEVELS AND DIRECTION AT HIGH DATA RATE (HIGH RESOLUTION)

ELECTRICAL INTERFACE

DAE COMMANDS	DISCRETE	QUANTITATIVE
POWER ON-OFF	1	
CALIBRATE A	1	
CALIBRATE B	1	
ENERGY COMMAND		6 BITS

DAE SIGNALS	LEADS	
A - DIRECTION AND SPECTRUM	2	6 BIT WORD 6 BIT/SEC
B - HIGH RESOLUTION PULSE DATA	2	6 BIT WORDS (AVAILABLE AT 60 BIT/SEC)

2 TEMPERATURE, 4 VOLTS ENGINEERING MEASUREMENTS

POWER

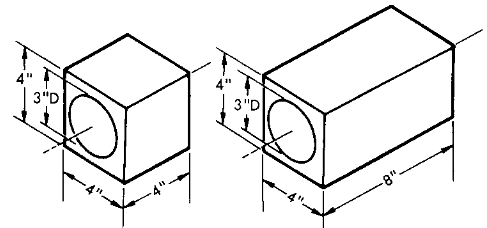
6 WATTS
0.6 SENSORS
5.4 REMOTE HARDWARE

SENSOR LEADS

A - 7
B - 6

MECHANICAL INTERFACE

SENSOR A — SENSOR B (REQUIRES COVER)



WEIGHT 1 LB — WEIGHT 3 LB

REMOTE HARDWARE

VOLUME 160 IN^3
WEIGHT 5.5 LB

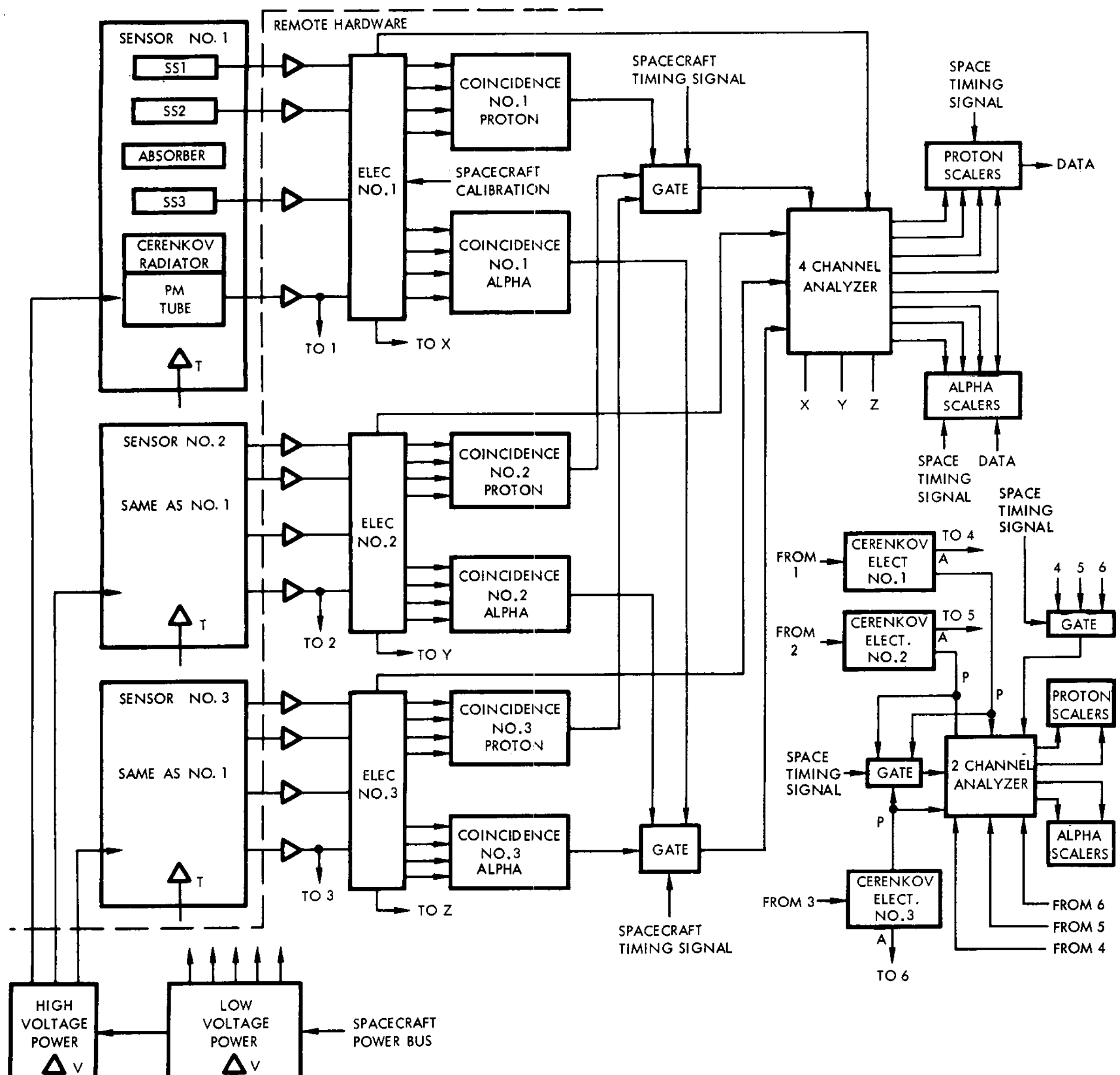
THERMAL INTERFACE

NOMINAL, VS-4-510

CONSTRAINTS

VS-3-180
1971 VOYAGER MISSION SPECIFICATION

Figure 11. Plasma Probe Data Sheet



ELECTRICAL INTERFACE

DAE COMMANDS

POWER ON-OFF	1
CALIBRATE	1
FLUX CHANGE	1

DAE SIGNALS (FROM EXP.)

DATA. MIN BIT RATE	1 BITS/SEC
DATA. MAX BIT RATE	4 BITS/SEC
ENGINEERING DATA	4 TEMPERATURE 2 VOLTAGE

POWER

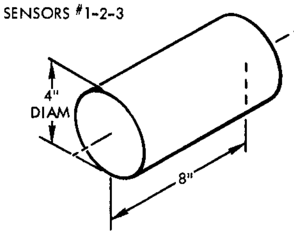
1 WATT SENSORS
4 WATTS REMOTE HARDWARE
5 WATTS TOTAL

SENSOR LEADS

8 PER SENSOR

MECHANICAL INTERFACE

SENSORS #1-2-3



SENSOR WEIGHT	1 1/2 LB EACH
REMOTE HARDWARE WEIGHT	8 LB, VOLUME 512 IN^3

THERMAL INTERFACE

VS-4-510

CONSTRAINTS

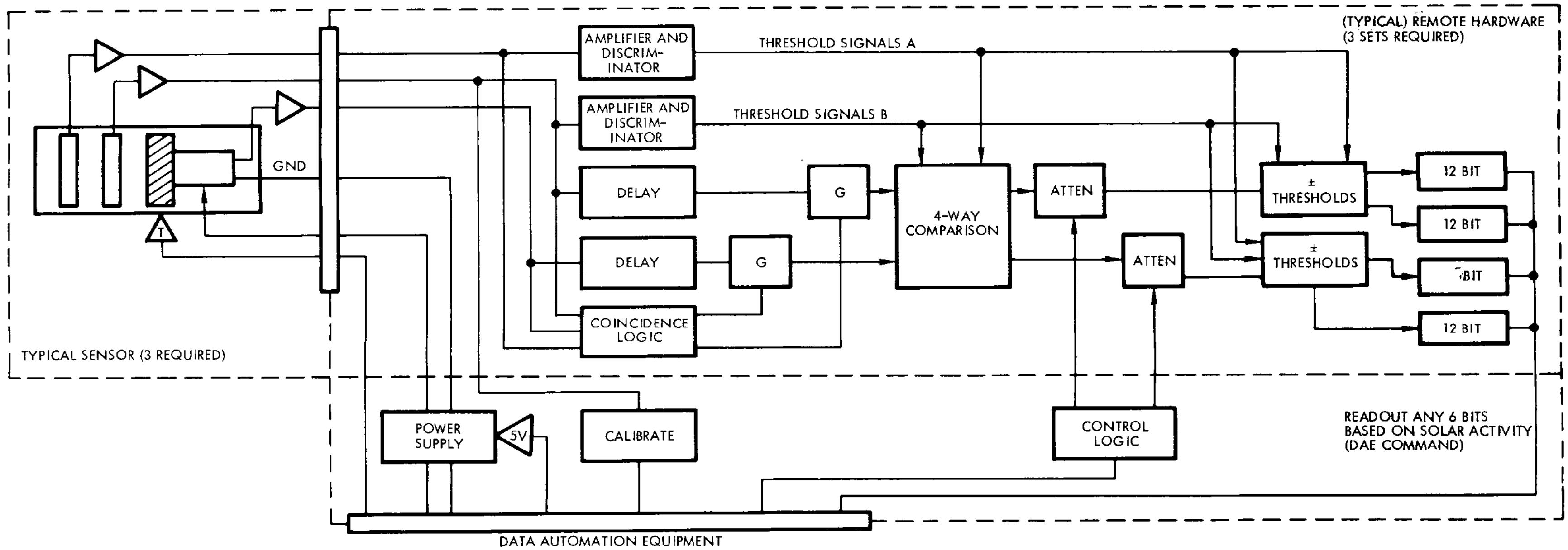
1 SENSOR TOWARDS SUN
1 SENSOR AWAY FROM SUN
1 SENSOR PERPENDICULAR TO SUN LINE

CHARACTERISTICS

OBTAIN DIRECTIONAL AND SPECTRAL INFORMATION ON THE PROTON FLUX AND ALPHA PARTICLES IN THE FOLLOWING RANGES

PROTONS:	100-150 MEV 150-250 MEV 250-500 MEV 500 MEV - 1 BEV 1 BEV - 2 BEV ABOVE 2 BEV	ALPHAS:	4 TIMES THAT OF THE PROTON RANGES

Figure 12. Cosmic Ray Experiment Data Sheet



CHARACTERISTICS:

- TWO SOLID STATE DETECTORS FOLLOWED BY SCINTILLATOR AND PHOTOMULTIPLIER TUBE
- PROTON ENERGY RANGE - 10 MEV < E < 200 MEV IN 16 CHANNELS WITH E > 100 MEV IN HIGHEST CHANNEL
- ELECTRON ENERGY RANGE - 50 KEV < E < 5 MEV IN 16 CHANNELS WITH E > 2 MEV IN HIGHEST CHANNEL
- LOOK ANGLE 45°

ELECTRICAL INTERFACE

DAE COMMANDS

POWER ON-OFF	1
CALIBRATE	2
SCALE CHANGE	1

DAE SIGNALS

16 6-BIT WORDS PER SAMPLE, 8 SAMPLES PER CYCLE
TOTAL 800 BITS FOR ONE READING (8 SEC) AVERAGE DUTY CYCLE 1/4
3 TEMPERATURE, 5 VOLTAGE ENGINEERING DATA MEASUREMENTS
6 LEADS PER SENSOR

POWER

5 WATTS
0.4 PER SENSOR OR 1.2
3.8 REMOTE HARDWARE

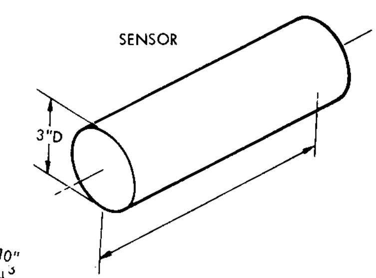
THERMAL INTERFACE NOMINAL, VS-4-510

CONSTRAINTS SAFETY-RADIOACTIVE MATERIALS PROTECTIVE COVERS REQUIRED

REMOTE HARDWARE

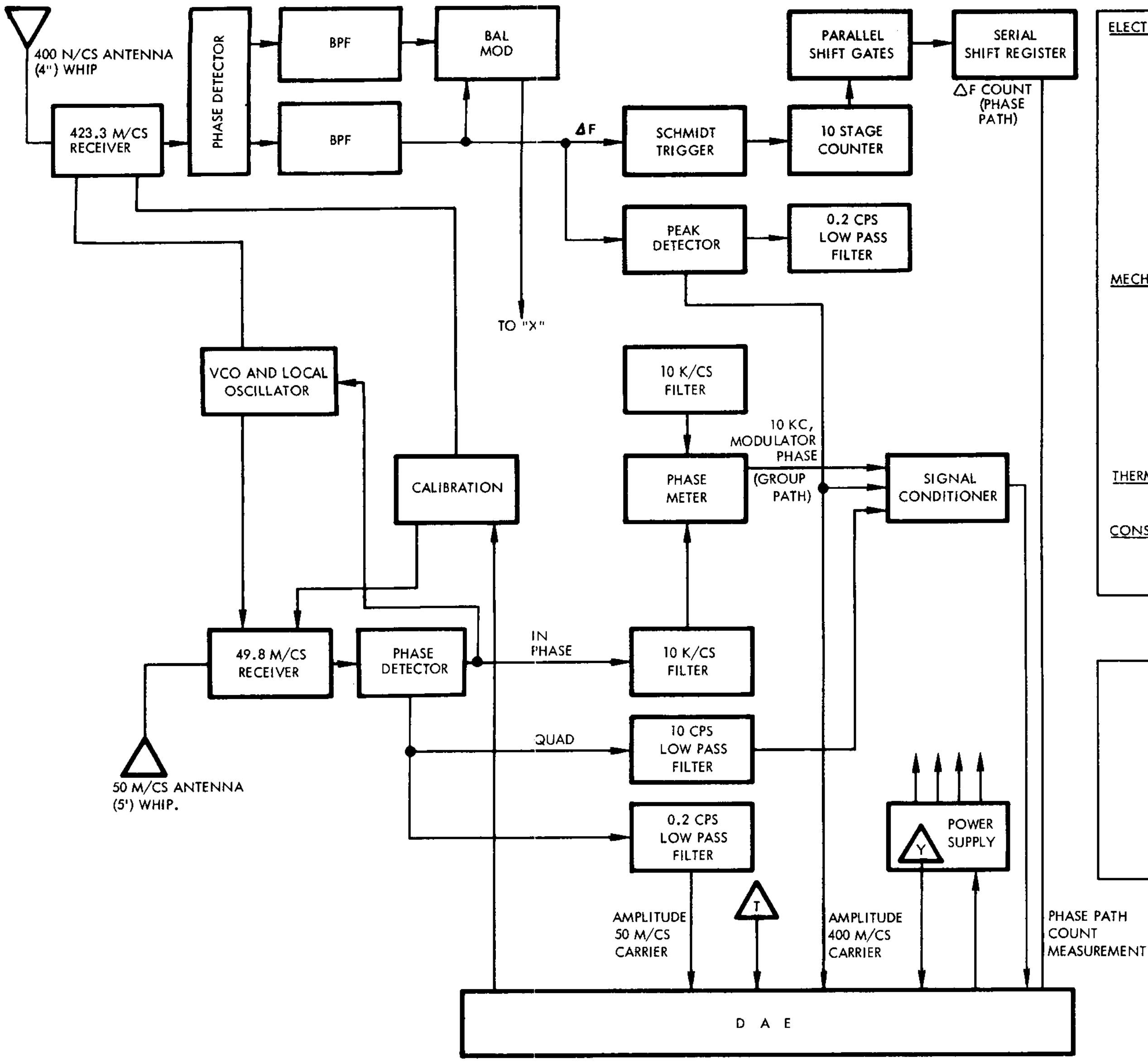
VOLUME 512 IN^3
WEIGHT 7.5 LB

MECHANICAL INTERFACE



WEIGHT 1.7 LB

Figure 13. Trapped Radiation Experiment Data Sheet



<u>ELECTRICAL INTERFACE</u>

<u>DAE COMMANDS</u>

POWER ON-OFF	1
CALIBRATE	1

<u>DAE SIGNAL</u>

PHASE COUNT MIN. 10 BITS/SEC PHASE DIFF. COUNT MAX. (OCCULT) 30 BITS CARRIER AMPLITUDES AND PHASE DIFF. COUNT

1 TEMPERATURE, 1 VOLTAGE — ENGINEERING MEASUREMENTS

<u>POWER</u>	1.5 WATTS

<u>MECHANICAL INTERFACE</u>

<u>ANTENNA</u>

AT 50 MC/S	60" WHIP
AT 400 MC/S	4" WHIP

<u>REMOTE HARDWARE</u>:

VOLUME	252 IN^3
WEIGHT	6 LB

<u>THERMAL INTERFACE</u>

VS-4-510

<u>CONSTRAINTS</u>

VS-3-180	DEPLOY ANTENNAS

CHARACTERISTICS

- TO MEASURE THE AVERAGE INTERPLANETARY ELECTRON DENSITY BETWEEN EARTH AND PROBE USING 2 PHASE LOCKED RECEIVERS, ONE AT 50 M/CS AND ONE AT 400 M/CS. SENSITIVE TO CHANGES IN ELECTRON CONTENT NO LARGER THAN 4 x 10^{14} ELECTRONS/M^2

FREQUENCIES

- 49.8 M/CS AND 422.3 M/CS

Figure 14. Ionosphere Experiment Data Sheet

COMMUNICATIONS

VS-4-310

CONTENTS

1. SCOPE

This document presents the design for the Voyager spacecraft communications subsystem. This subsystem, in conjunction with the ground system, implements the over-all functions of tracking, telemetering, ranging, and a ground to spacecraft command link.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows

JPL

V-MA-004-001-14-03	JPL Project Document No. 45, "Preliminary Voyager 1971 Mission Specification"
GMC-50109-DSN-A	
TM-33-83	

TRW 1971 Voyager Spacecraft Design Documents

VS-3-110	Configuration
VS-3-111	Components Design Parameters
VS-4-550	Electronic Packaging

3. DESCRIPTION

3.1 Functions

The Voyager spacecraft communications subsystem performs the following functions:

a) Receives simultaneous RF signals transmitted to the spacecraft from the DSIF and capsule

b) Coherently translates the frequency and phase of the received DSIF RF signal by a fixed ratio

c) Demodulates the received capsule VHF signal and sends the detected bits to the data handling subsystem

d) Demodulates the received DSIF RF signal and sends the detected commands bits to the spacecraft central sequencing and command subsystem

e) Phase-modulates the transmitter with a composite telemetry signal

f) Phase-modulates the transmitter with the demodulated ranging signal from the turn-around ranging channel

g) Transmits a modulated RF signal to the DSIF stations using the translated RF signal (b), above) or an independent frequency source

h) Provides appropriate antenna patterns for the functions described in (a) and (g), above.

3.2 Organization

The communications subsystem elements are listed below and their interrelationships are shown in Figure 1.

- S-band receivers
- Modulator-exciters
- RF power amplifiers
- Power monitors and power splitters (four port hybrid ring)
- RF switches and control circuits
- Diplexers and filters
- Low power transmitter
- S-band and VHF antennas
- Command detector
- VHF preamplifier
- VHF receivers
- Capsule demodulators

3.3 Mission Operation

At launch the low power transmitters, S-band receivers, and command detectors are turned on. The low power transmitter is connected to the low gain antenna and provides spacecraft telemetry independent of the launch vehicle communications system. It also provides a signal which is used by the DSN for tracking of the spacecraft. Command

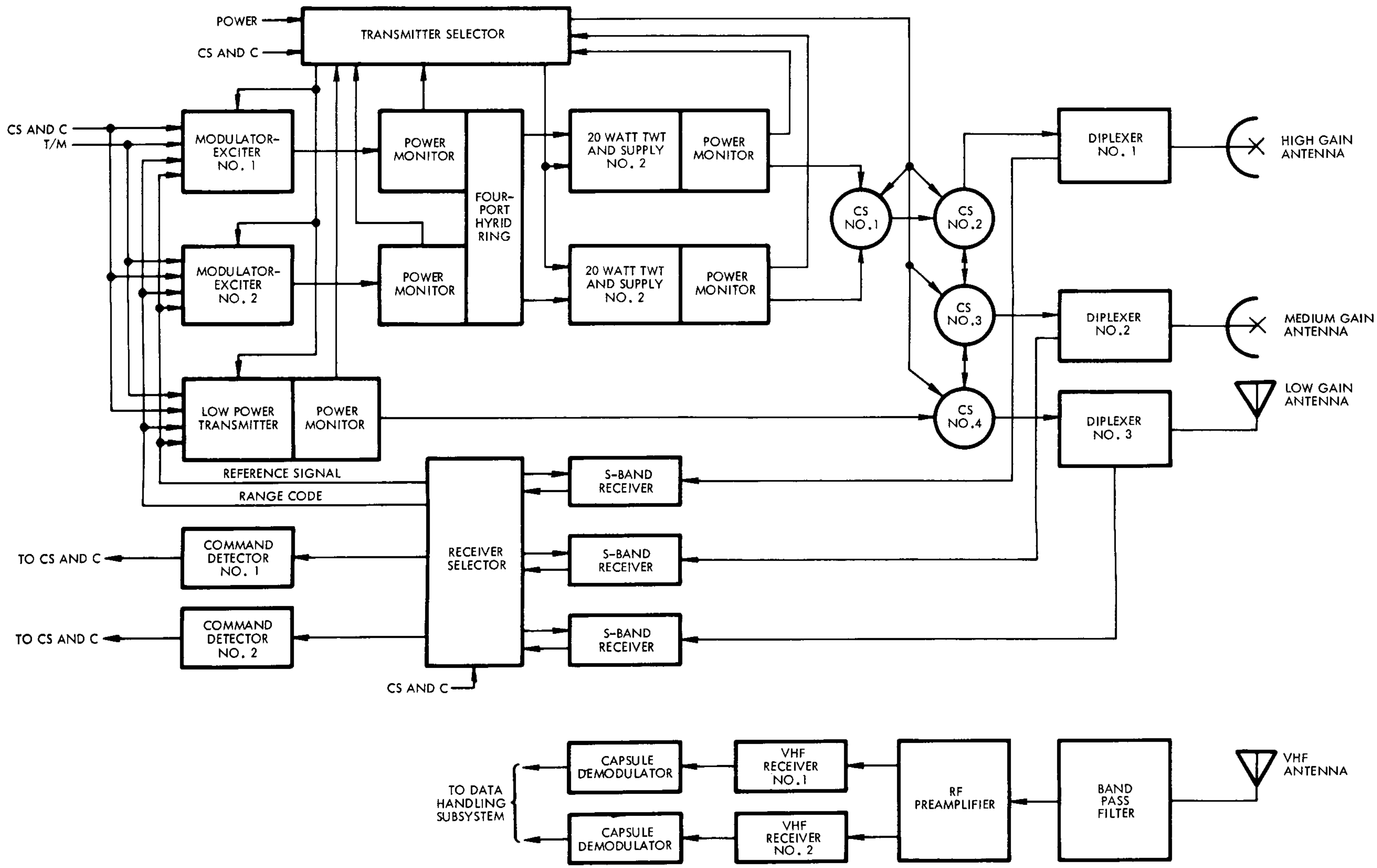


Figure 1. Communication Subsystem Block Diagram

signals transmitted by the DSN are normally received over the low gain antenna. The addressed command detector provides the command data bits to the CS&C. The application of prime power to the power amplifiers is inhibited until approximately 15 hours after launch. After sun-Canopus acquisition and high gain antenna deployment, the low power transmitter is switched to the high gain antenna. At this time the high telemetry data rate is available. A short while later the communications system is switched to the high power, high gain mode. This mode is used for the remainder of the mission except for maneuvers. The 1024 bit rate is selected during the cruise phase so that the standard, 10 kw, 85 ft diplexed DSIF stations can be used up to nominal encounter range. The receiver connected to the low gain antenna is normally used for command reception until the link budget for the 10 kw DSIF stations requires that commands and ranging signals be received over the high gain antenna.

For maneuvers, the normal sequence is to slew the high gain antenna to the position required so as to be earth pointing when in maneuver attitude. Confirmation of the antenna angles is accomplished over the low or medium gain antenna. After confirmation of antenna angles, the spacecraft maneuver enable command is transmitted and is received over either the low or medium gain antenna. After spacecraft maneuver attitude execution has been confirmed over the high gain antenna, the motor fire enabling command from the ground station is received over the high gain antenna. Upon completion of the propulsion maneuver, the spacecraft automatically goes through sun-Canopus acquisition and reorients the high gain antenna to earth. At this time the communications subsystem is returned to the high power, high gain mode.

3.4 Modes of Operation

3.4.1 Antenna Transfer

The low and high power transmitters are capable of transmitting over the low, medium, or high gain antennas. The primary transmission modes are:

a) Low power, low gain - is used from launch until after sun-Canopus lock and high gain antenna deployment.

b) Low power, high gain - is used during the early part of mission after sun-Canopus lock and high gain antenna deployment.

c) High power, high gain - is used for all other mission phases, except for a portion of the maneuver sequence as described in paragraph 3.3.

The antenna switch positions required to connect any transmitter to any antenna are given in Table 1.

Table 1. Antenna Transfer Switch Position

RF Power Source	Antenna	Circulator Switch Position			
		CS No. 1	CS No. 2	CS No. 3	CS No. 4
PA No. 1	Low	CW	CCW	CCW	CW
	Medium	CW	CCW	CW	-
	High	CW	CW	-	-
PA No. 2	Low	CCW	CCW	CCW	CW
	Medium	CCW	CCW	CW	-
	High	CCW	CW	-	-
Low Power Tx	Low	-	-	-	CCW
	Medium	-	-	CCW	CW
	High	-	CCW	CW	CW

3.4.2 Transmitter Transfer

The transmitter consists of two redundant power amplifiers, two redundant modulator-exciters, and a low power transmitter. On-board power monitors and control logic (subject to CS&C override) are used to switch power amplifiers, exciters, and low power transmitters. The following combinations can also be selected by ground command:

Modulator-Exciter No. 1, Power Amplifier No. 1

Modulator-Exciter No. 2, Power Amplifier No. 1

Modulator-Exciter No. 1, Power Amplifier No. 2

Modulator-Exciter No. 2, Power Amplifier No. 2

Low Power Transmitter

3.4.3 Receiver Transfer

Three receivers are provided; one permanently connected to each antenna. All receivers are operated continuously with the output of one selected by on-board logic (subject to CS&C override). There are two basic modes:

a) Maximum Coverage. The receiver that is in-lock and provides maximum area coverage is selected to provide signals to the command demodulators, the modulator-exciters, and the low power transmitter. The priority of the logic is:

- Use the receiver connected to the low gain antenna if it is in-lock
- Use the receiver connected to the medium gain antenna if it is in-lock and condition above is not satisfied
- Use the receiver connected to the high gain antenna if it is in-lock and both conditions above are not satisfied.

b) Maximum Gain. The receiver that provides maximum antenna gain and is in-lock is selected to provide signals to the command demodulators, modulator-exciters, and low power transmitter. The priority of the logic is:

- Use the receiver connected to the high gain antenna if it is in-lock
- Use the receiver connected to the medium gain antenna if it is in-lock and the condition above is not satisfied
- Use the receiver connected to the low gain antenna if it is in-lock and both conditions above are not satisfied.

A CS&C override is used to switch to the maximum coverage mode if the vehicle loses sun or Canopus lock. The override is inhibited during normal maneuvers. Provision is also made to turn off any receiver on ground command.

3.4.4 Command Detector Transfer

Two redundant command detectors are simultaneously operated and each is wired to recognize a different pseudo-noise (PN) sequence for synchronization. Thus, the command detectors are selected by using the corresponding PN generator at the DSIF. The outputs of both detectors are connected to the CS&C; only the detector that is in-lock will provide signals to the CS&C.

3.4.5 Capsule Receiver

Two redundant VHF receivers and capsule demodulators are provided; both are switched on at the proper time in the mission. Thus, two independent (non-switched) means are provided for recovering capsule data. Two parallel 10 bit/sec demodulated capsule data signals (one from each receiver-demodulator) are sent to the digital telemetry units; both are buffered, commutated with other data, and retransmitted to DSIF in real time.

3.4.6 Phase Control

The unmodulated phase and frequency of the transmitted RF carrier is controlled by either one of two sources; one coherent, the other non-coherent. When no S-band receiver is in-lock, the phase and frequency of the transmitted carrier is controlled by a crystal oscillator. When one of the S-band receivers is in-lock, the phase and frequency of the transmitted carrier is 240/221 times the phase and frequency of the received carrier.

3.5 Subsystem Elements

3.5.1 S-Band Modulator-Exciters

Two modulator-exciters are provided; the transmitter selector (subject to ground command override) controls the application of prime power to one unit or the other. Each modulator-exciter provides the drive to both power amplifiers through the four-port power splitter. It also phase modulates (PM) the telemetry or ranging signal onto the transmitted S-band carrier. The modulator-exciter operates either from the coherent reference (VCO) signal, supplied by the receiver, or from the self-contained auxiliary (crystal) oscillator. The choice is made by the presence or absence of the in-lock signal from the receiver.

3.5.2 Four-Port Hybrid Ring

The four-port hybrid provides one-half (-3db) of the power from either modulator-exciter to each of the two power amplifiers. In addition, the four-port assembly includes a power monitor in each of the input arms to be used by the transmitter selector logic on board the spacecraft.

3.5.3 Power Amplifiers

Two redundant 20-watt TWT's, each with its own power supply, are provided, however, only one power amplifier is operated at any one time. The transmitter selector (subject to ground command override) controls the application of prime power to the selected power amplifier and the circulator switch to connect the RF output to the diplexer-antenna system.

3.5.4 Low Power Transmitter

The low power transmitter is identical to the modulator-exciter except for the power output. The primary function of this transmitter is to transmit telemetry during the boost and early part of the mission.

3.5.5 Transmitter Selector

The transmitter selector provides the signals to switch modulator-exciters, power amplifiers, and antenna. Ground commands are used to select the desired power level and antenna. The transmitter selector logic then uses the output of on-board power monitors to switch modulator-exciters and power amplifiers in the event RF power output drops below a fixed threshold. Ground commands can also by-pass the automatic switching provisions and select the individual modulator-exciter and power amplifier.

3.5.6 RF Switches

The RF switches are voltage operated circulators. The switch offers a matched load impedance and a means of changing the RF path. Application of a positive or negative drive voltage causes the RF signal to circulate in a clockwise or counterclockwise direction from the input terminal to the adjacent terminal, which then becomes the output.

3.5.7 Rotary Joints

Coaxial rotary joints are utilized in all rotary actuator mechanisms which require RF transmission lines to pass through them, and which actuate more than once. The rotary joints, are of the noncontacting choke, single-channel, transmission line variety, and are broad-band enough to operate over the full transmission and reception frequency range without degradation.

3.5.8 RF Diplexers

Three diplexers are used; one for each antenna. The use of diplexers in the antenna transmission lines allows continuous monitoring of signals received on all three antennas while allowing transmission on any of the three antennas. The minimum isolation afforded between the transmitted signal path and the received signal path is 70 db. The isolation is afforded by band-pass characteristics in each of the transmission and reception lines.

3.5.9 S-Band Antennas

a. High Gain Antenna

The high gain antenna is a sectoral paraboloidal reflector with an elliptical aperture, and has two degrees of freedom. The major axis of the ellipse is 78.5 inches, and the minor axis of the ellipse is 66 inches. The antenna is equipped with a suitable radiating feed. The antenna is right-hand circularly polarized, and is capable of being oriented such that it scans through 3π steradians about the spacecraft.

b. Medium Gain Antenna

The medium gain antenna is a 36-inch circular aperature paraboloidal reflector and has one degree of freedom. The antenna has a suitable feed to provide a right-hand circularly polarized pattern. The antenna is capable of being oriented such that is scans through 180 degrees, in a single plane.

c. Low Gain Antenna

The low gain antenna is an assembly consisting of two antennas and one coupler mounted on the spacecraft at a 70 degree clock angle. Each antenna is a right-hand circularly polarized radiator having a radiation pattern which is symmetric about its axis. The "primary antenna" has its peak centered about the roll axis (zero degree cone angle). The "secondary antenna" has its peak centered about an axis at a cone angle of 135 degrees. The "secondary antenna" is decoupled from the primary antenna by an amount consistent with the gain requirements for the "primary antenna".

3.5.10 S-Band Receivers

Three redundant S-band receivers are provided. Each receiver is a narrow-band superhetrodyne receiver with an automatic phase tracking loop, and a "turn-around" ranging demodulator. The receiver locks onto, and tracks, the phase of the received S-band carrier. The oscillator (VCO) of the phase tracking loop, is used as the reference oscillator for the modulator-exciter. The receiver demodulates the composite command subcarrier and the PN range code, and supplies these signals to the command demodulator and modulator-exciter.

3.5.11 Receiver Selector

The receiver selector receives in-lock signals from each receiver and discrete commands from the CS&C. It uses the in-lock signals to select the receiver to be used based upon the mode of operation command from the CS&C. The logic output of the receiver selector enables the outputs of the selected receiver to be sent to the modulator-exciters and command detectors.

3.5.12 Command Demodulators

Two redundant command detectors, selectable by ground station coding are provided to demodulate the command data from the subcarrier outputs of the receivers in synchronization with a received synchronizing signal. The selected detector generates an output data bit stream, a bit sync pulse train, and an in-lock (good-data) signal.

3.5.13 VHF Antenna

The VHF antenna operates at 136 - 138 mc and consists of a right-hand circularly polarized radiator, with a radiation pattern approximately a cardioid of revolution. The antenna is mounted on the spacecraft at a clock angle of 105 degrees with maximum radiation at a cone angle of 140 degrees. The radiator is mounted on the solar cell array on the opposite side to the solar cells. The antenna consists of a turnstile reflector array in a plane tipped 50 degrees to the solar array.

3.5.14 VHF Preamplifier

The VHF preamplifier provides coupling of the received capsule signal from one antenna to two receivers. It has sufficient gain with a low noise figure to establish an over-all receiver noise figure of 4 db. Two parallel, buffered outputs are provided for the VHF receivers.

3.5.15 VHF Receivers

Two VHF receivers are provided; both are powered simultaneously. Each receiver operates in the 136 to 138 mc band and receives the FSK modulated signal from the VHF preamplifier. Each receiver provides a 10 mc IF output to a capsule demodulator.

3.5.16 Capsule Demodulators

Two units for demodulating the received capsule data are provided; one is connected to each VHF receiver. Each capsule demodulator reduces the FSK, 10 mc signal from the VHF receiver to a 10 bit/sec NRZ signal. Parallel mark (10.001 mc) and space (9.989 mc) filters with individual envelope detectors are used. The summed output of the envelope detectors is matched filter detected.

4. INTERFACE DEFINITION

4.1 Shroud-Antenna Interface

The shroud contains a radio window opposite the secondary low gain antenna to permit communications on the launch pad and until the shroud is ejected after launch.

4.2 Input Signals

4.2.1 Received Signals

The DSIF/DSN system provides a signal in the 2115 ± 5 mc band. The maximum bandwidth required for this signal is 3.3 mc.

The capsule system provides a signal in the 136 to 138 mc band. The maximum bandwidth required for this signal is 44 kc.

4.2.2 Data Handling Subsystem

The data handling subsystem supplies a composite telemetry signal to the input of the three phase modulators. The bandwidth of this signal is from 10 cps to 150 kc. Each phase modulator has an input impedance for this signal of 1000 ohms while the source impedance is less than 50 ohms. The relative amplitude of the data and sync signals is adjusted in the data handling subsystem to provide the desired modulation indices. The composite signal amplitude is 3 ± 0.3 volts peak-to-peak.

4.2.3 Central Sequencing and Command

Command signals supplied by the CS&C are given in Table 2.

4.3 Output Signals

4.3.1 Radiating Power

Two power levels are provided, 1 watt and 20 watts, in the 2295 ± 5 mc band. The maximum channel bandwidth required for this signal is 3.3 mc.

4.3.2 Data Handling Subsystem

Table 3 lists data monitoring points. The ground test and telemetry points are isolated through a high impedance from the internal circuitry, and can be shorted without degrading the normal system operation.

Table 2. Command Inputs

Function	Primary Source of Command	Backup Source of Command	Type Control	Notes
Transmitter-Antenna Selection				
Low Power, Low Gain	Ground	None	Discrete	
Low Power, High Gain	Ground	CS&C	Discrete	
High Power, High Gain	Ground	CS&C	Discrete	1
High Power, Medium Gain	Ground	None	Discrete	1
High Power, Low Gain	Ground	None	Discrete	1, 2
Low Power, Medium Gain	Ground	None	Discrete	
PA No. 1	Ground	None	Discrete	
PA No. 2	Ground	None	Discrete	
Mod Exciter No. 1	Ground	None	Discrete	
Mod Exciter No. 2	Ground	None	Discrete	
Receiver Selection				
Maximum Gain	Ground	CS&C	Discrete	
Maximum Coverage	Ground	CS&C	Discrete	2
Noncoherent Override - On	Ground	None	Discrete	
Noncoherent Override - Off	Ground	None	Discrete	
Receiver No. 1 Disconnect	Ground	None	Discrete	
Receiver No. 2 Disconnect	Ground	None	Discrete	
Receiver No. 3 Disconnect	Ground	None	Discrete	
All Receivers On	Ground	None	Discrete	
Antenna Angles				
Shaft Angle - High Gain	CS&C	Ground	Quantitative (12 bits)	3
Hinge Angle - High Gain	CS&C	Ground	Quantitative (12 bits)	3
Shaft Angle - Medium Gain	CS&C	Ground	Quantitative (12 bits)	3
Antenna Deployment				
High Gain	CS&C	Ground	Discrete	4
Medium Gain	CS&C	Ground	Discrete	4
Range Code				
On	Ground	None	Discrete	
Off	Ground	None	Discrete	
Capsule Data				
On	Ground	CS&C	Discrete	
Off	Ground	CS&C	Discrete	

Notes:
1. Interlocked until preset time after separation
2. Special mode selected in the event of loss of attitude control
3. CS&C increments command register as required
4. Timed from spacecraft separation

Table 3. Communication Subsystem Monitor Points

Subsystem Element	Function	Number of Points	A = Analog D = Discrete B = Binary	Primary Use T/M	Primary Use Ground Test
S-Band receiver	Static phase error	3	A	X	
	Signal strength	3	A	X	
	VCO temperature (or frequency)	3	A	X	
	In-lock	3	D	X	
	Regulated voltage	6	A		X
	Ranging modulation	3	A		X
	Command subcarrier	3	A		X
Power amplifier and power supply	Helix current	2	A	X	
	Cathode (or collector) current	2	A		X
	Cathode voltage	2	A		X
	Power output	2	A	X	
	Base plate temperature	2	A	X	
	Collector (or helix) voltage	2	A		X
Modulator-exciter and low power transmitter	Regulated voltage	6	A	X	
	Power output	3	A	X	
	Crystal drive	3	A		X
	Driver current	3	A		X
VHF receiver and preamplifier	Regulated voltage	2	A	X	
	Signal strength	2	A	X	
	Receiver output	2	A		X
	Preamplifier voltage	1	A		X
Caspule demodulator	In-lock	2	D	X	
	Regulated voltage	2	A		X
	Data output	2	A		X
Command demodulator	In-lock	2	D	X	
	PN sync	2	A		X
	Regulated voltage	4	A		X
	Data output	4	A		X
High gain antenna	Shaft angle	1	B(12 bits)	X	
	Hinge angle	1	B(12 bits)	X	
	Shaft motor current	1	A		X
	Hinge motor current	1	A		X
	Regulated voltage	1	A		X
	Temperature	2	A	X	
	Radiated power	1	A	X	
Medium gain antenna	Shaft angle	1	B(12 bits)	X	
	Shaft motor current	1	A		X
	Regulated voltage	1	A		X
	Temperature	1	A	X	
	Radiated power	1	A	X	
Low gain antenna	Radiated power	1	A	X	

In addition to monitor points listed, the telemetry system also receives two demodulated capsule data signals from the VHF receiver system. Each signal consists of: 10 bits/sec, NRZ data, asynchronous to spacecraft timing signals, and bit sync.

4.3.3 Central Sequencing and Command

The three signals sent to the CS&C are 1 bit/sec NRZ command signal, bit sync, and good-data signal.

4.4 Capsule Interface

4.4.1 Commands

No RF command link is provided.

4.4.2 Antenna

The VHF antenna on the capsule vehicle must have at least 0 db gain (right-hand circularly polarized) over a 90-degree cone angle and a front-to-back ratio of at least 6 db (right- and left-hand circularly polarized).

4.4.3 VHF Transmitter

The capsule VHF transmitter must be capable of providing a 20-watt power output with a 22 kc separation between the mark and space channels in the 136 to 138 mc band.

5. PARAMETERS AND LINK PERFORMANCE SUMMARY

5.1 Link Parameters

The values of the principal parameters that determine the over-all communication system performance as expressed in terms of the signal-to-noise ratio are given in Tables 4 through 12.

The high gain earth pointing antenna must be pointed to within ± 1.4 degrees, 3σ, to remain within the 1 db points of the beam. The pointing requirement reflected into a single-axis error, results in an allowable error of 1 degree.

Table 4. Spacecraft Radio Transmission Parameters (2295 ±5 Mc)

Parameter	Low Gain Antenna		Medium Gain Antenna		High Gain Antenna	
	Value	Tolerance	Value	Tolerance	Value	Tolerance
High Power Mode						
Total Transmitter Power[1]	+43.0 dbm	+1.0 db -0 db	+43.0 dbm	+1.0 db -0 db	+43.0 dbm	+1.0 db -0 db
Carrier Modulation Loss[2]	+10.5 db	+1.5 db -1.8 db	+10.5 db	+1.5 db -1.8 db	+10.5 db	+1.5 db -1.8 db
Transmission Circuit Loss[3]	+2.65 db	±0.8 db	+2.30 db	±0.8 db	+2.05 db	±0.8 db
Spacecraft Antenna Gain[4]	+4.0 db	+0.50 db -0.25 db	+24.0 db	+0.25 db -0.50 db	+30.0 db	+0.25 db -0.50 db
Low Power Mode						
Total Transmitter Power[5]	+30.0 dbm	±1.0 db	+30.0 dbm	±1.0 db	+30.0 dbm	±1.0 db
Carrier Modulation Loss[2]	+10.5 db	+1.5 db -1.8 db	+10.5 db	+1.5 db -1.8 db	+10.5 db	+1.5 db -1.8 db
Transmission Circuit Loss[3]	+1.60 db	±0.8 db	±1.95 db	±0.8 db	+2.40 db	±0.8 db
Spacecraft Antenna Gain	+4.0 db	+0.50 db -0.25 db	+24.0 db	+0.25 db -0.50 db	+30.0 db	+0.25 db -0.50 db

[1] 20 watts at the output of the TWT amplifier.

[2] Based upon carrier modulation indices (at bit rates 1024 and above) by:
a. Telemetry Data Subcarrier = 1.25 radians peak (square wave).
b. Telemetry Sync Subcarrier = 0.32 radians peak (square wave).

[3] Includes all circuitry between the output of the cavity amplifier and the input to the antenna.

[4] Referenced to perfectly circular isotropic, pattern maximum.

[5] 1 watt at the output of the Low Power Amplifier.

Table 5. DSIF Radio Reception Parameters (2295 ±5 Mc)

Parameter	Standard Deep Space Station (DSS)					
	Early Flight Acquisition		Tracking, Diplexed, Maser (85-ft dia)		Tracking, Listen Maser (210-ft dia)	
	Value	Tolerance	Value	Tolerance	Value	Tolerance
Antenna Gain[1]	+21.0 db	±1.0 db	+53.0 db	+1.0 db -0.5 db	+61.0 db	±0.5 db
Circuit Loss[2]	+0.5 db	±0.2 db	+0.2 db	±0.1 db	+0.2 db	±0.1 db
Effective System Noise Temperature[3]	270°K	±60°K	55°K	±10°K	24°K	±1°K
Antenna Ellipticity	Less than 1.5 db	-	+0.7 db	±0.3 db	±0.3 db	±0.2 db
Carrier APC Noise Bandwidth ($2B_{LO}$)	12 cps	+0.0 db -0.5 db	12 cps	+0.0 db -0.5 db	12 cps	+0.0 db -0.5 db
Carrier Threshold SNR in $2B_{LO}$						
a. One-way doppler tracking	0.0 db	-	0.0 db	-	0.0 db	-
b. Two-way doppler tracking[4]	+2.0 db	±1.0 db	+2.0 db	±1.0 db	+2.0 db	±1.0 db
c. Telemetering	+6.0 db	+0.5 db -1.0 db	+6.0 db	+0.5 db -1.0 db	+6.0 db	+0.5 db -1.0 db

[1] To matched polarization.

[2] Circuit losses includes diplexer, switch, and waveguide losses

[3] Includes contributions due to antenna zenith temperature, circuit losses, diplexer, low noise amplifier, and follow-on receiver

[4] When the carrier threshold SNR in $2B_{LO}$ is +3.8 db on the earth-to-spacecraft link, +2.0 db is required to overcome the ground receiver degradation.

Table 6. DSIF Radio Transmission Parameters (2115 ±5 Mc)

Parameter	Standard Deep Space Station				Non-Standard Deep Space Station	
	Early Flight Acquisition		Tracking, Diplexed 10 kw		Horn, Non-Diplexed 100 kw	
	Value	Tolerance	Value	Tolerance	Value	Tolerance
Total Transmitter Power	+70.0 dbm	+0.5 db -0.0 db	+70.0 dbm	+0.5 dbm -0.0 dbm	+80.0 dbm	-
Carrier Modulation Loss[1]	+2.13 db	+0.22 db -0.18 db	+2.13 db	+0.22 db -0.18 db	+2.13 db	+0.22 db -0.18 db
Transmission Circuit Loss[2]	+0.5 db	±0.2 db	+0.4 db	±0.1 db	+0.2 db	±0.1 db
Antenna Gain[3]	+20.0 db	±2.0 db	+51.0 db	+1.0 db -0.5 db	+53.0 db	+1.0 db -0.5 db
Antenna Ellipticity	Less than 1.5 db	-	+1.0 db	±0.5 db	+0.5 db	±0.3 db

[1]Based upon carrier modulation indices of:
a. Data Subcarrier = 0.584 radians peak (sine wave)
b. Sync Subcarrier = 0.547 radians peak (square wave).

[2]Circuit loss includes diplexer, switch, and waveguide losses.

[3]To matched polarization.

Table 7. Spacecraft Radio Reception Parameters (2115 ±5 Mc)

Parameter	Low Gain Antenna		Medium Gain Antenna		High Gain Antenna	
	Value	Tolerance	Value	Tolerance	Value	Tolerance
Antenna Gain (pattern maximum)[1]	+3.2 db	+0.50 db -0.25 db	+23.5 db	+0.25 db -0.50 db	+29.5 db	+0.25 db -0.50 db
Receiving Circuit Loss[2]	+1.26 db	±0.2 db	+1.26 db	±0.2 db	+1.36 db	±0.2 db
Effective System Noise Temperature[3]	2610°K	-	2610°K	-	2610°K	-
Carrier APC Noise Bandwidth ($2B_{LO}$)	20 cps	+0.46 db -0.41 db	20 cps	+0.46 db -0.41 db	20 cps	+0.46 db -0.41 db
Carrier Threshold SNR in $2B_{LO}$						
a. Two-way Doppler Tracking[4]	+3.8 db	-	+3.8 db	-	+3.8 db	-
b. Command Reception[5]	+8.0 db	±1.0 db	+8.0 db	±1.0 db	+8.0 db	±1.0 db

[1]Antenna gains referenced to perfectly circular isotropic.

[2]Includes all circuitry between the antenna and the input to the receiver.

[3]Includes contributions due to antenna temperature, circuit losses, and noise figure at input to pre-selector (10 db, maximum).

[4]+3.8 db SNR is required for +2.0 db ground receiver degradation.

[5]Based upon command threshold definition, see Table 10.

Table 8. Capsule Radio Transmission Parameters
(137 ±1 Mc)

Parameter	Value	Tolerance
Total Transmitter Power[1]	+43.0 dbm	±1.0 db
Transmission Circuit Loss[2]	0.5 db	±0.2 db
Antenna Gain[3]	0.0 db	+0.5 db -1.0 db

[1] 20 watts at output of capsule transmitter.

[2] Total circuit loss from transmitter output to antenna

[3] Referenced to perfectly circular isotropic, pattern maximum.

Table 9. Spacecraft Radio Reception Parameters
(137 ±1 Mc)

Parameter	Value	Tolerance
Antenna Gain[1]	+4.0 db	±1.0 db
Circuit Loss[2]	0.7 db	±0.1 db
Effective System Noise Temperature[3]	$1020^{\circ}K$	-
Mark and Space Channel Bandwidth	22K	±0.4 db

[1] Referenced to perfectly circular isotropic, pattern maximum.

[2] Total circuit loss from antenna to preamplifier.

[3] Includes contributions due to cosmic noise, low noise preamplifier, and follow-on receiver.

Table 10. Telemetry Parameters

1.	Type Subcarrier Modulation	Two-channel; square wave PSK data channel plus square wave PN sync channel
2.	Transmission Rates	4096, 2048, 1024, and 128 bit/sec
3.	Required ST/N/B for bit error probability $P_e^b = 5 \times 10^{-3}$	+7.3 db-cps/bps $\pm$1.0 db
4.	Modulation Losses	
	a. Bit rates of 4096, 2048, and 1024 bit/sec	
	Date Channel (θ_D = 1.25 radians)	0.91 db $^{+0.21\text{ db}}_{-0.025\text{ db}}$
	Sync Channel (θ_S = 0.32 radians)	20.07 db $^{+1.93\text{ db}}_{-2.29\text{ db}}$
	Carrier Channel	10.48 db $^{+1.49\text{ db}}_{-1.80\text{ db}}$
	b. Bit rates of 128 bps	
	Data Channel (θ_D = 1.13 radians)	1.74 db $^{+0.30\text{ db}}_{-0.35\text{ db}}$
	Sync Channel (θ_S = 0.44 radians)	14.81 db $^{+1.38\text{ db}}_{-1.58\text{ db}}$
	Carrier Channel	8.27 db $^{+1.11\text{ db}}_{-1.28\text{ db}}$
5.	Sync Channel Threshold SNR in $2B_{LO}$	+18.4 db $\pm$1.0 db
6.	Sync Channel Effective Noise Bandwidth	0.5 cps $\pm$0.4 db

Table 11. Command Parameters

1.	Type Subcarrier Modulation	Two-channel; sine wave PSK command channel plus square wave PN sync
2.	Transmission Rate	1 bit/sec
3.	Required Carrier SNR in 20 cps bandwidth at Command Threshold	8.0 db $\pm$1.0 db
4.	Required Command Channel ST/N/B for bit error probability $P_e^b = 1 \times 10^{-5}$	13.7 db $\pm$1.0 db
5.	Command Channel Modulation Loss	9.43 db $^{+0.53\text{ db}}_{-0.50\text{ db}}$
6.	Required Sync Channel SNR at Threshold	13.7 db $\pm$1.0 db
7.	Sync Channel Effective Noise Bandwidth	2.0 cps $\pm$0.8 db
8.	Sync Channel Modulation Loss	6.44 db $^{+0.35\text{ db}}_{-0.47\text{ db}}$
9.	Carrier Channel Modulation Loss	2.13 db $^{+0.22\text{ db}}_{-0.18\text{ db}}$

Table 12. Ranging Channel

1. Type Subcarrier Modulation	Single-channel, square wave
2. Clock Rate	Approximately 500K bit/sec
3. Required Range Code SNR in $2B_{LO}$ = 0.8 cps (threshold)	+22 db $\pm$1.0 db
4. Required Carrier SNR in $2B_{LO}$ (threshold)	6.0 db $^{+0.5\text{ db}}_{-1.0\text{ db}}$
5. Up-Link Modulation Losses (θ_u = 1.25 radians)	
a. Carrier	10.02 db $^{+1.42\text{ db}}_{-1.75\text{ db}}$
b. Range Code	0.46 db $\pm$0.18 db
6. Down-Link Modulation Losses[1]	
a. Carrier	3.87 db $^{+1.84\text{ db}}_{-3.32\text{ db}}$
b. Range Code	10.34 db $^{+4.32\text{ db}}_{-6.72\text{ db}}$
7. Loop Noise Bandwidth	
a. Carrier	12 cps $^{+0.45\text{ db}}_{-0.0\text{ db}}$
b. Range Code	0.8 cps $^{+0.46\text{ db}}_{-0.0\text{ db}}$
8. Sync Channel Loss During Acquisition[2]	12 db

[1]Based on effective range code modulation (θ_D) at encounter range of 0.444 radians and modulation loss due to up-link noise (σ_n^2 = 0.69 rad^2) of 3.00 db.

[2]Occurs during acquisition of X component.

The primary error sources are:

Attitude reference accuracy	± 0.1 degree, 3σ
Limit cycle error	± 0.499 degree, 3σ
Antenna drive program quantization	± 0.5 degree, 3σ
Antenna drive dynamic error	± 0.2 degree, 3σ
Antenna drive pickoff error	± 0.15 degree, 3σ
Antenna boresight error	± 0.2 degree, 3σ
Antenna alignment to spacecraft	± 0.25 degree, 3σ

The RSS total is ± 0.822 degree.

5.2 Link Performance Definitions

5.2.1 Typical Voyager Mission Communications Profile

In order to determine the link performance it is necessary that the critical events in the mission be identified and the parameters affecting the communication capability be determined for each of these events, e.g., range, available antennas, etc. Figure 2 shows communication range versus time from launch for a typical trajectory and is used to determine the mission profile shown in Figure 3. Three midcourse correction maneuvers, a capsule vehicle separation maneuver, and the deboost maneuver at encounter are indicated with typical times and ranges for each. Table 13 gives a more detailed breakdown for each of the maneuver sequences and shows the spacecraft antennas available during each phase of the maneuver. The times given in Table 13 are not critical, from the communication point of view, but are used in plotting the performance margin in Section 5.3 for the various maneuvers.

It should be noted that the ranges (particularly at encounter) indicated in Figures 2 and 3 do not represent worst case maximum values. The maximum range values are:

Encounter	1.85×10^8 km
Encounter plus 1 month	2.3×10^8 km
Encounter plus 6 months	3.9×10^8 km

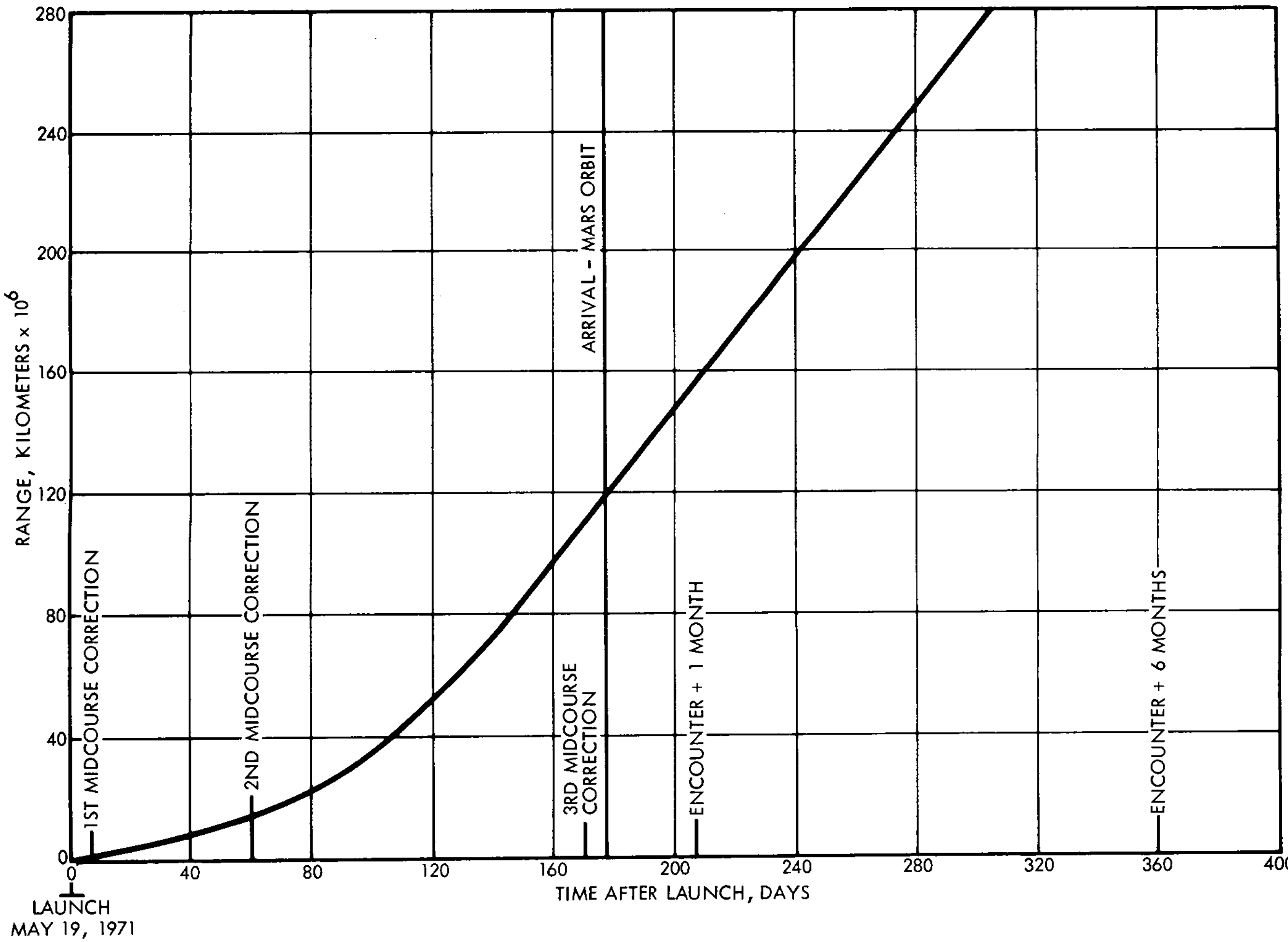


Figure 2. Earth-Spacecraft Range vs. Time After Launch May 19, 1971

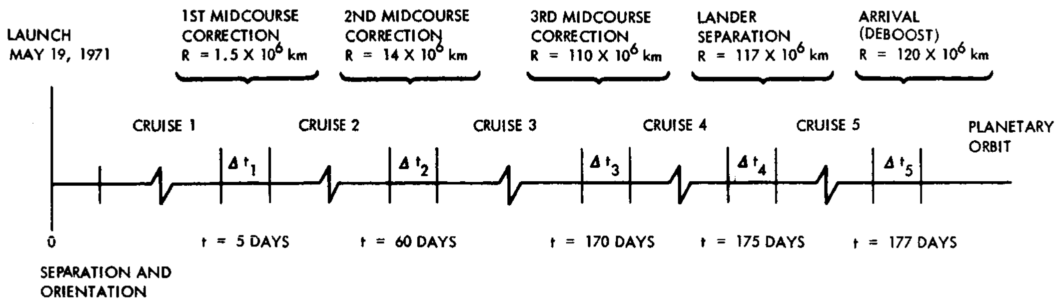


Figure 3. Typical Voyager Mission Profile

Table 13. Maneuver Sequence

	Antennas Available	First Midcourse	Second Midcourse	Third Midcourse	Lander Separation	Arrival (Deboost)
Maneuver preparations	Low Medium High	35.4 min	38.9 min	65.6 min	67.5 min	68.5 min
Point high gain antenna	Low Medium	15.0 min	15.0 min	15.0 min	15.0 min	15.0 min
Verify antenna position	Low Medium	5.1 min	5.8 min	11.1 min	11.5 min	11.7 min
Enable maneuver	Low Medium	5.1 min	5.8 min	11.1 min	11.5 min	11.7 min
Spacecraft maneuver	None	38.0 min	38.0 min	38.0 min	38.0 min	38.0 min
Verify spacecraft position	High only	5.1 min	5.8 min	11.1 min	11.5 min	11.7 min
Enable burn	High only	5.1 min	5.8 min	11.1 min	11.5 min	11.7 min
Burn	High only	10 sec	10 sec	10 sec	10 sec	90 sec
Reorient spacecraft and reacquire sun/Canopus	None	46.0 min	46.0 min	46.0 min	46.0 min	46.0 min
Post maneuver sequence (verification)	Low Medium High	5.1 min	5.8 min	11.1 min	11.5 min	11.7 min
Total maneuver time		Δt_1 = 160 min	Δt_2 = 167 min	Δt_3 = 220 min	Δt_4 = 224 min	Δt_5 = 228 min

The effect on link performance of encountering at maximum range is considered in Section 5.3.

Figures 2 and 3, with the telecommunications design control table presented in Section 5.3, enable a normal mode of operation to be defined for the entire mission. The selection of the normal modes is based largely on the desirability of using the 85-foot antennas in diplex operation at the DSIF. The modes are defined below for the command and telemetry links.

a) Command Link

Spacecraft - low gain antenna up to 50 x 10^6 km (≈120 day) high gain antenna above 50 x 10^6 km (≈ 120 days) medium gain antenna - backup only

DSIF - 85-feet - 10 kw diplex operation throughout

b) Telemetry Link

Spacecraft - low gain until sun/Canopus acquisition, high gain antenna during cruise and orbit, high gain and medium or low gain during maneuvers
1024 bit/sec up to encounter
4096 bit/sec after encounter as long as possible

DSIF - 85-feet - diplex operation 100 x 10^6 km (≈ 150 days) and as backup thereafter to 135 x 10^6 km
210 feet after 100 x 10^6 km (≈ 150 days)

When turn-around ranging is desired it will also be necessary to use the 100 kw transmitter after about 60 x 10^6 km.

5.2.2 Modulation Loss

The modulation loss for a given channel is that fraction of the total received power which does not appear as useful signal in the channel due to the modulation process. For the telemetry link where square-wave subcarriers are used for both the data and sync channels these losses are given by:

$$\text{carrier modulation loss} = \cos^2\beta_D \cos^2\beta_S$$

$$\text{data modulation loss} = \cos^2\beta_S \sin^2\beta_D$$

$$\text{sync modulation loss} = \cos^2\beta_D \sin^2\beta_S$$

and for the command link which has a sinusoidal data subcarrier and squarewave sync subcarrier the losses are given by:

$$\text{carrier modulation loss} = J_0^{\,2}(\beta_D) \cos^2\beta_S$$

$$\text{data modulation loss} = 2J_1^{\,2}(\beta_D) \cos^2\beta_S$$

$$\text{sync modulation loss} = 2J_0^{\,2}(\beta_D) \sin^2\beta_S$$

where β_D and β_S are the modulation indices due to the data and sync channels respectively.

The carrier modulation loss for the PRN turn-around ranging link is a function of the signal-to-noise ratio received at the spacecraft and is treated in Volume 5, Section III, Paragraph 1.3.1.

5.2.3 Performance Margin

The performance margin for each mode of operation is the nominal available signal power in a given channel (carrier, data, or sync) minus the nominal power required in that channel for threshold. Since the various channels do not necessarily threshold together, their performance margins may differ. In this case it is the performance margin for the controlling channel which is plotted in Section 5.3.

5.2.4 Nominal Received Signal Level

The nominal received signal level for a given channel is the total received power (item 11), determined from the nominal values in the telecommunications design control table, minus the nominal modulation loss for that channel - both expressed in decibels.

5.2.5 Nominal Threshold Signal Level

The nominal threshold signal level is the signal level required to satisfy some minimum performance criteria as determined from the nominal values for noise spectral density, bandwidth, communication efficiency, etc., in the telecommunications design control table.

5.2.6 Parameter Tolerances

In tabulating the parameter values in the telecommunications design control tables presented in Section 5.3, the positive and negative tolerances have been assigned in a manner such that tolerances appearing in the negative column always denote a decrease in the signal-to-noise ratio and those appearing in the positive column denote an increase. Thus, a numerical increase in parameters such as the system noise temperature or various losses appears as a negative tolerance in the table. This facilitates the use of the control tables since the tolerances listed in the negative column are really the adverse tolerances and may be added directly to find the sum of the adverse tolerances.

5.3 Link Performance Summary

In this section the link performance for the various modes is summarized in the form of telecommunications design control tables and plots of performance margin versus time or range. All of the control tables except the capsule-spacecraft link have been calculated at a range of 250×10^6 km for convenience and consequently negative performance margins are shown in some cases. This merely implies a nonstandard operation for these modes at the given range and does not affect the performance margin plots for standard operation. The telecommunications design control data is presented in Tables 14 through 21 and performance margin plots are presented in Figures 4 through 15.

5.3.1 Spacecraft-to-Earth Link

Telecommunications design control data is presented in Tables 14 to 17 for the spacecraft-to-earth telemetry link. Figure 4 shows the telemetry link performance margin versus range for the various bit rates and spacecraft antennas. These curves are scaled from a range of 250×10^6 km and do not include the variation in pointing loss with range that will exist for the different antennas. All the curves assume reception is in the diplexed mode with the 85-foot antennas (maser front end). The difference in the sum of the adverse tolerances for 128 bits/sec and the higher bit rates occur because the sync channel threshold is controlling

Table 14. Telecommunications Design Control

Channel: Spacecraft to DSIF
Mode: Two-Channel Telemetry at 128 bit/sec (low gain antenna)

No.	Parameter	Value	Tolerance		Source
1	Total Transmitter Power	43.0 dbm	+1.0	-0.0	VS-4-310
2	Transmitting Circuit Loss	2.65 db	0.8	0.8	VS-4-310
3	Transmitting Antenna Gain (at 45° cone)	2.6 dbi	0.5	0.5	VS-4-310
4	Transmitting Antenna Pointing Loss	(included in 3)			
5	Space Loss 2295 Mc R = 250 x 10^6 km	267.63 db			
6	Polarization Loss	0.01 db	0.01	0.09	
7	Receiving Antenna Gain	61.0 dbi	0.5	0.5	TM-33-83
8	Receiving Antenna Pointing Loss	(included in 7)			
9	Receiving Circuit Loss	0.2 db	0.1	0.1	TM-33-83
10	Net Circuit Loss	206.89 db	1.91	1.99	
11	Total Received Power	-163.89 dbm	2.91	1.99	
12	Receiver Noise Spectral Density (N/B) T System = 24°K ±1°K	-184.81 dbm/cps	0.18	0.18	TM-33-83
13	Carrier Modulation Loss	8.27 db	1.11	1.28	Vol. 5, Sec. III, par. 1.3.2
14	Received Carrier Power	-172.16 dbm	4.02	3.27	
15	Carrier APC Noise BW ($2B_{LO}$ = 12 cps)	10.79 db	0.45	0.0	MC-4-310A
Carrier Performance Tracking (one-way)					
16	Threshold SNR in $2B_{LO}$	0.0 db			MC-4-310A
17	Threshold Carrier Power	-174.02 dbm	0.63	0.18	
18	Performance Margin	10.13 db	3.54	2.17	
Carrier Performance Tracking (two-way)					
19	Threshold SNR in $2B_{LO}$	2.0 db*	+1.0	-1.0	MC-4-310A
20	Threshold Carrier Power	-172.02 dbm	1.63	1.18	
21	Performance Margin	8.13 db	4.54	3.17	
Carrier Performance (telemetry)					
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	MC-4-310A
23	Threshold Carrier Power	-168.02 dbm	1.13	1.18	
24	Performance Margin	-4.14 db	5.15	4.45	
Data Channel					
25	Modulation Loss (1.13 ±5% radian deviation)	1.74 db	0.30	0.35	Vol. 5, Sec. III, par. 1.3.2 VC-4-310
26	Received Data Subcarrier Power	-165.63 dbm	3.21	2.34	
27	Bit Rate (1/T) 128 bit/sec	21.07 db			
28	Required ST/N/B ($P_e = 5 \times 10^{-3}$)	7.3 db	1.0	1.0	Vol. 5, Sec. III, par. 1.3.2
29	Threshold Subcarrier Power	-156.44 dbm	1.18	1.18	
30	Performance Margin	-9.19 db	4.39	3.52	
Sync Channel					
31	Modulation Loss (0.44 ±5% radian deviation)	14.81 db	1.38	1.58	Vol. 5, Sec. III, par. 1.3.2 VC-4-310
32	Received Sync Subcarrier Power	-178.70 dbm	4.29	3.57	
33	Sync APC Noise BW ($2B_{LO}$ = 0.5 cps)	-3.01 db	0.4	0.4	MC-4-310A
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	TR-32-495
35	Threshold Subcarrier Power	-169.42 dbm	1.58	1.58	
36	Performance Margin	-9.28 db	5.87	5.15	

*Based on 3.8 db SNR in spacecraft carrier tracking loop.

Table 15. Telecommunications Design Control

Channel: Spacecraft to DSIF
Mode: Two-Channel Telemetry at 1024 bit/sec (medium gain antenna)

No.	Parameter	Value	Tolerance		Source
1	Total Transmitter Power	43.0 dbm	+1.0	-0.0	VS-4-310
2	Transmitting Circuit Loss	2.30 db	0.8	0.8	VS-4-310
3	Transmitting Antenna Gain	24.0 dbi	0.25	0.5	VS-4-310
4	Transmitting Antenna Pointing Loss	0.2 db	0.2	0.2	Vol. 5, Sec. III, par. 1.5
5	Space Loss 2295 MC R = 250 x 10^6 km	267.63 db			
6	Polarization Loss	0.01 db	0.01	0.09	
7	Receiving Antenna Gain	61.0 dbi	0.5	0.5	TM-33-83
8	Receiving Antenna Pointing Loss	(included in 7)			
9	Receiving Circuit Loss	0.2 db	0.1	0.1	TM-33-83
10	Net Circuit Loss	185.34 db	1.86	2.19	
11	Total Received Power	-142.34 dbm	2.86	2.19	
12	Receiver Noise Spectral Density (N/B) T System = $24^{o}K \pm 1^{o}K$	-184.81 dbm/cps	0.18	0.18	TM-33-83
13	Carrier Modulation Loss	10.48 db	1.49	1.80	Vol. 5, Sec. III, par. 1.3.2
14	Received Carrier Power	-152.82 dbm	4.35	3.99	
15	Carrier APC Noise BW ($2B_{LO}$ = 12 cps)	10.79 db	0.45	0.0	MC-4-310A
Carrier Performance Tracking (one-way)					
16	Threshold SNR in $2B_{LO}$	0.0 db			MC-4-310A
17	Threshold Carrier Power	-174.02 dbm	0.63	0.18	
18	Performance Margin	31.68 db	3.49	2.37	
Carrier Performance Tracking (two-way)					
19	Threshold SNR in $2B_{LO}$	2.0 db*	1.0	1.0	MC-4-310A
20	Threshold Carrier Power	-172.02 dbm	1.63	1.18	
21	Performance Margin	29.68 db	4.49	3.37	
Carrier Performance (Telemetry)					
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	MC-4-310A
23	Threshold Carrier Power	-168.02 dbm	1.13	1.18	
24	Performance Margin	15.20 db	5.48	5.17	
Data Channel					
25	Modulation Loss (1.25 $\pm$5% radian deviation)	0.91 db	0.21	0.25	Vol. 5, Sec. III, par. 1.3.2
26	Received Data Subcarrier Power	-143.25 dbm	3.07	2.44	
27	Bit Rate (1/T) 1024 bit/sec	30.10 db			
28	Required ST/N/B ($P_e = 5 \times 10^{-3}$)	7.3 db	1.0	1.0	Vol. 5, Sec. III, par. 1.3.2
29	Threshold Subcarrier Power	-147.41 dbm	1.18	1.18	
30	Performance Margin	4.16 db	4.25	3.62	
Sync Channel					
31	Modulation Loss (0.32 $\pm$5% radian deviation)	20.07 db	1.93	2.29	Vol. 5, Sec. III, par. 1.3.2
32	Received Sync Subcarrier Power	-162.41 dbm	4.79	4.48	
33	Sync APC Noise BW ($2B_{LO}$ = 0.5 cps)	-3.01 db	0.4	0.4	MC-4-310A
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	TR-32-495
35	Threshold Subcarrier Power	-169.42 dbm	1.58	1.58	
36	Performance Margin	7.01 db	5.37	6.06	

*Based on +3.8 db SNR in the spacecraft carrier tracking loop.

Table 16. Telecommunications Design Control

Channel: Spacecraft to DSIF
Mode: Two-Channel Telemetry at 2048 bit/sec (high gain antenna)

No.	Parameter	Value	Tolerance		Source
1	Total Transmitter Power	43.0 dbm	+1.0	-0.0	VS-4-310
2	Transmitting Circuit Loss	2.05 db	0.8	0.8	VS-4-310
3	Transmitting Antenna Gain	30.0 dbi	0.25	0.5	VS-4-310
4	Transmitting Antenna Pointing Loss	0.5 db	0.5	0.5	Vol. 5, Sec. III, par. 1.5
5	Space Loss 2295 Mc R = 250 x 10^6 km	267.63 db			
6	Polarization Loss	0.01 db	0.01	0.09	
7	Receiving Antenna Gain	61.0 dbi	0.5	0.5	TM-33-83
8	Receiving Antenna Pointing Loss	(included in 7)			
9	Receiving Circuit Loss	0.2 db	0.1	0.1	TM-33-83
10	Net Circuit Loss	179.39 db	2.16	2.49	
11	Total Received Power	-136.39 dbm	3.16	2.49	
12	Receiver Noise Spectral Density (N/B) T System = $24^oK \pm 1^oK$	-184.81 dbm/cps	0.18	0.18	TM-33-83
13	Carrier Modulation Loss	10.48 db	1.49	1.80	Vol. 5, Sec. III, par. 1.3.2
14	Received Carrier Power	-146.87 dbm	4.65	4.29	
15	Carrier APC Noise BW ($2B_{LO}$ = 12 cps)	10.79 db	0.45	0.0	MC-4-310A
	Carrier Performance Tracking (one-way)				
16	Threshold SNR in $2B_{LO}$	0.0 db			MC-4-310A
17	Threshold Carrier Power	-174.02 dbm	0.63	0.18	
18	Performance Margin	37.63 db	3.79	2.67	
	Carrier Performance Tracking (two-way)				
19	Threshold SNR in $2B_{LO}$	2.0 db*	+1.0	-1.0	MC-4-310A
20	Threshold Carrier Power	-172.02 dbm	1.63	1.18	
21	Performance Margin	35.63 db	4.79	3.67	
	Carrier Performance (Telemetry)				
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	MC-4-310A
23	Threshold Carrier Power	-168.02 dbm	1.13	1.18	
24	Performance Margin	21.14 db	5.78	5.47	
	Data Channel				
25	Modulation Loss (1.25 ±5% radian deviation)	0.91 db	0.21	0.25	Vol. 5, Sec. III, par. 1.3.2
26	Received Data Subcarrier Power	-137.30 dbm	3.37	2.74	
27	Bit Rate (1/T) 2048 bit/sec	33.11 db			
28	Required ST/N/B ($P_e = 5 \times 10^{-3}$)	7.3 db	1.0	1.0	Vol. 5, Sec. III, par. 1.3.2
29	Threshold Subcarrier Power	-144.40 dbm	1.18	1.18	
30	Performance Margin	7.10 db	4.55	3.92	
	Sync Channel				
31	Modulation Loss (0.32 ±5% radian deviation)	20.07 db	1.93	2.29	Vol. 5, Sec. III, par. 1.3.2
32	Received Sync Subcarrier Power	-156.46 dbm	4.09	4.78	
33	Sync APC Noise BW ($2B_{LO}$ = 0.5 cps)	-3.01 db	0.4	0.4	MC-4-310A
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	TR-32-495
35	Threshold Subcarrier Power	-169.42 dbm	1.58	1.58	
36	Performance Margin	12.96 db	5.67	6.36	

*Based on +3.8 db SNR in the spacecraft carrier tracking loop.

Table 17. Telecommunications Design Control

Channel: Spacecraft to DSIF
Mode: Two-Channel Telemetry at 4096 bit/sec (high gain antenna)

No.	Parameter	Value	Tolerance		Source
1	Total Transmitter Power	43.0 dbm	+1.0	-0.0	VS-4-310
2	Transmitting Circuit Loss	2.05 db	0.8	0.8	VS-4-310
3	Transmitting Antenna Gain	30.0 dbi	0.25	0.5	VS-4-310
4	Transmitting Antenna Pointing Loss	0.5 db	0.5	0.5	Vol. 5, Sec. III, par. 1.5
5	Space Loss 2295 Mc R = 250 x 10^6 km	267.63 db			
6	Polarization Loss	0.01 db	0.01	0.09	
7	Receiving Antenna Gain	61.0 dbi	0.5	0.5	TM-33-83
8	Receiving Antenna Pointing Loss	(included in 7)			
9	Receiving Circuit Loss	0.2 db	0.1	0.1	TM-33-83
10	Net Circuit Loss	179.39 db	2.16	2.49	
11	Total Received Power	-136.39 dbm	3.16	2.49	
12	Receiver Noise Spectral Density (N/B) T System = $24^{\circ} \pm 1^{\circ}$K	-184.81 dbm/cps	0.18	0.18	TM-33-83
13	Carrier Modulation Loss	10.48 db	1.49	1.80	Vol. 5, Sec. III, par. 1.3.2
14	Received Carrier Power	-146.87 dbm	4.65	4.29	
15	Carrier APC Noise BW ($2B_{LO}$ = 12 cps)	10.79 db	0.45	0.0	MC-4-310A
Carrier Performance Tracking (one-way)					
16	Threshold SNR in $2B_{LO}$	0.0 db			MC-4-310A
17	Threshold Carrier Power	-174.02 dbm	0.63	0.18	
18	Performance Margin	37.63 db	3.79	2.67	
Carrier Performance Tracking (two-way)					
19	Threshold SNR in $2B_{LO}$	2.0 db*	1.0	1.0	MC-4-310A
20	Threshold Carrier Power	-172.02 dbm	1.63	1.18	
21	Performance Margin	35.63 db	4.79	3.67	
Carrier Performance (Telemetry)					
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	MC-4-310A
23	Threshold Carrier Power	-168.02 dbm	1.13	1.18	
24	Performance Margin	21.14 db	5.78	5.47	
Data Channel					
25	Modulation Loss (1.25 ±5% radian deviation)	0.91 db	0.21	0.25	Vol. 5, Sec. III, par. 1.3.2
26	Received Data Subcarrier Power	-137.30 dbm	3.37	2.74	
27	Bit Rate (1/T) 4096 bit/sec	36.12 db			
28	Required ST/N/B ($P_e = 5 \times 10^{-3}$)	7.3 db	1.0	1.0	Vol. 5, Sec. III, par. 1.3.2
29	Threshold Subcarrier Power	-141.39 dbm	1.18	1.18	
30	Performance Margin	4.09 db	4.55	3.92	
Sync Channel					
31	Modulation Loss (0.32 ±5% radian deviation)	20.07 db	1.93	2.29	Vol. 5, Sec. III, par. 1.3.2
32	Received Sync Subcarrier Power	-156.46 dbm	4.09	4.78	
33	Sync APC Noise BW ($2B_{LO}$ = 0.5 cps)	-3.01 db	0.4	0.4	MC-4-310A
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	TR-32-495
35	Threshold Subcarrier Power	-169.42 dbm	1.58	1.58	
36	Performance Margin	12.96 db	5.67	6.36	

*Based on +3.8 db SNR in the spacecraft carrier tracking loop.

for 128 bits/sec whereas the data channel threshold controls for all the higher rates. Note that only the performance margin for the controlling channel is plotted, the other channels automatically having better performance.

Figures 4 and 2 are used together to obtain Figure 5 which shows the performance margin versus time from launch for a typical Voyager trajectory. Several operational modes are shown with the normal mode indicated by a heavy line. The effect of antenna pointing errors (Section 7) for the medium gain and low gain antennas is included in the curves shown in Figure 5. Pointing error of the double gimballed high gain antenna is kept within ± 0.5 db independently of range.

The 1024 bits/sec high power, high gain to 85-foot diplex mode will give satisfactory performance out to about 130×10^6 km. However, during the third midcourse correction, capsule vehicle separation and encounter retropropulsion maneuver when only the medium gain spacecraft antenna is available for confirmation of high gain antenna angles, it is necessary to use the 210-foot antenna at the DSIF to attain satisfactory performance at 1024 bits/sec. After encounter, use of the 210-foot antenna permits satisfactory performance at 4096 bits/sec out to about 240×10^6 km and at 1024 bits/sec out to about 490×10^6 km which includes the worst case encounter plus 1 month and encounter plus 6 months ranges, respectively.

Figures 6 through 9 show the performance margin during the first, second, and third midcourse correction and encounter deboost maneuvers, respectively, using the mission profile shown in Figure 3 and Table 13. Alternate operational modes are shown in addition to the normal mode (indicated by a heavy line).

5.3.2 Earth-to-Spacecraft Link

Tables 18 and 19 present the telecommunications design control tables for the earth-to-spacecraft command link for the low gain spacecraft antenna. Performance is improved by 21.2 db with the medium gain antenna and by 26.9 db with the high gain antenna. Figure 10 shows

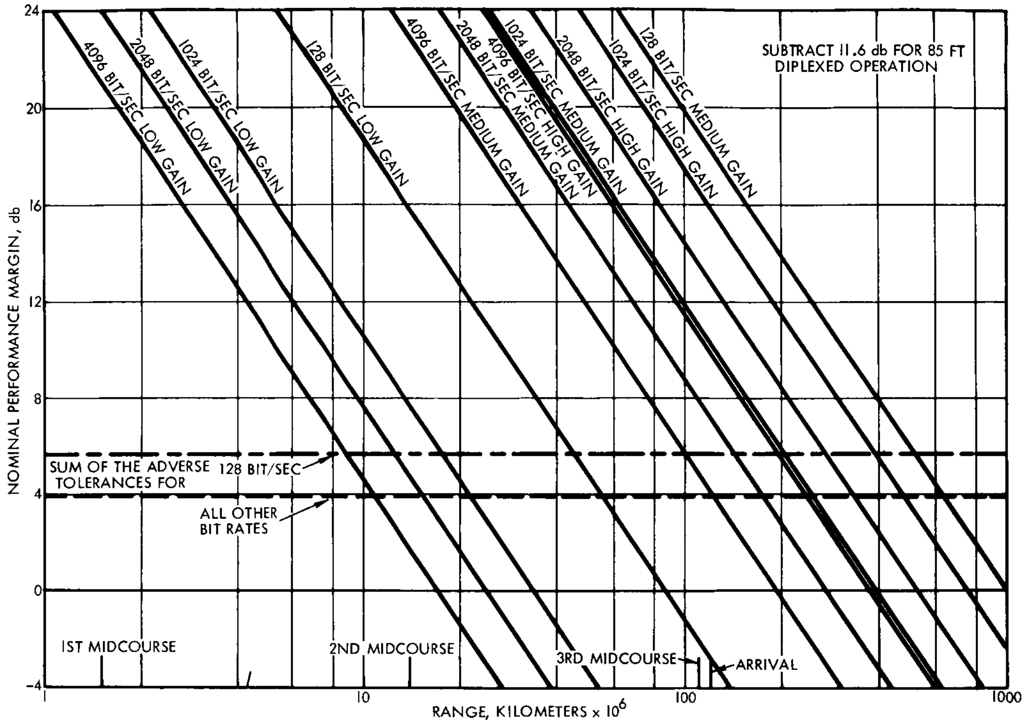


Figure 4. Performance Margin vs. Range Telemetry Link 210' Antenna

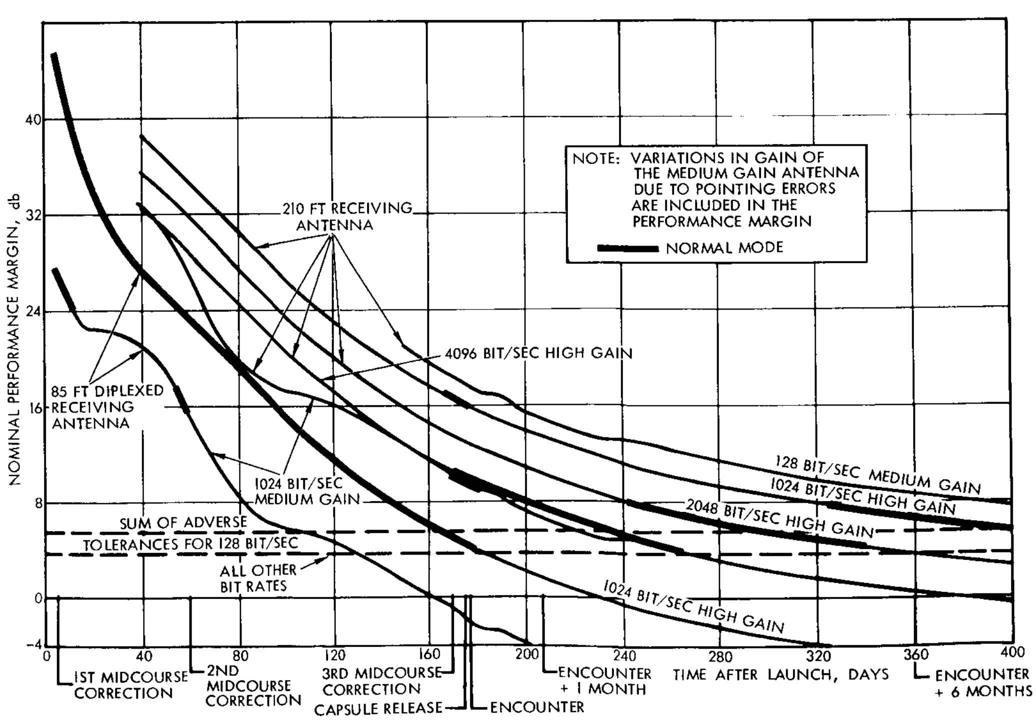


Figure 5. Typical Voyager Performance Margin vs. Time Telemetry Link Spacecraft-to-Earth

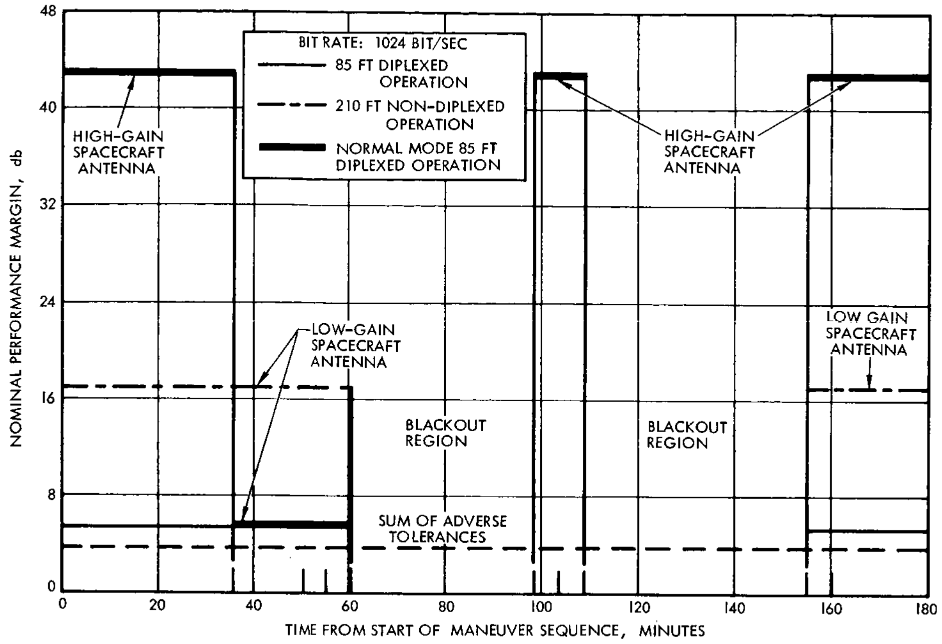


Figure 6. Typical Performance Margin for First Midcourse Correction Telemetry Link

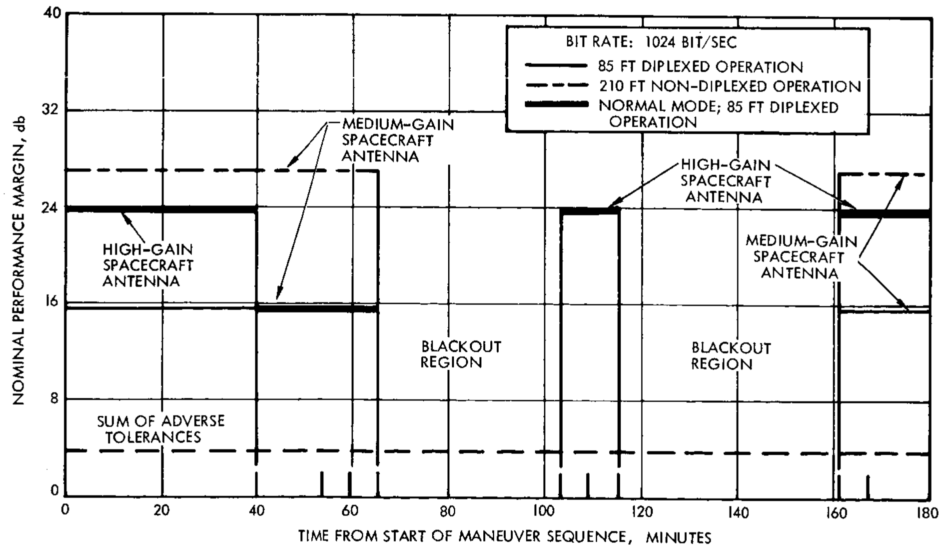


Figure 7. Typical Performance Margin for Second Midcourse Maneuver Telemetry Link

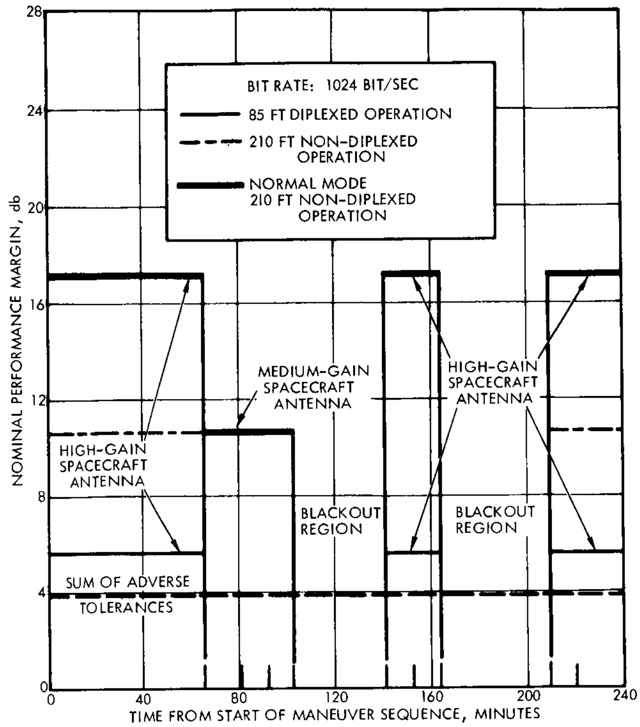


Figure 8. Typical Performance Margin for Third Midcourse Maneuver Telemetry Link

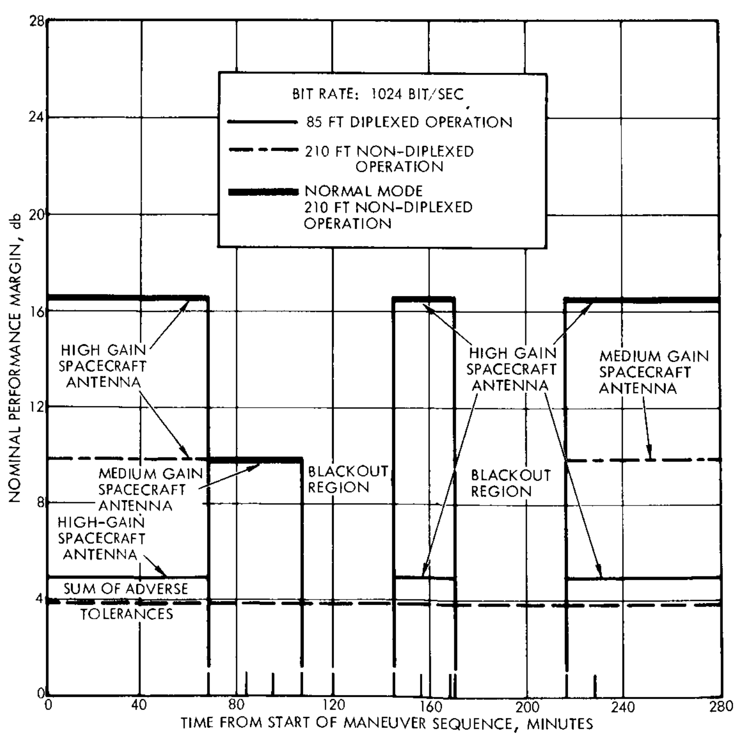


Figure 9. Typical Performance Margin for Encounter Deboost Maneuver Telemetry Link

Table 18. Telecommunications Design Control

Channel: DSIF to Spacecraft (10 kw diplexed)
Mode: Two-Channel PSK with PN Sync

No.	Parameter	Value	Tolerance		Source
1	Total Transmitter Power	70 dbm	+0.5	-0.0	MC-4-310A
2	Transmitting Circuit Loss	0.4 db	0.1	0.1	MC-4-310A
3	Transmitting Antenna Gain	51.0 dbi	1.0	0.5	MC-4-310A
4	Transmitting Antenna Pointing Loss	(included in 3)			
5	Space Loss 2115 MC $R = 250 \times 10^6$km	266.92 db			
6	Polarization Loss	0.15 db	0.15	0.58	
7	Receiving Antenna Gain (at 45° cone)	2.1 dbi	0.5	0.5	VS-4-310
8	Receiving Antenna Pointing Loss	(included in 7)			
9	Receiving Circuit Loss	1.26 db	0.2	0.2	VS-4-310
10	Net Circuit Loss	215.63 db	1.95	1.88	
11	Total Received Power	-145.63 dbm	2.45	1.88	
12	Receiver Noise Spectral Density (N/B) T System = 2610°K (10 db noise figure)	-164.44 dbm/cps	1.12	2.18	VS-4-310
13	Carrier Modulation Loss	2.13 db	0.22	0.18	Vol. 5, Sec. III, par. 1.3.3
14	Received Carrier Power	-147.76 dbm	2.67	2.06	
15	Carrier APC Noise BW ($2B_{LO}$ = 20 cps)	13.01 db	0.46	0.41	MC-4-310A
Carrier Performance Tracking (one way)					
16	Threshold SNR in $2B_{LO}$				
17	Threshold Carrier Power				
18	Performance Margin				
Carrier Performance Tracking (two-way)					
19	Threshold SNR in $2B_{LO}$	3.8 db*			MC-4-310A
20	Threshold Carrier Power	-147.63 dbm	1.58	2.59	
21	Performance Margin	2.00 db	3.43	4.47	
Carrier Performance (Command)					
22	Threshold SNR in $2B_{LO}$	8.0 db	1.0	1.0	MC-4-310A
23	Threshold Carrier Power	-143.43 dbm	2.58	3.59	
24	Performance Margin	-4.33 db	5.25	5.65	
Data Channel					
25	Modulation Loss (0.584 $\pm$5% radian deviation)	9.43 db	0.53	0.50	Vol. 5, Sec. III, par. 1.3.3
26	Received Data Subcarrier Power	-155.06 dbm	2.98	2.38	
27	Bit Rate (1/T) 1 bit/sec	0.0 db			
28	Required ST/N/B	13.7 db	1.0	1.0	MC-4-322B
29	Threshold Subcarrier Power	-150.74 dbm	2.12	3.18	
30	Performance Margin	-4.32 db	5.10	5.56	
Sync Channel					
31	Modulation Loss (0.547 $\pm$5% radian deviation)	6.44 db	0.35	0.47	Vol. 5, Sec. III, par. 1.3.3
32	Received Sync Subcarrier Power	-152.07 dbm	2.80	2.35	
33	Sync APC Noise BW ($2B_{LO}$ = 2 cps)	3.0 db	0.8	0.8	MC-4-310A
34	Threshold SNR in $2B_{LO}$	13.7 db	1.0	1.0	MC-4-322B
35	Threshold Subcarrier Power	-147.74 dbm	2.92	3.98	
36	Performance Margin	-4.33 db	5.72	6.33	

*+3.8 db SNR is required to limit the ground receiver degradation to +2.0 db.

Table 19. Telecommunications Design Control

Channel: DSIF to Spacecraft (100 kw)
Mode: Two-Channel PSK with PN Sync

No.	Parameter	Value	Tolerance		Source
1	Total Transmitter Power	80.0 dbm	-	-	MC-4-310A
2	Transmitting Circuit Loss	0.2 db	+0.1	-0.1	MC-4-310A
3	Transmitting Antenna Gain	53.0 dbi	1.0	0.5	MC-4-310A
4	Transmitting Antenna Pointing Loss	(included in 3)			
5	Space Loss 2115 MC $R = 250 \times 10^6$ km	266.92 db			
6	Polarization Loss	0.13 db	0.13	0.48	
7	Receiving Antenna Gain (at 45° cone)	2.1 dbi	0.5	0.5	VS-4-310
8	Receiving Antenna Pointing Loss	(included in 7)			
9	Receiving Circuit Loss	1.26 db	0.2	0.2	VS-4-310
10	Net Circuit Loss	213.41 db	1.93	1.78	
11	Total Received Power	-133.41 dbm	1.93	1.78	
12	Receiver Noise Spectral Density (N/B) T System = 2610°K (10 db noise figure)	-164.44 dbm/cps	1.12	2.18	VS-4-310
13	Carrier Modulation Loss	2.13 db	0.22	0.18	Vol. 5, Sec. III, par. 1.3.3
14	Received Carrier Power	-135.54 dbm	2.15	1.96	
15	Carrier APC Noise BW ($2B_{LO}$ = 20 cps)	13.01 db	0.46	0.41	MC-4-310A
	Carrier Performance Tracking (one-way)				
16	Threshold SNR in $2B_{LO}$				
17	Threshold Carrier Power				
18	Performance Margin				
	Carrier Performance Tracking (two-way)				
19	Threshold SNR in $2B_{LO}$	3.8 db*			MC-4-310A
20	Threshold Carrier Power	-147.63 dbm	1.58	2.59	
21	Performance Margin	14.22 db	3.51	4.37	
	Carrier Performance (Command)				
22	Threshold SNR in $2B_{LO}$	8.0 db	1.0	1.0	MC-4-310A
23	Threshold Carrier Power	-143.43 dbm	2.58	3.59	
24	Performance Margin	7.89 db	4.73	5.55	
	Data Channel				
25	Modulation Loss (0.584 $\pm$5% radian deviation)	9.43 db	0.53	0.50	Vol. 5, Sec. III, par. 1.3.3
26	Received Data Subcarrier Power	-142.84 dbm	2.46	2.28	
27	Bit Rate (1/T) 1 bit/sec	0.0 db			
28	Required ST/N/B	13.7 db	1.0	1.0	MC-4-322B
29	Threshold Subcarrier Power	-150.74 dbm	2.12	3.18	
30	Performance Margin	7.90 db	4.58	5.46	
	Sync Channel				
31	Modulation Loss (0.547 $\pm$5% radian deviation)	6.44 db	0.35	0.47	Vol. 5, Sec. III, par. 1.3.3
32	Received Sync Subcarrier Power	-139.85 dbm	2.28	2.25	
33	Sync APC Noise BW ($2B_{LO}$ = 2 cps)	3.01 db	0.8	0.8	MC-4-310A
34	Threshold SNR in $2B_{LO}$	13.7 db	1.0	1.0	MC-4-322B
35	Threshold Subcarrier Power	-147.74 dbm	2.92	3.98	
36	Performance Margin	7.89 db	5.20	6.23	

*+3.8 db SNR is required to limit the ground receiver degradation to +2.0 db.

the command link performance margin versus range for the different modes of operation. As in the case of the telemetry link, the curves are scaled from 250 x 10^6 km and do not account for variations if the antenna pointing loss. Figure 11 shows the performance margin versus time from launch for the same trajectory used in determining telemetry link performance. Again, the performance for several modes is shown with the normal mode indicated by a heavy line. The variation in pointing loss for the medium and low gain spacecraft antennas is included in the curves of Figure 11.

In the case of the command link all the channels threshold together so that the performance margin plotted is equally valid for the carrier, data, and sync channels. However, the sum of the adverse tolerances differs somewhat between channels with the sync channel being the highest and it is this value which is plotted for the sum of the adverse tolerances in Figures 10 through 15.

It can be seen from Figures 10 and 11 that 85-foot 10 kw diplexed operation permits a satisfactory command link with the medium gain spacecraft antenna to beyond 600 x 10^6 km. Command can be effected to about 250 x 10^6 km with the low gain antenna if the 100 kw transmitter is used. This is beyond the worst case encounter plus 1 month range.

Performance margins for the first, second, and third midcourse corrections and encounter deboost maneuvers are shown in Figures 12 through 15, respectively.

5.3.3 Capsule-to-Spacecraft Link

Table 20 presents the telecommunications design control table for the lander capsule-to-spacecraft link. This link is analyzed in detail in Appendix F-2 of Volume 5. Table 20 summarizes the results of that analysis. The 6 db negative tolerance indicated for the space loss (item 5) is the allowance for signal fading. The 40, 000 km range is the maximum range between the capsule and spacecraft just before capsule impact.

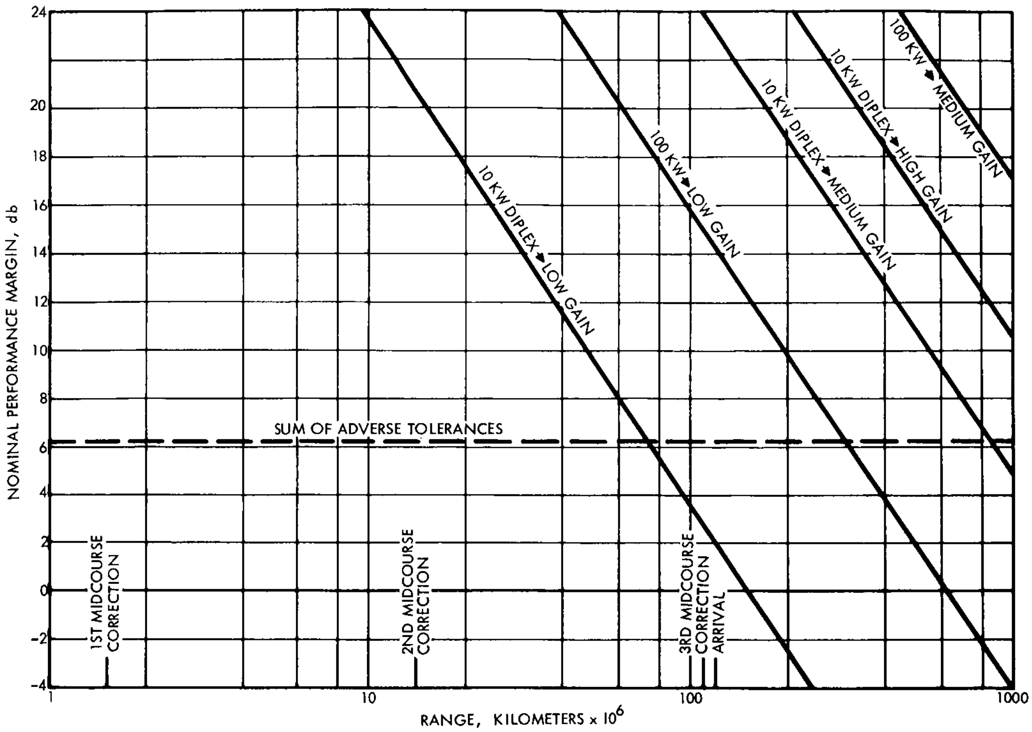


Figure 10. Performance Margin vs. Range Command Link

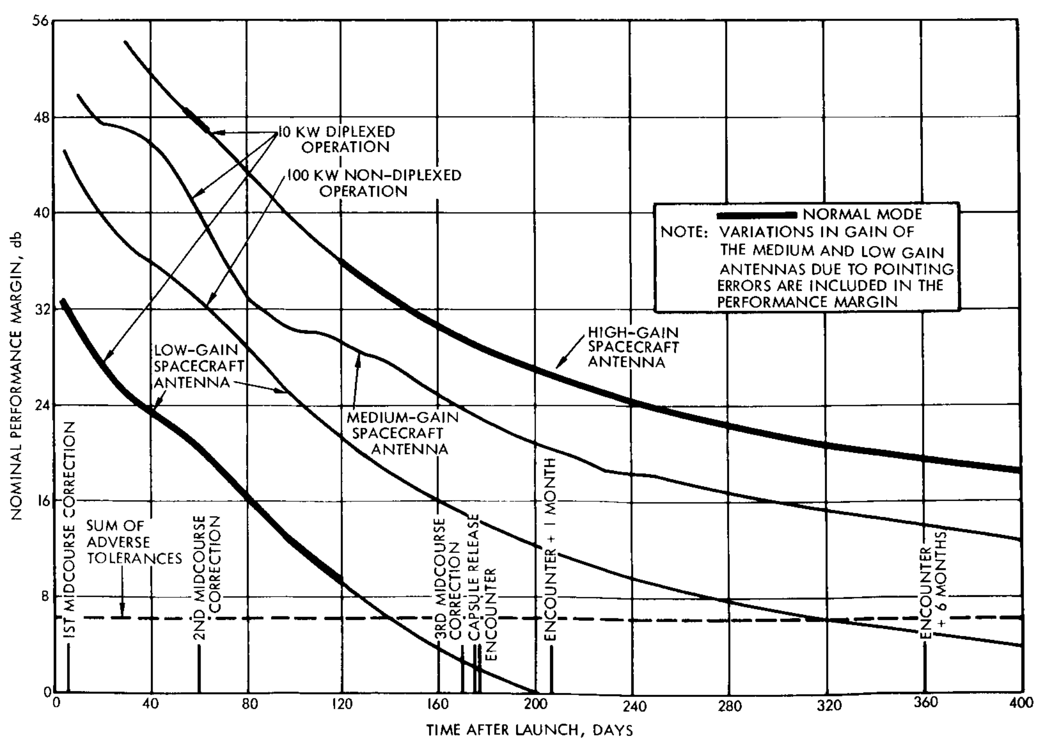


Figure 11. Typical Voyager Performance Margin vs. Time Command Link Earth-to-Spacecraft

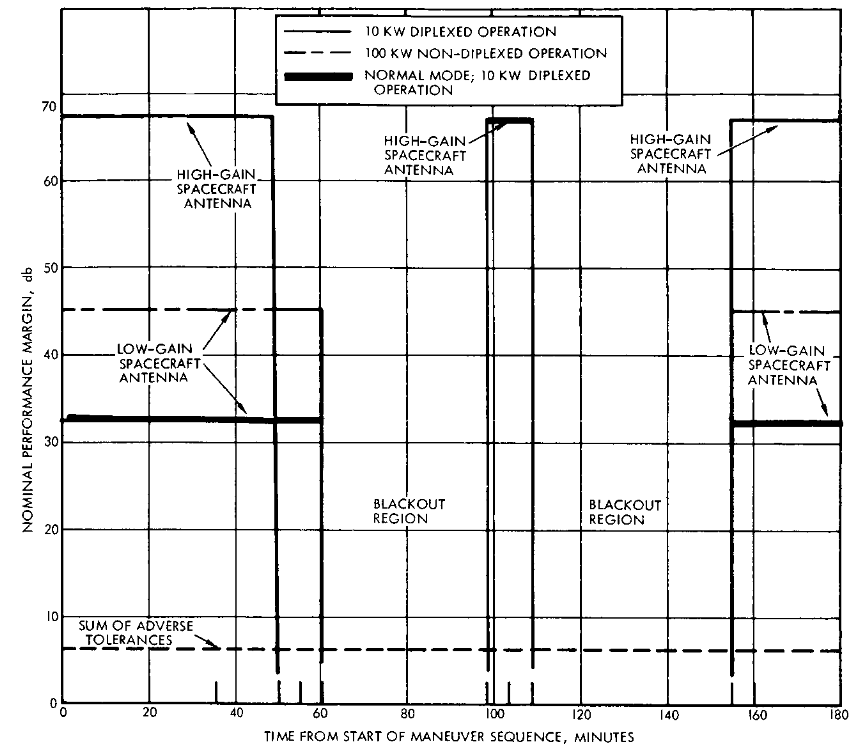


Figure 12. Typical Performance Margin for First Midcourse Maneuver Command Link

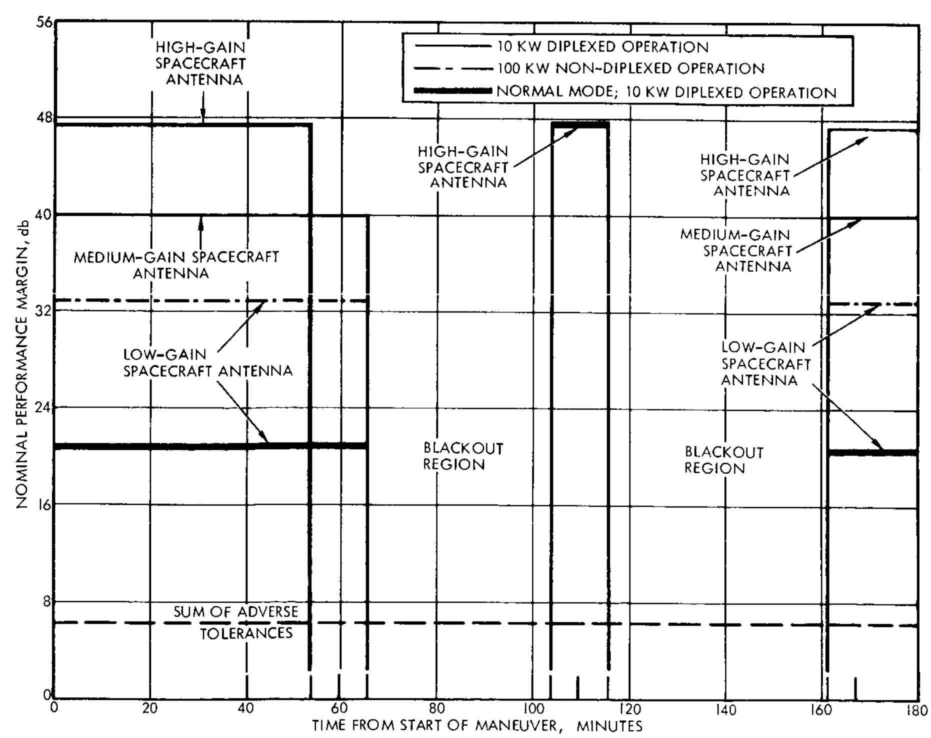


Figure 13. Typical Performance Margin for Second Midcourse Maneuver Command Link

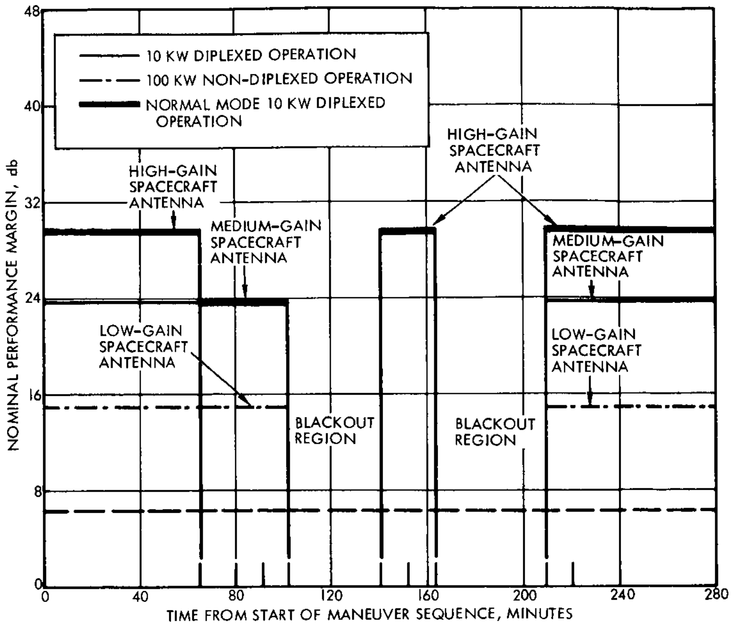


Figure 14. Typical Performance Margin for Third Midcourse Maneuver Command Link

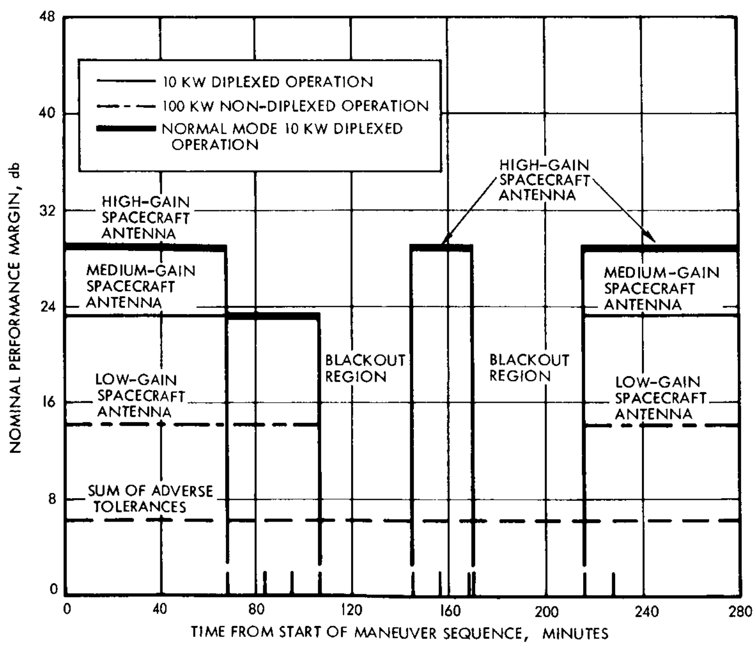


Figure 15. Typical Performance Margin for Encounter Deboost Maneuver Command Link

Table 20. Telecommunications Design Control

Channel: Capsule to Spacecraft
Mode: Noncoherent FSK

No.	Parameter	Value	Tolerance		Source
1	Total Transmitter Power	43.0 dbm	+1.0	-1.0	VS-4-310
2	Transmitting Circuit Loss	0.5 db	0.2	0.2	Appendix F-2, Vol. 5
3	Transmitting Antenna Gain	0.0 dbi	0.5	1.0	Appendix F-2, Vol. 5
4	Transmitting Antenna Pointing Loss	(included in 3)			
5	Space Loss 136 Mc R = 40,000 km	167.2 db	0.0	6.0*	Appendix F-2, Vol. 5
6	Polarization Loss	0.2 db	0.1	0.9	
7	Receiving Antenna Gain	4.0 dbi	1.0	1.0	VS-4-310
8	Receiving Antenna Pointing Loss	0.5 db	0.5	0.5	
9	Receiving Circuit Loss	0.7 db	0.1	0.1	VS-4-310
10	Net Circuit Loss	165.1 db	2.4	9.7	
11	Total Received Power	-122.1 db	3.4	10.7	
12	Receiver Noise Spectral Density (N/B) T System = 1020°K	-168.5 dbm/cps	1.0	1.0	Appendix F-2, Vol. 5
13	Receiver Predetection Noise Bandwidth (22 kc)	43.4 db	0.4	0.4	Appendix F-2, Vol. 5
14	Received C/N (in 22 kc)	3.0 db	4.8	12.1	
15	Post-detection S/N (in 11 kc)	2.1 db	5.3	16.0	Appendix F-2, Vol. 5 Figure 2
16	Bit Rate (1/T) 10 bit/sec	10 db			
17	Received ST/N/B	32.5 db	5.3	16.0	
18	Required ST/N/B ($P_e^b = 10^{-3}$)	14.0 db	1.0	2.0	Appendix F-2, Vol. 5
19	Performance Margin	18.5 db	6.3	18.0	

*Allowance for fading

The antenna pointing loss (item 8) is for the capsule-spacecraft vector associated with the maximum range (see Volume 4, Section 5.6). The relatively high value for the sum of the adverse tolerances results from the allowance required for fading and the low input signal-to-noise ratio at the input of the envelope detector. However, the control tables shows that satisfactory performance can be attained at 10 bits/sec in the worst case.

5.3.4 PRN Turn-around Ranging Link

Table 21 shows the telecommunications design control table for the PRN turn-around ranging (Mark I system) at a range of 250 x 10^6 km. The performance margin shows that satisfactory ranging is possible to this range with the 100 kw transmitter and 210-foot receiving antenna at the DSIF. Diplex operation with the 85-foot antenna and 10 kw transmitter will permit ranging to about 60 x 10^6 kw. Beyond this range the 100 kw transmitter is required for satisfactory performance.

5.3.5 Launch Mode Link

Table 22 shows the telecommunications design control table for the telemetry link during the launch phase. The calculations assume a 30-foot diameter antenna at the spacecraft monitoring station at Cape Kennedy (DSIF Station 71). The effective gain of the spacecraft transmitting antenna is assumed to be -30 dbi in the direction of the launch site for this phase. It is seen from Table 22 that there is an ample performance margin for 1024 bits/sec at 1000 km even with a 10-foot antenna. However, the effects of flame attunuation have not been included in the performance margin. The exhaust plume can easily cause the signal to be attunuated by 20 db or more depending on the aspect angle for a particular launch. This may cause the system performance to be marginal or inadequate near the maximum visible range or near burnout. If this turns out to be the case, it may be necessary to reduce the data rate to 128 bits/sec or require the spacecraft antenna gain to be greater than -30 dbi.

Table 21. Telecommunications Design Control

Channel: DSIF (100 kw 85 ft) to Spacecraft to DSIF (210 ft)
Mode: PRN Turnaround Ranging (Mark I System)

No.	Parameter	Value	Tolerance		Source
Up-Link					
1	Total Transmitter Power	80.0 dbm	-	-	MC-4-310A
2	Transmitting Circuit Loss	0.2 db	+0.1	-0.1	MC-4-310A
3	Transmitting Antenna Gain	53.0 dbi	1.0	0.5	MC-4-310A
4	Transmitting Antenna Pointing Loss	(included in 3)			
5	Space Loss 2115 Mc $R = 250 \times 10^6$ km	266.92 db			
6	Polarization Loss	0.13 db	0.12	0.33	
7	Receiving Antenna Gain	29.5 dbi	0.25	0.5	VS-4-310
8	Receiving Antenna Pointing Loss	0.5 db	0.5	0.5	Vol. 5, Sec. III, par. 1.5
9	Receiving Circuit Loss	1.36 db	0.2	0.2	VS-4-310
10	Net Circuit Loss	186.51 db	2.17	2.13	
11	Total Received Power	-106.51 dbm	2.17	2.13	
12	Receiver Noise Spectral Density (N/B) T System = 2610°K (10 db noise figure)	-164.44 dbm/cps	1.12	2.18	VS-4-310
13	Receiver Noise Bandwidth (3.3 mc)	65.18 db	0.45	0.42	MC-4-320A
14	Total Received SNR (in 3.3 mc) = $1/\beta$	-7.35 db	2.74	4.73	
15	Limiter Signal Suppression Factor $\alpha_s^2 = 1/(1 + \frac{4}{\pi}\beta)$	8.99 db	2.99	4.34	
16	Limiter Noise Suppression Factor $\alpha_n^2 = \frac{\beta}{\beta + 2}$	1.37 db	0.92	0.87	
Up-Link Carrier Tracking Performance					
17	Carrier Modulation Loss	10.02 db	1.42	1.75	
18	Received Carrier Power	-116.63 dbm	3.59	3.88	
19	Carrier APC Noise BW ($2B_{LO}$ = 20 cps)	13.01 db	0.46	0.41	
20	Threshold SNR in $2B_{LO}$	8.0 db*	1.0	1.0	
21	Threshold Carrier Power	-143.43 dbm	2.58	3.59	
22	Performance Margin	26.80 db	6.17	7.47	
Down-Link					
23	Total Transmitter Power	43.0 dbm	+1.0	-0.0	VS-4-310
24	Transmitting Circuit Loss	2.05 db	+0.8	0.8	VS-4-310
25	Transmitting Antenna Gain	30.0 dbi	+0.25	0.5	VS-4-310
26	Transmitting Antenna Pointing Loss	0.5 db	+0.5	-0.5	VS-4-310
27	Space Loss 2295 Mc $R = 250 \times 10^6$ km	267.63 db			
28	Polarization Loss	0.01 db	0.01	0.09	
29	Receiving Antenna Gain	61.0 dbi	0.5	0.5	TM-33-83
30	Receiving Antenna Pointing Loss	(included in 29)			
31	Receiving Circuit Loss	0.2 db	0.1	0.1	TM-33-83
32	Net Circuit Loss	179.39 db	2.16	2.49	
33	Total Received Power	-136.39 dbm	3.16	2.49	
34	Receiver Noise Spectral Density (N/B) T System = $24^\circ K \pm 1^\circ K$	-184.81 dbm/cps	0.18	0.18	TM-33-83
35	Up-Link Contribution to Noise Spectral Density	0.07 db	0.05	0.19	Vol. 5, Sec. III, par. 1.3.1
36	Total Effective Noise Spectral Density	-184.74 dbm/cps	0.23	0.37	
37	Carrier Modulation Loss $\theta_D' = 0.444$; $\sigma_n^2 = 0.690$	3.87 db	1.84	3.32	Vol. 5, Sec. III, par 1.3.1
38	Received Carrier Power	-140.26 dbm	5.00	5.81	
39	Carrier APC Noise BW ($2B_{LO}$ = 12 cps)	10.79	0.45	0.0	TM-33-83

*The threshold SNR in $2B_{LO}$ is assumed to include any limiter effects.

Table 21. Telecommunications Design Control (Continued)

Carrier Performance					
40	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	MC-4-310A
41	Threshold Carrier Power	-167.95	1.18	1.37	
42	Performance Margin	27.69 db	6.18	7.18	
Ranging Performance					
43	Modulation Loss $\theta_D' = 0.444$, $\sigma_n^2 = 0.690$	10.34 db	4.32	6.72	Vol. 5, Sec. III, par. 1.3.1
44	Received Ranging Power	-146.73 dbm	7.48	9.21	
45	Ranging APC Noise BW ($2B_{LO}$ = 0.8 cps)	-0.97 db	0.46	0.0	
46	Threshold SNR in $2B_{LO}$	22.0 db	1.0	1.0	Vol. 5, Sec. III, par. 1.3.1
47	Threshold Ranging Power	-163.71 dbm	1.69	1.37	
48	Performance Margin	16.98 db	9.17	10.58	

Table 22. Telecommunications Design Control

Channel: Spacecraft to DSIF (Station 71)
Mode: Two-Channel Telemetry at 1024 bit/sec (launch phase)

No.	Parameter	Value	Tolerance		Source
1	Total Transmitter Power	30 dbm	+1.0	-0.0	
2	Transmitting Circuit Loss	1.60 db	0.8	0.8	
3	Transmitting Antenna Gain	-30.0 dbi	1.0	1.0	
4	Transmitting Antenna Pointing Loss	(included in 3)			
5	Space Loss	159.7 db			
	2295 Mc R = 1000 km				
6	Polarization Loss	0.5 db	0.3	0.3	
7	Receiving Antenna Gain (30 ft)	44.2 dbi	0.5	0.5	
8	Receiving Antenna Pointing Loss	(included in 7)			
9	Receiving Circuit Loss	0.5 db	0.2	0.2	
10	Net Circuit Loss	147.1 db	2.8	2.8	
11	Total Received Power	-117.1 dbm	3.8	2.8	
12	Receiver Noise Spectral Density (N/B)	-174.3 dbm/cps	1.1	0.9	
	T System = $270^{o}K \pm 60^{o}K$				
13	Carrier Modulation Loss	10.48 db	1.49	1.80	
14	Received Carrier Power	-127.58 dbm	5.29	4.6	
15	Carrier APC Noise BW ($2B_{LO}$ = 12 cps)	10.79 db	0.45	0.0	
Carrier Performance Tracking (one-way)					
16	Threshold SNR in $2B_{LO}$	0.0 db			
17	Threshold Carrier Power	-163.51 dbm	1.55	0.9	
18	Performance Margin	46.41 db*	5.35	3.8	
Carrier Performance Tracking (two-way)					
19	Threshold SNR in $2B_{LO}$				
20	Threshold Carrier Power				
21	Performance Margin				
Carrier Performance (Telemetry)					
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	
23	Threshold Carrier Power	-157.51 dbm	2.05	1.9	
24	Performance Margin	29.93 db*	7.3	6.5	
Data Channel					
25	Modulation Loss	0.91 db	0.21	0.25	
26	Received Data Subcarrier Power	-118.01 dbm	4.0	3.0	
27	Bit Rate (1/T) 1024 bit/sec	30.10 db			
28	Required ST/N/B ($P_e = 5 \times 10^{-3}$)	7.3 db	1.0	1.0	
29	Threshold Subcarrier Power	-136.9 dbm	2.1	1.9	
30	Performance Margin	18.91 db*	6.1	4.9	
Sync Channel					
31	Modulation Loss	20.07 db	1.93	2.29	
32	Received Sync Subcarrier Power	-137.17 dbm	5.7	5.1	
33	Sync APC Noise BW ($2B_{LO}$ = 0.5 cps)	-3.01 db	0.4	0.4	
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	
35	Threshold Subcarrier Power	-158.91 dbm	2.5	2.3	
36	Performance Margin	21.74 db*	8.2	7.4	

*In the case of a 10-ft dish these margins will be reduced by 9.5 db.

5.4 Equipment Parameters

5.4.1 S-Band Modulator-Exciter

a. Frequency

The nominal center frequency is 2295 Mc (DSIF channel 14a). The exciter is capable of operation in the range 2290 to 2300 Mc with the installation of the proper crystals.

b. Power Output

The power output is 100 milliwatts minimum into a 50-ohm load having VSWR no greater than 1.6 to 1. The variation in output power due to all environmental conditions, input voltage variations, and RF load variations, is less than ± 0.6 db.

The exciter is capable of operating into a 50-ohm load with a VSWR of 5 to 1 or less. It is also designed to operate into an open or short circuit without permanent degradation.

c. Phase Characteristics

Phase Control. The unmodulated phase of the carrier is determined by the phase of the receiver voltage controlled oscillator (VCO) or an auxiliary crystal controlled oscillator as selected by the "in-lock" signal or ground command.

Phase Stability. The phase jitter of the unmodulated carrier when operating in the auxiliary oscillator mode is such as to cause not more than 6 degrees peak phase error in a noise-free, phase-coherent receiver having a strong signal bandwidth ($2\ B_L$) of 20 cps and a transfer function of:

$$H(S) = \frac{\theta_{out}}{\theta_{in}} = \frac{1 + \frac{3}{4B_L} S}{1 + \frac{3}{4\ B_L} S + \frac{9}{32B_L} S^2}$$

Under the same test conditions but operating in a coherent mode the phase jitter is 12 degrees peak.

The additional phase error above residual due to vibration is within the tracking capabilities of a receiver with a strong signal bandwidth ($2B_L$) of 100 cps and transfer function having the same form as above.

d. Frequency Stability

Auxiliary Oscillator

a) The short term stability is 1 part in 10^7 (for 20 minutes).

b) The long term stability if 1 part in 10^6 (for 12 hours).

c) The output frequency of the unmodulated carrier is 2295 Mc ± 0.002 per cent under all conditions.

e. Modulation Characteristics

The exciter is phase modulated. The phase modulator is capable of modulating the carrier ± 4 radians at the output frequency.

Linearity. The phase modulation characteristic does not deviate from a linear characteristic by more than ± 2.5 per cent of the deviation for peak deviations of up to ± 4 radians.

Modulation Sensitivity. There are two modulation inputs which are isolated from each other by at least 30 db. The modulation sensitivity is 1 rad ± 5 per cent per volt peak for the telemetry imput and 2 rad ± 5 per cent per volt peak for the ranging input.

Modulation Stability. The modulation sensitivity is stable to within ± 5 per cent of the nominal value.

Bandwidth. The over-all bandwidth of the exciter is such that the first sideband of a phase modulated output signal remains within $\pm 1/2$ db from 10 cps to 1.0 Mc and is down no more than 3 db at 1.8 Mc.

Modulator Input Impedance. The input impedance is 1000 ohms for telemetry input and 50 ohms for the ranging input. The modulator is designed to operate from a source impedance of less than 70 ohms.

f. Spurious Emission Limits

The maximum level of any spurious outputs is 50 db below the unmodulated carrier.

g. Input Impedance

The RF input impedance of the exciter is 50 ohms with a VSWR not exceeding 1.2:1.

h. Output Impedance

The RF output impedance of the exciter is 50 ohms with a VSWR not exceeding 1.4:1.

5.4.2 Four-Port Hybrid Ring and Exciter Power Monitors

a. Impedance

The input impedance of any port is 50 ohms with a VSWR not exceeding 1.3:1.

b. Insertion Loss

The relative power level between either output port and each input port does not exceed -3.5 db at 2295 Mc (including power monitor losses).

c. Isolation

The isolation between the two input ports is at least 25 db at 2295 Mc

d. RF Power Capability

The power handling capability exceeds 500 milliwatts, CW, at 2295 Mc.

e. Power Monitors

There is one power monitor integral with each input port.

Source Impedance. The power monitor source impedance is 10K ohms.

Sensitivity. A positive 2.0 VDC monitoring signal is provided when 100 milliwatts of incident power is applied to the power monitor.

5.4.3 RF Power Amplifiers

a. Power Output

The power amplifier provides a minimum of 20 watts of RF power at 2295 Mc.

b. Input Impedance

The input impedance is 50 ohms with a VSWR not exceeding 1.4:1.

c. Output Impedance

The output impedance is 50 ohms with a VSWR not exceeding 1.5:1.

d. Noise Figure

The noise figure is less than 30 db.

e. Gain

The power amplifier provides a 20-watt output for input power levels of 40 milliwatts ± 1.0 db.

f. Bandwidth

The over-all bandwidth of the power amplifier is flat $\pm 1/2$ db for the 2290 to 2300 Mc band.

g. Spurious Radiation

The internally generated spurious radiations are at least 45 db below the desired output signal.

5.4.4 Low Power Transmitter

The characteristics of the low power transmitter are the same as the modulator-exciter, except that the power output is 1 watt.

5.4.5 Power Monitors

The power monitor is capable of measuring incident power levels of 30 watts (± 44.8 dbm). The monitor output indicates incident power with variations in load VSWR of 20:1 due to an RF load mismatch.

a. Sensitivity

A positive 2.0 VDC monitoring signal is provided when 20 watts of incident power is applied to the transmitter arm, and 3.0 VDC at the 30-watt input level.

5.4.6 RF Switches

Impedance	50 ohms nominal
Insertion Loss	0.35 db/section at 2295 mc/s
VSWR	1.1:1 maximum
Reverse Isolation	25 db minimum at 2295 mc/s
Power Handling Capacity	50 watts CW minimum
Actuating Signal	13 ma at 2 VDC to switch
Memory	Bistable solid state

5.4.7 RF Diplexer

Nominal Frequency	The nominal frequencies are 2295 and 2115 Mcs
Bandwidth	The RF bandwidth of the transmit and receive channels is 10 Mcs
Insertion Loss	The insertion loss is 0.36 db maximum
VSWR	The VSWR is 1.2:1 maximum
Isolation	The transmit to receive channel isolation is 70 db minimum at 2115 Mcs
Weight	Each diplexer weighs not more than 0.82 pound.

5.4.8 S-Band Antennas

a. High Gain Antenna

Weight:	Elliptical 5'6" x 6'6-1/2" dish	6.7	lb
	Dish structure and stowing structure	6.32	lb
	Feed	1.4	lb
	Total dish	14.42	lb
	Actuator assembly (including coax and rotary joints)	29.0	lb
	Total assembly	43.42	lb

Maximum gain (relative to circular isotropic) $30.0^{+0.25}_{-0.50}$ db at 2295 Mcs

$29.5^{+0.25}_{-0.50}$ db at 2115 Mcs

Ellipticity on axis	1.0 ±1.0 db at 2295 Mcs
	3.0 ±2.0 db at 2115 Mcs
VSWR	Less than 1.2:1 at 2295 Mcs
	Less than 1.4:1 at 2115 Mcs
Rotary joints - weight (integral with actuator shafting structure and therefore included in actuator weight)	0.38 lb each

Actuators

The parameters for the high gain antenna drive are as follows:

Slew rate:	5.3 milliradians/sec (max) about either axis
Angular acceleration:	0.5 milliradian/sec^2 (max)
Angular freedom:	
Hinge axis:	±90 degrees
Shaft axis:	±180 degrees
Acceleration loading:	3 g steady-state acceleration with antenna deployed
Weight:	Maximum weight of the drive including connectors and the internal rotary RF joints is estimated at 29 lb
Power:	Power requirements for the antenna drive are estimated as follows:
1. Drive motors	Hinge axis drive: 26 watts peak, 0.78 watt average
2. Drive Electronics:	4.1 watts peak, 1.6 watts average
3. Heaters:	10 watts maximum

b. Medium Gain Antenna

Weight:	Circular 3-foot dish	1.52	lb
	Dish support and stowing structure	2.2	lb
	Feed	1.4	lb
	Total dish	5.12	lb
	Actuator assembly (including coax and rotary joint)	10.0	lb
	Total assembly	15.2	lb

Maximum gain (relative to circular isotropic)	$24.0^{+0.25}_{-0.50}$ db at 2295 Mcs
	$23.5^{+0.25}_{-0.50}$ db at 2115 Mcs
Ellipticity on axis	1.0 $\pm$2.0 db at 2295 Mcs
	3.0 $\pm$2.0 db at 2115 Mcs
VSWR	Less than 1.2:1 at 2295 Mcs
	Less than 1.4:1 at 2115 Mcs
Rotary joint - weight (integral with actuator shafting structure and therefore included in actuator weight)	0.38 lb

Actuators

The parameters for the medium gain antenna drive are as follows:

Slew rate:	5.3 milliradians (max)
Angular acceleration:	0.5 milliradian/sec^2 (max)
Angular freedom:	$\pm$90 degrees (deployed)
Acceleration loading:	3 g steady-state acceleration with antenna deployed
Weight:	The maximum weight of the drive including connectors and the internal rotary RF joint is estimated at 10 lb
Power:	Power requirements for the antenna drive are estimated as follows:
1. Drive motor;	7.0 watts peak, 0.20 watt average
2. Drive Electronics:	1.1 watts peak, 0.80 watt average
3. Heaters:	5 watts max

c. Low Gain Antenna

Weight:	1 lb maximum

Gain of primary antenna relative to circular isotropic

Main beam peak:	$4.0^{+0.5}_{-0.25}$ db at 2295 Mcs
45-degree off peak:	$2.6^{+0.5}_{-0.5}$ db at 2295 Mcs
Main beam peak:	$3.2^{+0.5}_{-0.25}$ db at 2115 Mcs
45-degree off peak:	$2.1^{+0.5}_{-0.5}$ db at 2115 Mcs

Primary Antenna

Ellipticity - beam peak	3.0 ±1.0 db at 2295 Mcs
45-degree off peak:	3.0 ±1.0 db at 2295 Mcs
Beam peak:	3.0 ±2.0 db at 2115 Mcs
45-degree off peak:	3.0 ±3.0 db at 2115 Mcs

Input Total

VSWR	Less than 1.5:1 at 2115 Mcs
(includes two antennas and coupler)	Less than 1.5:1 at 2295 Mcs

d. High Gain Antenna Cable

Weight:	1.5 ±0.1 lb
VSWR:	Less than 1.2:1
Insertion Loss	1.0 ±0.1 db including mismatch loss

e. Medium Gain Antenna Cable

Weight:	1.5 ±0.1 lb
VSWR:	Less than 1.2:1
Insertion loss	0.9 ±0.1 db including mismatch loss

Impedance 50 ohms nominal

f. Low Gain Antenna Cable

Weight:	0.3 $\pm$0.1 lb
VSWR:	Less than 1.2:1
Insertion loss	0.9 $\pm$0.1 db including minmatch loss

Impedance 50 ohms nominal

5.4.9 S-Band Receivers

a. Frequency

The nominal center frequency is 2113 Mc (DSIF Channel 14B) with a capability of tracking ± 5 parts in 10^5. The receiver is capable of operation in the 2110 to 2120 Mc with the installation of the proper crystals.

b. Noise Figure

The noise figure at the receiver input is 10 db maximum.

c. Threshold

Threshold is defined at +6 db signal-to-noise ratio in the effective loop bandwidth ($2B_L$) of the receiver. The threshold signal level is -143 dbm.

d. Range of Input Signal Level

The range of input signal levels is -50 dbm to threshold.

e. RF Carrier APC Loop Characteristics

Transfer Function

$$H(s) = \frac{\theta \text{ out}}{\theta \text{ in}} = \frac{240}{221} \quad \frac{1 + \frac{3S}{4B_L}}{1 + \frac{3S}{4B_L} + \frac{9S^2}{32B_L{}^2}}$$

where $2B_L$ = loop noise bandwidth.

Noise Bandwidth. The design value of the noise bandwidth ($2B_L$) at SNR = 6 db is 32 cps with damping factor of 0.707 (α = 0.103). The noise bandwidth varies from 32 cps at SNR = 6 db to 200 cps at strong signal levels in excess of -70 dbm (α = 1).

Residual Phase Modulation. The residual phase modulation (noise) at an input signal level of -70 dbm due to all sources other than vibration, frequency offset, frequency rate, and modulation is less than 3 degrees peak.

Static Phase Error - SNR = 6.0 db in $2B_L$. The loop static phase error at SNR = 6.0 db in $2B_L$ is less than 0.1 radian for a frequency offset of 20 kc.

Static Phase Error - Strong Signal. The loop static phase error at -70 dbm is less than 0.05 radian for a frequency offset at 100 kc.

Dynamic Phase Error - SNR = 6.0 db in $2B_L$. The loop dynamic phase error at threshold is less than 0.1 radian for an input signal whose maximum rate of change is 11.7 cps/sec.

Reference (VCO) Signal Output. The receiver provides a coherent output to the modulator exciter. The signal is coherent in phase and frequency with the input S-band signal in a 2/221 ratio when the receiver is in lock as indicated in the "in-lock" detector. The power level of this reference signal is 2 $\pm$1 dbm.

f. In-Lock Detector

The receiver provides a DC level when a signal above threshold is present.

g. Modulation Characteristics

Frequency Response. The predetection noise bandwidth is 4.5 kc. The predetection filter is monotonic, such that the phase modulation response at the command output is flat within 1.5 db from 120 cps to 1500 cps, for an input signal range of -70 dbm to threshold.

Command Subcarrier Output Impedance. The nominal output impedance is 10 K ohms.

h. Input Impedance

The receiver input impedance is 50 ohms with a VSWR of 1.6 to 1 or less.

i. Turn-Around Ranging Channel

1) The bandwidth from the receiver input to the ranging detector input is 3.3 Mc minimum at the 3 db points.

2) The maximum ranging channel video signal rise and fall times are 0.3 microseconds.

3) The ranging video channel output is a 3.0-volt peak-to-peak signal.

5.4.10 Command Demodulator

a. Input

Bandwidth	Approximately 1500 cps
Impedance	10 K Ω max
Bit rate	1 BPS
PN clock rate	511 cps

b. Modulation Type

Data	PSK on sine-wave subcarrier
Sync	PSK on square-wave subcarrier

c. Clock Loop

Bandwidth	2.0 cps ± 0.8 db
Threshold	13.7 db ± 1.0 db

Transfer function $$H(s) = \frac{B_o^2 + \sqrt{2}\, B_o S}{B_o^2 + \sqrt{2}\, B_o S + S^2}$$

d. Data Detector

Bandwidth	10 cps approximately
Threshold	13.7 db $\pm$1.0 db for $P_e^b = 1 \times 10^{-5}$
Type detection	Coherent phase detection followed by matched filter.

e. Outputs

Data	NRZ level shift: 0 to +3 volts
Bit Sync	Pulse: 3 V p-p
Sync Lock	Level: 0 to +3 volts

5.4.11 VHF Antenna

Weight:	Turnstile and balun	2.8 lb
	Ground plane structure	1.7 lb
		4.5 lb

Maximum gain:	4 db $\pm$1 db on main beam peak
	1 db $\pm$1 db at 60 degrees from peak
Ellipticity:	1 db $\pm$1.0 db on main beam peak
	3 db $\pm$2.0 db at 60 degrees from peak
VSWR:	2.0:1 maximum

5.4.12 VHF Preamplifier

Frequency:	136 to 138 Mc
Noise figure:	3.5 db maximum
Gain (single channel):	10 db minimum
Range of input signals:	-55 dbm to -125 dbm (noise level)
Bandwidth (3 db):	1 Mc
Input impedance:	50 ohms nominal
Input VSWR:	1.7:1 maximum
Number of output channels	Two
Isolation between output channels:	30 db
Output impedance:	50 ohms nominal

5.4.13 VHF Receiver

a. Frequency

The nominal frequency is in the 136 to 138 Mc range. The operating frequency is determined by a crystal controlled local oscillator operating on the low side of the input signal. The local oscillator long term stability is ± 3 parts in 10^5 under all environmental conditions. The nominal frequency is set to within ± 1 in 10^5.

b. Noise Figure

The noise figure of the receiver is 4 db.

c. Limiter Operation

The receiver has sufficient gain so as to hard limit on input noise.

d. Range of Input Signal Level

The range of the input signal level is -45 dbm to -115 dbm (noise level).

e. Bandwidth

The RF bandwidth as determined by the RF bandpass filter is 2 Mc at the 3 db points and 20 Mc at the 60 db points.

The IF bandwidth is 44 kc.

f. Input Impedance

The nominal impedance is 50 ohms with a VSWR not exceeding 1.7:1.

g. Limiter Characteristics

Hard limiting is provided which is capable of handling a dynamic range of 70 db.

h. IF Frequency Response

For a 40 Kc bandwidth centered at 10 Mc, the frequency response of the IF amplifier is flat to within ± 1 db.

i. Output Impedance

The nominal output impedance is 50 ohms.

j. Output Signal Level

The output signal level is 0 $\pm$0.5 dbm.

5.4.14 Capsule Demodulator

a. Input Frequency

The input signal frequency is in the range of 10 $\pm$0.022 Mc.

b. Input Impedance

The input impedance is 50 ohms resistive.

c. Signal Input Power

The signal input power is 0 dbm $\pm$0.5 db.

d. Threshold

The demodulator provides a data bit error probability of 10^{-3} or less at the threshold SNR.

e. Mark and Space Filter Characteristics

Bandwidth. The 3 db bandwidth of the mark and space filter is 22 Kc.

Center Frequency. The center frequency (f_c) of the mark filter is 10.011 Mc. The center frequency of the space filter is 9.989 Mc.

Flatness. The filters are flat to within $\pm$0.5 db in the band f_c $\pm$9 kc.

f. Output Signal Characteristcis

The output signal is a 10 bit/sec NRZ wave of peak-to-peak amplitude of 3 volts. The output impedance is 10 k ohms.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Mission Constraints

6.1.1 Communication Range and Rates

The earth-spacecraft communication ranges are: 0.9 to 1.85 x 10^8 km at encounter, 1.3 to 2.3 x 10^8 km at encounter plus 1 month, and 3.3 to 3.9 x 10^8 km at encounter plus 6 months.

A 4096 bit/sec telemetry link can be supported to encounter plus 1 month and a 1024 bit/sec telemetry link can be supported to encounter plus 6 months.

A command (1 bit/sec) can be received over the low gain antenna from the 100 kw Venus site transmitter through encounter plus 1 month when the spacecraft is sun oriented. For command reception through encounter plus 6 months, the medium or high gain antenna must be pointed toward the earth.

Turn-around ranging can be supported to encounter plus 1 month with the use of the spacecraft high gain antenna and the 100 kw Venus site transmitter.

The maximum required capsule-spacecraft communication range is 4 x 10^4 km during the entry phase. The communication system will support a 10 bit/sec link at this range.

6.1.2 Antenna Pointing

The high gain antenna has sufficient freedom that it can be oriented toward the earth whenever the spacecraft is in sun-Canopus lock. The high gain antenna can be oriented to provide 3π sterradian coverage. The peak radiation vector of the medium gain antenna can be aligned within 2 degrees of the earth-spacecraft line after encounter.

The capsule-spacecraft line during entry must lie within 60 degrees of the radial extending from the spacecraft at 105 degrees clock angle and 140 degrees cone angle.

6.1.3 Spacecraft Occultation

During the period when earth-spacecraft communication is required, neither the target planet nor an appreciable part of its stmosphere is to pass between the spacecraft and earth. During periods when capsule-spacecraft communications are required, the spacecraft is to be at least 5 degrees above the horizon.

6.1.4 Maneuvers

For the normal maneuver sequence both high and medium gain antennas are used. This allows complete freedom in orienting the spacecraft if a three-maneuver sequence (i.e., roll, pitch, roll) is used. The angular freedom available with the high gain antenna permits a two-maneuver sequence for spacecraft orientations over approximately 3π sterradians.

Normally, the high gain antenna is slewed to the proper coordinates and the angles confirmed over the medium (or low gain) antenna before a command is sent to change spacecraft orientation. When the spacecraft is properly oriented for a maneuver, the high gain antenna is pointed toward earth, permitting confirmation of spacecraft orientation. After confirmation of the spacecraft orientation, the maneuver enable command is transmitted. The acceleration generated during a midcourse maneuver will not cause either spacecraft or ground receivers to unlock. The 3 g acceleration for orbit injection will not cause spacecraft or ground receivers to unlock provided that the 210-foot receiving station is utilized and the 100 kw Venus site utilized for transmission.

6.2 DSN Compatibility

The communications subsystem is to be compatible with the Deep Space Instrumentation Facility (DSIF) equipment as defined in TM-33-83. In particular, operation is to be in accordance with the following.

6.2.1 Channels

Separate frequency channels are required by each Voyager spacecraft. The channel separation should be at least 200 kc.

6.2.2 Antenna, Receivers, and Transmitters

From launch to encounter, 85-foot, 10 kw diplexed ground stations can provide 24-hour operational coverage. During the cruise phase, 24-hour coverage is desirable for continuous monitoring of spacecraft performance and the collection of cruise science data.

After encounter, 210-foot receiving stations are required for 24-hour operational coverage at the high telemetry rates (4096 bits/sec). The 10 kw 85-foot ground station is required for up link communications.

The 100 kw Venus site is required for two functions:

a) Up-link command transmission at Mars ranges when only the spacecraft low gain antenna is available.

b) Two-way range code transmission at Mars ranges.

6.2.3 Command Equipment

Command equipment is to be compatible with the command verification equipment (see GMG-50109-DSN-A).

6.3 Error Probability

The bit error probability for the telemetry link at the threshold is to be less than $P_e^b = 5 \times 10^{-3}$. The bit error probability for the command link at threshold is to be less than $P_e^b = 1 \times 10^{-5}$.

6.4 Spacecraft Equipment Constraints

6.4.1 Command Demodulator Acquisition Procedure

Demodulator acquisition is accomplished in one of two ways, depending on the availability of telemetry data:

a) If telemetry data are availabel indicating the 2fs frequency and the lock condition of the demodulator, the transmitted 2fs is adjusted to a 1.0 cps offset from the demodulator 2fs frequency, and dummy command data (all zeros) is transmitted until a lock condition is received.

b) If telemetry is not available, the transmitted 2fs frequency is swept through the range of possible demodulator 2fs frequencies at a slow enough rate to insure demodulator lock-up some time during the scan.

6.4.3 Radio Frequency Interference

For satisfactory operation of the VHF and S-band receivers, it will be necessary to limit broadband noise and interference signals from other spacecraft subsystems to -175 dbm/cps at 2115 $\pm$50 Mc and -180 dbm/cps at 137 $\pm$10 Mc. The Voyager RFI test plan will include special tests in these frequency bands to assure that this constraint is satisfied rather than limit system performance by the -150 dbm/cps environment specified in the Preliminary Voyager Specification, JPL Project Document No. 45, p. 86.

6.5 Configuration and Packaging

The communications equipment is to be compatible with the spacecraft configuration given in VS-3-110. Equipment installation and packaging are to be compatible with VS-4-550.

6.6 Environment

Allowable temperature limits for the communications equipment are given in VS-3-111.

6.7 Weight, Power and Volume

Weight, power, and volume are given in VS-3-111.

7. SAFETY

7.1 High Voltage

Warning signs are used to indicate high voltage circuits for personnel safety. In addition, all terminals are covered with an insulating material to prevent accidental shorts and possible damage to the equipment.

7.2 Test Points

All telemetry and ground test points are isolated from the operational circuitry in such a manner that they can be shorted without degrading the communication subsystem performance.

7.3 High RF Power

Proper procedures and ground test interlocks are used to prevent accidental turn-on of RF power amplifier in the enplosive safe area, on the launch pad, or in a partial pressure environment. After launch prime power to the power amplifiers is interlocked until approximately 15 hours after launch to allow adequate time for outgassing.

DATA HANDLING

VS-4-311

CONTENTS

1. SCOPE

This document presents the design for the Voyager spacecraft data handling subsystem, which includes spacecraft telemetry. This subsystem conditions, commutates and encodes spacecraft engineering data, combines these data with scientific data from the DAE, and outputs this combined data as a modulated subcarrier linearly mixed with a bit synchronization subcarrier.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-3-111	Components Design Parameters
VS-4-312	Tape Recorder
VS-4-550	Electronic Equipment Packaging

3. SUBSYSTEM DESCRIPTION

3.1 Functions

The Voyager spacecraft data handling subsystem performs the following functions:

- Accepts analog, discrete and binary signals from the Voyager spacecraft subsystems
- Stores scientific data and corresponding calibration data
- Stores and handles capsule data
- Conditions and encodes the engineering analog data into a 7-bit word format
- Time-multiplexes (commutates) engineering, scientific and capsule measurements
- Adds unique codes into the data to identify the beginning of main frames and prime frames of commutated data

- Generates an appropriate pseudo-noise (PN) code sequence which is synchronous with the data. Combines the PN code with the sync subcarrier to produce the synchronization code

- Bi-phase modulates a square wave subcarrier with the binary coded data and linearly mixes this signal with the synchronization code. Conditions this resultant signal to suitably modulate the Voyager transmitter

- Provides change of data rate to make use of available bandwidth in the RF communications system

3.2 System Organization

Figure 1 is a block diagram of the data handling subsystem. The major subsystem elements are the signal conditioning unit, two PCM encoders, the buffer and two bulk storage units. The bulk storage unit is described in VS-4-312.

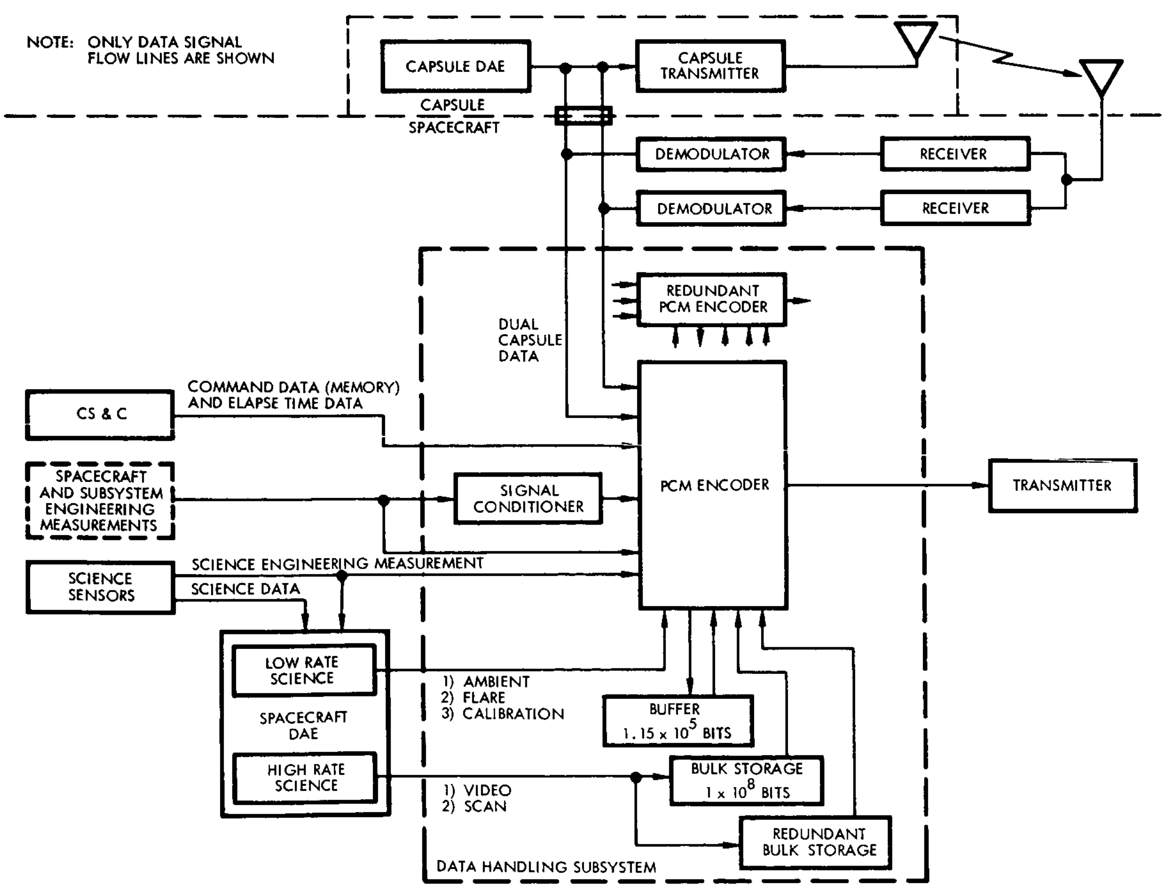


Figure 1. Data Handling Subsystem Block Diagram

3.3 Data Modes

There are seven operational data modes, as shown in Figure 2, and their relationship to the six data formats and the flight phases are tabulated in Table 1. The seven data gathering modes provide unique combinations of the six formats.

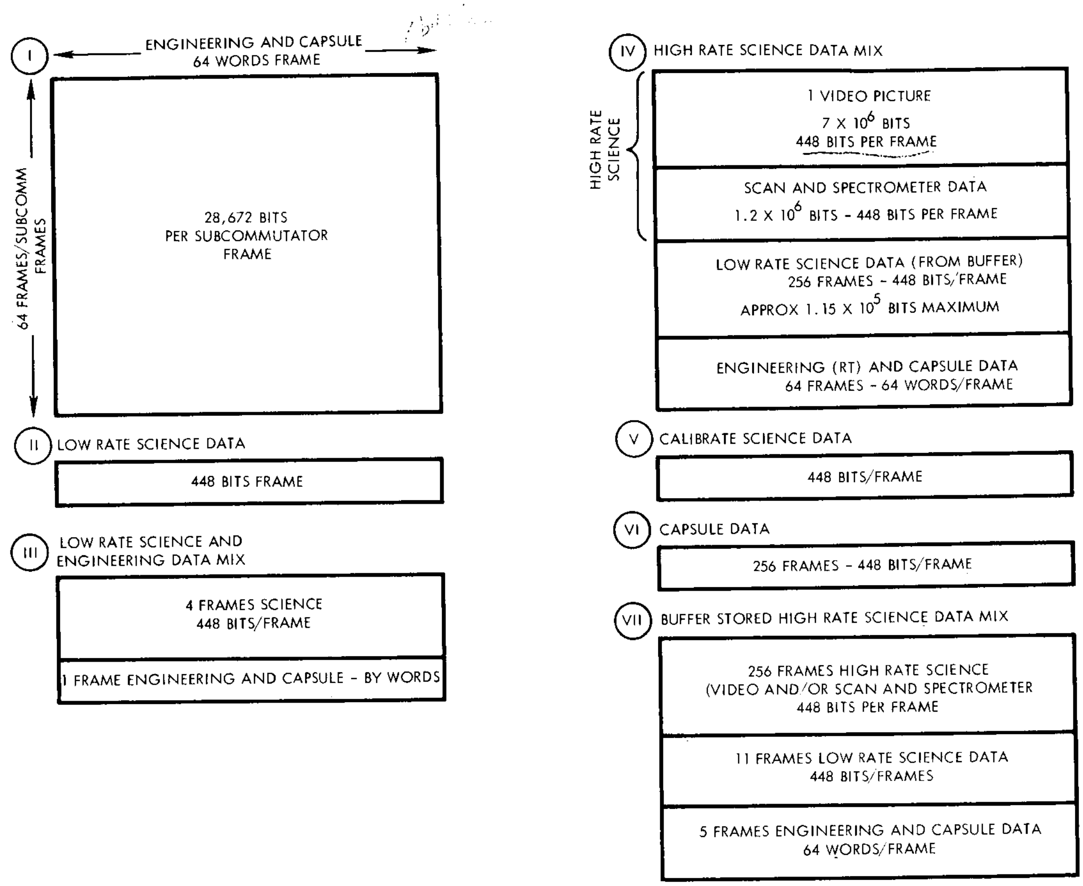


Figure 2. Data Handling Mode

3.3.1 Engineering and Capsule Data (Mode I)

Data mode I consists of transmission of real time engineering data combined with capsule data from the PCM encoder buffer.

3.3.2 Low Rate Science Data (Mode II)

Data mode II consists of transmission of the low rate science data format. During a solar flare, the data automation equipment (DAE) will sense the flare and switch its operation to provide selected solar flare science data within this mode.

Table 1. Formats and Modes versus Flight Phases

Format

- A - High rate (video and scan) science data
- B_1 - Low rate science data
- B_2 - Solar flare selected science data
- B_3 - Science data calibration
- C - Engineering and capsule data (pre - separation)
- D - Capsule data

Mode

- I - Engineering and capsule data (Format C)
- II - Low rate scientific data (Formats B_1, B_2 or B_3)
- Main Cruise III - 4 frames science data and 1 frame engineering and capsule data (Formats B_1, B_2 or B_3 and format C)
- Prime playback IV - High rate science data (1 video picture + 1.2×10^6 bits scan data Format A)
  256 frames low rate science (Format B_1)
  64 frames engineering data (Format C)
- V - Calibrate science data (Format B_3)
- VI - Capsule data (Format D) (Relay)
- VII - High rate science data sequential readout of tape in groups of 256 frames through buffer (Format A)
  11 science data frames (B_1, B_2, or B_3)
  5 engineering and capsule data frames (Format C)
  Repeat until bulk store empties or new mode command

Flight Phase	Modes and Formats	Remarks
Launch	I-C	
Injection, acquisition	I-C	
Interplanetary cruise	III-B_1 and C	
	V-B_3	Science data calibrate
	II-B_2	Solar flare - initial period
	III-B_2 and C	Solar flare
Trajectory correction	I-C	
Capsule separation	I-C	
Capsule descent mode	VI-D	
Orbital insertion	I-C	
	III-B_1 and C	
	IV-A, B, and C	
Orbital operations	IV-A, B, and C	
	V-B_3	Science data calibrate

3.3.3 Low Rate Science and Engineering Data Mix (Mode III)

Data mode III consists of four frames of low rate science data format followed by one frame of the engineering and capsule data format.

3.3.4 High Rate Science Data Mix (Mode IV)

During data mode IV, high rate science data is first read into the bulk storage unit from the DAE vidicon and buffers. Each data message consists of one complete video picture followed by scan and other high rate science data. After each data message, a data gap is

would take about 3 hrs to read 1 TV picture into the BULK STORAGE

provided before the next data message. The recording process continues until 12 data messages have been recorded. Upon completion of the recording of 12 data messages, the bulk storage unit starts to play back the data into the PCM encoder, and the second bulk store starts to record. Low rate science data is stored in the buffer during each data message transmission. During each data gap in the recording, the PCM encoder inserts the low rate science data from the buffer, and one or more complete engineering and capsule data sequences. At the end of readout of the 24 data messages, the data handling subsystem reverts to mode III operation, unless otherwise commanded. Mode IV is utilized at 4096, 2048, and 1024 bps data rates.

3.3.5 Calibrate Science Data (Mode V)

Data mode V consists of the transmission of the science payload calibration format.

3.3.6 Capsule Data (Mode VI)

During data mode VI, 256 frames of capsule data are stored in the buffer during capsule descent. This data is then played back through the PCM encoder upon command. At the end of readout the data handling reverts to mode III operation.

3.3.7 Buffer Stored High Rate Science Data Mix (Mode VII)

During this data mode, a portion of a high rate science data message is fed from the bulk storage unit into the buffer. During the buffer load cycle, frames of low rate science, engineering, and capsule data are transmitted. Upon completion of the buffer loading, the tape recorder is stopped and the high rate science data mix is transmitted. This sequence is continued until the end of tape signal switches to mode III operation, unless otherwise commanded. Mode VII is utilized at 128 bps data rate. Why? What for? Backup

3.4 System Elements

3.4.1 Transducers

Transducers are used to convert pressures and temperatures into electrical parameters compatible with the analog-to-digital converter (ADC) or signal conditioner.

3.4.2 Signal Conditioning Equipment

The signal conditioning equipment conditions voltages to a standard range for entry into the analog-to-digital converter.

3.4.3 Commutator

The commutator is a multispeed, multiposition device, which time division multiplexes engineering measurements into the analog-to-digital converter, as well as binary data and discretes.

3.4.4 Analog-to-Digital Converter

The ADC is a device which accepts the commutator output and converts the analog input voltages into 7-bit binary words. The ADC converter output modulates the data subcarrier through a phase shift keyed modulator.

3.4.5 Phase Shift Keyed Modulators

These modulators phase shift key two subcarriers ± 90 degrees with pseudonoise (PN) and data, respectively. Logically, this is accomplished by half-adding the PN and data with the corresponding square-wave subcarriers.

3.4.6 Pseudo-Noise Generator

The PN generator is synchronized with the subcarriers and modulates the sync subcarrier with a 63-bit code. This modulation is used for bit and word synchronization through the use of coherent code detection in the ground equipment.

3.4.7 Power Supply Units

The transformer-rectifier units supply the required power voltages for the PCM encoder, signal conditioning, buffer, and bulk storage units.

3.4.8 Buffer

The buffer is a core storage device used to accumulate low rate scientific data during the transmission of other data. The buffer is also used as a means of low rate (128 bps) playback of science data from the bulk storage unit, and to store capsule data during entry and landing.

3.4.9 Bulk Storage Unit

The bulk storage unit is a tape recorder of 10^8 bits capacity which provides a recording means for a large quantity of high rate data. This data is played back at the telemetry bit rate, except in the buffer stored mode described above. (See VS-4-312 for description of bulk storage.)

4. INTERFACE DEFINITION

4.1 Input Signals

4.1.1 Engineering Data

The engineering data is received or sensed as analog, discrete or binary measurements.

a. Analog Measurements

Analog measurement voltages are received from or sensed within each of the spacecraft subsystems. The analog voltages are to be 0 to 3 volts DC at a source impedance of 10 k or less. The measurements and data frequencies utilized for preliminary design purposes are tabulated in Volume 5, Appendix F, Section 7. The input impedance of the data encoder exceeds 1.3 megohms during sampling and exceeds 10 megohms when not sampling.

b. Discrete Measurements

Discrete measurements are accepted as either of two states at a source impedance of 25 k or less.

c. Binary Measurements

Binary measurements are accepted as a nonreturn-to-zero level (NRZ) serial train of coded data synchronous with the telemetry bit rate and received during a gated time interval. Alternately, binary measurements are accepted as a coordinated group of discrete measurements. A "1" is represented by a 4.0 ± 1.0 volt level and a "0" is 0 ± 0.6 volt. Source impedance is less than 25 k. Rise and fall times are less than 3 milliseconds.

4.1.2 Low Rate Science Data

The low rate science data is an NRZ serial train of coded data synchronous with the telemetry bit rate. It has amplitude, source impedance, and rise and fall times as described in 4.1.1.c, above.

4.1.3 High Rate Science Data

The high rate science data is an NRZ serial train of coded data synchronous with a multiple of the telemetry bit rate. The bit rate is 163,812 ± 16 bits per second. A "1" level is indicated by 4.0 ± 1.5 volts with a maximum load current of ± 200 microamperes. A "0" level is indicated by 0 ± 0.6 volt with a maximum current of 2 milliamperes into the source. The rise and fall times are less than 0.5 millisecond.

4.1.4 Capsule Data

The capsule data is a dual input each of which is an NRZ serial train of coded data at a rate of 10 bits per second. Amplitude, source impedance, and rise and fall times are as described in 4.1.1.c, above.

4.1.5 Elapsed Time

The elapsed time signal is received from the CS and C as a time gated serial data train of characteristics described in 4.1.1.c, above.

4.2 Controls and Synchronization

Control is exerted on the data handling subsystem by the CS and C subsystem and/or the DAE. The data handling subsystem in turn provides control inputs to the DAE. All spacecraft data is synchronized to clock rate from the CS and C.

4.2.1 CS and C Inputs

The CS and C provides the command input for each of the telemetry modes, telemetry bit rates, and redundant equipment switching as given in Table 2. A total of 18 commands are required. In addition, the CS and C provides a group of four synchronized square wave frequencies as given in Table 3. The group of frequencies supplied is a function of the selected bit rate.

Table 2. Discrete Command Inputs to Data Handling

Item	Function	Primary Source of Command	Backup Source of Command
Mode Selection	Mode I	Ground	None
	II	DAE	Ground
	III	Ground	None
	IV	Ground	CS and C
	V	Ground	CS and C
	VI	Ground	None
	VII	Ground	CS and C
Telemetry Bit Rate	4096 bits/sec	Ground	None
	2048 bits/sec	Ground	None
	1024 bits/sec	Ground	None
	128 bits/sec	Ground	None
Redundancy Switching	Switch PCM encoder (2)	Ground	None
	Switch bulk storage (2) and inhibit sequenced switching	Ground	None
Controls	Bulk storage on/off (2)	DAE	Ground
	Store capsule data	Ground	None

Table 3. CS and C Synchronization and Subcarrier Frequencies

Telemetry Bit Rate	PN Bit Rate	PN Sync Subcarrier Oscillator Frequency	Data Subcarrier Oscillator Frequency
4096	36,864	73,728	147,456
2048	18,432	36,864	73,728
1024	9,216	18,432	36,864
128	1,152	2,304	4,608

4.2.2 Inputs from the DAE

The DAE provides the following command inputs to the PCM encoder:

a) A discrete signal to switch telemetry data mode to low rate science data upon sensing a solar flare. (A discrete signal may also be provided to switch the data mode to low rate science and engineering data mix after the solar flare has existed for a period of time.)

b) A discrete signal to return the data handling subsystem to its original state at the end of the solar flare.

The DAE also provides the following command inputs to the bulk storage:

a) A discrete signal to start the bulk storage operation. This signal is provided in advance of transmission of the data for recording.

b) A discrete signal to stop the bulk storage operation. This signal is provided whenever a time gap of 4 or more seconds exists between data messages.

c) A discrete signal signifying the end of each data message (tape data gap).

d) A clock rate signal equal to the bit rate

e) A word rate signal.

4.2.3 Outputs to the DAE

The PCM encoder provides the following signals:

a) Five discrete commands corresponding to the telemetry modes that contain spacecraft science

b) A signal indicating the current telemetry bit rate

c) Telemetry bit rate, word rate and frame rate

The bulk storage provides a discrete signal at the end of tape.

4.3 Output Signal

A composite (two channel) telemetry subcarrier is supplied to the communication subsystem. This signal is the result of the linear addition of the data and sync square wave signals. The amplitudes are adjusted to provide the correct relative carrier modulation index for each signal.

4.4 Test Connector

A test connector provides the following outputs:

a) Modulation output

b) Data serial outputs (4)

c) Word sync

d) Frame sync

5. PERFORMANCE PERAMETERS

5.1 Data Capacity

The number and type of inputs are adequate to handle all signals listed in the Telemetry Measurement List, Volume 5, Appendix F, Section 7, plus a minimum growth factor of 12 per cent for the engineering measurements. Figure 3, the commutation planning format, shows the number of channels available at each sampling rate. Further variations can be accomplished by cross strapping. The data bandwidth is a function of the input data rates and the signal sampling arrangement of the commutator assignments.

5.2 Data Bit Rates

Four data bit rates of 4096, 2048, 1024, and 128 bits per second are provided. These data rates are selectable in the PCM encoder on commands from the CS and C. The data rate selected is provided to the DAE.

5.3 Data Mode Selection and Signal Interlacing

The telemetry data mode selection and signal interlacing is accomplished by the PCM encoder on command from the CS and C or the DAE. The interlacing required by each of the data modes is shown in Figure 2.

	0	1	2	3	4	5	6	7
0	00 FS	$\overline{FS}$	$\overline{FS}$	MODE ID	VID SID	VID	A	07 SUPER COMM
1	SUB COMM	SUB COMM	SUB COMM	SUB COMM	CAP-SULE		B	
2							C	
3							D	
4							A'	
5							B'	
6				CAP-SULE			C'	
7							D'	77 SUPER COMM

MAIN FRAME

MAIN FRAME DATA @ 2048

4-4/7 S/SEC	48
9-1/7 S/SEC	4
36-4/7 S/SEC	1
	53 CHANNELS

REQUESTED

4-4/7	27
9-1/7	2
36-4/7	1
	30 CHANNELS

SUBCOMMUTATOR FRAME

	0	1	2	3	4	5	6	7
0	00 ET							07
1								
2								
3								
4								
5								
6								
7	70							77

SUBCOMMUTATED DATA @ 2048

	AVAIL	REQUESTED
1 SAMPLE/14 SEC	256	229
TOTAL CHANNELS	309	259

NOTE:

1. ELAPSED TIME (ET) OCCUPIES CHANNEL OD IN SUB-COMMS 10, 11 AND 12 ONLY

Figure 3. Commutation Planning Format

5.4 Analog Data Encoding

The analog data encoding is performed in the PCM encoder.

5.4.1 Linearity

Linearity of the encoder as read from measurement voltage input (with a source impedance of 0 to 10 k) to the analog-to-digital (A/D) converter output is within $\pm$ 0.5 per cent of full scale to the least squares defined straight line.

5.4.2 Accuracy

The accuracy of a measurement as read from the voltage input (with a source impedance of 0 to 10 k) to the data encoder to the A/D converter output is within $\pm$ 1 quantization level (0.78 per cent). This accuracy is maintained for the mission lifetime and under all specified environmental conditions.

5.4.3 Quantization Levels

The data is encoded to 7 binary quantization levels ($2^7 = 128$).

5.5 Data Sampling

The PCM encoder commutator provides time division data sampling for the engineering and capsule data. The commutator provides the interlacing of analog, discrete, and binary data inputs.

5.5.1 Commutation Formats

One cycle of the high rate commutator deck is defined as a main frame. It contains 64 seven-bit words or 448 bits. A group of main frames is a prime frame. It contains 64 main frames or 28,672 bits. The commutation format is shown in Figure 4. The channel numbers shown in Figure 4 are octal numbers.

5.5.2 Rate of Sampling

Several sampling rates are available and are a function of bit rate. Figure 3 illustrates the sampling rate at a 2048 bit rate. At other bit rates, the sampling rates are modified in proportion to the bit rate change.

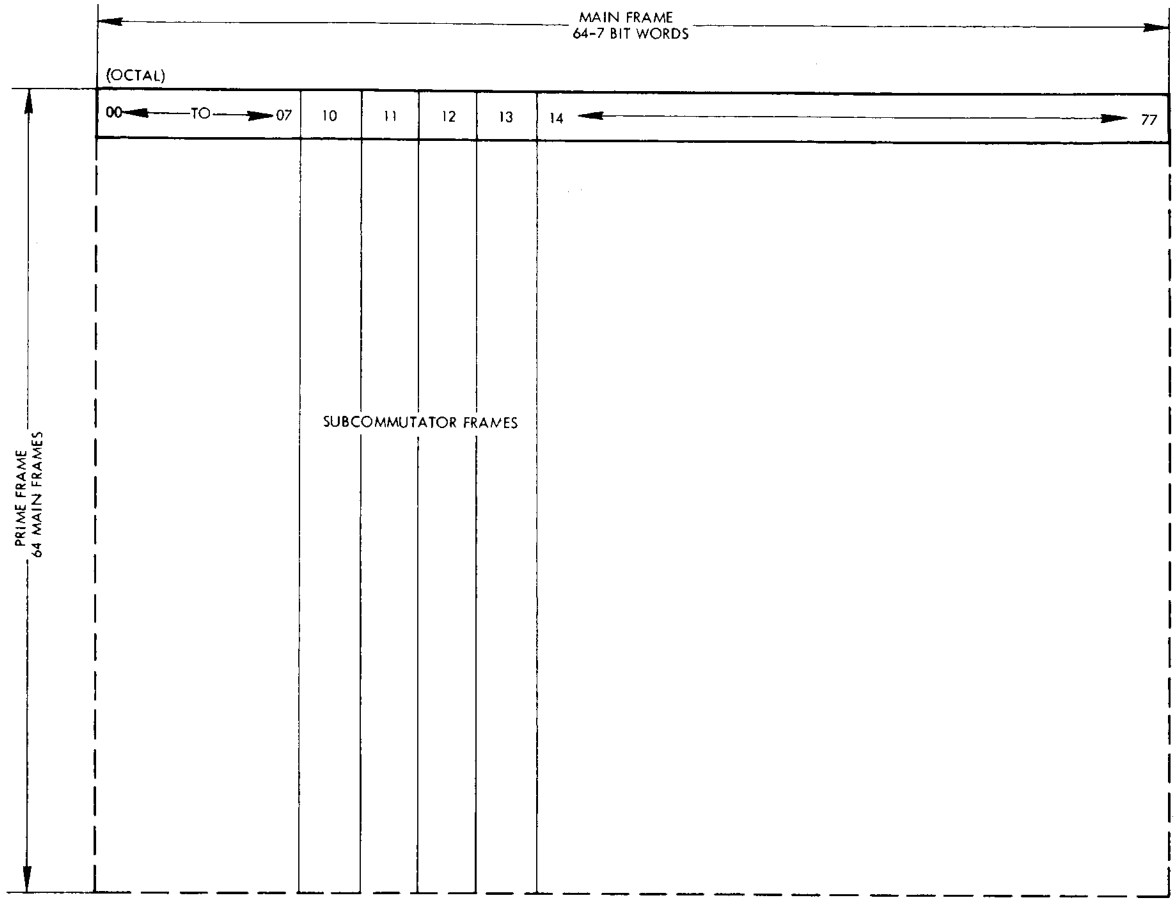


Figure 4. Commutation Format

5.5.3 Synchronization

The commutator and encoder logic is synchronized to the bit rate received from the CS and C. A 21-bit code is used to provide main frame synchronization. It consists of a 7-bit Barker code followed by two 7-bit Barker code complements. Subcommutator frame synchronization is provided by the subcommutator location identification word in each main frame, channel 04. This word is the binary equivalent of the subcommutator deck position.

5.5.4 Format Identification

Format, bit rate, and redundancy information is provided by a 7-bit word in each main frame, channel 03.

5.6 Bit and Word Sync

To provide bit and word synchronization, pseudo-noise techniques are used. A pseudo-noise generator is used to derive the 63-bit length pseudo-noise code. This code is applied to the sync modulator, where it is modulo 2 added to the sync subcarrier ($2f_s$).

5.6.1 Pseudo-Noise Code

The pseudo-noise sync code is as follows: 1-0-0-0-0-0-1-1-1-1-1-1-0-1-0-1-0-1-1-0-0-1-1-0-1-1-1-0-1-1-0-1-0-0-1-0-0-1-1-1-0-0-0-1-0-1-1-1-1-0-0-1-0-1-0-0-0-1-1-0-0-0-0.

5.6.2 Word Sync

Word sync is derived from the pseudo-noise code, and is coincident with the 100000 condition of the generating register where the "I" is the next digit to be read out.

5.6.3 Bit Sync

Bit sync is derived from the pseudo-noise code and is coincident with every ninth pseudo-noise bit starting with word sync. This gives 7-bit sync words for each data word sync. The conditions of the generating register for bit sync are:

1	100000	Word and bit sync
2	111010	Bit sync
3	100110	Bit sync
4	011010	Bit sync
5	011100	Bit sync
6	111100	Bit sync
7	000110	Bit sync

Bit sync is used to clock the information bits out of the data encoder into the data subcarrier modulator.

5.7 Output Mixing

The output of the data encoder is a mixed signal which is composed of the linear mixing of the data and sync subcarriers.

5.7.1 Mixing Ratio

The voltage mixing ratio of data subcarrier to sync subcarrier is 3.9/1 for data bit rates of 1024 and higher. It is 2.5/1 for the 128 data bit rate.

5.7.2 Frequency

The frequencies of the sync and data subcarriers at each bit rate are given in Table 3.

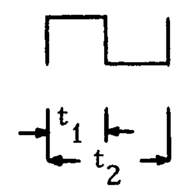
5.7.3 Waveform

The modulated subcarriers are symmetrical square waves with the following characteristics:

Rise time: μsec

Fall time: μsec

Symmetry: $\frac{t_2}{t_1} = 2 \pm 0.25\%$



5.7.4 Amplitude

The output mixed signal amplitude shall be adjustable through the range of 2.0 to 3.0 volts peak-to-peak.

5.8 Over-all Performance

5.8.1 Acquisition Time

The mean time for acquisition of frame synchronization at the specified bit error rate is 1.006 frames.

5.8.2 Word Errors

The mean word error rate at the specified bit error rate is 3.4×10^{-2}.

5.8.3 Frame Error

The mean time to loss of frame synchronization at the specified bit error rate is 250,000 frames.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Configuration

Equipment packaging and installation is to be as shown in VS-4-550.

6.2 Environment

The data handling equipment must be capable of operating satisfactorily for the specified life under all expected Voyager environmental conditions. Allowable temperature limits are given in VS-3-111.

6.3 Weight, Power, and Volume

Weight, power, and volume data are given in VS-3-111.

7. SAFETY

There are no known safety considerations peculiar to the data handling subsystem.

TAPE RECORDER

VS-4-312

1. SCOPE

This document presents the design for the Voyager tape recorder which provides bulk storage for the data handling subsystem described in VS-4-311. This tape recorder records high rate science data in digital form and plays the data back on command in synchronization with the spacecraft system clock.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

<u>TRW 1971 Voyager Spacecraft Design Documents</u>

VS-3-111	Spacecraft Components Design Parameters
VS-3-120	Reliability Design Objectives
VS-4-550	Electronic Packaging

3. DESCRIPTION

The tape recorder unit is a reel-to-reel magnetic tape recorder capable of recording and reproducing serial digital information. It is a sealed unit containing a tape transport mechanism, magnetic tape and reels, record and reproduce circuits, motor drive circuits, and other electronics required to control or buffer inputs and outputs.

3.1 <u>Tape Transport</u>

The tape transport is an electromechanical device for handling the magnetic recording tape. It includes the tape reels, tensioning system, tape guides, end-of-tape sensors, and motor and associated servo control.

3.2 <u>Record (Write) System</u>

The record system is composed of line receivers, serial to parallel conversion logic and write amplifiers capable of reliably driving the record heads.

3.3 Playback (Read) System

The playback system is composed of playback amplifiers capable of recovering data from the playback heads and driving the synchronization system.

3.4 Record and Playback Heads

The record head, when correctly driven, is capable of saturating the magnetic material of the tape to a degree suitable for storage of the digital data. The playback head is capable of recovering data stored on the magnetic storage tape.

3.5 Synchronization System

The synchronization system consists of a phase-locked servo loop and a buffer storage register capable of ensuring that the playback data is synchronized with the spacecraft system clock.

3.6 Data Gap Indicator

The data gap indicator is used to indicate absence of digital high rate science data during playback to allow the insertion of other scientific and engineering data when high rate science information is not available.

3.7 End-of-Tape Sensors

The end-of-tape sensors detect either the read or the write end-of-tape and stop the tape recorder. Once activated, the end-of-tape sensors prevent operation in the mode which would allow "running off" the end-of-tape.

4. INTERFACE DEFINITION

4.1 Write Interface

The data automation equipment (DAE) supplies the following:

a) Data input in the form of a serial bit train digital signal at 163, 812 bits per second non-return to zero (NRZ) to be stored by the tape recorder

23,401

b) A data word clock input in the form of a 163, 812 divided by 7 bits per second clock to control writing of information on the tape and to store a clock track

c) A 163, 812 bit per second clock to be used to shift the serial data into a register for parallel transfer to tape.

d) A tape data gap signal is supplied to the tape recorder by the DAE to indicate when there is a gap between high rate science data.

4.2 Motor Synchronization

The CS and C supplies a motor sync signal in the form of a 409.6-cycle per second square wave that is used to control the motor speed during the record mode.

4.3 Read Interface

4.3.1 Tape Recorder Data Output

The data output is sent to the PCM encoder in the form of a 4096, 2048, or 1024 bit per second NRZ signal. The rate is determined by sync input signals.

4.3.2 PCM Encoder Input

The following signals are furnished to the recorder by the PCM encoder.

a) A 4096, 2048, or 1024 pulse per second bit sync signal to be used for clocking the output from the recorder

b) A 4096 divided by 7, 2048 divided by 7, or 1024 divided by 7 bit per second word sync signal to be used in synchronizing the output data.

4.3.3 Data Gap Indicator

The data gap indicator provides discrete signals to the PCM encoder when a gap in high rate science data occurs so that other scientific and engineering data may be inserted.

4.4 Command, Read, and Write

Command signals are supplied from the CS and C to select the read or write mode of operation of the tape recorder.

3401 +
163,812

4.5 Telemetry Monitor Points

4.5.1 Analog Signals

Suitably conditioned analog signals for measurement of recorder temperature, pressure, and motor rotation speed are furnished to telemetry.

4.5.2 Discrete Signals

Four discrete signals are furnished to telemetry to indicate the operating status of the recorder:

- Read mode
- Write mode
- End of tape, read
- End of tape, write.

4.6 Test Interface

A separate connector is provided to make critical signals available at the output of the recorder for test purposes.

5. SYSTEM PARAMETERS

5.1 Record Rate

The recorder records data at the single input rate of 163,812 ± 16 bits per second.

5.2 Playback Rate

The three selectable playback rates are 4096, 2048, and 1024 bits per second.

The playback rate is determined by input clock rates supplied to the recorder during the playback mode. The output data rate is synchronized to the spacecraft system clock.

5.3 Data Capacity

The recorder has a storage capacity of 1×10^8 bits.

5.4 Bit Dropouts

The maximum bit dropout is 1 bit in 10^5 at start-of-life and 1 bit in 10^4 at end-of-life.

5.5 Reliability

Reliability data is given in VS-3-120.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Configuration

The recorder is housed in a single container and is designed to permit rapid and easy removal and replacement as a complete unit. The installation is to be as shown in VS-4-550.

6.2 Magnetic Fields

Magnetic fields generated by the recorder will, as a design goal, not exceed 1 gamma at 3 feet from the unit.

6.3 Overload Protection

Overload protection is to be provided to protect the power source in cases where the input current exceeds twice the maximum expected input current. The overload protection device will be resettable.

6.4 Life

The unattended minimum operating life of the tape recorder is to be 4000 hours of nonoperation in a space environment followed by 4000 hours of operating life in a space environment.

6.5 Angular Momentum

Angular momentum induced into the spacecraft is not to exceed 0.08 foot-pound seconds.

6.6 Tape Fastening

The tape is to be fastened to the reel hub such that the end of the tape cannot be inadvertently driven off.

6.7 Environment

The recorder is capable of operating satisfactorily for the specified life under all expected Voyager environmental conditions. Allowable temperature limits are given in VS-3-111.

6.8 Weight, Power, and Volume

Data on weights, power, and volume are given in VS-3-111.

STABILIZATION AND CONTROL

VS-4-410

CONTENTS

1. SCOPE

This document presents the design for the stabilization and control subsystem. This subsystem controls the orientation of the Voyager flight spacecraft at all times after separation of the spacecraft from the launch vehicle.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

<u>TRW 1971 Voyager Spacecraft Design Documents</u>

VS-2-110	Design Characteristics and Restraints
VS-3-111	Components Design Parameters
VS-3-120	Reliability Design Objectives
VS-4-311	Data Handling
VS-4-550	Electronic Equipment Packaging
VS-4-610	Midcourse Propulsion
VS-4-611	Retropropulsion

3. FUNCTIONAL DESCRIPTION

3.1 Functions

During flight the stabilization and control subsystem (SCS) must perform various functions described below.

3.1.1 Three-Axis Stabilization

The SCS establishes and maintains three-axis stabilization throughout spaceflight operation. The sun and the star Canopus are used as references except during velocity correction maneuvers, capsule vehicle separation maneuver, retropropulsion maneuver, and commanded maneuvers. For such maneuvers an inertial gyro reference is utilized. Two-axis sun stabilization aligns the spacecraft roll (Z) axis with the spacecraft-sun line, thereby maintaining the solar array surface normal to sun incidence. Roll stabilization and control about the Z axis, using the star Canopus as a reference, provides spacecraft reference attitude relative to the sun-Canopus system.

3.1.2 Command Reorientation

a. Velocity Adjustment

Based on ground commands the SCS reorients the flight spacecraft to the attitude required for velocity adjustment and maintains that attitude during engine firing. After each maneuver, the cruise orientation is re-established.

b. Capsule Vehicle Separation

The SCS reorients the flight spacecraft to the attitude required for capsule separation and injection into impacting trajectory. After capsule vehicle separation, the cruise orientation is re-established.

c. Ground Control of Roll Position

The capability is provided for ground override control of spacecraft roll position.

3.1.3 Spacecraft Control Torque Sources

The flight spacecraft is maintained in a known orientation or is rotated at prescribed angular rates through the application of appropriate torques. During cruise, these torques are obtained by the expulsion of heated nitrogen gas through jet nozzles, located so as to produce couples about each of the principal control axes. During periods of midcourse rocket motor firing, pitch and yaw axis stabilizing torques are obtained by positioning jet vanes which deflect a portion of the rocket exhaust. During the retropropulsion firing, pitch and yaw axis stabilizing torques are obtained by the injection of inert liquid into the boundary layer of the rocket exhaust and roll axis control is augmented by high thrust cold gas jet torques.

3.2 Normal Operating Modes

The SCS provides several modes of operation, incorporating various combinations of sensors and torque sources. The SCS modes and the associated equipment elements are summarized in Table 1 and described in detail in the following sections.

Table 1. Voyager SCS Modes

	Sensors					Actuators			
Control Mode	Coarse Sun Sensor	Fine Sun Sensor	Star Sensor	Gyros Rate Mode	Gyros Position Mode	Earth Sensor	Gas Jets	Jet Vanes	LITVC
Initial Acquisition	o	o	o	o		o	o		
Cruise		o	o				o		
Inertial and Maneuver					o		o		
Midcourse Correction					o		o	o	
Reacquisition	o	o	o	o			o		
Retropropulsion					o		o		o

3.2.1 Acquisition Mode

a. Yaw and Pitch Sun Acquisition Mode

Stabilization and control gyro power is turned on prior to launch and the gyros are in the caged or rate output configuration. When the planetary vehicle separates from the Centaur the SCS is placed in the acquisition mode. Since the vehicle is expected to be in eclipse at separation, the lack of sun sensor outputs will put the SCS in the inertial hold mode (see Section 3.2.3). When the sun is sensed, control reverts to the acquisition mode and alignment of the roll axis with the spacecraft-sun line begins, using attitude error signals from the sun sensors. The sun sensor attitude error signals cause switching amplifiers to actuate the appropriate gas jets to bring the sun within the field of view of the precise sun sensor.

Angular velocity signals from the gyros operating in the rate mode are mixed with the attitude error signals to provide stabilization during acquisition. Sun stabilization occurs in 20 minutes or less, depending on initial attitude error and angular velocity of the flight spacecraft.

When the roll axis is aligned with the spacecraft-sun line, the sun acquisition gate is energized and the coarse sun sensor is switched out of the circuit. Stabilization is maintained by the fine sun sensor signals. The sun acquisition gate signal also enables the roll-spin or roll-search mode.

The pitch or yaw channel configuration for the acquisition mode is shown in Figure 1.

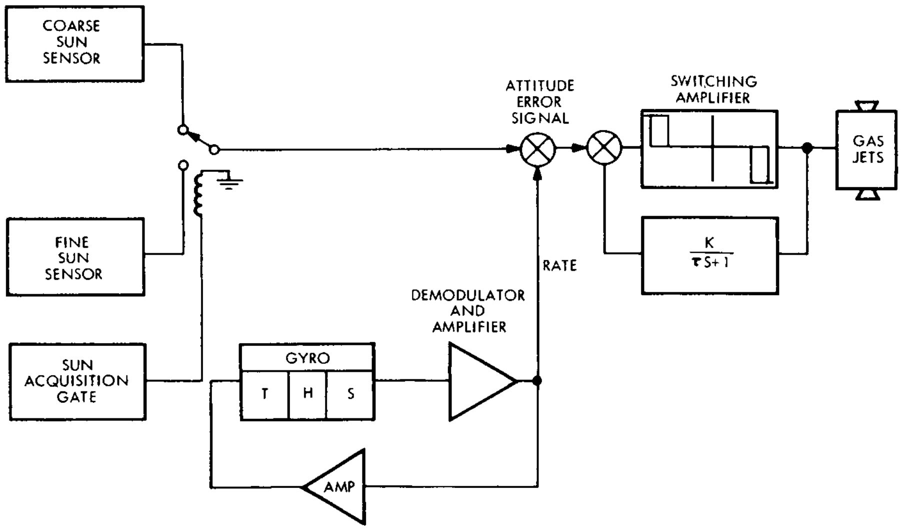


Figure 1. Pitch and Yaw Sun Acquisition Mode

b. Roll-Spin or Roll-Search Mode

After sun acquisition has been accomplished, a controlled spin about the roll axis is executed. A 0.22 deg/sec spin signal is fed to the roll switching amplifier, actuating the roll gas jets and accelerating the spacecraft. When a roll rate of 0.22 deg/sec is achieved, the roll gyro output cancels the spin signal and the jets are turned off. The flight spacecraft continues to rotate at 0.22 deg/sec. (The roll-spin described here is for calibration of the magnetometer, assuming an instrument similar to that employed on Mariner C. If another type of magnetometer is selected for Voyager, an altogether different sequence of maneuvers may be required for calibration.)

After magnetometer calibration, the central sequencing and command subsystem (CS&C) turns on the Canopus sensor. This command also changes the input to the roll switching amplifier from the spin signal of 0.22 deg/sec to the search signal of 0.1 deg/sec. The resulting roll rate error signal causes the gas jets to fire to decelerate the flight spacecraft from the spin rate to the search rate. The search mode is maintained until a star of sufficient brightness appears in the field of view of the Canopus sensor. At this time, the star acquisition gate initiates a switch to the cruise mode.

Canopus acquisition is verified by prepositioning the high gain antenna so that the beam inersects 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, is provided for telemetry. The intensity of the Canopus sensor output, telemetered to earth, provides additional verification of acquisition.

Should the roll gyro fail to provide a rate signal to null the spin or search rate command, the spacecraft would continue to accelerate until gas exhaustion or an override command were issued. To prevent this, the roll gas jet firing signals are electrically integrated and after a jet firing time slightly in excess of normal, the input signal to the switching amplifier is interrupted, allowing backup modes of roll control to be exercised.

The roll channel configuration in the roll-spin or search mode is shown in Figure 2.

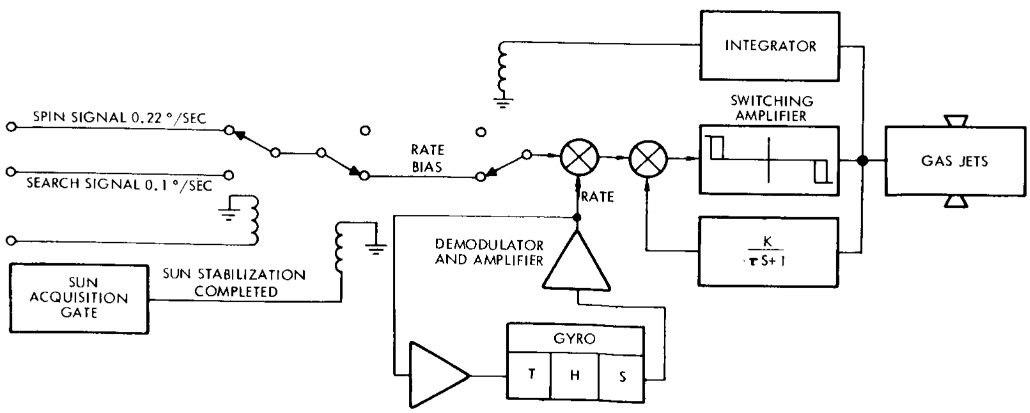


Figure 2. Roll-Spin and Search Mode

3.2.2 Cruise Mode

At the conclusion of the roll search, the Canopus acquisition gate switches the SCS into the cruise mode. The gyros are turned off and the pitch and yaw derived-rate network parameters are switched to the cruise configuration.

Pitch and yaw stabilization is maintained by fine sun sensor attitude error signals which actuate the gas jets through the jet driver switching amplifiers. Damping is provided by mixing the derived-rate signals with the attitude error signals.

The pitch or yaw channel configuration for the cruise mode is shown in Figure 3.

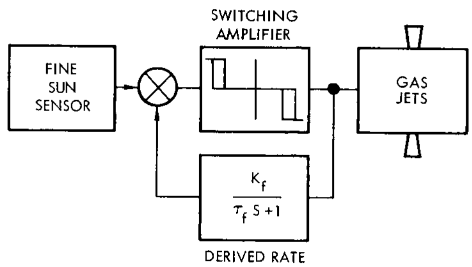


Figure 3. Pitch or Yaw Cruise Mode

The star acquisition signal switches the roll attitude error signal from the Canopus sensor into the roll channel gas jet driver switching amplifier and the roll derived rate network is switched to the cruise configuration.

Following normal star acquisition, a ground command provides a means of interrupting the star acquisition gate signal, thereby commanding further roll search for another star.

Two ranges of control deadband are provided for the cruise mode. A deadband of $\pm$ 0.5 degree normally is used in all channels for both the transit and orbit phases of the mission. Prior to maneuvers and periods of photographic operation, the CS&C commands the SCS to switch to a $\pm$ 0.25 degree deadband in all channels.

The roll channel configuration for the cruise mode is shown in Figure 4.

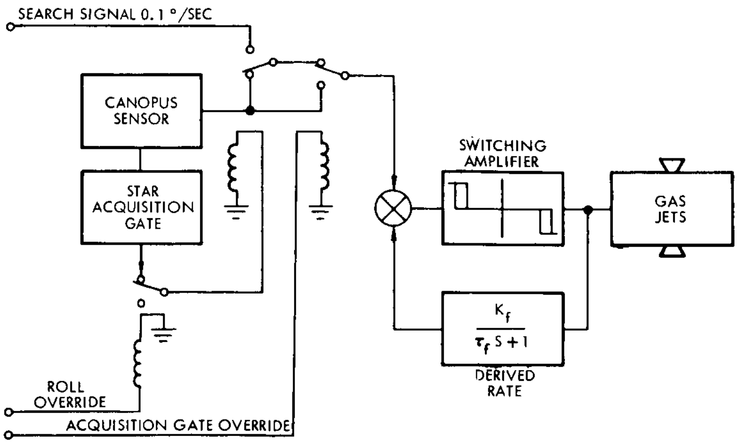


Figure 4. Roll Cruise Mode

3.2.3 Inertial and Maneuver Mode

The inertial mode maintains the flight spacecraft orientation in correspondence with the null orientation of the gyros. Attitude error signals for all channels are provided by the gyros operating in the rate-integrating configuration. This mode is commanded by the CS&C prior to a reorientation sequence and after 60 minutes for gyro warmup. It is maintained for each axis throughout a reorientation sequence except during periods of commanded turns.

Commanded turns are used to reorient the flight spacecraft to align the spacecraft axes for midcourse and retropropulsion velocity changes and for separation of the capsule vehicle. In all cases, the desired orientation relative to the sun-Canopus system is computed on the ground, based on tracking data. Turn data in the form of turning rate polarity and turning time is transmitted by radio command and stored in the CS&C. Prior to execution of a maneuver, the control deadband is switched to the fine range by command from the CS&C.

The turning rate commands for maneuvers are provided by precision current generators which torque the gyros at 0.2 deg/sec. The polarity of the current, hence the direction of the turn, is determined determined by the presence or absence of a CS&C signal. Torquing the gyros produces an error signal which causes the appropriate gas jets to actuate. When the flight spacecraft achieves the commanded rate, the error signal is nulled, the gas jets close, and the commanded turning rate is maintained. The roll and pitch turns are initiated by the CS&C in that order. Following completion of a reorientation maneuver, the new orientation is maintained by inertial control. The SCS automatically returns to the acquisition mode when the CS&C removes the inertial and maneuver mode signal.

The inertial and maneuver mode for the pitch and yaw axes is also used for periods of sun occulation which occur during extended missions while in orbit about Mars. The SCS is returned to the cruise mode when the sun sensors again detect the sun.

The SCS configuration for the inertial and maneuver mode is shown in Figure 5.

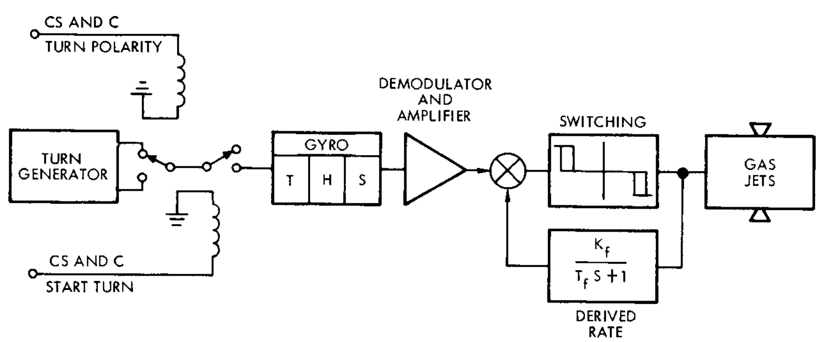


Figure 5. Roll or Pitch Inertial and Maneuver Mode

3.2.4 Midcourse Correction Mode

The midcourse correction mode is used during periods of velocity correction when the midcourse rocket engine is firing. Attitude reference signals are provided by the gyros and the SCS operates in the inertial mode. Pitch and yaw control torques from the gas jets are augmented by jet-vane-actuated thrust vector control driven by lead compensated signals from the rate-integrating gyros. The cold gas jets are used for stabilization about the roll axis.

After the midcourse rocket engine has been shut down by the CS&C, the SCS returns to the acquisition mode and reorients to the cruise attitude. The SCS configuration for the modcourse correction mode is shown in Figure 6.

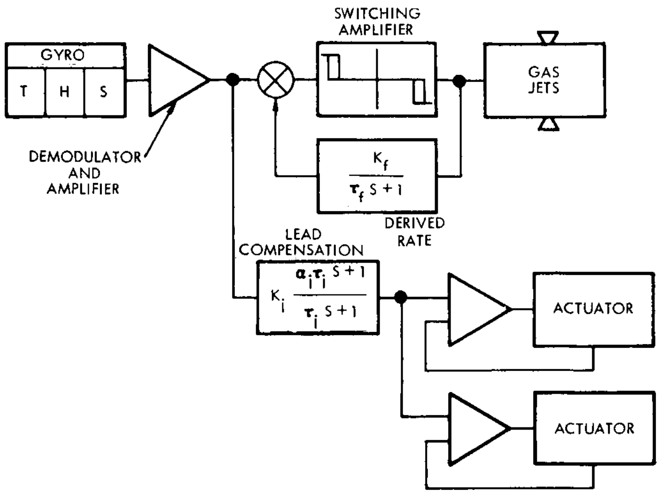


Figure 6. Midcourse or Correction Mode Retropropulsion

3.2.5 Retropropulsion Mode

The retropropulsion mode is similar to the midcourse correction mode except that thrust vector control for the pitch and yaw axes is provided by liquid injection thrust vector control. Roll control torques are augmented by high thrust cold gas jets.

After the retropropulsion engine has fired, the CS&C commands the cruise mode to be re-established via the acquisition mode.

3.2.6 Incremental Roll Turn Mode

An additional mode provides inertial control of roll attitude with the capability for incremental adjustment of roll position in ± 2-degree steps by ground command.

This mode is similar to the roll reorientation mode. The flow of precision torquing current is controlled by fixed timers rather than by the variable timing from the CS&C.

The incremental roll control mode is available in the event of a failure of a Canopus sensor. The absence of a star acquisition gate signal normally causes the gyros to remain on or to turn on. Ground command is available to turn off the gyros, leaving the spacecraft to drift in roll until the incremental roll control mode is initiated.

A ground command turns on the gyros and places the roll control channel in the inertial mode. Subsequent commands start the appropriate timer. The torquing current polarity is determined by choosing the appropriate timer. The gyros are torqued at 0.2 deg/sec for about 13.2 seconds. The gyro output actuates the gas jets and the spacecraft accelerates. Since the angular velocity does not reach 0.2 deg/sec, the jets remain on for the entire 13.2-second period. At the end of the time interval, the torquing current is interrupted and the resultant gyro output causes the appropriate jets to actuate, decelerating the spacecraft. The net result is a 2-degree turn. The incremented attitude is held until the next incremental turn command is received.

The incremental roll control mode is shown in Figure 7.

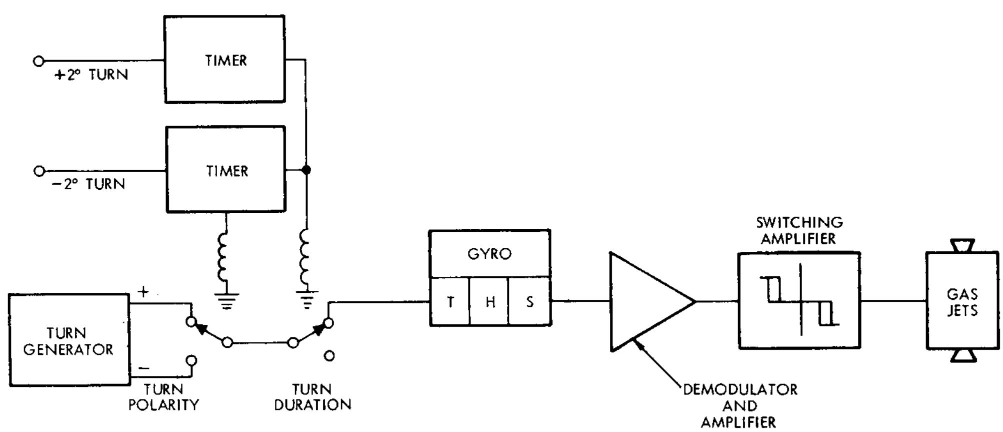


Figure 7. Incremental Roll Turn Mode

3.3 Functional Reliability

3.3.1 Jet Redundancy

The gas jet system utilizes a coupled pair of jets for each direction in all three control axes. Each valve of a coupled pair is supplied from completely separate gas supply systems to assure attitude control operation in spite of failure in one of the systems. In case of failure of a valve to open, the angular acceleration constant is halved and a linear momentum component is generated in addition to a control torque, when the remaining jet is actuated.

3.3.2 Acquisition Command Backup

The possibility of a CS&C lockup in the maneuver sequence can be made noncatastrophic (except for trajectory errors introduced or uncorrected) by use of a ground command which allows the automatic acquisition system to override the CS&C maneuver sequence.

3.3.3 Ground Initiated Backup Control Modes for the Roll Axis

Two backup roll axis control modes, are available via ground command:

a) Normal roll control may be turned off and restored through the use of ground commands. In this way the spacecraft may be left free to rotate about the roll axis.

b) The roll axis may be placed under gyro-referenced inertial control through use of a ground command. Incremental adjustment to the inertially maintained position may be accomplished by subsequent ground command transmissions.

3.3.4 Roll Search Inhibit Logic

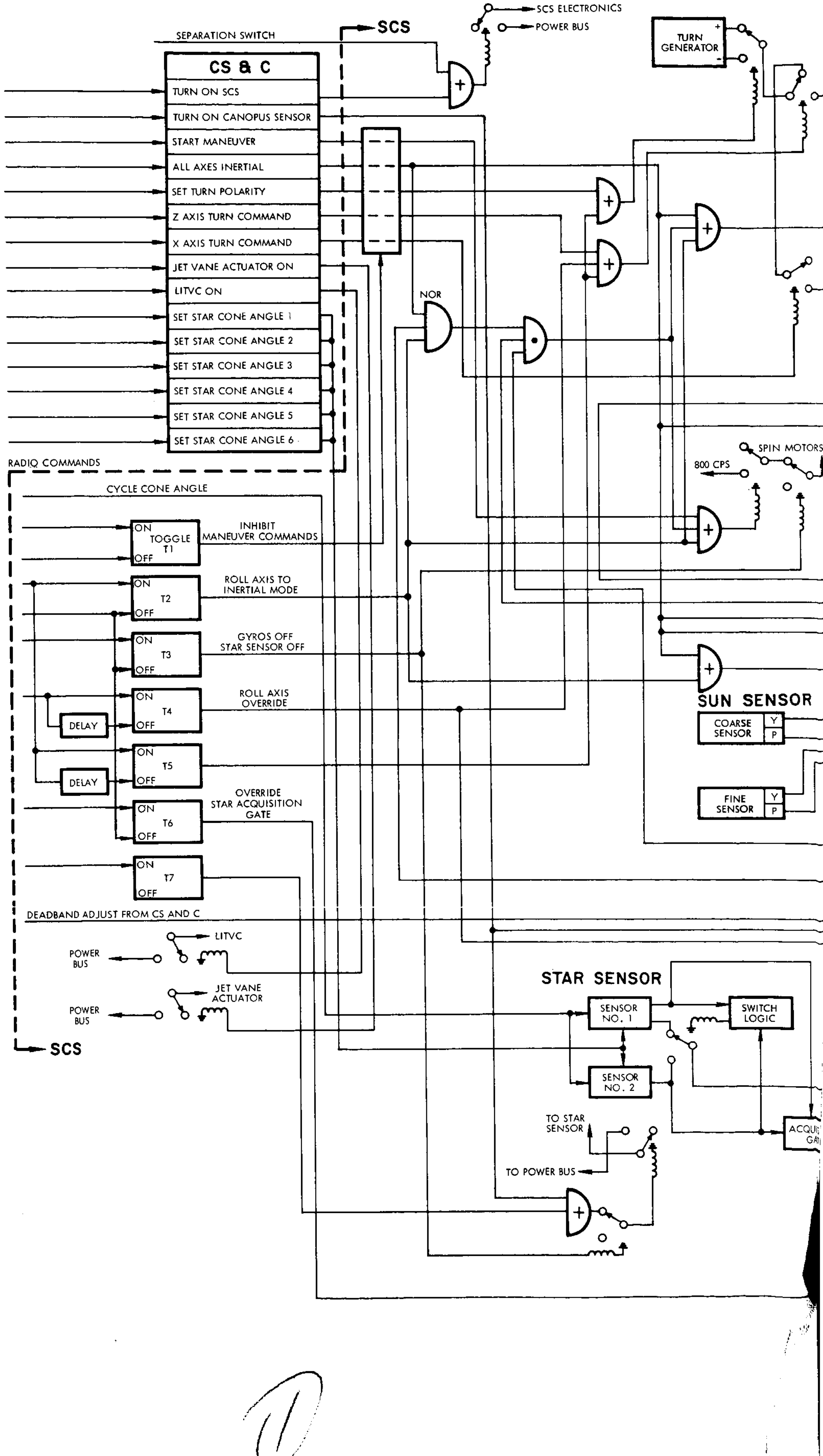
In the event of an inoperative roll gyro (and if the gyro output is near zero), the roll search command is inhibited after a slightly greater than nominal jet firing. This logic prevents continuous spinup of the spacecraft and thus allows emergency backup modes to be invoked by ground command.

3.3.5 Canopus Sensor Redundancy

Two completely separate Canopus sensors are provided for increased reliability. The sensors are selected by switching logic activated by comparing the intensity signals. In the event of the failure of the primary sensor, the backup sensor is switched into the circuit to prevent loss of Canopus lock.

3.3.6 Electronic Redundancy

The signal electronics elements contain circuit level redundancies in the mode control circuits, valve driver elements, and thrust vector control actuator servoamplifiers to enhance the reliability of the stabilization and control subsystem.


SCS ELECTRONICS
POWER BUS
SCS
SEPARATION SWITCH
TURN GENERATOR
CS & C
TURN ON SCS
TURN ON CANOPUS SENSOR
START MANEUVER
ALL AXES INERTIAL
SET TURN POLARITY
Z AXIS TURN COMMAND
X AXIS TURN COMMAND
JET VANE ACTUATOR ON
LITVC ON
SET STAR CONE ANGLE 1
SET STAR CONE ANGLE 2
SET STAR CONE ANGLE 3
SET STAR CONE ANGLE 4
SET STAR CONE ANGLE 5
SET STAR CONE ANGLE 6
NOR
RADIO COMMANDS
SPIN MOTORS
800 CPS
CYCLE CONE ANGLE
TOGGLE T1
INHIBIT MANEUVER COMMANDS
T2
ROLL AXIS TO INERTIAL MODE
T3
GYROS OFF STAR SENSOR OFF
T4
ROLL AXIS OVERRIDE
DELAY
T5
DELAY
T6
OVERRIDE STAR ACQUISITION GATE
T7
ON
OFF
SUN SENSOR
COARSE SENSOR
FINE SENSOR
Y
P
DEADBAND ADJUST FROM CS AND C
LITVC
POWER BUS
JET VANE ACTUATOR
POWER BUS
SCS
STAR SENSOR
SENSOR NO. 1
SENSOR NO. 2
SWITCH LOGIC
TO STAR SENSOR
TO POWER BUS

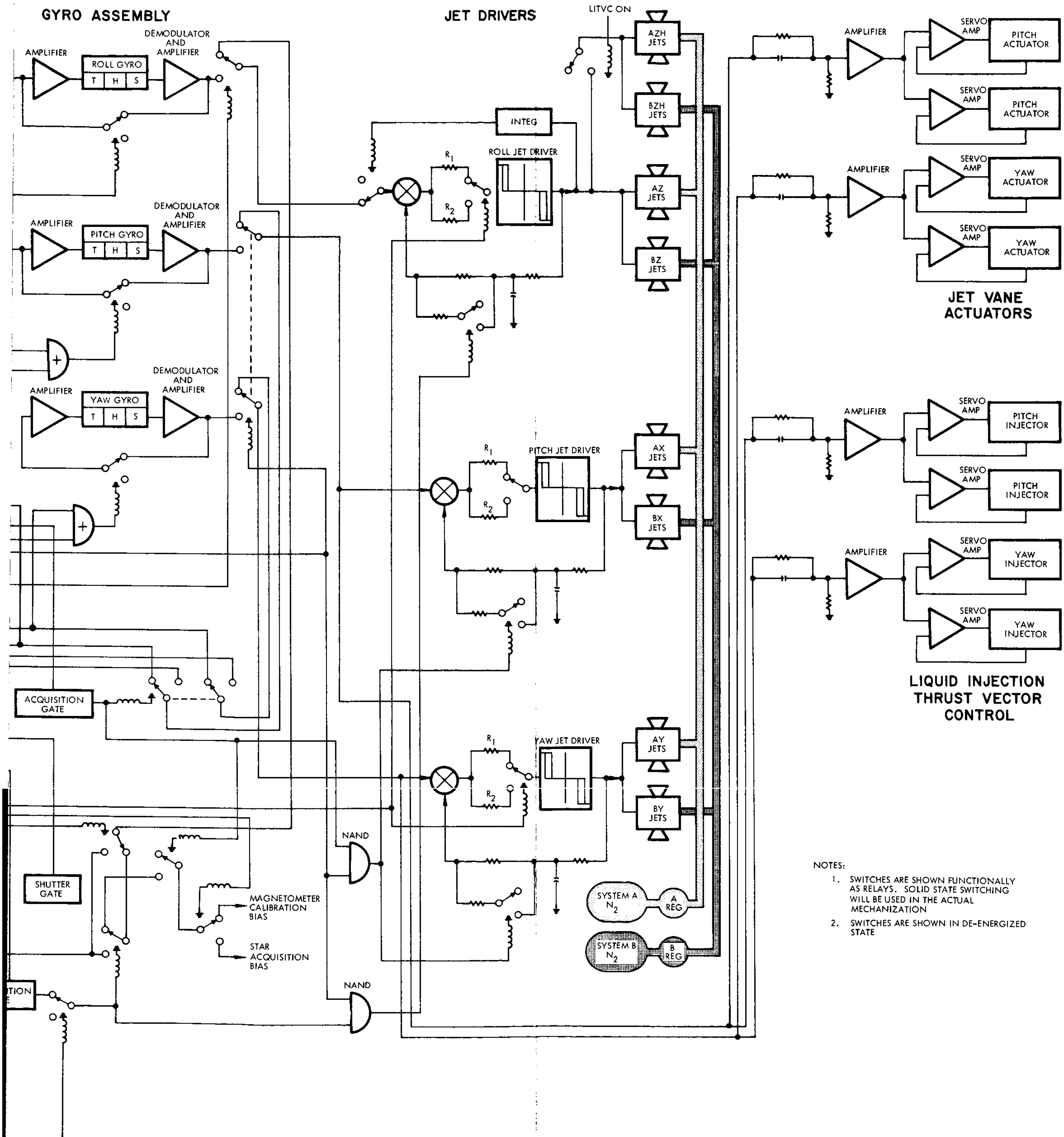


Figure 8. Voyager Stabilization and Control Subsystem Diagram

3.4 System Elements

The stabilization and control subsystem consists of inertial and optical sensors, a reaction control system, and signal processing and mode control circuitry. A system diagram is shown in Figure 8 and the various elements are described below.

3.4.1 Inertial Sensors

The inertial sensors are single degree of freedom, rate-integrating gyroscopes with hydrodynamic spin motor bearings. Three instruments are block mounted with a single temperature controller for the inertial reference package. The gyros are operated in the caged mode during acquisition to give rate signals. A current generator provides torquing signals for reorientation maneuvers. A functional diagram of one channel of the inertial reference is shown in Figure 9.

3.4.2 Sun Sensors

Two coarse sun sensors having a combined field of view 4π steradians and a fine sun sensor for precise reference are utilized. Each coarse sun sensor consists of a pair of silicon solar cells back-to-back, with plano convex lenses in optical contact with the cell surfaces. The elements of the coarse sensor are mounted on the periphery of the flight spacecraft to obtain an unobstructed field of view.

The fine sun sensor consists of a silicon photovoltaic quad cell, mounted behind a mask which acts as a shadowing structure. The field of view of the fine sensor is a cone of approximately 10 degrees half angle. The sensor is mounted inside the spacecraft to obtain a temperature-controlled environment.

The scale factors for the coarse and fine sun sensors are made approximately equal when the sun line is between 8 and 10 degrees from the fine sun sensor axis.

3.4.3 Star Sensor

The star sensor is essentially the same as the Mariner C Canopus tracker. The sensor has a total field of view of 4 x 30 degrees and an

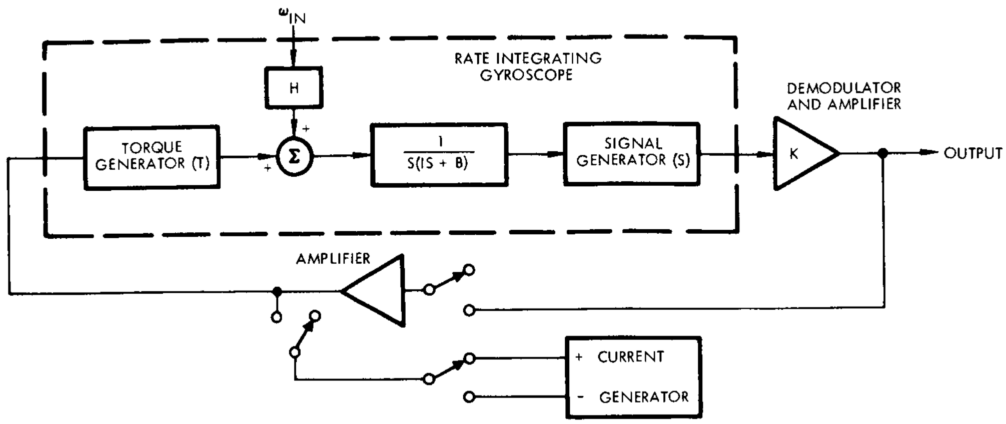


Figure 9. Voyager Gyro Loop

instantaneous field of view of 0.86 x 11 degrees. The sensor consists of a wide field of view lens, a diaphragm, filter, image dissector tube, deflection yoke, and signal processing electronics. A simple sun sensor controls a shutter to protect the star sensor from solar radiation above a selected threshold value.

3.4.4 Reaction Control System

The reaction control system consists of two identical, independent, heated gaseous nitrogen systems, each sized for the Voyager mission requirements. The dual systems provide equipment redundancy and also minimize the probability of mission failure due to gas leakage. The gas jets are mounted on the periphery of the flight spacecraft to maximize the control moment arm. Each system consists of six normal level heated thrusters, two high level thrusters, solenoid valves, plumbing, fill valve, pressure regulator, and a spherical storage vessel. Pressure and temperature transducers are provided for telemetry purposes. The solenoid valves and the pressure regulators use redundant seats to minimize.

3.4.5 Control Electronics

The control electronics contains sensor signal processing circuits, mode control circuits, and drive circuits for the reaction control valves, jet vane actuators, and liquid injector valves. Integrated circuits are used extensively to enhance reliability. Redundant circuits are employed in the mode control, valve drivers, and thrust vector control actuator servo amplifier circuits.

3.5 Coordinate System

The stabilization and control system sensing and control axes are given in VS-3-110.

The nominal cruise orientation is with the negative Z axis toward the sun and the Canopus sensor axis in the plane containing the sun, the spacecraft, and Canopus.

4. INTERFACE DEFINITION

4.1 Configuration and Packaging

SCS equipment is located on the spacecraft as shown in VS-3-110. The electronic equipment is mounted to spacecraft panel III as shown in VS-4-550.

4.2 Electrical Power

The stabilization and control system receives 4096 cps square wave power, 820 cps 2 step square wave power, and DC power from the power system as summarized in VS-3-111. Excess power is generally available for heating gas for the reaction control jets except possibly near end of life if solar cell degradation is high.

4.3 Central Sequencing and Command

Command requirements on the CS&C are given in Table 2.

4.4 Telemetry

Various analog outputs are utilized for measurements within the subsystem. These measurements are conditioned such that the measurement falls within data handling subsystem ranges as given in VS-4-311.

4.5 Thrust Vector Control

The SCS is compatible with the midcourse propulsion jet vane system presented in VS-4-610 and the retropropulsion liquid injection TVC system presented in VS-4-611.

Table 2. Stabilization and Control Requirements

Function	Number of Separate Commands Required	Initial Source of Commands	On-Board Command Source	Quantitative or Discrete Output	CS&C Memory Required	Logic for On-Board Command
Pitch start and stop time	2	Ground	CS&C	Discrete	yes	CS&C control gyro torquing
Pitch turn polarity	1	Ground	CS&C		yes	-
Roll start and stop times	4	Ground	CS&C		yes	CS&C control gyro torquing
Roll turn polarity	2	Ground	CS&C		yes	-
Canopus sensor select	1	Ground and On-Board	CS&C		no	CS&C selects sensor based on star ac with ground override
Canopus sensor cone angle update	6	Ground	CS&C	Discrete	yes	-
Command acquisition Mode	1	On-Board	CS&C		yes	See text
TVC test signal	1	On-Board	CS&C		no	See text
TVC electronics select	4	Ground	CS&C		no	
Gyros on/off	2	Ground	CS&C		no	
Roll incremental maneuver	2	Ground	CS&C	Discrete		
Maneuver start time	1			Internal		
Mode control and override	3	Ground	CS&C	Discrete		
Dead-zone select 65° and 25°	2					
Heater control for reaction control gas on/off	2					

5. PERFORMANCE PARAMETERS

5.1 Coarse Sun Sensor

Performance parameters for the coarse sun sensor are:

Field of view:	4π steradians
Linearity:	$\pm$ 10 per cent for $\pm$ 20 degrees about null plane (each axis)
Null accuracy:	1.0 degree

5.2 Fine Sun Sensor

Performance parameters for the fine sun sensor are:

Field of view:	10-degree half angle cone
Linearity:	± 10 per cent within field of view
Null accuracy:	0.1 degree per axis

5.3 Gyros

Characteristics and parameters for the gyros are:

Input axis angular freedom	± 5 degrees (minimum)
Gimbal freedom	± 3 degrees (output axis)
Gimbal bearing	Pivot and jewel or flexure
Motor	
Type	Synchronous hysteresis, four-pole
Excitation	Two-phase square wave 820 cps
Starting power	10.0 watts peak for less than 45 seconds
Running power	2.0 watts (maximum)
Signal generator power	4096 cps, 0.5 watts (maximum)
Long term drift	
G-insensitive	Initial value ± 0.3 deg/hr (maximum) Stability 0.4 deg/hr, 3 σ, 1 year
G-sensitive	Initial value ± 1π0 deg/hr-g (maximum) Stability 0.7 deg/hr-g, 3 σ, 1 year

5.4 Reaction Control Parameters

The normal control deadband is ± 0.5 degree from the nominal zero position about all three control axes. During photographic operation while in orbit and prior to reorientation maneuvers, the CS&C commands the control deadband to switch to ± 0.25 degree.

The nominal control angular acceleration is about 0.2 mr/sec^2 prior to flight capsule separation and about 0.8 mr/sec^2 after flight capsule separation about each of the three control axes when the gas jets are actuated.

The high level control angular acceleration is about 9 mr/sec^2 about each of two axes, roll (Z) and pitch (X).

5.5 Jet Vane Actuator Parameters

The jet vane actuator servo loops have a bandpass in excess of ten cycles per second.

Full deflection of the jet vane actuator from null deflects the mid-course correction rocket motor thrust direction by at least 5 degrees.

For operation during midcourse correction rocket motor burning, the SCS operates so that the worst expected combination of steady and transient disturbances does not cause the gyros or actuators to operate at greater than 50 per cent of their saturation limits.

5.6 Liquid Injection Actuators

The liquid injection actuator servo loops have a bandpass in excess of 10 cycles per second.

Full deflection of a liquid injection actuator from null deflects the retropropulsion rocket motor thrust direction by at least 4 degrees.

For operation during retropropulsion rocket motor burning, the SCS operates so that the worst expected combination of steady and transient disturbances does not cause the gyros or actuators to operate at greater than 50 per cent of their saturation limits.

5.7 Errors

5.7.1 Error Sources

Error sources within the stabilization and control system are as follows:

a) Coasting attitude control: Errors due to control dead-band that correspond to a rectangular probability distribution about a nominal zero point for each control axis.

b) Powered flight attitude control: Errors due to misalignment and offset between the nominal propulsion system thrust axis and the actual propulsion system thrust axis. For this purpose the actual propulsion System thrust axis is defined to be a line connecting the flight spacecraft actual center of mass with the centroid of thrust in the rocket motor nozzle exit plane.

c) Alignment errors: Null offsets and drifts in sensors that result in a shift of the zero point for the errors of (a) above.

d) System noise and short term variable, randomly distributed errors.

5.7.2 Over-all Errors

Pointing requirements for the Voyager mission involve interaction between the SCS, the high-gain antenna, the POP, velocity corrections, and the flight capsule.

a) Cruise Mode Pointing

Spacecraft accuracy during cruise mode operation, reflected into a single-axis error, is as follows:

Attitude reference accuracy	0.1 deg, 3σ
Limit cycle error	
(normal deadband)	0.5 deg, 3σ
(reduced deadband)	0.25 deg, 3σ

b) Pointing for Propulsive Maneuver

The pointing accuracy required during propulsive maneuver is 0.5 degree, 1σ. The most critical condition is during retrofire where the effects of thrust offset are greatest.

The primary sources of error during retrofire velocity correction are:

Gyro torquing error	$\pm$ 0.2 deg, 3σ
Gyro alignment error	$\pm$ 0.1 deg, 3σ
Gyro drift error	$\pm$ 0.4 deg, 3σ
Limit cycle error	$\pm$ 0.5 deg. 3σ

TVC limit cycle error	$\pm$ 0.3 deg, 3σ
Control error from thrust misalignment	$\pm$ 0.5 deg. 3σ
Control error from center of mass offset	$\pm$ 1.1 deg. 3σ

The RSS total for the retrofire correction is 0.465 deg, 1σ

c) Capsule Separation Pointing

For the reorientation maneuver prior to capsule separation the required accuracy is 0.75 degree, 3σ.

The primary error sources are:

Gyro torquing error	$\pm$ 0.2 deg, 3σ
Gyro alignment error	$\pm$ 0.1 deg. 3σ
Gyro drift error	$\pm$ 0.4 deg, 3σ
Limit cycle error	$\pm$ 0.25 deg, 3σ
Capsule to spacecraft alignment	$\pm$ 0.4 deg, 3σ

The RSS total is $\pm$ 0.66 degree, 3σ

5.8 Reliability

The reliability block diagram and assessment for the stabilization and control system are discussed in VS-3-120.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Turns

The stabilization and control subsystem places no limits on the magnitude of spacecraft commanded turns. Turns are made sequentially, with a roll turn first followed by a pitch turn. A second roll turn can be connected, if necessary, to allow the high gain antenna to point to earth in any maneuver attitude.

6.2 Settling Time

The system is capable of stabilizing the spacecraft and acquiring its references in a period not to exceed 2 hours after initial turn on or emergence into the sun. This assumes angular rates on separation from the boost vehicle of up to 50 mr/sec about all three axes.

6.3 Reflected Light Into Canopus Sensor

Shades are incorporated wherever stray light about 0.1-foot candle may be reflected into the Canopus sensor during cruise and orbit:

a) The upper thermal shield is extended to shade louvers adjacent to sensor.

b) Shades are provided between the solar panel spars adjacent to the sensor.

c) Shades are provided on reaction control jets adjacent to sensor (reaction control jet shade).

In the event that light into the Canopus sensor causes the shutter to close, the roll axis is put under inertial control automatically if it is not already in that mode.

6.4 Environment

The SCS is compatible with the environment described in 4.2 of VS-2-110. Specific allowable temperature ranges are given in VS-3-111.

7. SAFETY CONSIDERATIONS

The stabilization and control subsystem, is designed with factors of safety so that when the reaction control gas system is fully pressurized, it is safe for personnel to work around it. In addition, the reaction control system is designed so that it may be loaded or unloaded when the flight spacecraft is fully assembled with the deployable components in the launch configuration.

High voltage circuitry, such as that required for the star sensor, is shielded so that there is no hazard to personnel working with it.

CENTRAL SEQUENCING AND COMMAND

VS-4-450

CONTENTS

1. SCOPE

This document presents the design for the central sequencing and command (CS&C) subsystem. This subsystem accepts command data and provides timing and sequencing functions for the spacecraft.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-2-110	Design Characteristics and Restraints
VS-3-111	Components Design Parameters
VS-3-120	Reliability Design Objectives
VS-4-470	Electrical Distribution
VS-4-550	Electronic Equipment Packaging

3. FUNCTIONAL DESCRIPTION

3.1 System Elements

The functional elements for the CS&C are the input decoder, command decoder, sequencer, and a power converter. Two of each of these units are used in a redundant fashion. A block diagram is shown in Figure 1.

3.2 Functions

The CS&C accepts messages from the command detector of the telecommunications subsystem and routes discrete signals to other spacecraft subsystems or stores command data as indicated by the command format. Acting on such data, the CS&C performs timing and sequencing functions for the spacecraft.

In addition to controlling spacecraft operations, the CS&C provides a synchronizing function by distributing clock frequences to the electric power and data handling subsystems.

3.3 System Operation

The input decoder is used to process command messages and route the information to the appropriate units. The decoder receives a serial

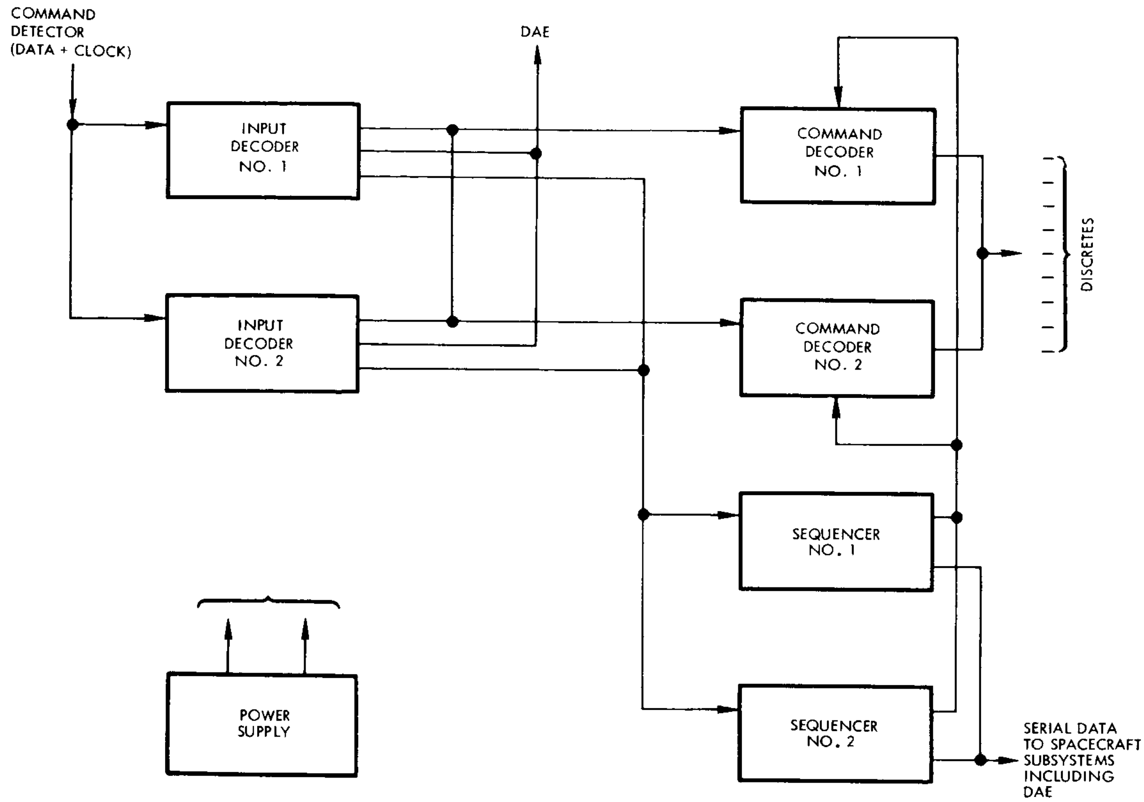


Figure 1. SC and C Subsystem Redundancy with Centralized Memory System

bit stream from the telecommunications command detector, decodes valid messages, routes direct commands to the command decoder or to the data automation equipment decoder for immediate action, and routes delayed commands and numerical data to the sequencer for storage and later use. The input decoder inspects the incoming command for a valid synchronization code and subsystem address using parity checks to increase the error rejection probability. Parity checks are available in the CS and C subassembly address, and memory cell or direct command address. A subassembly address in the form of a code in the incoming message selects the particular unit of each redundant pair that is being used. This address also distinguishes between direct commands to be decoded immediately and delayed commands to be stored in the sequencer memory.

The command decoder is used to interpret all commands, whether they issue from the sequencer memory or directly through the input decoder. The command decoder distinguishes up to 128 commands and issues discrete pulses to appropriate spacecraft subsystems. It also accepts discretes from the sequencer and issues them through its output circuits. The operative command decoder must receive an enable signal as both of the command decoders in the redundant system are receptive to the same commands. This enable must be issued by either the input decoder or the sequencer, depending on whether the command is direct or delayed. For the sequencer to issue an enable, identifying information must have been previously stored in the sequencer, setting a mode bit in the sequencer by a ground command.

The sequencer contains a centralized, random access memory whose word locations are function-oriented. It stores time-tagged commands, quantitative data and data to be transferred to the data automation equipment, scans memory continually for action items, and at the proper times delivers commands to the command decoder or issues serial data to the appropriate subsystem. Serial data which is issued to the data automation equipment may be either command or numerical data. The bits are shifted into the DAE decoder and data register and an enable signal is issued. The DAE then parity checks and decodes the data into a control pulse or constant level signal to a science subsystem.

A normal reset of the input decoder occurs when a message is complete. Since there are two message lengths, the input decoder interrogates the address register at specified times to determine the type of command. This information is necessary to reset the input decoder as well as to provide the command decoders or sequencers with an enable signal signifying that the subassembly in question has received the complete message and can now act on its content. The command decoders and sequencers each receive a separate enable signal.

Logic is provided at the bit sync input to the address register to sense a "1" as the lead bit in a message. After a reset, bits are shifted into the address register only after a "1" has been detected. However, after a reset due to a failure condition, bits are shifted into the address register only if the "1" is preceded by 12 successive "0's," thus increasing the sync code requirements prior to acceptance of new messages.

A logic diagram of the operations not including the redundancy is shown in Figure 2.

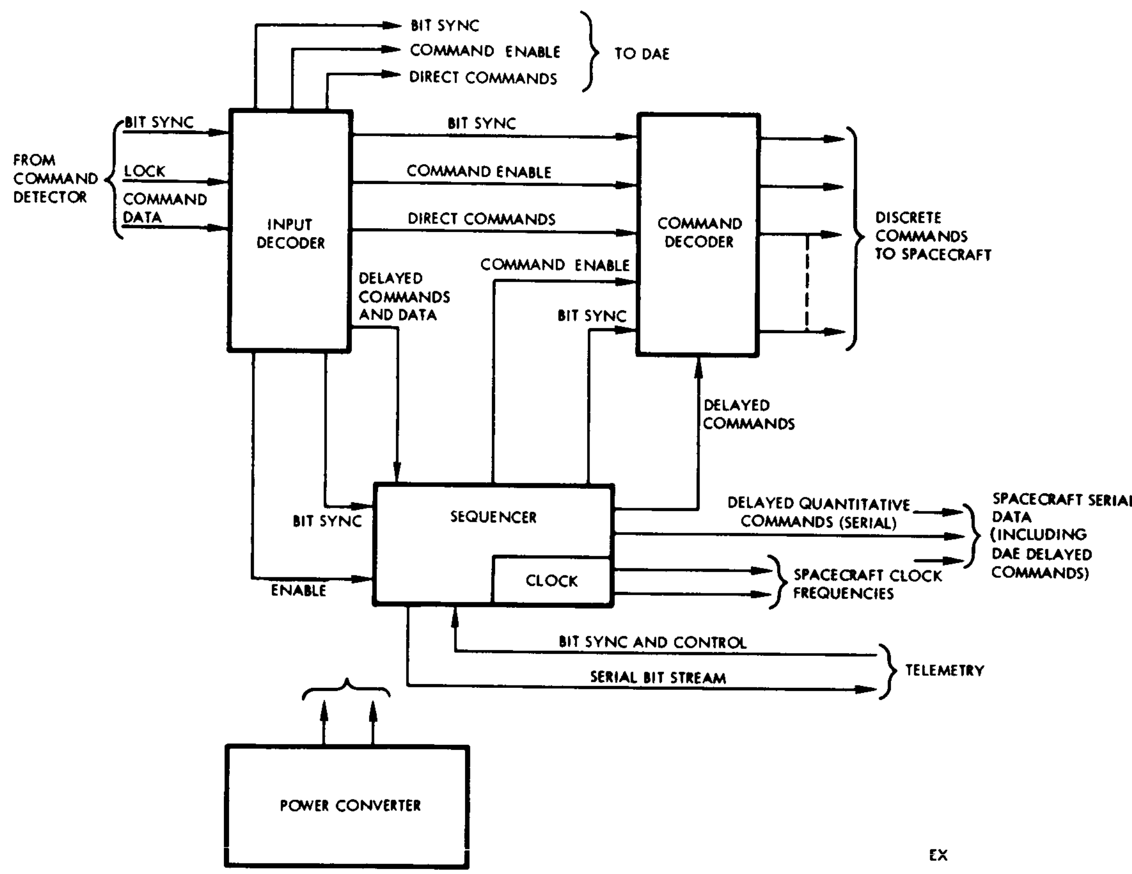


Figure 2. CS and C Centralized Memory System

3.4 Input Decoder

The input decoder is a low power digital command processor and routing device. A diagram for the unit is given in Figure 3. An address register is shifted and loaded by the command detector's command data line and bit sync line. The bit sync pulses also pulse a bit counter which is used to establish the times at which specific portions of the message are in the address register. The address register is 8 bits long. At specified bit counts, e.g., spacecraft address time and synchronization code time, the outputs of the address register are sampled to establish the validity or presence of the message portion in question.

In the event that a test or condition fails, such as parity [redacted], the input decoder is reset. The reset immediately places the input decoder in a condition ready to accept the next command. Loss of receiver lock at the command detector, as indicated by the lock line, also resets the input decoder.

The outputs of the redundant input decoders include OR gates such that either input decoder can accomplish the task.

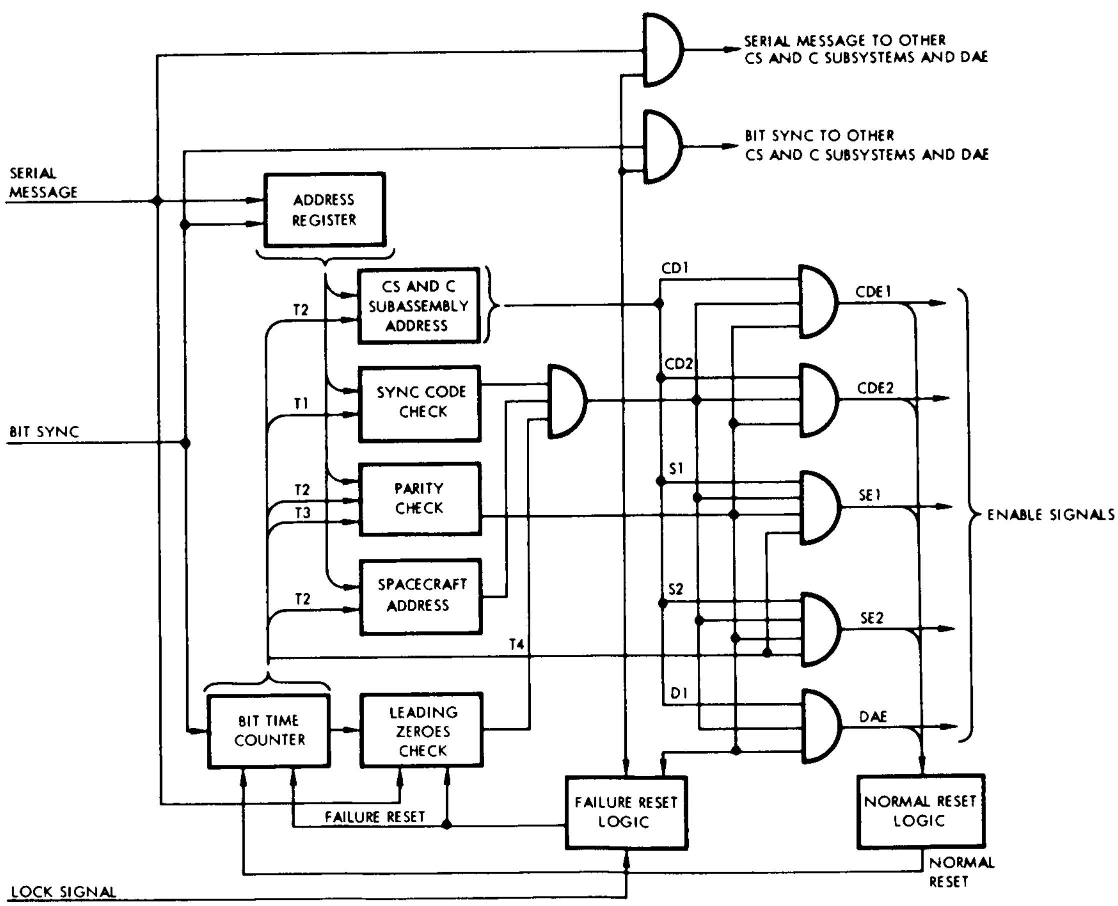


Figure 3. Input Decoder with Redundancy

3.5 Command Decoder

Figure 4 is a block diagram for the command decoder. The redundant decoders are nominally in continuous operation. Should a failure in a command decoder be detected by the ground station a command format change is executed to avoid using the failed command decoder. Each command therefore contains the code which specifies the appropriate command decoder to be employed. No distinction is made between command decoders in the case of sequencer-issued commands. Sequencer-issued discretes have already been decoded upon arrival at the command decoder and are shunted to the output buffers of both command decoders, which supply the necessary driving power to the subsystems. The decoder bit in the mode register defines the enable condition of the sequencer.

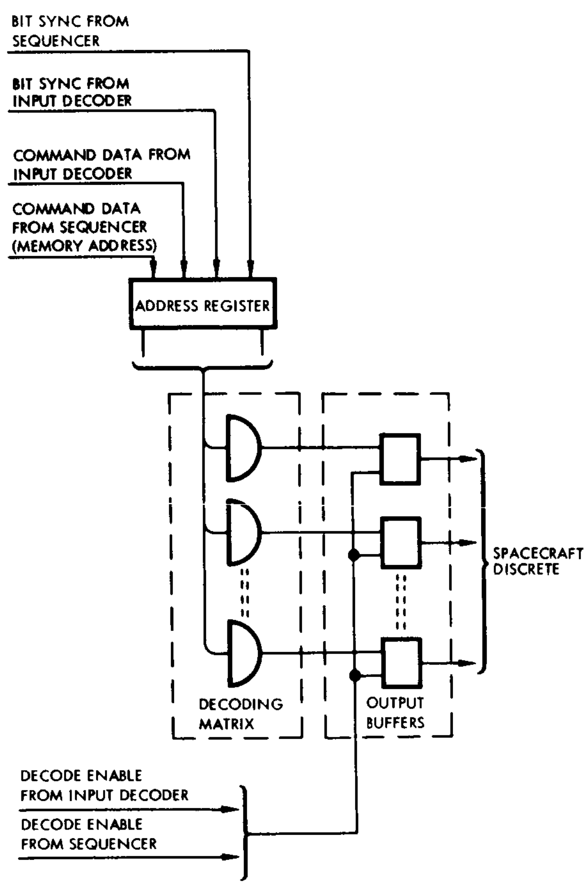


Figure 4. Command Decoder Centralized Memory System

The incoming command data and bit sync pulses from the command detector load and shift, respectively, the address register. Since the command decoders are continuously sampling these lines, the register is always activated and the decoding matrix associated with the register is constantly decoding. To ensure that only the portion of the message intended for a command decoder is decoded as a discrete, a decode enable must be issued by either the sequencer or the input decoder. This enable conditions the output buffers to respond to the decoding matrix output.

3.6 Sequencer

The sequencer consists of a random-access core memory, a clock, an output decoder, parity checking logic, and the logic necessary to address and store information in the memory. A block diagram is shown in Figure 5. The command inputs in the form of time-tagged direct commands and quantitative commands are stored in a 256-word coincident-current core memory. The sequencer sequentially scans the contents of memory looking for a time match of the data with the mission elapsed time. A valid match indicates that the command specified by the particular location is to be executed. The sequencer provides the ground with telemetry data reflecting the contents of memory and the elapsed time register.

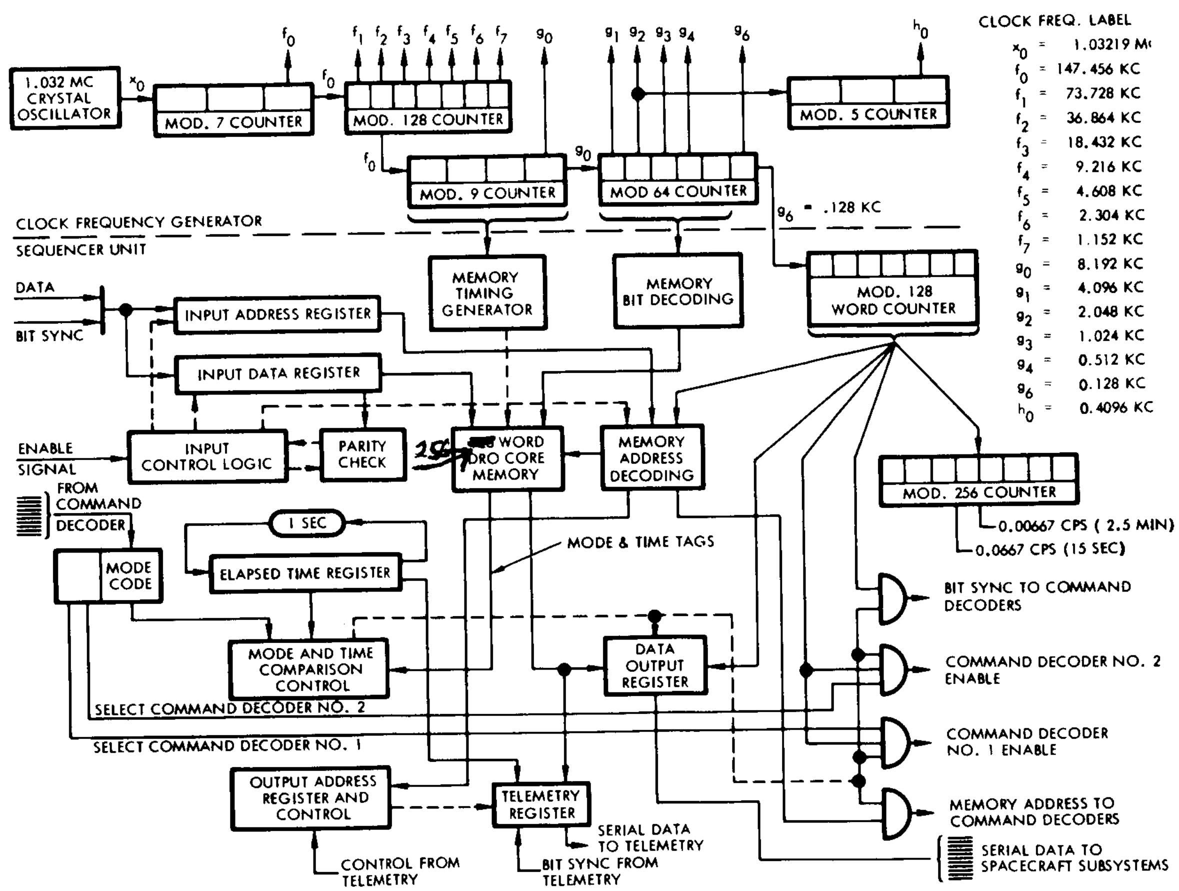


Figure 5. Sequencer Logic Diagram

3.6.1 Clock Generation

The clock generation portion of the sequencer provides the frequencies necessary to operate the memory as well as other spacecraft subsystems. The clock generation process starts with a 1032 kc crystal controlled oscillator which provides an input to 6 counters. These counters divide the oscillator frequency down to 18 required output frequencies.

A 26-bit basic clock provides a continuous elapsed time record for 400 days with a granularity of 1 second. Secondary timing is available by sampling portions of the master clock. These samples are compared regularly with the 14-bit time portion in the memory word to establish coincidence. A mode register is also sampled to determine which secondary clock is being used as the reference. Figure 6 presents the clock format.

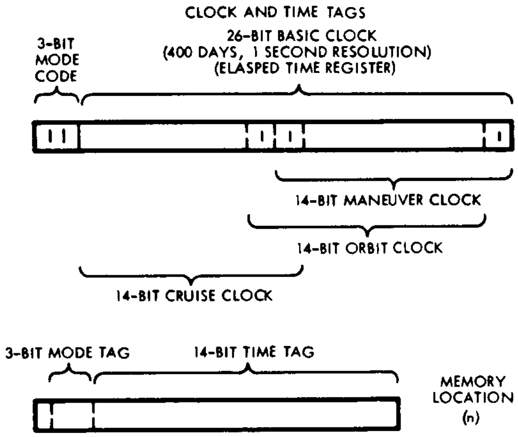


Figure 6. Clock and Time Tags

3.6.2 Memory Addressing

The input addressing register decodes the 8-bit address portion of the command and energizes the appropriate drivers to store the input data. The input data is stored in an input data register prior to being stored in the core memory. The command data line and bit sync line from the command detector fill the address register and input data register. Upon receipt of an enable from the input decoder the address register performs the memory storage cell selecting process and the input data register is checked for correct parity. The correct parity indication along with the received enable permit storage of the input data register in the specified storage cell.

The memory bit and memory address decoding block provide the bit timer and the word location times necessary to address the memory.

3.6.3 Time Scanning and Output Decoding

The sequencer continuously scans the memory for an execution time match between the time and mode tags in each storage cell and the elapsed time register and mode code registers. When a successful match is detected the memory address counter is shifted out to the command decoder. The decoded address defines either one subsystem to receive a discrete pulse or one subsystem to receive quantitative data stored in an adjacent memory cell. In the case of a quantitative command the memory address decoding logic initiates a transfer of data from the adjacent storage cell to the output register. From the output register this data is transferred to the designated subsystem in a serial stream. The data is clocked out of the output register at a 73 kc rate. This clock rate is provided to telemetry to maintain synchronization in the data transfer.

Upon command verification the delayed direct command in storage is enabled. A verify-enable command is transmitted to the sequencer to alter the verification bit in the stored word. This enable function is accomplished without altering the time tag associated with the command. The verification commands contain a memory address and a zero time tag. The sequencer recognizes the zero time tag as an instruction to alter the verification bit in the specified memory address.

3.7 Power Converter

The power converter provides appropriate power to the CS&C elements and carries out power switching between the redundant units. The regulated 50 VAC primary power is converted to the working voltages required. It provides continuous power to both input and command decoder and provides power switching to the redundant sequencer in the event of failure in the operating unit. It also provides power switching to a standby power converter in the event of a power converter failure.

3-1

4. INTERFACES

4.1 Mechanical

The CS&C subsystem is panel mounted as described in VS-4-550.

4.2 Electrical Distribution

The central sequencing and command function imposes requirements for electrical distribution of signals to many spacecraft units. This signal flow is illustrated in VS-4-470. All outputs to the spacecraft subsystems are DC, transformer isolated.

4.3 Subsystem Discretes

The command decoder provides one discrete pulse to a particular subsystem on the basis of the command address. Up to 128 discretes can be decoded from the command address. Each of these decoded discretes provides the subsystem interface with a 120-ma, 28-volt signal. The discrete pulse is enabled for a minimum of 1 second in the case of direct ground commands or for a minimum of 8 milliseconds in the case of sequencer issued commands.

4.4 Telecommunications Command Detector

The input decoder accepts the serial bit stream from the command detector. Along with the serial data the command detector provides a bit synchronization clock and a separate enable signal which assures the input decoder that the message is in lock at the command detector. The moment that lock is lost, the command detector removes this enable signal, thereby informing the input decoder that the incoming message is no longer locked with the receiver in frequency, phase, and PN code.

The outputs of redundant command detectors include OR gates so as to provide only one set of inputs to the CS&C subsystem for the input command.

The frequency and logic levels of the three input lines from the command detector unit are:

	Frequency	Logic Level
Command line:	1 pulse/sec	0, + 3v
Bit synchronization:	1 pulse/sec	0, + 3v
Lock:	DC	0, + 3v

4.5 Telemetry

4.5.1 Memory Read Out

For the purpose of verifying specific stored commands prior to their execution or for a general check on the contents of the sequencer's memory, an output is provided to telemetry. An 18-bit register holds a memory word until it is sampled by telemetry and an end-of-sample message is given to the CS&C. At this time the next word in line is transferred to that register. Each 18-bit group contains one entire storage cell content comprised of a time tag plus verification bits and mode tag, or data. Each cycle through memory starts or ends with a sampling of the elapsed time register, which resynchronizes the bookkeeping process of tracking the cell location with the cell contents. The elapsed time register is available to the telemetry subsystem at all times.

Maneuver data words that have just been transmitted by command link and stored in the memory are placed in the telemetry line ahead of other data to provide rapid verification.

4.5.2 Data Rate

The telemetry subsystem is also provided with a discrete signal indicating one of four data rates. These discretes exhibit a logic level swing between 0 ± 0.6 and 4 ± 1 volt, rise time less than 1 millisecond, and a 10 ma drive capability at the 0 state. Along with the discrete, the CS&C supplies the appropriate group of frequencies for the particular data rate. The list below gives the four groups of frequencies associated with the data rates.

Data Rate Group	1	2	3	4
Frequencies (in kc)	4.096	2.048	1.024	0.128
	147.456	73.728	36.864	4.608
	73.728	36.864	18.432	2.304
	36.864	18.432	9.216	1.152

These frequencies are provided to the telemetry subsystem with a 0.005 per cent long term accuracy.

4.6 Mission Operations System

4.6.1 Command Words

There are four types of messages which must be processed by the CS&C:

- Direct Commands. Provide a designated subsystem with a discrete pulse. The discrete pulse is issued by the CS&C or by the DAE as soon as the incoming command is decoded.

- Delayed Direct Command. Provides a designated subsystem with a discrete pulse at some time after the receipt of the command. The time of execution depends on the time tag value accompanying the command. The command is stored in memory until its execution time; a discrete pulse is then issued to the designated subsystem. The memory location identifies the designated subsystem. Maneuver commands cannot be executed until they have been verified on the ground.

- Verified-Enable Command. Accesses the memory and changes the verification bit in any particular delayed direct command location. The same format as delayed direct commands is used with a zero time tag. The zero time tag identifies the function of the command while the storage cell address specifies which command is to be changed (hence which command is verified).

- Quantitative Data. Provides a designated spacecraft subsystem with serial data. Before the data is transmitted to the subsystem it is placed in the CS&C memory with an associated delayed direct command which determines when this data is to be provided to the designated subsystem. The same format is used for quantitative data as for delayed direct commands. However, the memory cell to which it is addressed insures that the data cannot be interpreted as a command.

4.6.2 Command Word Format

Figure 7 illustrates the message formats for the four message types identified above. Constituent parts are described below:

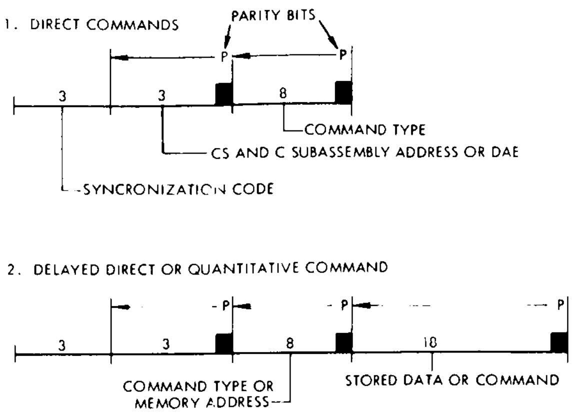


Figure 7. Command Formats

- Synchronization Code (3 bits). A synchronization code is examined by the CS&C to determine the validity (format bit position) of the incoming message. A 3-bit code is used to check the sync between successive messages in a continuous bit stream; however, a longer unique pattern is required to initially establish sync after a detection of sync loss. This can easily be handled by requiring the ground station to send a constant pattern (e.g., 12 or more 0's) followed by the normal 3-bit sync code at the beginning of message transmission, even when loss of sync is detected (e.g., via telemetry).

- CS&C Subassembly Address (3 bits). This address code provides the ground station with a code capable of selecting the particular subassembly to be used during the command processing. It also distinguishes among direct commands to redundant CS&C decoders, direct commands to the DAE, and quantitatives to the sequencer.

- Command Type (8 bits). These 8 bits are used directly in the command decoder to determine which subsystem or experiment (which output line) is to be selected for execution of direct commands. For other message types these bits determine the memory address where the command, data or verification bit is to be stored. In the case of stored commands, the memory address (the same 8 bits) identifies the subsystem (output line) which is to be pulsed at the appropriate time.

- Stored Data or Command (18 bits). These 18 bits are stored in the memory for stored quantitative data and stored command messages. For the stored commands the 18 bits consist of 1 verify bit, 3 mode bits and 14 time tag bits. The exact format of the stored quantitative data is not critical since it does not require all 18 bits.

- Parity Bits. Three parity bits are checked by the CS&C during receipt of a quantitative command. The parity bits are distributed as shown in Figure 7. The distribution has been selected on the basis of the number of message bits being decoded at any given time in the decoding process.

5. PERFORMANCE

5.1 Command Word Bit Rate

The command word bit rate is 1 bit per second.

5.2 Timing

The basic timing tolerance of the CS&C clock is one part in 2×10^4 over the allowable temperature range. Power system frequencies and event initiation are subject to this tolerance. A basic 400-day clock is provided with 1 second resolution.

5.3 Output Power

The outputs of the CS&C are 150 ma, 28 volt pulses.

5.4 Reliability

Reliability data is given in VS-3-120.

5.5 Memory

The CS&C memory consists of function-oriented centralized cores. It has a capacity for 256 words with 18 bits per word.

5.6 Decoding Capacity

The CS&C can decode up to 128 commands.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Spacecraft Commands

Up to 128 distinct commands to the spacecraft subsystems will be provided together with the capability for initiating discretes and numerical data from the sequencer to the spacecraft subsystems. Direct commands, delayed direct commands and numerical data can be issued to the DAE.

6.2 Out-of-Lock Condition

No command word will be executed during periods when the command detector is out of lock. Upon detector lock-up with the command signal, the CS and C must receive at least 12 consecutive zero bits before command word decoding can be accomplished by the decoder.

6.3 Environment

The CS and C environment is compatible with the 4.2 of VS-2-110. The allowable temperature limits are given by VS-3-111.

6.4 Weight, Volume, Power

Weight, volume, and power data are given in VS-3-111.

ELECTRICAL POWER

VS-4-460

CONTENTS

1. SCOPE

This document presents the design for the electrical power subsystem. This subsystem provides a central supply of electrical power to operate spacecraft electrical equipment.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-3-111	Components Design Parameters
VS-3-120	Reliability Design Objectives
VS-4-550	Electronic Packaging

3. FUNCTIONAL DESCRIPTION

3.1 Function

The function of the electrical power subsystem is twofold. First, it provides a central supply of electrical power in the proper forms to operate the electrical equipment onboard the spacecraft. Second, it provides the required switching and control functions for the effective management of that power.

Specifically, the power subsystem satisfies the following functional requirements:

- Provides a photovoltaic power source using an oriented solar cell array.
- Provides for energy storage in the form of secondary batteries to energize the spacecraft loads during the launch-to-solar acquisition phase, during maneuvers away from the cruise orientation, and during eclipses in orbit around Mars.
- Conditions power to the required voltages, frequencies, waveforms, noise levels, etc., for spacecraft use.
- Provides power to the flight capsule as required.

- Provides solar array and battery controls as required.
- Provides for synchronization of switching type power control and conditioning equipment.
- Provides for telemetry monitoring of selected power system parameters.

3.2 Description and Operation

The power subsystem shown in Figure 1 consists of the following major functional elements: the solar array, solar array shunt elements assembly, secondary batteries, battery regulators, power conditioning equipment, and a power control unit (PCU) which provides the monitoring, synchronization, control, switching, and logic functions for the power subsystem elements.

3.2.1 Solar Array and Controls

The solar array is divided into 12 identical electrical sections connected in parallel through suitable isolating diodes. The solar array output voltage is limited to 50 VDC ± 1 per cent by shunt regulation of a portion of the series connected solar cells in each section.

Power dissipation in the shunt elements assembly (SEA) is minimized by a unique sequential shunt configuration. The SEA is controlled from bus voltage sensing and error signal amplifier circuitry located in the PCU.

a. Operation of Shunt Regulator

A block diagram of the sequential shunt regulator is shown in Figure 2. Each section of the solar array is provided with a suitable tap point, and a simple transistor power amplifier is connected between this point and the return bus to shunt a portion of the series-connected solar cells. The shunt elements are controlled to draw current from the shunted solar cells in direct proportion to the magnitude of an error voltage determined by comparing output bus voltage with a suitable reference. The transconductance of each shunt is linear between zero and short circuit current of the shunted solar cells. The combined transconductance of the total shunt regulator is sufficiently large to saturate all of the shunt power amplifiers with a bus voltage error signal within the regulation tolerance

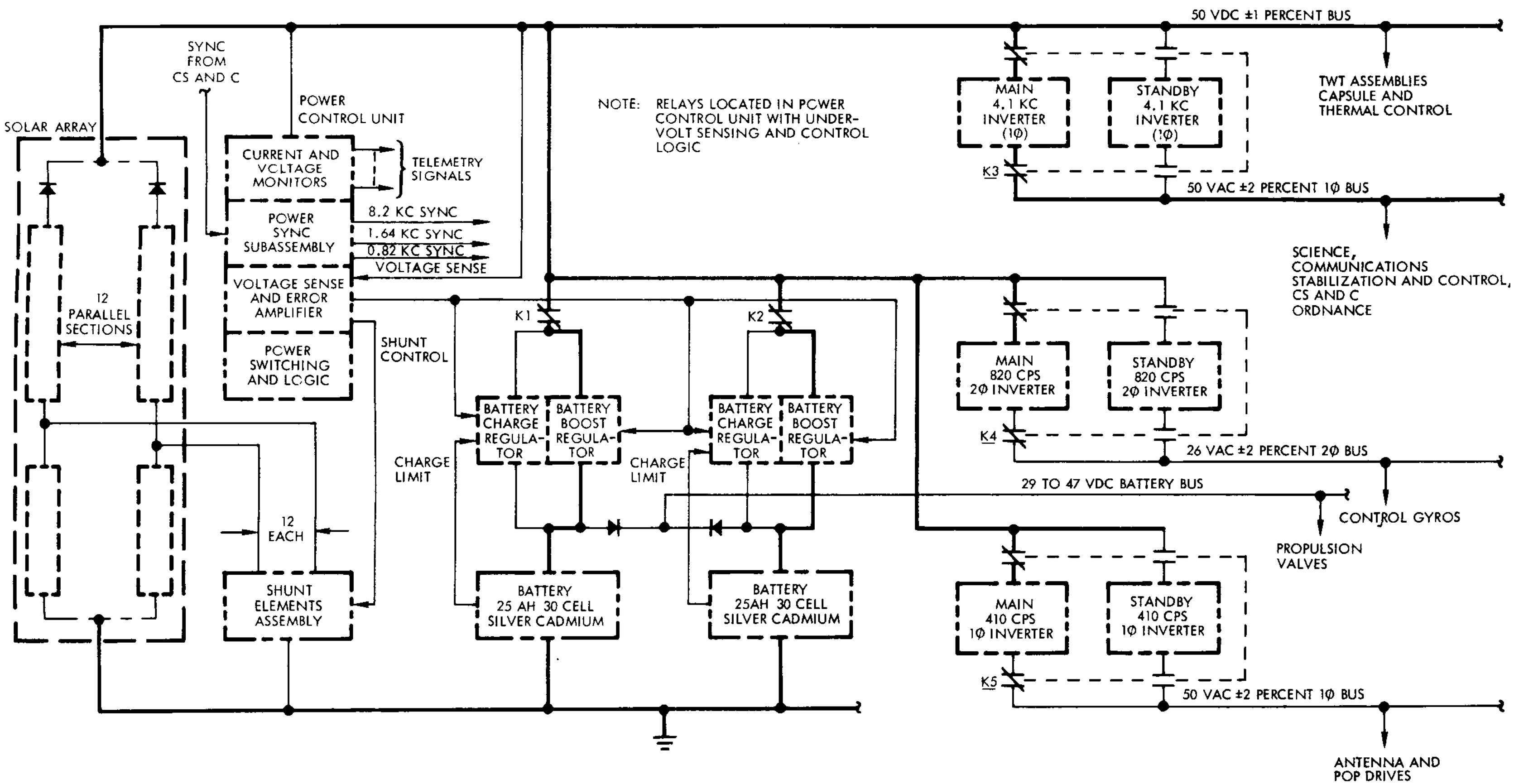


Figure 1. Power Subsystem Block Diagram

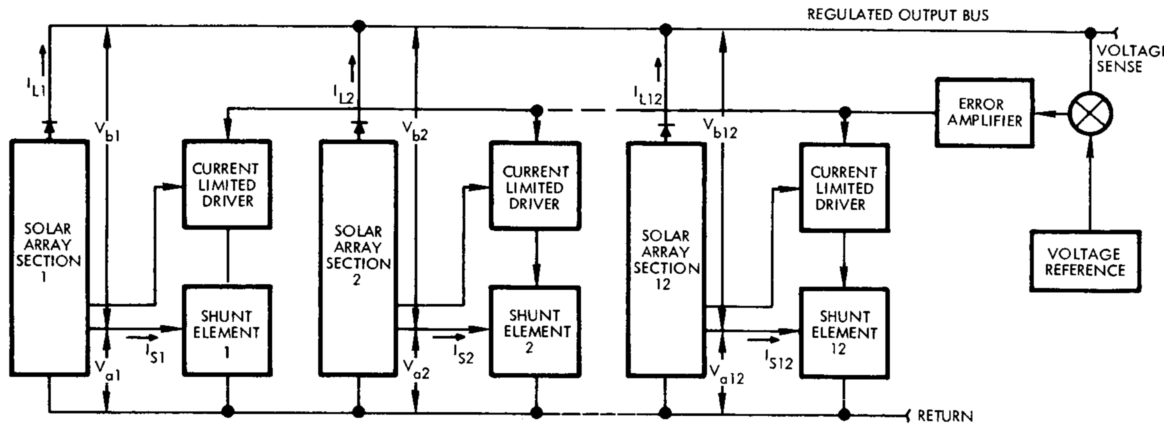


Figure 2. Sequential Shunt Regulator Block Diagram

limits. Each shunt element is designed to begin conducting at a higher error signal than the preceding element. Each shunt element is saturated at an error signal which causes the succeeding element to begin conducting. In this way, only one shunt power amplifier is linearly conducting as a function of error signal at a time. The other shunt elements are either nonconducting or saturated.

Base drive current to the shunt elements is limited to reduce total drive current after several power amplifiers saturate. To reduce over-all drive power dissipation, an additional solar array tap point, approximately 2 volts above the shunt element tap point, is brought out to supply shunt drive current at a voltage significantly less than the bus voltage.

b. Operating Conditions

Figure 3 illustrates a typical operating condition for the various solar array sections. The load current required is equal to $I_{L1} + I_{L2}$, and is assumed to have a magnitude which requires full output from array section 1 and partial output from array section 2. For each section, the solar cell operating conditions are indicated by the current voltage plots for cells both above the tap point and below the tap point (shunted). To maintain bus voltage regulation, the sum of the solar cell voltages, including diode voltage drops, must equal the desired bus voltage for those cells which contribute to the output bus current.

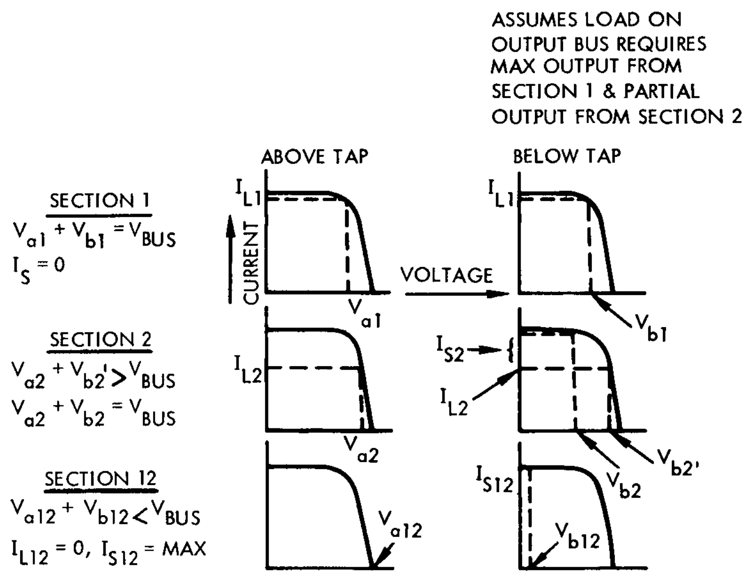


Figure 3. Typical Operation of Sequential Shunt Regulator

Section 1 operates with no shunt current and delivers maximum current (I_{L1}) at regulated bus boltage (V_{bus}). Section 2 delivers the remainder of the load current (I_{L2}). The solar cells above the tap point produce a resultant voltage of V_{a2}. The shunted solar cells are then controlled to provide a current $I_{L2} + I_{S2}$, which produces an additional voltage, V_{b2}, such that $V_{a2} + V_{b2}$ is equal to V_{bus}. The remaining sections, 3 through 12, are operated with the shunt elements in a saturated condition to reduce their voltage below V_{bus} and prevent them from contributing to the load.

The total shunt power dissipation for the assumed operating condition is:

$$P_{shunt} = I_{S2} V_{b2} + (12 - 2) \; (I_{S12} V_{b12})$$

where V_{b12} is small, representing the saturated voltage drop of the shunt power amplifier.

3.2.2 Batteries and Controls

The batteries are charged from the 50 VDC bus through simple dissipative current limiters. Charging is terminated by a control signal from individual temperature-compensated cell voltage sensors mounted on the battery cells.

Whenever the solar array is incapable of supporting the system load, as during maneuvers and eclipses, the batteries discharge through boost regulators to maintain the regulated 50 VDC bus.

The two 30-cell, 25 ampere-hour, silver-cadmium batteries, each with a charge-discharge regulator, are operated in parallel under normal conditions. In the event of a battery or regulator malfunction, the associated battery and regulator are disconnected by the power switching and logic circuitry in the PCU. The remaining battery is capable of supporting essential spacecraft loads through maneuver and eclipse operations. Command capability is provided to permit charging at an increased rate in the event of single battery operation.

3.2.3 System Outputs

The two main outputs from the system are the regulated 50 VDC bus and a 50 VAC, 4.1 kcps, single-phase square-wave bus. A simple unregulated inverter is used to supply this AC output. Sequential inverter redundancy is provided in event of an inverter failure by sensing AC bus undervoltage and switching to the standby inverter. This sensing and switching function is performed by the PCU power switching and logic circuitry. The majority of the loads are energized through transformer rectifier units from the AC bus. These transformer rectifier units are considered part of the load equipment and may also include supplementary output regulators where required.

Additional 410 cps single-phase and 820 cps two-phase inverters are provided to supply AC power to the antenna and planet-oriented package drive motors and to the control gyros, respectively. Sequentially redundant units are provided in the same manner as in the case of the 4.1-kilocycle inverter.

4. INTERFACE DEFINITION

4.1 Structural Support

The solar array is installed and supported as shown in Figure 1 of VS-4-520.

4.2 Orientation

Solar array orientation to within ± 10 degrees of normal to the solar vector is necessary to provide specified output.

4.3 Electrical Inputs

Power subsystem inputs are as follows:

a) External power to operate the spacecraft during prelaunch and checkout

b) Control signals to operate redundant unit switching circuits during prelaunch and checkout

c) 4096 cps synchronization signal from CS&C

d) Command signals to provide for in-flight override of automatic switching functions and to select battery charging rates.

4.4 Electrical Outputs

Power subsystem outputs are as follows:

a) Main 50 VDC ± 1 per cent bus

b) 50 V ± 2 per cent, rms, 4.1 kcps, single-phase, square-wave bus

c) 50 V ± 2 per cent rms, 410 cps, single-phase, square wave bus

d) 26 V ± 2 per cent rms, line-to-neutral, 820 cps, two-phase, square wave bus

e) 29-47 VDC battery bus

f) Conditioned analog telemetry signals representing power subsystem voltages and currents.

4.5 Central Sequencing and Command

Command requirements for electrical power are given in Table 1.

Table 1. Power Command Requirements

Unit	Status	No. of Commands
Battery No. 1, Charge Control Override	On/Off	2
Battery No. 2, Charge Control Override	On/Off	2
Battery No. 1, Disconnect	Open/Close	2
Battery No. 2, Disconnect	Open/Close	2
4.1 kc Inverter Transfer	Main inverter on/ standby inverter on	2
820-cycle Inverter Transfer	Main inverter on/ standby inverter on	2
410-cycle Inverter Transfer	Main inverter on/ standby inverter on	2
Charge Rate Select	Charge Rate 1 Regulator No. 1 Charge Rate 2 Regulator No. 1 Charge Rate 1 Regulator No. 2 Charge Rate 2 Regulator No. 2	4

5. PERFORMANCE

5.1 Power Output

The power subsystem is designed to supply the following electrical power outputs during periods of normal orientation in sunlight at a maximum sun-spacecraft distance of 1.67 astronomical units:

a) 215 watts maximum at 50 volts, 4.1 kcps, single-phase, square wave for science, stabilization and control, telecommunications, CS&C , and ordnance.

b) 47 watts maximum at 50 volts, 410 cps, single-phase, square wave for antenna and planet-oriented package drive motors.

c) 27 watts maximum at 26 volts, line-to-neutral, 820 cps, two-phase, quasi-square wave for gyro motors.

d) 432 watts maximum at 50 VDC for traveling wave tube assemblies, stabilization and control, flight capsule, and thermal control heaters.

5.2 Battery Discharge

During battery discharge in the launch-to-acquisition, maneuver and eclipse phases, the required power subsystem outputs are the same as in 5.1 above, except that the 50 VDC requirement is reduced to 300 watts maximum. The maximum continuous duration of battery discharge is 2.0 hours during maneuvers and 2.3 hours during eclipses. The minimum recharge period in orbit is 12.2 hours.

5.3 Power Availability

Comparisons of solar array current availability and average load current requirements are illustrated in Figure 4 for the various mission phases. The average load requirements include:

a) All essential spacecraft electrical loads

b) All losses within the power control and conditioning equipment

c) Battery recharging at maximum 2.5 ampere rate following periods of battery discharge

d) 4 amps maximum at 50 VDC to the flight capsule during cruise only

e) 0.8 amps at 50 VDC for reaction control gas heaters. This load is not essential and is energized only when excess power is available.

Battery discharge currents during launch, maneuvers, and eclipses are shown by the shaded portions of Figure 4.

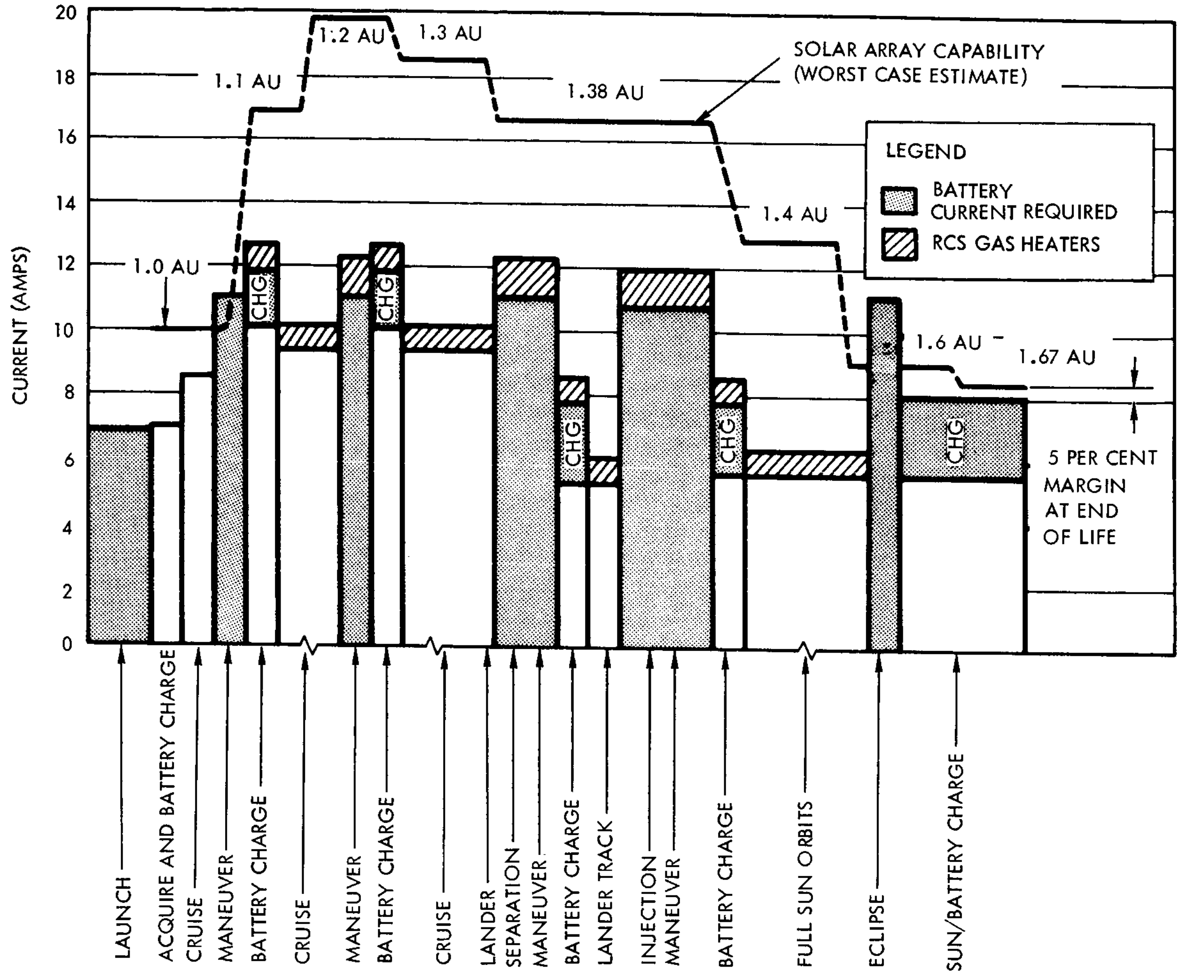


Figure 4. Solar Array Current Availability and Average Load Current Requirements

The solar array currents available at various sun-spacecraft distances represent the minimum array output at 50 VDC expected for the 1971 Voyager mission. A power margin of 5 per cent is estimated at the end of 6 months in orbit at Mars, exclusive of the nonessential reaction control gas heating load. The power margin at encounter is 100 per cent.

Solar array degradation factors used in estimating the minimum array output are discussed in Volume 5, Section IV. Electron and proton degradation of solar cells in Mars orbit is conservatively estimated assuming that the radiation at Mars is equal to that near earth, with peaks at 5000 and 16,000 km for the inner and outer belts, respectively.

5.4 Operating Life

The power subsystem is designed to supply the required outputs for a minimum period of 14 months including 6 months of operation in a Mars orbit.

5.5 Reliability

Reliability data for the electrical subsystem is given in VS-3-120.

6. PHYSICAL CHARACTERISTICS

Pertinent physical characteristics for each component of the power subsystem are presented below.

6.1 Solar Array

The solar array consists of six identical panels. Each panel is divided into two electrical sections. The 12 sections are connected in parallel to the main 50 VDC bus. Each section contains two strings of 116 series-connected 10-cell modules and one string of 116 series-connected 8-cell modules. These three strings are connected in parallel through blocking diodes. The solar cells comprising each module are connected in parallel. The resultant solar cell configuration for each section is 116 series by 28 parallel cells. The cells are of the 2 x 2 cm N-on-P type with nominal 10 ohm-cm base resistivity. A 0.006-inch-thick cover slide of Corning fused silica 7940 is provided on each cell.

6.2 Batteries

Each battery contains 30 series-connected, 25 ampere-hr silver-cadmium cells, cell voltage sensors and a battery temperature sensor.

6.3 Packaging

The electrical power equipment consisting of the PCU, SEA batteries, battery regulators and power inverters, is packaged and installed as described in VS-4-550.

6.4 Weights and Temperature Limits

Weights data and temperature limits for the electrical power equipment are given in VS-3-111.

ELECTRICAL DISTRIBUTION

VS-4-470

CONTENTS

1. SCOPE

This document provides the functional description of the system which implements the electrical integration of the Voyager electrical assemblies and subsystems and defines the specific equipments to be supplied.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

<u>TRW 1971 Voyager Spacecraft Design Documents</u>

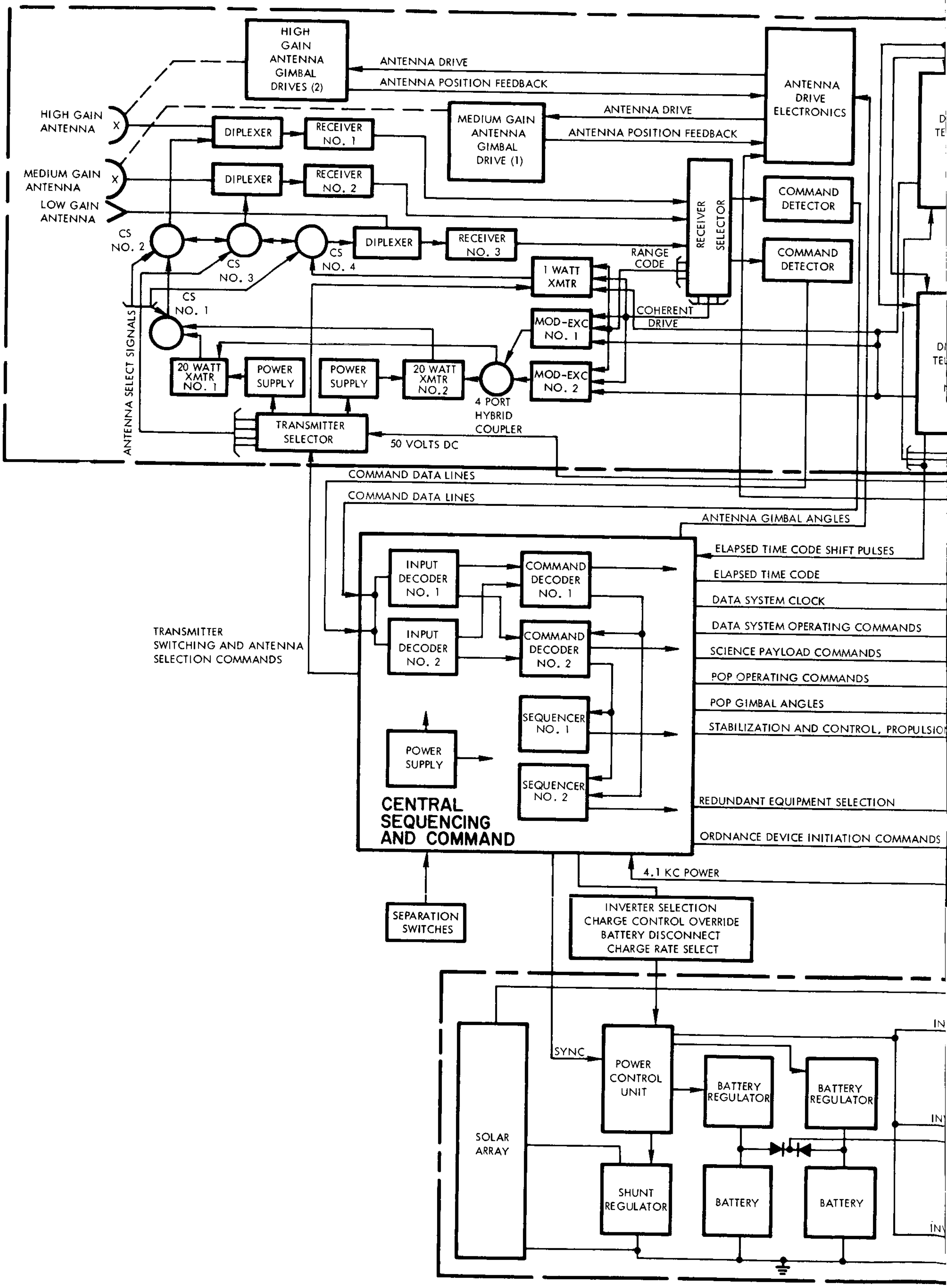
VS-2-110	Design Characteristics and Constraints
VS-3-111	Components Design Parameters
VS-3-112	Weights and Mass Properties
VS-3-130	Spacecraft-Launch Vehicle Interface

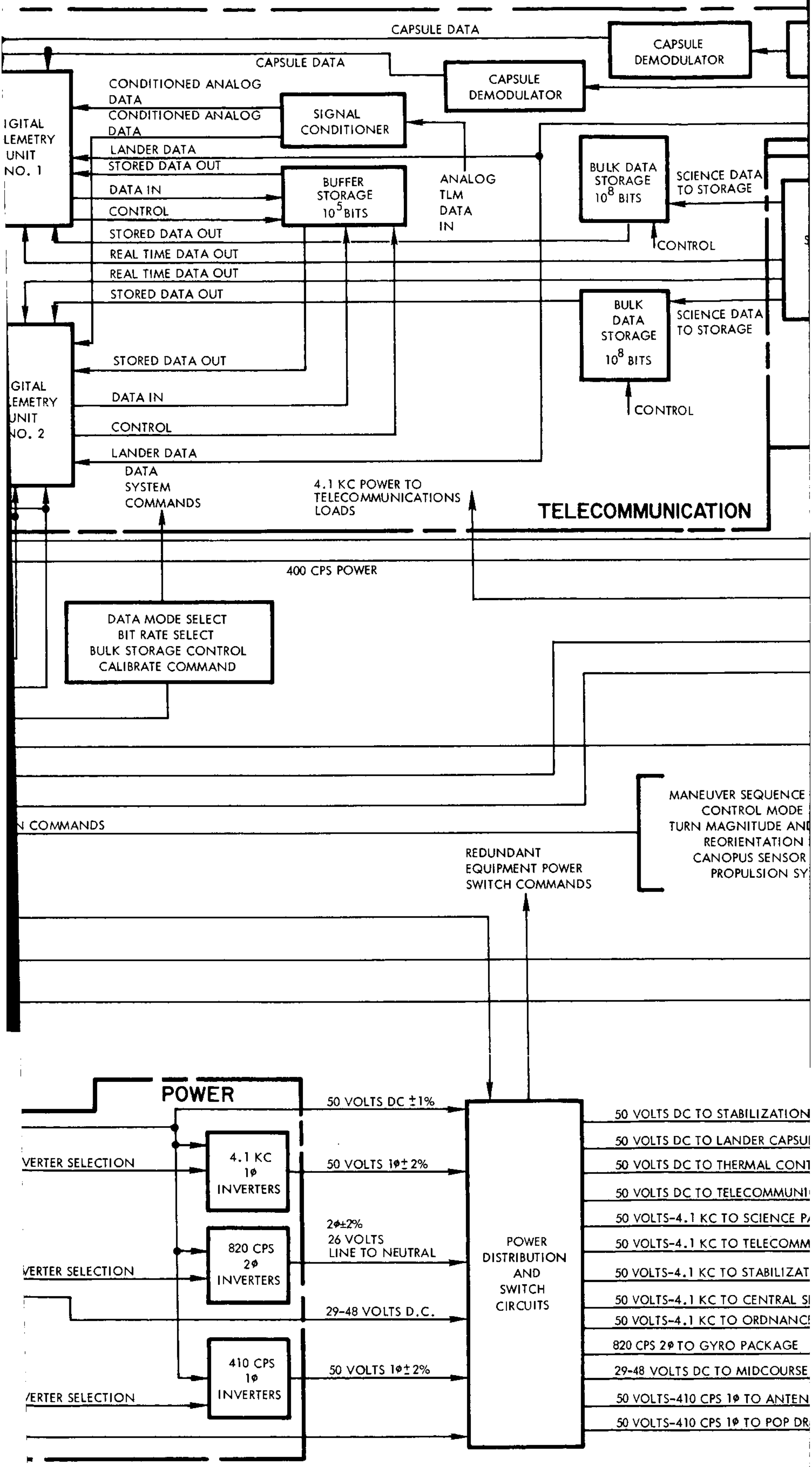
3. FUNCTIONAL DESCRIPTION

The electrical distribution equipment consists of the following components: (1) all interconnect cabling throughout the spacecraft, (2) junction boxes for installation of producible cabling configurations and for multiple distribution of electrical functions, (3) power switching circuitry, (4) circuit protective devices, and (5) circuitry for the control and initiation of pyrotechnic devices. This equipment is utilized for integration of the electrical subsystems into the over-all spacecraft. In addition, the equipment integrates the science payload, the flight capsule, and the launch vehicle electrically with the spacecraft and implements the system level test points. A spacecraft system schematic diagram illustrating the power and signal flow to be implemented by the electrical distribution subsystem is shown in Figure 1.

3.1 <u>Electrical Interconnect Cabling and Junction Boxes</u>

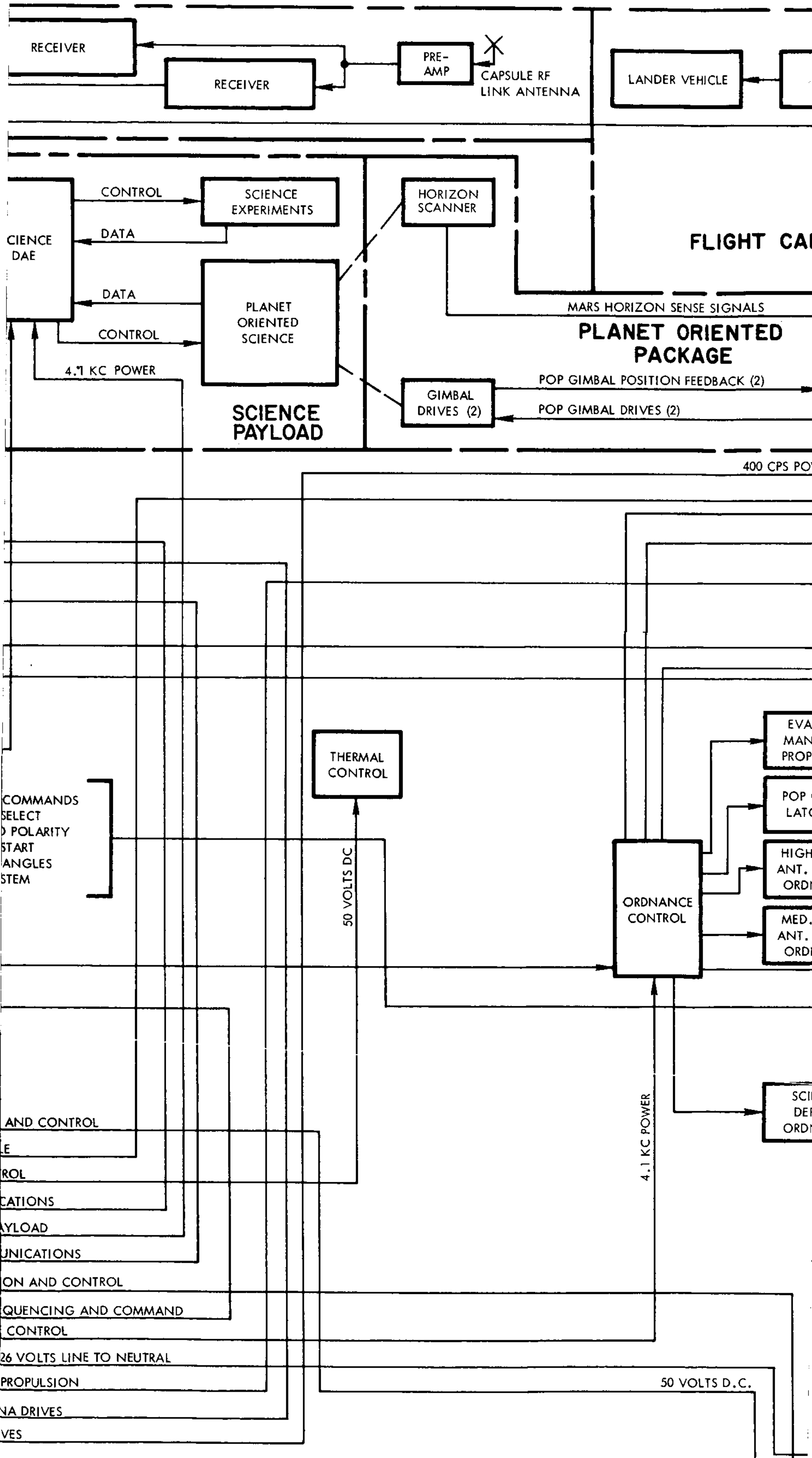
In accordance with the spacecraft packaging concept, the majority of the electronic equipment is mounted on structural panels which form the six facets of the spacecraft. Each panel is hinged at the bottom to allow access to and mating of connectors which interface with the panel assembly.


HIGH GAIN ANTENNA GIMBAL DRIVES (2)
ANTENNA DRIVE
ANTENNA POSITION FEEDBACK
ANTENNA DRIVE ELECTRONICS
HIGH GAIN ANTENNA
MEDIUM GAIN ANTENNA
LOW GAIN ANTENNA
DIPLEXER
RECEIVER NO. 1
MEDIUM GAIN ANTENNA GIMBAL DRIVE (1)
ANTENNA DRIVE
ANTENNA POSITION FEEDBACK
DIPLEXER
RECEIVER NO. 2
RECEIVER SELECTOR
COMMAND DETECTOR
CS NO. 2
CS NO. 3
CS NO. 4
DIPLEXER
RECEIVER NO. 3
COMMAND DETECTOR
RANGE CODE
1 WATT XMTR
CS NO. 1
COHERENT DRIVE
MOD-EXC NO. 1
ANTENNA SELECT SIGNALS
20 WATT XMTR NO. 1
POWER SUPPLY
POWER SUPPLY
20 WATT XMTR NO. 2
4 PORT HYBRID COUPLER
MOD-EXC NO. 2
TRANSMITTER SELECTOR
50 VOLTS DC
COMMAND DATA LINES
COMMAND DATA LINES
ANTENNA GIMBAL ANGLES
ELAPSED TIME CODE SHIFT PULSES
INPUT DECODER NO. 1
COMMAND DECODER NO. 1
ELAPSED TIME CODE
DATA SYSTEM CLOCK
TRANSMITTER SWITCHING AND ANTENNA SELECTION COMMANDS
INPUT DECODER NO. 2
COMMAND DECODER NO. 2
DATA SYSTEM OPERATING COMMANDS
SCIENCE PAYLOAD COMMANDS
POP OPERATING COMMANDS
POP GIMBAL ANGLES
SEQUENCER NO. 1
STABILIZATION AND CONTROL, PROPULSIO
POWER SUPPLY
SEQUENCER NO. 2
CENTRAL SEQUENCING AND COMMAND
REDUNDANT EQUIPMENT SELECTION
ORDNANCE DEVICE INITIATION COMMANDS
4.1 KC POWER
SEPARATION SWITCHES
INVERTER SELECTION
CHARGE CONTROL OVERRIDE
BATTERY DISCONNECT
CHARGE RATE SELECT
SYNC
POWER CONTROL UNIT
BATTERY REGULATOR
BATTERY REGULATOR
SOLAR ARRAY
SHUNT REGULATOR
BATTERY
BATTERY
D



CAPSULE DATA
CAPSULE DEMODULATOR
CAPSULE DATA
CAPSULE DEMODULATOR
CONDITIONED ANALOG DATA
CONDITIONED ANALOG DATA
SIGNAL CONDITIONER
IGITAL LEMETRY UNIT NO. 1
LANDER DATA
STORED DATA OUT
DATA IN
CONTROL
BUFFER STORAGE 10^5 BITS
ANALOG TLM DATA IN
BULK DATA STORAGE 10^8 BITS
SCIENCE DATA TO STORAGE
CONTROL
STORED DATA OUT
REAL TIME DATA OUT
REAL TIME DATA OUT
STORED DATA OUT
BULK DATA STORAGE 10^8 BITS
SCIENCE DATA TO STORAGE
STORED DATA OUT
GITAL LEMETRY UNIT NO. 2
DATA IN
CONTROL
CONTROL
LANDER DATA
DATA SYSTEM COMMANDS
4.1 KC POWER TO TELECOMMUNICATIONS LOADS
TELECOMMUNICATION
400 CPS POWER
DATA MODE SELECT
BIT RATE SELECT
BULK STORAGE CONTROL
CALIBRATE COMMAND
N COMMANDS
MANEUVER SEQUENCE
CONTROL MODE
TURN MAGNITUDE AN
REORIENTATION
CANOPUS SENSOR
PROPULSION SY
REDUNDANT EQUIPMENT POWER SWITCH COMMANDS
POWER
50 VOLTS DC ±1%
VERTER SELECTION
4.1 KC 1φ INVERTERS
50 VOLTS 1φ ± 2%
2φ±2%
26 VOLTS
LINE TO NEUTRAL
820 CPS 2φ INVERTERS
VERTER SELECTION
29-48 VOLTS D.C.
410 CPS 1φ INVERTERS
50 VOLTS 1φ ± 2%
VERTER SELECTION
POWER DISTRIBUTION AND SWITCH CIRCUITS
50 VOLTS DC TO STABILIZATION
50 VOLTS DC TO LANDER CAPSU
50 VOLTS DC TO THERMAL CONT
50 VOLTS DC TO TELECOMMUNI
50 VOLTS-4.1 KC TO SCIENCE P
50 VOLTS-4.1 KC TO TELECOMM
50 VOLTS-4.1 KC TO STABILIZAT
50 VOLTS-4.1 KC TO CENTRAL S
50 VOLTS-4.1 KC TO ORDNANC
820 CPS 2φ TO GYRO PACKAGE
29-48 VOLTS DC TO MIDCOURSE
50 VOLTS-410 CPS 1φ TO ANTEN
50 VOLTS-410 CPS 1φ TO POP DR

2



RECEIVER
RECEIVER
PRE-AMP
CAPSULE RF LINK ANTENNA
LANDER VEHICLE
CONTROL
SCIENCE EXPERIMENTS
DATA
CIENCE DAE
DATA
PLANET ORIENTED SCIENCE
CONTROL
4.1 KC POWER
SCIENCE PAYLOAD
HORIZON SCANNER
FLIGHT CA
MARS HORIZON SENSE SIGNALS
PLANET ORIENTED PACKAGE
GIMBAL DRIVES (2)
POP GIMBAL POSITION FEEDBACK (2)
POP GIMBAL DRIVES (2)
400 CPS PO
THERMAL CONTROL
COMMANDS
SELECT
POLARITY
START
ANGLES
STEM
50 VOLTS DC
ORDNANCE CONTROL
EVA
MAN
PROP
POP
LAT
HIGH
ANT.
ORD
MED.
ANT.
ORD
SCI
DE
ORD
4.1 KC POWER
AND CONTROL
LE
ROL
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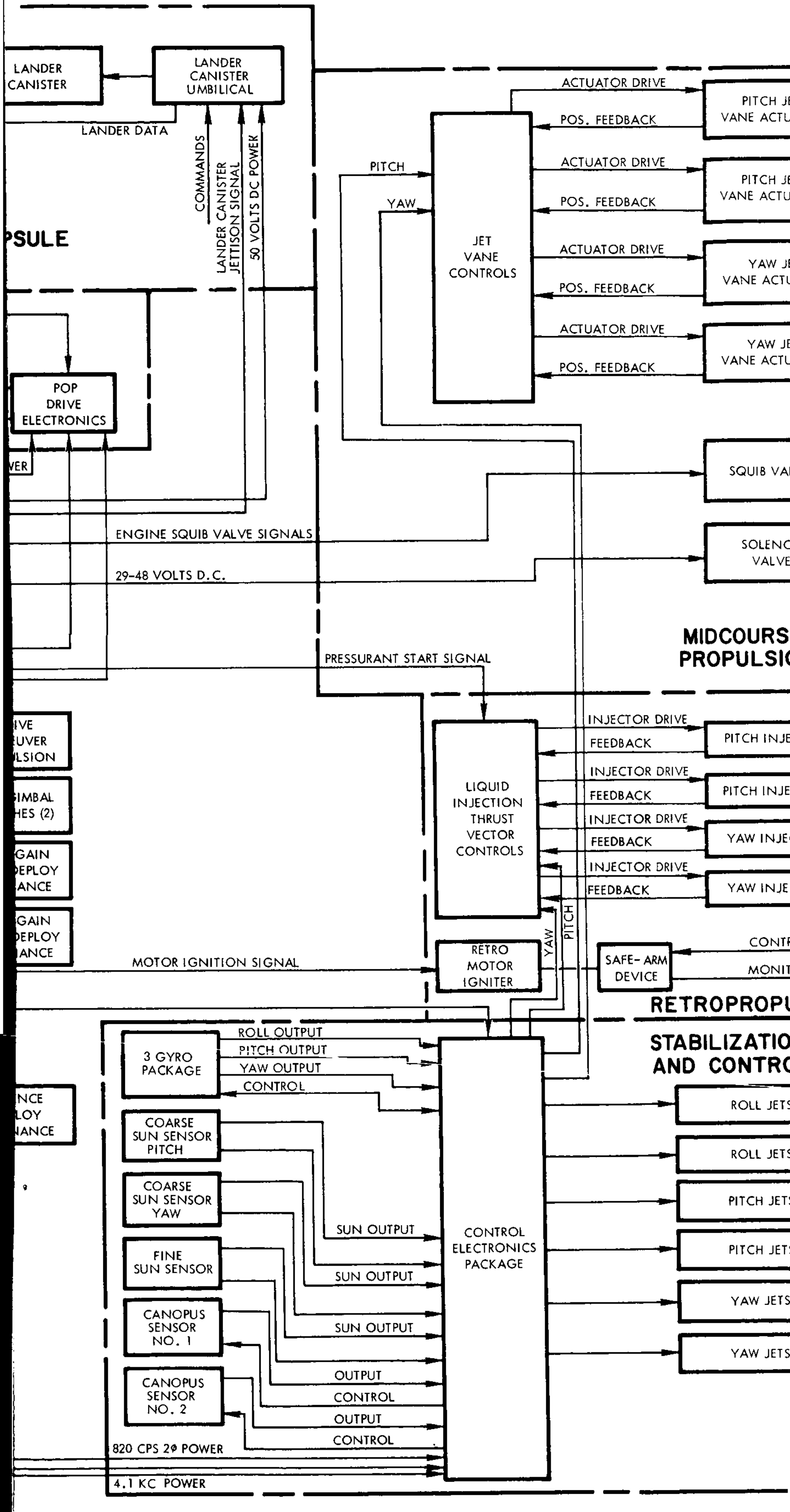
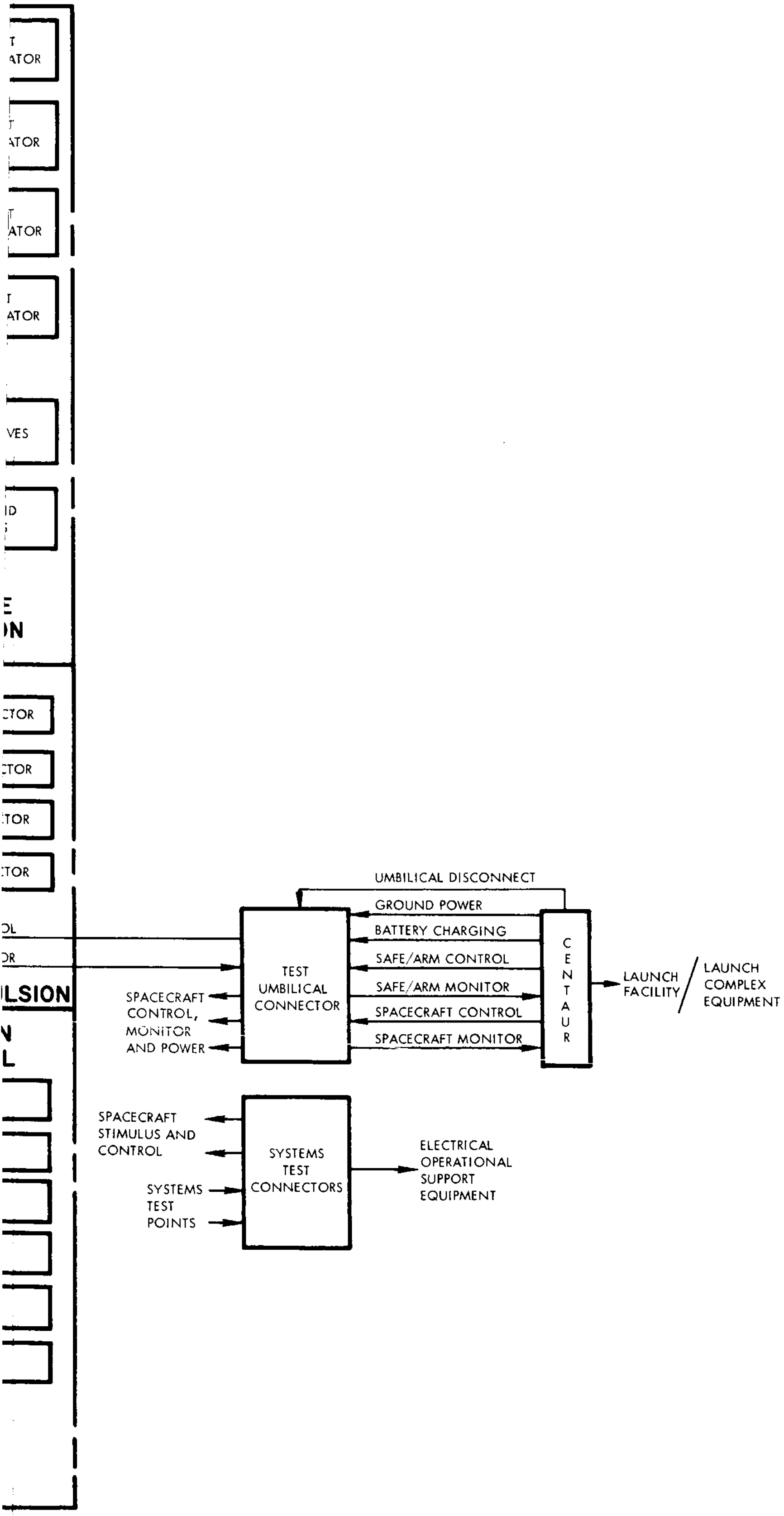
AYLOAD
UNICATIONS
ON AND CONTROL
QUENCING AND COMMAND
CONTROL
26 VOLTS LINE TO NEUTRAL
PROPULSION
50 VOLTS D.C.
NA DRIVES
VES

3


LANDER CANISTER
LANDER CANISTER UMBILICAL
LANDER DATA
COMMANDS
LANDER CANISTER JETTISON SIGNAL
50 VOLTS DC POWER
PSULE
POP DRIVE ELECTRONICS
WER
ENGINE SQUIB VALVE SIGNALS
29-48 VOLTS D.C.
PRESSURANT START SIGNAL
ACTUATOR DRIVE
POS. FEEDBACK
PITCH
YAW
JET VANE CONTROLS
PITCH JE VANE ACTU
YAW JE VANE ACTU
SQUIB VA
SOLENO VALVE
MIDCOURS PROPULSIO
INJECTOR DRIVE
FEEDBACK
LIQUID INJECTION THRUST VECTOR CONTROLS
PITCH INJE
YAW INJE
RETRO MOTOR IGNITER
SAFE-ARM DEVICE
CONTR
MONIT
RETROPROPU
MOTOR IGNITION SIGNAL
IVE EUVER LSION
IMBAL HES (2)
GAIN DEPLOY ANCE
NCE LOY NANCE
STABILIZATIO AND CONTRO
3 GYRO PACKAGE
ROLL OUTPUT
PITCH OUTPUT
YAW OUTPUT
CONTROL
COARSE SUN SENSOR PITCH
COARSE SUN SENSOR YAW
FINE SUN SENSOR
CANOPUS SENSOR NO. 1
CANOPUS SENSOR NO. 2
SUN OUTPUT
OUTPUT
820 CPS 2Φ POWER
4.1 KC POWER
CONTROL ELECTRONICS PACKAGE
ROLL JETS
PITCH JETS
YAW JETS

Fi

4



ure 1. Spacecraft System Schematic Diagram

Each panel contains inter-assembly harnesses to interconnect the assemblies on the panel and an associated junction box mounted at the bottom of the panel. The inter-panel electrical connections interface with the lower ring harness which runs in a cable trough mounted below the hinge line for the panels. Nonpanel-mounted equipment interface with the ring harness directly or through subsidiary harnesses.

The individual cables are fabricated from nonmagnetic materials with limited outgassing qualities. In order to minimize cable weight, the cables are not fabricated with a molded jacket.

3.2 Electrical Power Distribution

Some functions of the Voyager spacecraft require switching electric power loads in a manner that is beyond the power handling capability of the central sequencing and command subsystem. These normally consist of the switching on and off of electrical equipment which do not operate continuously or the switching selection of redundant assemblies. The electrical distribution equipment provides this switching upon receipt of a switching command from the initiating element.

3.3 Electromagnetic Compatibility

The distribution of electrical signals and power throughout the spacecraft must be implemented properly to assure electromagnetic compatibility among all the spacecraft operating systems.

Grounding, shielding, and bonding are given below.

3.3.1 Primary DC Power

Primary DC power connects the primary battery or solar array power to the DC-AC inverters. It is grounded to the spacecraft structure at or adjacent to the source of DC power and the primary DC power returns are carried to this point on separate conductors.

3.3.2 Primary AC Power

Primary AC power is the output from the inverters which is distributed within the spacecraft to the various using equipments. The AC power neutrals are grounded to the vehicle within or immediately adjacent to each of the inverters and the power returns are carried back to the inverter(s) in a twisted pair containing the return and the individual power distribution line. The AC loads at the using end are treated as balanced circuits. DC and AC power returns within each inverter are electrically isolated.

3.3.3 Secondary DC Power

Secondary DC power is the DC output of power supplies, operated from the primary AC power buses, which supplies the individual assemblies with the required DC power. These power supplies may be individual assemblies or integrally packaged within the using electrical assembly. The using assemblies, which place loads on each power supply, avoid the use of common power returns and do not combine signal and power returns. Coupling via return lines is precluded by grounding the DC power returns to the chassis in each using assembly.

3.3.4 Shielding

Electromagnetic shielding is employed on all cables connecting circuitry which is capable of generating or is susceptible to electromagnetic interference. All primary AC power is distributed by twisted, shielded pairs. Radio frequency and video signals, transmitting and receiving, are distributed by solid-jacketed or double-shielded cables. Pulse, digital, and state signals employ signal-shielded cables. Analog signals are shielded as the individual circuit requirement dictates.

All coaxial cables, shielded multiple conductor cables, and individually shielded wires employ grounding of the shield to the unit or chassis at each connector point.

3.3.5 Ordnance Circuits

All electroexplosive devices are fully shielded and electrically bonded to the basic spacecraft structure. Connectors are used to provide electrical grounding of the ordnance circuit cable around the entire periphery of the connector shell and to ensure electrical bonding of the connector shell and the device. Wiring to each individual ordnance device is made up of twisted-shielded pairs with the shields grounded at each end. Circuitry for actuating the ordnance devices is adequately shielded and decoupled to preclude response to electrical transients and radio frequency energy.

Initiation circuitry is designed to protect the spacecraft systems, and in particular the power subsystem, from permanently shorted ordnance devices.

3.4 Systems Test Points

System level test points are to be defined for testing the integrated spacecraft and will include subsidiary test points to support assembly and checkout operations for the spacecraft.

3.4.1 Test Connections

Within the design constraints established for the spacecraft, including the electromagnetic compatibility and test circuit isolation requirements, the system test points are to be integrated into hard-mounted connectors on the spacecraft structure or on the individual assembly mounting panel. These connectors define the hardline electrical interface with the electrical OSE.

In addition, an umbilical test connector, carrying the launch-stand power, signal, control, and monitor circuits, is to be implemented. This defines the hardline interface between the spacecraft and the launch complex equipment via the Centaur and the launch facility equipment.

3.4.2 Test Circuit Isolation

Individual circuit and equipment characteristics will be considered in meeting the requirements for the test circuit isolation given in 5.3.4 of VS-2-110. Isolation of sensing test points will be accomplished by isolation resistors. Isolation diodes or amplifiers will be used in stimulus lines, such as for battery charging, when necessary. The electrical distribution subsystem will provide circuit protection to prevent marginal operation or failure of elements within the power subsystem in the event of equipment failures causing electrical overloads; that is, whenever the power-using assembly is not self-protecting. Circuit protective devices such as those in current applications on the OGO, Vela, and Pioneer spacecraft will be applied to the Voyager spacecraft.

4. INTERFACE DEFINITION

4.1 Spacecraft-Science Payload Interface

It is a function of the electrical distribution subsystem to implement the electrical interfaces between the spacecraft and the science payload and, in addition, to implement the inter-assembly interfaces within the science payload. Both of these areas impose bilateral design constraints between the spacecraft and the science payload.

A major factor in the design implementation of these interfaces concerns not only the 1971 Voyager but the degree of variability of the science payload configuration as the 1971 design proceeds, as well as the bounding of requirements for future missions. The design is to attain the flexibility to react to total payload variability and changes within a specified payload with minimum changes in the spacecraft hardware.

It is assumed, for preliminary design purposes, that a considerable portion of this capability will be contained within the science DAE and an additional portion will be contained within a junction box mounted on the equipment panel assigned to the science assemblies.

The inter-assembly interfaces, and resulting interconnect cabling, is dependent on the specific payload configuration. Some flexibility is obtained by the addition of spare wires within the individual cables. Certain cable runs between an outboard appendage and the experiment panel and the flexible wiring configuration which carries the science information across the planet oriented package gimbal rotary joints may be standardized.

4.2 Spacecraft-Flight Capsule Interface

The spacecraft-flight capsule interface consists of application of spacecraft DC power to the flight capsule, transmission of hardline commands from the spacecraft, and receipt of hardline data from the flight capsule.

A separable umbilical connector is required to provide for jettison of the capsule adapter and canister. Only the hardmounted connector remains with the spacecraft; all other portions are jettisoned.

Design constraints stated in 4.1 above are applicable to the control of the spacecraft-flight capsule interface in order to maintain over-all spacecraft electromagnetic compatibility.

4.3 Spacecraft-Launch Vehicle Interface

The spacecraft-launch vehicle electrical interface corresponds to a single umbilical and the connectors to the separation devices as described in 3.5 of VS-3-130. No in-flight disconnect is required.

4.3.1 Umbilical Connector

The umbilical connector, which is separated late in the launch terminal countdown sequence, carries the ground power and battery charging functions as well as the control and monitor functions required for prelaunch on-stand spacecraft test operations. These include control and monitoring of the retropropulsion motor safe and arm device.

A device such as the Lockheed explosively actuated umbilical connector which is used on the Agena vehicle and the OGO spacecraft will be used from this application.

Implementation of the spacecraft-launch vehicle interface in this manner accomplishes separation of the interface prior to launch vehicle liftoff (and verifies the separation). Although it is an explosively actuated device, this technique has already been classified for the OGO program as a Class B ordnance application, i.e., one whose inadvertant actuation will not damage government property or endanger life.

The Class B application removes some of the safety constraints from the design of the launch complex equipment.

4.3.2 Separation Signal Cabling

The connectors to the separation devices are on the launch vehicle side of the interface and remain with the launch vehicle after separation.

5. PERFORMANCE

The electrical distribution subsystem provides electrical interconnections in a compatible manner for all spacecraft equipment.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Design Criteria

The electrical distribution subsystem is to be in accordance with VS-2-110 and the various subsystem design documents.

6.2 Junction Boxes

Four 6" x 6" x 6" junction boxes are utilized in the distribution system to be mounted at the bottom of each equipment panel.

6.3 Weight

Weight data is given in VS-3-111.

7. SAFETY CONSIDERATIONS

The major safety consideration for the electrical distribution subsystem is the proper implementation of ordnance devices in accordance with AFMTC Pamphlet No. 80-2, General Range Safety Plan.

THERMAL CONTROL

VS-4-510

CONTENTS

1. SCOPE

This document presents the design of the Voyager spacecraft bus thermal control subsystem, which provides continuous temperature control of spacecraft equipment as required throughout the life of the spacecraft.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

<u>TRW 1971 Voyager Spacecraft Design Documents</u>

VS-3-110	Layout and Configuration
VS-3-104	Flight Sequence
VS-3-111	Spacecraft Components Design Parameters
VS-4-410	Stabilization and Control
VS-4-460	Electrical Power
VS-4-550	Electronic Equipment Packaging

3. FUNCTIONAL DESCRIPTION

3.1 General

The thermal control subsystem limits the effects of the spacecraft thermal environments on spacecraft equipment to permit reliable operation of such equipment throughout the mission of the spacecraft. This is accomplished through on-board equipment, thermal control coatings, thermal insulation, structural heat paths, and thermal control louvers.

3.1.1 Thermal Control Coverage

The thermal control subsystem provides an adequate thermal environment for all on-board equipment by controlling base plate or external temperatures, but does not control equipment internal temperature distributions. It does not control temperature for the flight capsule nor for the midcourse rocket engine or retropropulsion solid propellant motor during firings. Prelaunch thermal control is accomplished by the launch vehicle system.

3.1.2 Thermal Control Concept

A functional flow diagram for the basic thermal control subsystem is shown in Figure 1. This system utilizes an insulated compartment concept whereby the external spacecraft bus surfaces are covered with insulation except for radiating areas. Radiation from these areas is controlled by louvers which reject dissipated power to space as required to maintain the internal spacecraft temperature within desired limits. Various surface coatings and finishes are used to achieve proper heat transfer between spacecraft elements. Electrical heaters and thermostats are utilized for local control in externally mounted equipment.

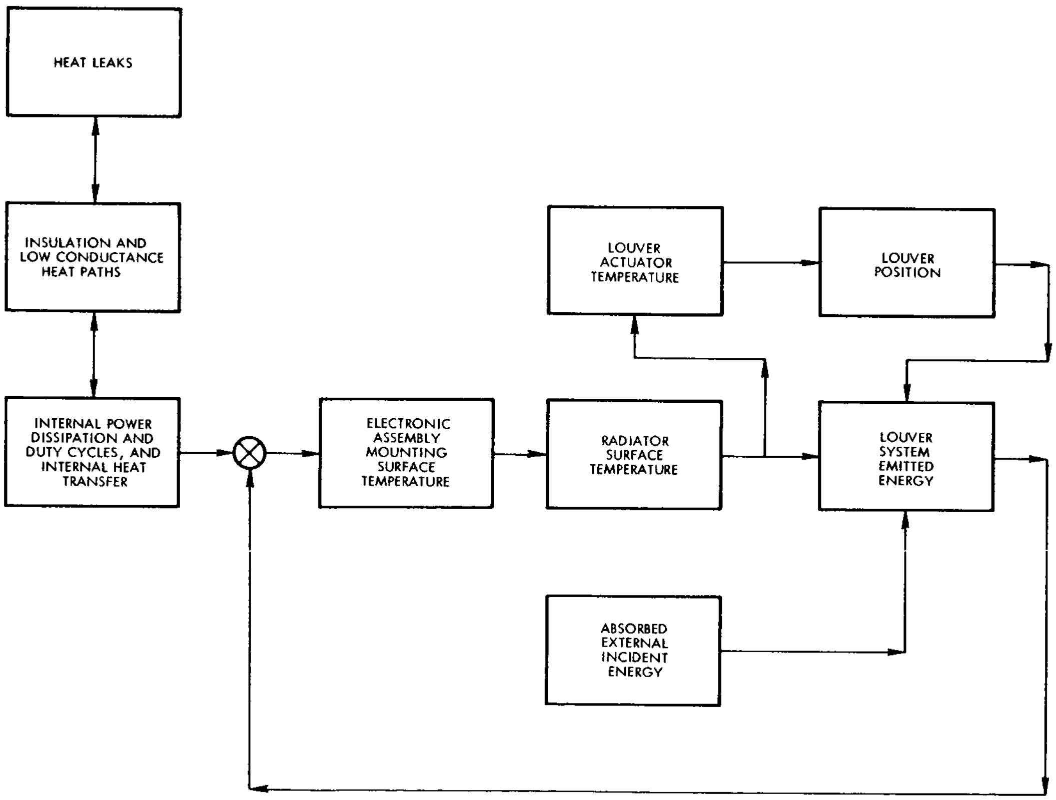


Figure 1. Thermal Control Subsystem Functional Block Diagram

3.2 Insulation

3.2.1 Spacecraft Bus Exterior

Multilayer insulation, consisting of 20 layers of 1/4-mil aluminized Mylar sandwhich between two layers of 3-mil Mylar, is utilized on the bus surface (which faces the sun during cruise), on the sides around the louver banks, and on the forward surface. This limits heat gain or loss to or from the bus so that the heat to be rejected is essentially that due to internal dissipation and remains within the control range of the louvers. The Mylar side of the aluminized Mylar outer layer faces space; the inner layer is clear Mylar

The forward bus surface is covered by high-temperature insulation in the form of 1/2-inch thick layer of Refrasil batt. This protects the spacecraft from thermal radiation and convection from the exhaust plume of the solid propellant motor.

3.2.2 Rocket Engines

The solid and midcourse rocket engines are covered with 1/2-inch thick blankets of alternating layers of aluminum foil and fiberglass paper to limit radiative heat transfer from the hot engines to spacecraft equipment and structure. The nozzle exit plane of the solid propellant motor is covered with a blanket of 20 layers of aluminized Mylar. This limits heat loss to achieve proper propellant grain temperature prior to firing.

3.2.3 Externally-Mounted Equipment

Blankets of 20-layer Mylar are also utilized on the planet-oriented package and external experiment packages, except for openings required for sensor viewing.

Gimbals are insulated with aluminized Mylar to minimize heat exchange with the sun, space, and the solar array, and are isolated from surrounding structure with low conductivity attachment fittings.

3.3 Louver System

There are four banks of louvers per spacecraft, one bank on each of the honeycomb side panels to which electronic equipment is mounted.

The louvers open so that heat from the solar array is reflected to space rather than into the spacecraft. The area of each louver bank is sized appropriately for the amount of power dissipation on that panel. Such a system of louver-covered radiating surfaces affords active temperature control to provide a suitable temperature environment for internally-mounted equipment during mission life.

In addition to accommodating relatively large, predictable, local and distributed changes in internal and external thermal environments during the mission, a louver system also has the capability for accommodating uncertainties in spacecraft thermal loads such as those occasioned by unpredictable degradation of surface properties, heat leaks, and failure mode power dissipation.

A number of permissible variations in louver system configuration are readily accomplished, e.g., choices can be made relative to specific louver blade length, louver rotation direction, actuator temperature sensitivity, etc.

3.3.1 Configuration

The louver system is comprised of the four principal elements described below.

a. Louver Assemblies

Each assembly contains a louver blade 2 inches wide and 15 inches in length. The blade is made from two pieces of thin aluminum suitably shaped to provide rigidity. A small diameter sintered-silver bearing pin is attached on the central longitudinal axis at each end of the louver blade through interposed insulation blocks and other fixtures. Lubrication for rotation about the longitudinal axis is provided by treating the bearing pin with a dry film lubricant (molybdenum disulphide).

b. Louver Support Brackets

Louver support brackets formed from small gage aluminum contain the louver bearings and are made long enough to span the width of the panel to which the louver is attached. The brackets have provisions which serve to secure one end of each bimetal actuator and to provide a stop to limit louver angle excursions (0 to 90 degrees). The span between brackets, when they are mounted to the panel surface, is about 2 inches greater than the length of the louver blade to permit space for the actuator shields.

c. Bimetal Actuators

Actuators for the louver blades employ alloy S992 and are of the type used on the Pioneer spacecraft (available from W. M. Chase Spring Co. of Detroit).

Each actuator is a spiral coil, the inner end of which is secured to the louver axis by a device which also permits setting the louver angle. The other end is formed to extend down to the base of the support bracket where it is held fixed. In a bank of louvers, one actuator per blade is utilized, with these placed at alternate ends for neighboring louvers.

d. Actuator Shields

To permit the actuator to be strongly responsive to the local temperature of the radiator surface, a radiation shield is provided. It is comprised of multilayer aluminized Mylar, formed in a hat section, which covers the actuators and the entire length of the support brackets. When installed, the only openings present are those required to permit free rotation of the louver shaft.

3.3.2 Thermal Characteristics

a. Louver Angle-Temperature Setting

On the radiator panel surfaces the louvers are set to close at a nominal local surface temperature of $40^{\circ}F$, and to be fully open at $85^{\circ}F$. Development tests conducted under space simulation conditions for variously distributed heat sources (and magnitudes) show that the temperature of actuators is approximately $5^{\circ}F$ lower than local temperatures of the radiator surface. Therefore, in setting the louver angle, a $5^{\circ}F$ bias is employed. That is, when installed under isothermal conditions the louvers will close at $35^{\circ}F$ and will be fully open at $80^{\circ}F$.

b. Effective Emissivity

The effective emissivity for the louver system is presented in Figure 2. The emissive power of the louver system is the product of the effective emissivity of the louver system and the black body emissive power of the radiator.

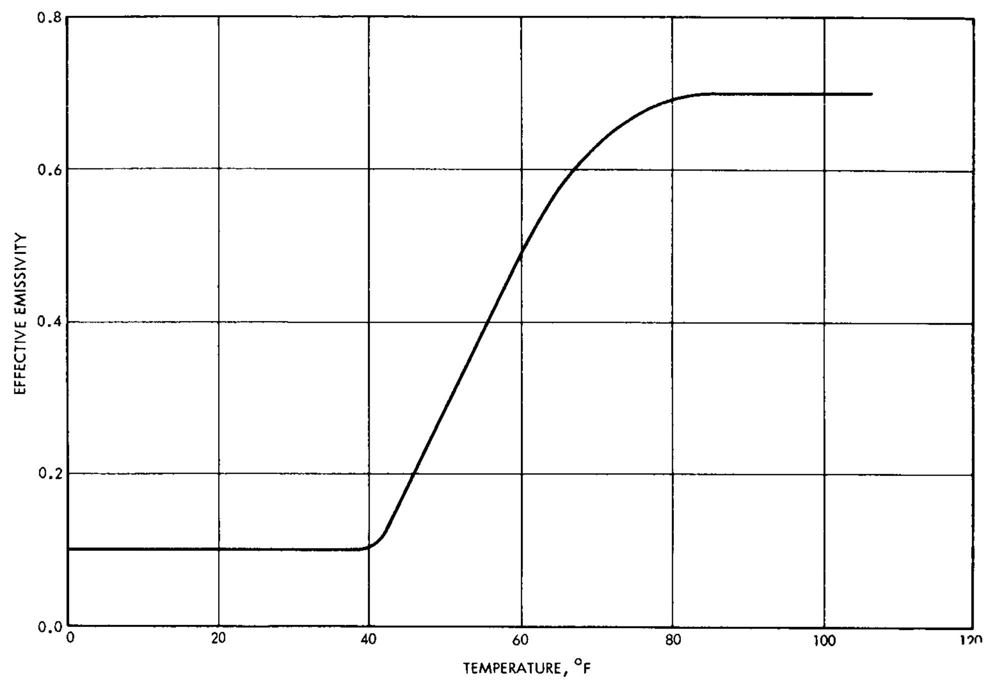


Figure 2. Effective Emissivity of Louver-Covered Radiator

3.4 Coatings and Finishes

The surface area directly beneath the louvers and the back of the solar array is coated with potassium-zirconium silicate paint (TRW process MT-6-2). This provides thermal radiation properties of α = 0.2 (after 100 hours of normal solar exposure, initially 0.17), and ϵ = 0.85. The high emissivity on the back of the solar array provides cool cell temperatures for electrical efficiency and the low absorptivity protects the array from the radiant heating of the solid particle plume.

The internal surfaces of the main compartment are coated with high emissivity materials such as black Cat-a-lac, $\epsilon = 0.86$, or anodize as appropriate to enhance radiative heat transfer between large areas of the spacecraft. Faying surfaces of the component-mounting surface interfaces are masked to remain bare metal for good thermal and electrical contact.

3.5 Heaters and Thermostats

Electrical strip heaters and thermostats are utilized for thermal control of equipment external to the main compartment where power dissipation is inadequate or widely varying. They are used on the planet-oriented package (POP), the external experiment packages, the POP gimbal and the antenna gimbals. Bifilar wound heaters are used which possess low magnetic characteristics. With DC power applied the magnetic field does not exceed 2 gamma at 2 inches.

3.6 External Equipment Temperature Control

Externally-mounted equipment requires a thermal design which is independent of the main compartment. These thermal designs are based upon minimizing thermal coupling between the equipment and their support structure, as well as minimizing the effects of the widely varying external radiant environment. Thermal isolation is accomplished through the use of low conductance mounts (stacked washers, phenolic blocks, etc.), multilayer radiation insulation, and where possible minimizing radiator view angles of the spacecraft and the sun. Thermostatically controlled heaters are used to prevent excessively low temperatures.

3.6.1 Planet-Oriented Package

The POP houses a variety of experiments, television systems, ultraviolet and infrared spectrometers, radiometers and Mars scanning radiometers. Of these experiments, the TV optical systems place the greatest demands upon thermal control design. The TV optical system requires a nearly uniform temperature environment in order to maintain optical resolution and pointing accuracy. A Mars scanning radiometer also requires a fairly uniform temperature environment since it has an image forming optical system. The remaining experiments have temperature level but not temperature gradient requirements.

The basic concepts employed in the thermal design are:

a) Insulate the entire package with multilayer aluminized Mylar insulation, except for viewing ports.

b) Provide low conductance mounts.

c) Use local electrical heaters and thermostats to minimize temperature gradients in the region of the fine resolution TV camera lens systems and for the Mars scanning radiometer.

d) Use thermal shutters, if required, over the TV lens aperture to reduce heat leaks. The shutter, an insulated door, would be open during picture taking modes, estimated to be less than 15 minutes per orbit. The rotating light filter for the TV optical system could be used to reduce heat leaks, instead of a separate shutter, by having a portion of the filter opaque to thermal radiation. The sequencer could be programmed to rotate the opaque portion of the filter over the lens aperture after every picture taking sequence.

e) If required, thermal doors will be placed over experiment apertures, similar to those proposed for the TV system, to reduce heat leaks.

The POP is shaded from the sun by the solar array, except for the low resolution infrared and ultraviolet spectrometers, which for Martian orbital operations may view the sun directly. During the transit phases of the mission, the package is positioned in such a manner that the sun does not view experiment apertures. The package has a total opening of 1 square foot, with power levels ranging from 0 to 16 watts. The package, because of the large number of openings and no appreciable view of the sun, does not require a radiator; hence it is completely insulated except for viewing ports.

3.6.2 External Experiment Packages

The external experiment packages house various science equipment such as meteoroid sensors and plasma detectors. These experiments are low-powered, on the order of 3 watts total power dissipation, and thus are completely insulated except for experiment apertures. Heater power is used to make up for heat leaks and heat losses out of experiment apertures. Where possible, windows which are opaque to thermal radiation but transparent to the frequencies of interest will be placed over the detector apertures. Bare metal windows which view the sun have sufficient conduction coupling to their housing to prevent excessively high temperatures. The packages are insulated from the 4- and 8-inch diameter plasma detector plates since the temperature limits of the plates are those of the metallic plate material.

3.6.3 Gimbals

Antenna and POP gimbal systems are maintained between 20 and 120°F. The gimbal systems are essentially electrically passive and must rely on heater power to maintain acceptable temperature limits. The gimbal mechanisms, motors and bearing, are insulated with low conductivity mounts to further reduce heat leaks, and cable leads are also insulated. The thermal capacity of the gimbal systems is used to absorb drive motor power dissipation during their brief periods of operation.

3.6.4 Antenna Temperature Control

The antenna surfaces facing the sun are painted black to eliminate the problem of focusing the sun's rays on the antenna feed. The aft surface is painted white to reduce solid rocket plume heating during the retrothrust phase of the mission. The antenna feed is isolated from the antenna by a phenolic insert and the leads are insulated to insure that the upper temperature limit of 200°F of the feed is not exceeded during rocket firing.

3.6.5 Magnetometer

The magnetometer, located on a boom, is insulated from the sun with multilayer insulation and has a radiator which faces outboard and does not view the sun except possibly during midcourse orientations and deboost maneuvers. The radiator is painted white to reduce solar heating during maneuvers and plume heating from the solid engine during deboost. Electric heaters are used to insure that the power rejected by the radiator is much greater than heat leaks. This is of concern for a low continuous power dissipating magnetometer where power dissipation is the same order of magnitude as heat leaks, since heat leaks are difficult to predict and cannot be controlled. Heater power requirements are such that heat leaks are less than 20 per cent of the total power to be dissipated by the radiator.

4. INTERFACE DEFINITION

The thermal subsystem imposes the following functional requirements.

4.1 Mechanical

4.1.1 Configuration

The spacecraft configuration provides 13.2 square feet of external surface area to serve as the main radiating areas. These areas are nominally shaded from the sun and have a wide view of space.

4.1.2 Structure

The structure provides a heat conduction path through the honeycomb panels having a conductance of 5 Btu/hr-ft^2-$^\circ$F from the power dissipating equipment to the external radiating areas. Lateral conductance is provided through the 25-mil face sheets of 7075-T6 aluminum having a conductivity of 103 Btu/hr-ft-$^\circ$F. A resistance of 4.2 hr-$^\circ$F/Btu between the solid motor flange and the thrust cone is provided through the use of a phenolic fiberglass gasket to protect the spacecraft from conduction soak after firing. A fiberglass angle fitting is provided for solar array attachment to give a resistance of 1.24 hr-$^\circ$F/Btu between the spacecraft and the array.

The structure provides insulation attachments and surface coatings as given in Section 5.

4.1.3 Propulsion

The propulsion system is designed to have the operating and standby temperatures given in VS-3-111.

4.2 Electrical

4.2.1 Electronic Packaging

The packaging design limits the heat dissipation per unit baseplate area to a nominal value of 0.2 and a maximum value of 0.3 watt per square inch. In areas where large design penalties are incurred to meet this requirement, such as the 20-watt transmitter, local power densities may exceed the nominal. In no case is the power density to exceed 1.4 watts per square inch, and this high density is to be associated with higher operating temperatures.

4.2.2 Electronic Assembly and Subassembly Design

Electronic assemblies and subassemblies are designed to have operating and storage temperatures given in VS-3-111.

4.2.3 Electrical Heater Power Distribution

The electrical heater power distribution for the thermal subsystem is depicted in Figure 3. Heater power is DC directly off the bus with heater resistance sized for the lowest expected voltage.

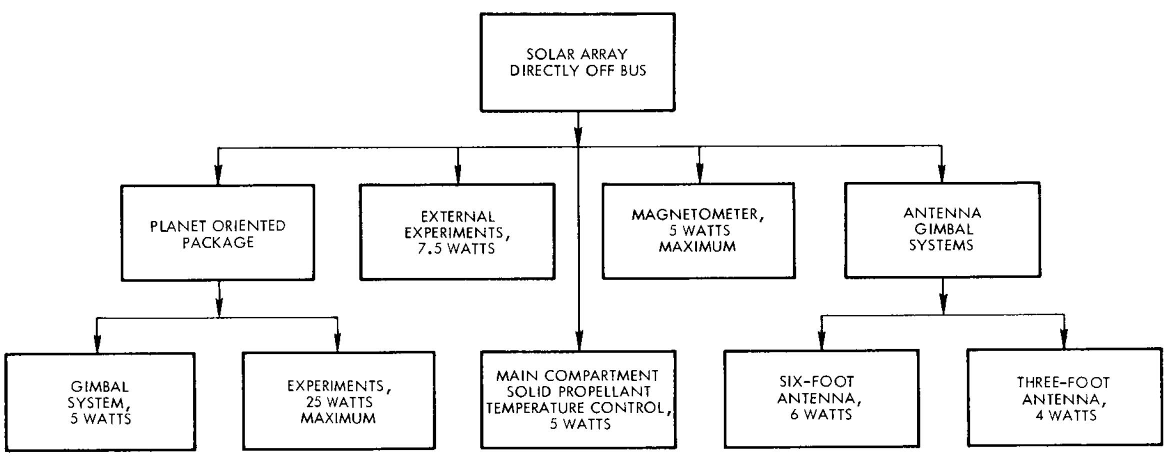


Figure 3. Thermal Subsystem Electrical Heater Power Distribution

4.2.4 Solar Array Design

The solar array is designed to have operating and eclipse temperatures given in Section 5.

4.3 Flight Capsule

The flight capsule design provides a nearly adiabatic interface with the spacecraft.

4.4 Launch Vehicle

The launch vehicle provides conditioned air at a specified temperature, relative humidity and flow rate prior to liftoff for cooling during on-stand operation. The launch vehicle limits nose fairing heating input to 40 watts/ft^2 and retains the fairing until peak aerodynamic heating rates are less than 24.2 watts/ft^2.

5. PERFORMANCE

5.1 Insulation

Thermal control design is based on heat flux from multilayer insulation as given below (see Volume 5, Appendix D):

$$\sigma \epsilon_x T_x^4 = \alpha_x G + \frac{\sigma \left(T_H^4 - T_x^4\right)}{\left(\frac{1}{\epsilon_1} + \frac{1}{\epsilon_2}\right) (N - 1)} + \frac{8.16 \times 10^{-5} (D^{1.186}) (T_H - T_x)}{(N - 1)}$$

where

σ = Stefan-Boltzmann constant (0.173×10^{-8} Btu/hr-ft^2-$^oR^4$)

ϵ_x = Exterior surface emissivity (variable)

T_x = Exterior surface temperature

α_x = Exterior surface solar absorptivity (variable)

G = Solar constant (433 Btu/hr ft^2 at 1 AU)

T_H = Interior temperature (variable)

ϵ_1 = 0.03 for NRC-2 insulation

ϵ_2 = 0.30 for NRC-2 insulation

N = Number of sheets of insulation (variable)

D = Number of sheets per inch (55)

For the 20 layers on the external surfaces the heat loss is 0.1 watt/ft^2 with an internal temperature of 40^{o}F and no solar irradiation. Other insulation conductivities are:

7×10^{-5} Btu/hr-ft^2-$^\circ$F/ft - Alternating layers of aluminum foil and fiberglass paper

0.04 Btu/hr-ft^2-$^\circ$F/ft - Refrasil batt

0.18 Btu/hr-ft^2-$^\circ$F/ft - Fiberglass

5.2 Louvers

The louvers are actuated from a fully-closed position through 90 angular degrees when the actuator temperature changes from 40 to 85°F. This changes the effective emissivity of the radiating area from 0.1 to 0.7 for specularly reflecting louver blades. The corresponding emitted energies are 3.1 and 30 watts/ft^2. The effective emissivity of the louver system and the absorptivity to solar array emitted radiation are shown in Figures 2 and 4 as a function of opening angle.

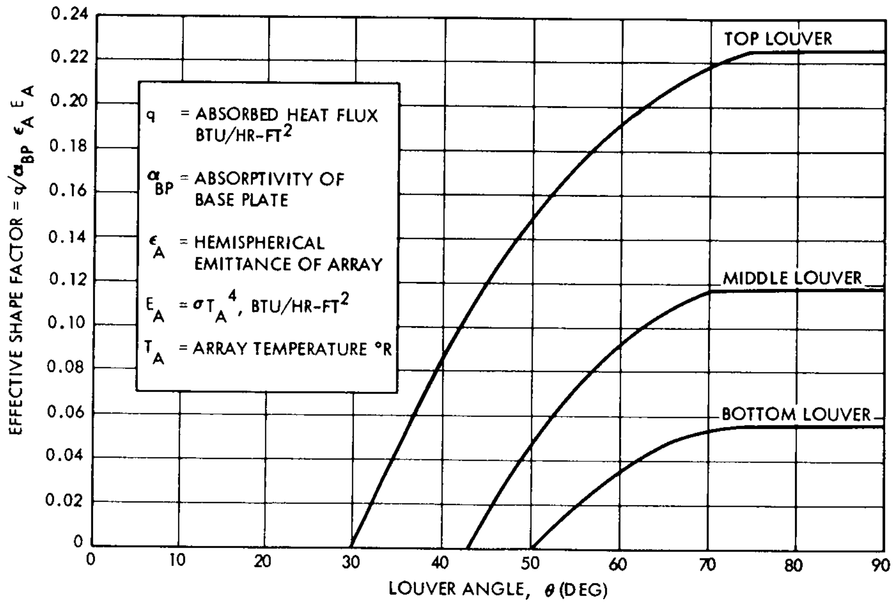


Figure 4. Heat Input to Louver System from Solar Array

5.3 Coatings and Finishes

Thermal radiation properties of coatings and finishes are:

	α	ϵ (hemispherical)
Potassium-zirconium silicate paint (MT-6-2)	0.2*	0.85
Black Cat-a-lac paint	0.95	0.86
Hard anodize on 6061 (0.9-mil thick)	0.87	0.8
Vacuum deposited aluminum on resin	0.1	0.03
Solar cells with quartz cover glass	0.74	0.8
3-mil aluminized Mylar (Mylar side out)	0.24	0.78
Refrasil batt	0.13	0.85

*After 100 hours of normal solar exposure, initially 0.17.

5.4 Heat Balance for Main Compartment

The main compartment heat balance for the near-earth cruise mode of power dissipation, the hot condition, determines the size of the louvered area required to limit the rise in compartment temperature. The Martian encounter condition with minimum power dissipation presents the cold case. For both cases margins are required to compensate for heat transfer that cannot be either analyzed or established fully during space simulation tests. The resulting parameters are as follows:

Louver-covered radiator area with 10 per cent margin	13.2 ft^2
Power dissipation margin over heat leaks with closed louvers for the cold condition.	60 watts

5.4.1 Hot Condition

a. Performance Data

The hot condition corresponds to near-earth cruise. Preliminary design data for this case is given below:

• Loads in the Main Compartment	Watts
Electrical power subsystem (100-watt shunt dissipation)	137.0
Communication subsystem (less 20 watts RF)	78.4
Stabilization and control	13.7
Command and sequencing	8.5
Science	55.5
Total power dissipated	293.1
• Heat Fluxes in Main Compartment	
Solar array attachment fittings	14.0
Solar array struts	-15.0
Reaction jet lines	1.0
Aft insulation	2.0
Forward insulation (heat flux assumed zero with capsule on)	0.0
Net heat flux (q net)	2.0

b. Heat Balance Calculation

The amount of louver area required may be determined from the over-all heat balance given below:

$$A_L E_L + (A_S - A_L)\, q_i = p + q_{net} + \alpha_L F_{L-SA} \epsilon_{SA} E_{SA} A_L$$

where A_L and A_S are the louvered and total areas of the trapezoidal sides; q_i is the heat loss from the insulation on the side areas (0.126 watt/ft^2, section 2.4*) for 20 layers of aluminized Mylar; E_L and E_{SA} are the emissive powers of a fully opened louver system (30 watts/ft^2 at 85°F) and the solar array (57 watts/ft^2); and the product $\alpha_L F_{L-SA} \epsilon_{SA}$ is the effective absorptivity of the louver system to emitted solar array radiation (Figure 4).

5.4.2 Cold Condition

The worst case steady-state cold conditions may be either prior to capsule separation when the power supply shunt dissipation is down to 25 watts, or in a Mars orbit with full sun at 1.67 AU without the capsule, but with power dissipations higher because of the orbital mode of operations.

Eclipses are not expected to present a worst case because of the increased internal dissipations. The battery charge and recharge cycle compensates for added heat loss through solar array attachments. Also in this transient condition the energy stored as sensible heat in the on-board equipment is available to maintain adequate temperatures compared to a steady state condition where only the dissipated power is available to compensate for heat leaks.

a. Performance Data for Encounter (Precapsule Separation)

- Loads in the Main Compartment

Loads in the Main Compartment	Watts
Electrical power subsystem (25 watt shunt dissipation)	62
Communications subsystem (less 20 watts RF)	76.8
Stabilization and control (cruise mode)	13.7
Command and sequencing (cruise mode)	8.5

*This heat loss is derived from vendor data and Douglas calorimeter tests showing a conductance across the blanket of 2.9 x 10^{-3} Btu/hr-ft^2-°F. Experiment package tests at TRW show a conductance of 5 x 10^{-3}. For conservatism the low conductance value is utilized in the hot case and the high conductance value in the cold cases.

Science (cruise mode)	55.5
Total power dissipated	216.0

- **Heat Fluxes in Main Compartment** — Watts

Heat loss through closed louvers (effective emissivity 0.1, panel temperature 40°F)	41.5
Heat loss through solar array attachment fittings	10
Heat loss through solar array struts	15
Heat loss through aft insulation -67 ft^2 (solar constant 57.2 watt/ft^2, conductivity degraded 25 per cent, nominally 0.076 watt/ft^2)	6.5
Heat loss through side panel insulation - 111.3 ft^2 (conductivity degraded 25 per cent, nominally 0.47 watt/ft^2)	65.5
Heat loss through Canopus sensor and earth detector openings (two 5 x 5 inch and one 4 x 4 inch openings)	14.3
Heat loss from reaction jet lines	4
Total heat loss	156.8
Margin for heat leak (above insulation uncertainty)	59.2

b. Performance Data for 1.67 AU (Capsule Off, Orbital Mode)

- **Loads in the Main Compartment** — Watts

Electrical power subsystem (shunt dissipation assumed zero)	65
Communication subsystem (less 20 watts RF, bulk storage on)	98.4
Stabilization and control (orbital mode)	19.7
Command and sequencing (orbital mode)	12.0
Science (some science loads turned off)	63.0
Total power dissipated	258.1

Heat Fluxes in Main Compartment	Watts
Heat loss through closed louvers	41.5
Heat loss through solar array attachment fittings	14
Heat loss through solar array struts	15
Heat loss through aft insulation - 67 ft^2 (solar constant 46.7 watts/ft^2, 25 per cent degradation)	8.6
Heat loss through side insulation (same as encounter)	65.5
Heat loss through forward insulation - 26 ft^2 (25 per cent degraded, nominally 0.47 watts/ft^2)	12.2
Heat loss through Canopus and earth sensors (same as encounter)	14.3
Heat loss through reaction jet lines	4
Heat loss through spacecraft lander interface fitting	3
Heat loss through expended rocket case	12
Total heat loss	190.1
Margin (above degraded insulation)	68

5.5 Equipment Panel Average Temperatures

5.5.1 Nominal Orientation

Using the spacecraft thermal control inboard profile shown in Figure 5 in conjunction with a simplified, lumped node, analytical thermal model of the spacecraft yields the following steady-state equipment temperatures for a nominal spacecraft orientation.

Average equipment panel temperatures, °F	1 AU "Hot Case"	1.38 AU "Cold Case"	1.67 "Orbital Case"
Communication Panel	62	53	57
Power Panel	69	50	55
Bus Science Equipment Panel	62	52	53
SCS/CS&C Panel	62	51	54

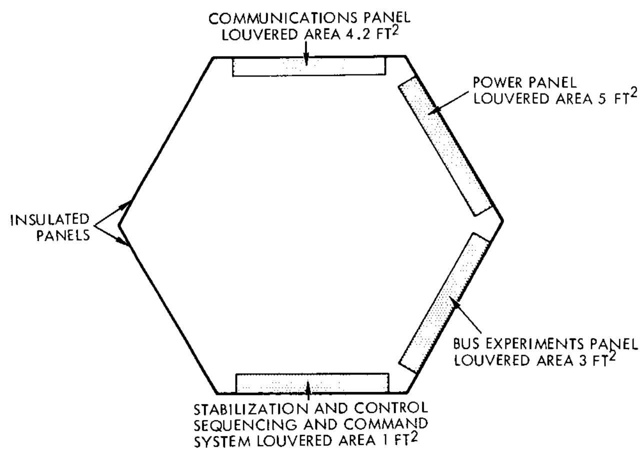


Figure 5. Spacecraft Thermal Control Inboard Profile

5.5.2 Nonnominal Conditions

The effect of a nonnominal orientation, such as that occasioned by midcourse and deboost maneuvers, with the sun normal to the communication panel for 1.0 and 1.38 AU is presented in Figures 6 and 7. The effect of a 2.3-hour eclipse, utilizing 1.67 AU steady-state temperatures for initial conditions is presented in Figure 8.

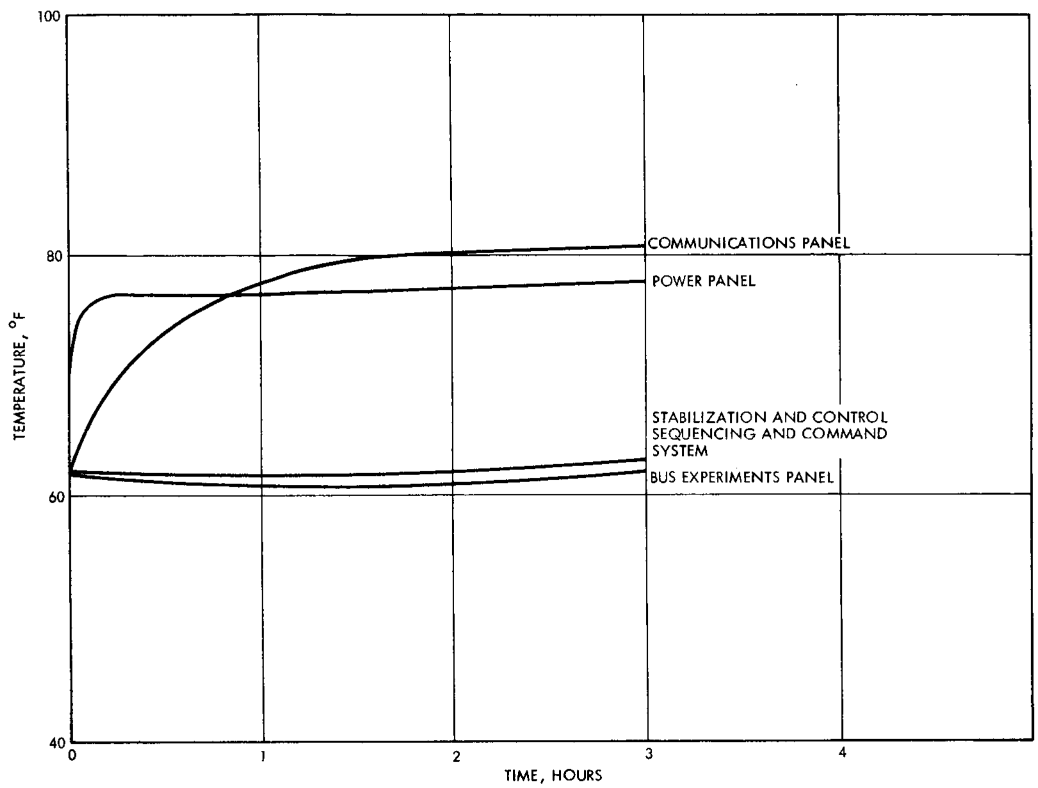


Figure 6. Temperature Histories of Equipment Panels at 1 AU Sun Normal to the Communications Panel

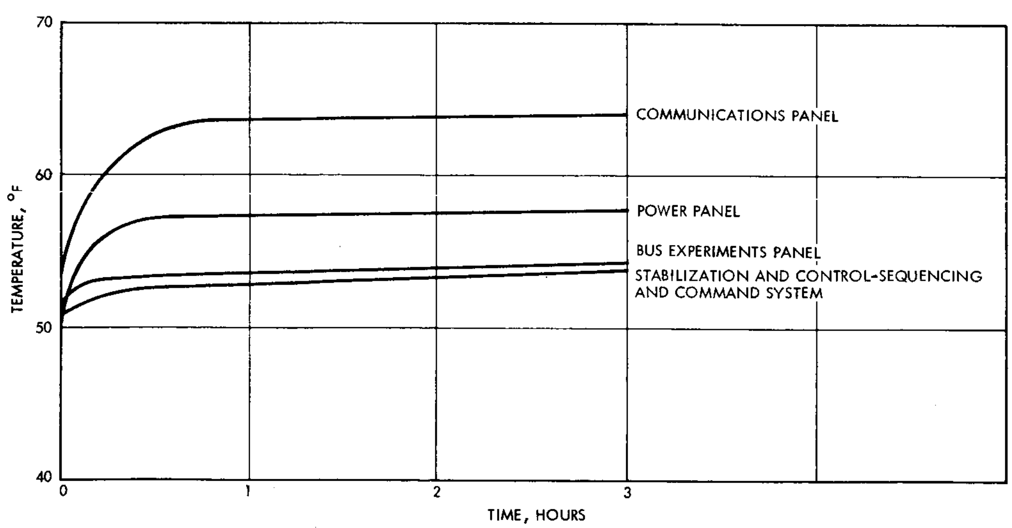


Figure 7. Temperature Histories of Equipment Panels at 1.38 AU with the Sun Normal to the Communications Panel and the Deboost Engine Firing

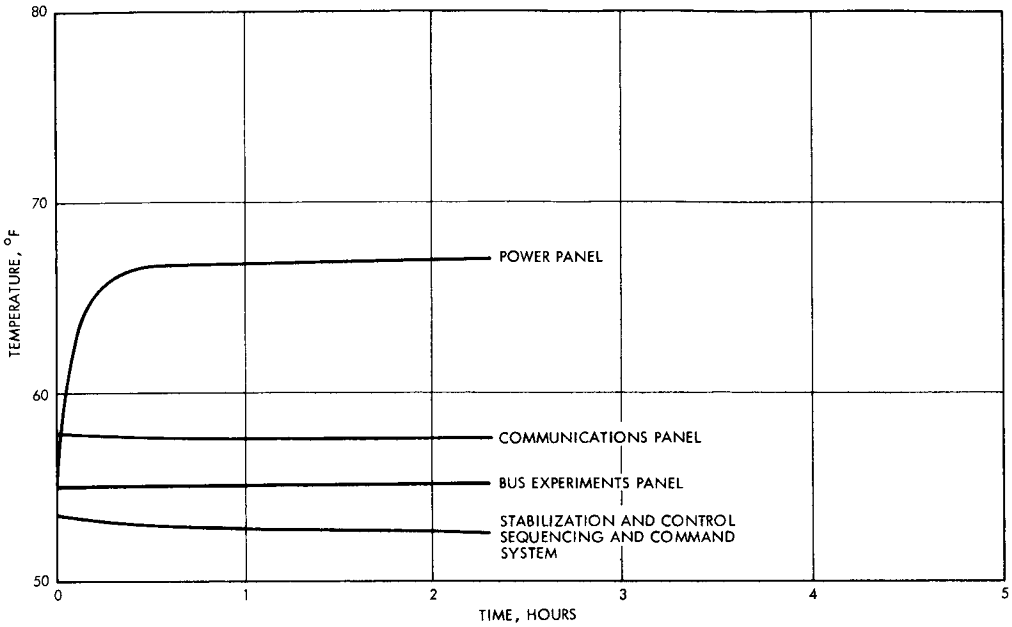


Figure 8. Temperature Histories of Equipment Panels During Mars Eclipse at 1.67 AU

5.5.3 Equipment Power Dissipation

The power dissipations on the respective panels are given in Table 1.

5.6 Solar Array Temperatures

Solar array temperature as a function of solar distance under normal solar radiation is presented in Figure 9.

In Figure 10 the array temperature is presented as a function of eclipse time for eclipses occurring at the maximum solar distance of 1.67 AU.

Figure 11 presents array temperature during injection firing. The absorptivity of the white coating to the plume radiation is 0.33.

5.7 External Equipment Thermal Control

5.7.1 Planet-Oriented Package Thermal Control

For the selected spacecraft configuration, the planet-oriented package is placed near the edge of the solar array and has a limited view of the sun and this only through the Martian atmosphere.

Table 1. Dissipated Power Distribution Within Main Compartment

Steady state 1 AU and transient sun normal to communications panel:

Equipment Panel	Dissipated Power (Watts)
1. Communications panel	78.4
2. Power panel	137.0
3. Bus experiments panel	55.5
4. Stabilization and control/sequencing and command system panel	22.2

Steady state 1.38 AU and transient sun normal to communications panel:

Equipment Panel	Dissipated Power (Watts)
1. Communications panel	78.4
2. Power panel	62.0
3. Bus experiments panel	55.5
4. Stabilization and control/sequencing and command system panel	22.2

Steady state 1.67 AU:

Equipment Panel	Dissipated Power (Watts)
1. Communications panel (bulk storage on)	98.4
2. Power panel	87.0
3. Bus experiments panel	63.0
4. Stabilization and control/sequencing and command system panel	31.7

1.67 AU eclipse transient:

Equipment Panel	Dissipated Power (Watts)
1. Communication panel	98.4
2. Power panel (eclipse)	198.0
3. Bus experiments panel	63.0
4. Stabilization and control/sequencing and command system panel	31.7

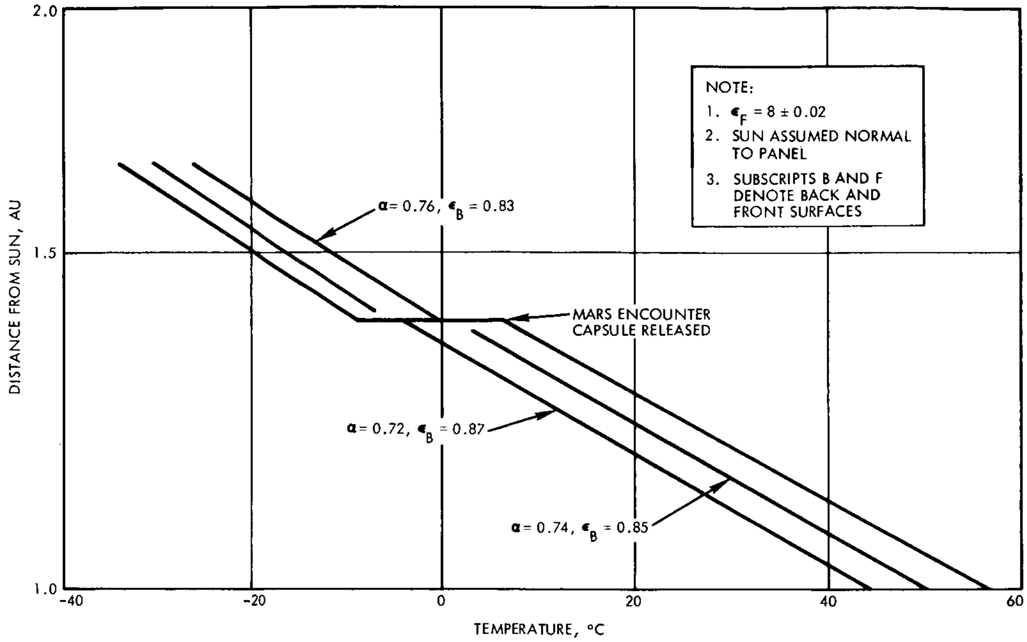


Figure 9. Solar Panel Temperatures Including Tolerances ϵ_B = 85

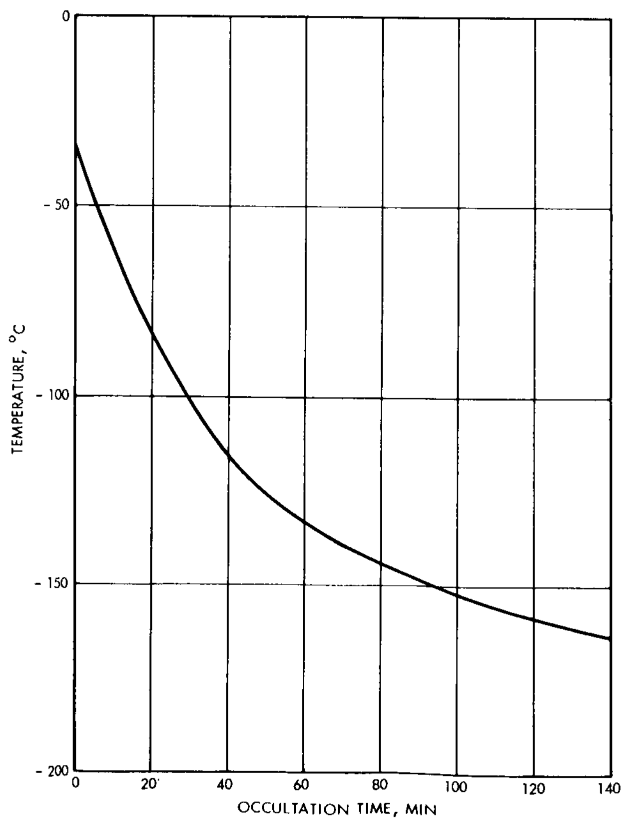


Figure 10. Solar Panel Temperature During Occulation

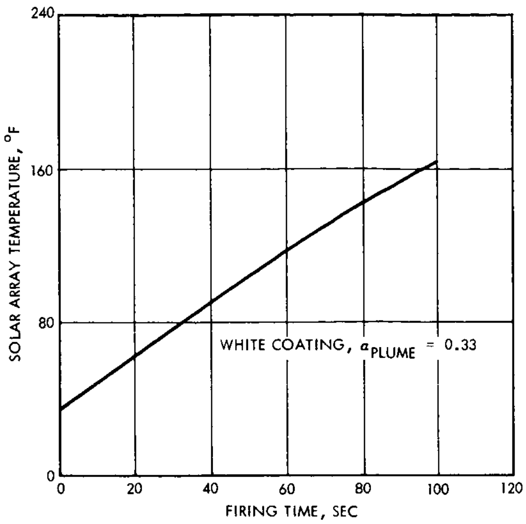


Figure 11. Solar Array Temperature Rise due to Radiant Heating from the Solid Rocket Particle Plume

The package is well insulated except for the square foot of experiment viewing ports, and requires up to 25 watts of heater power to maintain average package temperatures above $0^{\circ}F$. The maximum condition occurs when the sun can view 75 square inches of experiment openings at an angle of 45 degrees. This, with 16 watts of internal power dissipation and 6 watts of heat leak, causes an upper temperature limit of $85^{\circ}F$.

5.7.2 External Experiment Packages

The external experiments are housed within two packages on the edge of the solar array. With the packages insulated from the array and their surroundings, a total of 7.5 watts of heater power is required to maintain average package temperatures between zero and $85^{\circ}F$.

5.7.3 Gimbals

Antenna and POP gimbal systems are maintained between 20 and $120^{\circ}F$. The gimbals and bearings are conduction-insulated from their mounts and radiation insulated from their surroundings. The gimbal system, essentially electrically passive, requires 15 watts of heater power.

5.7.4 Antenna Temperature Control

The steady-state near-earth temperature for the antenna is $143^{\circ}F$, while the Martian maximum and minimum temperatures are expected to be +54 and -350°F, respectively. The antenna disk is expected to rise to 354°F due to the solid rocket plume heating. The antenna feed is insulated from the antenna disk and does not exceed its 200°F upper temperature limit during engine firing.

5.7.5 Magnetometer Temperature Control

The magnetometer is well insulated except for the 25 square inch radiator and requires a maximum of 5 watts of heater power to maintain average temperature between 20 and 70°F. The radiator faces outboard and hence could be solar irradiated during midcourse and orientation maneuvers, which would cause a maximum steady-state near-earth temperature of 140°F. Heating effects due to solid engine firing are estimated to cause a temperature rise of less than 10°F.

5.8 Rocket Engine Operation

Radiation energy during firing of the solid propellant motor impinging upon the forward (Refrasil batt) surface is 13.5 Btu/sec-ft^2. The absorptivity of the batt to plume radiation is 0.17. The temperature rise of inner surface of the batt after 90 seconds of exposure is less than 30°F.

Energy impinging on the array is 0.92 Btu/sec-ft^2. The array temperature history is presented in Section 5.6.

Convection heating of the forward surface is 0.016 Btu/sec-ft^2 which is negligible. Convection heating due to the midcourse motor is taken as 0.023 Btu/sec-ft^2. This heating coupled with normal solar irradiation causes the external layer of the Mylar insulation to rise to 152°F, well under the allowable limit of 300°F.

Internally, the equipment is protected from the hot case radiation by a blanket of alternating layers of aluminum foil and fiberglass paper insulation around the case. Conduction isolation is achieved by a phenolic fiberglass gasket between the thrust cone and the motor flange which limits the peak heat flux into the spacecraft to 36 watts, assuming a peak case temperature of 600°F, which is well within the radiating capability of the louver system.

After the case has cooled it represents a heat leak to space. The gasket and insulation limit the loss to 12 watts.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The thermal control subsystem provides temperature control for all phases of the mission consistent with allowable equipment temperature ranges as given in VS-3-111. Specific constraints or requirements associated with various design conditions for the mission are given below along with physical characteristics.

6.1 Parameters and Characteristics Affecting the Thermal Subsystem

The parameters and characteristics affecting the performance of the thermal subsystem are as follows.

a) The configuration layout as given in VS-3-110, in particular, the layout and placement of equipment panels, internal equipment location and method of attachment such as for the propulsion and attitude control systems, the placement and attachment of the solar array, and the placement and method of attaching external experiments including the planetary oriented package.

b) Electronic equipment installation as given in VS-4-550; the placement and method of attaching electronic power dissipating equipment on equipment panels.

c) Electronic equipment power profile as given in VS-4-460; the power dissipation and duty cycle of all electronic equipment.

d) Equipment allowable temperature limits as given in VS-3-111; the allowable minimum and maximum temperatures for both standby and operational modes for all equipment.

e) Flight sequence characteristics and spacecraft orientation as given in VS-4-410 and VS-3-104; the duration and attitude for nonnominal attitudes during midcourse and deboost maneuvers.

6.2 Design Conditions

Proper thermal control for a finite number of design conditions (that is, discrete worse case of power dissipation and external thermal environment) demonstrates satisfactory temperature control for the complete mission. Thermal control design conditions for the Voyager spacecraft preliminary design are given below. Power dissipation profiles for the various cases can be found in Volume 5, Section IV, Paragraph 3.1.

6.2.1 On-Stand Prelaunch

During on-stand checkout the spacecraft is convectively cooled by conditioned air supplied by the launch vehicle support equipment. Power dissipation ranges from standby to full power modes.

6.2.2 Boost Phase

During ascent, with the fairing on, the spacecraft is exposed to fairing radiant heating of 40 watts/ft^2, and following fairing jettison to peak aerodynamic heating loads of 24.2 watts/ft^2.

6.2.3 Centaur Coast and Injection Phases

After fairing jettisoning and exit of the earth atmosphere, the spacecraft is placed in a 100 n mi earth parking orbit having a maximum duration of 25 minutes. Following the parking orbit, the Mars injection phase of the mission begins and has a solar eclipse of up to 30 minutes.

6.2.4 Transit Phase

During the transit phase the spacecraft is nominally solar oriented. Nonnominal orientations lasting up to 3 hours can be experienced during midcourse and deboost maneuvers. The solar vector may be normal to any of the equipment panels during maneuvers, and has an irradiance of 127 watts/ft^2 near earth and a minimum value of 57.2 watts/ft^2 at Mars.

6.2.5 Solid Engine Firing

The spacecraft is exposed to heating effects produced from the solid motor firing lasting for approximately 100 seconds. This may follow a nonnominal orientation of up to 3 hours. The solid particle plume is estimated for preliminary design purposes to radiate 13.5 Btu/sec-ft^2 to a surface with a configuration factor of unity to the plume.

6.2.6 Mars Orbital Phase

The Mars orbital phases of the mission are characterized by solar eclipses of varying duration. The spacecraft is solar-oriented during Martian orbital phases of the mission. Solar irradiation varies between 46.7 and 67 watts/ft^2 at Mars.

6.3 Weight and Electrical Power Constraints

The thermal subsystem weight and power estimates are given in VS-3-111.

6.4 Spacecraft Envelope Constraints

The thermal subsystem is housed within the allowable spacecraft envelope as given in VS-3-110.

STRUCTURE

VS-4-520

CONTENTS

1. SCOPE

This document presents the design of the 1971 Voyager spacecraft structural subsystem. The structure serves as the unifying element for the vehicle, and provides load and thermal paths, support points for equipment installation, mechanical attachment at the interfaces with the launch vehicle and the capsule, and protection against environmental factors as required.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-3-110	Configuration
VS-3-111	Components Design Parameters
VS-3-130	Spacecraft-Launch Vehicle Interface
VS-3-140	Spacecraft-Capsule Interface
VS-4-510	Thermal Control
VS-4-521	Structural Design Criteria
VS-4-550	Electronic Equipment Packaging
VS-4-571	Planet-Oriented Package

3. FUNCTIONAL DESCRIPTION

3.1 Function and Design Objectives

The primary function of the structure is to integrate, with a minimum of weight, the many subsystems comprising the spacecraft. It provides sufficient strength, rigidity, and other physical characteristics necessary to maintain adequate alignment between components, acceptable static and dynamic load environments, and to support spacecraft 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 accept changes in the mission and subsystem requirements.

3.2 Structural Arrangement

The spacecraft structure is composed of three major components: (1) the basic bus module including the support structure for the solid propellant motor, the solar panels, and externally mounted equipment; (2) the midcourse propulsion module; and (3) the solar panels. The spacecraft structural arrangement is presented in Figure 1 and the over-all configuration and equipment layout is given in VS-3-110.

The module contains the spacecraft primary structure, which is a hexagonal framework extending from the launch vehicle interface at Saturn station 2048 (spacecraft station 0) to the flight capsule interface at spacecraft station 59. This module contains all electronic equipment and certain science instruments. The electronic equipment is located on four of the six panels of the bus. The high gain and medium gain antennas, and the planet oriented package (POP) are mounted on supports attached to the sides of the basic bus module. The midcourse propulsion module has a main element in the form of a transverse panel at spacecraft station 2. The entire midcourse propulsion system and the gas system for stabilization and control are mounted to this panel. Six nondeployed solar panels are attached to the aft end of the bus module to form the solar array.

3.3 Bus Module Structure

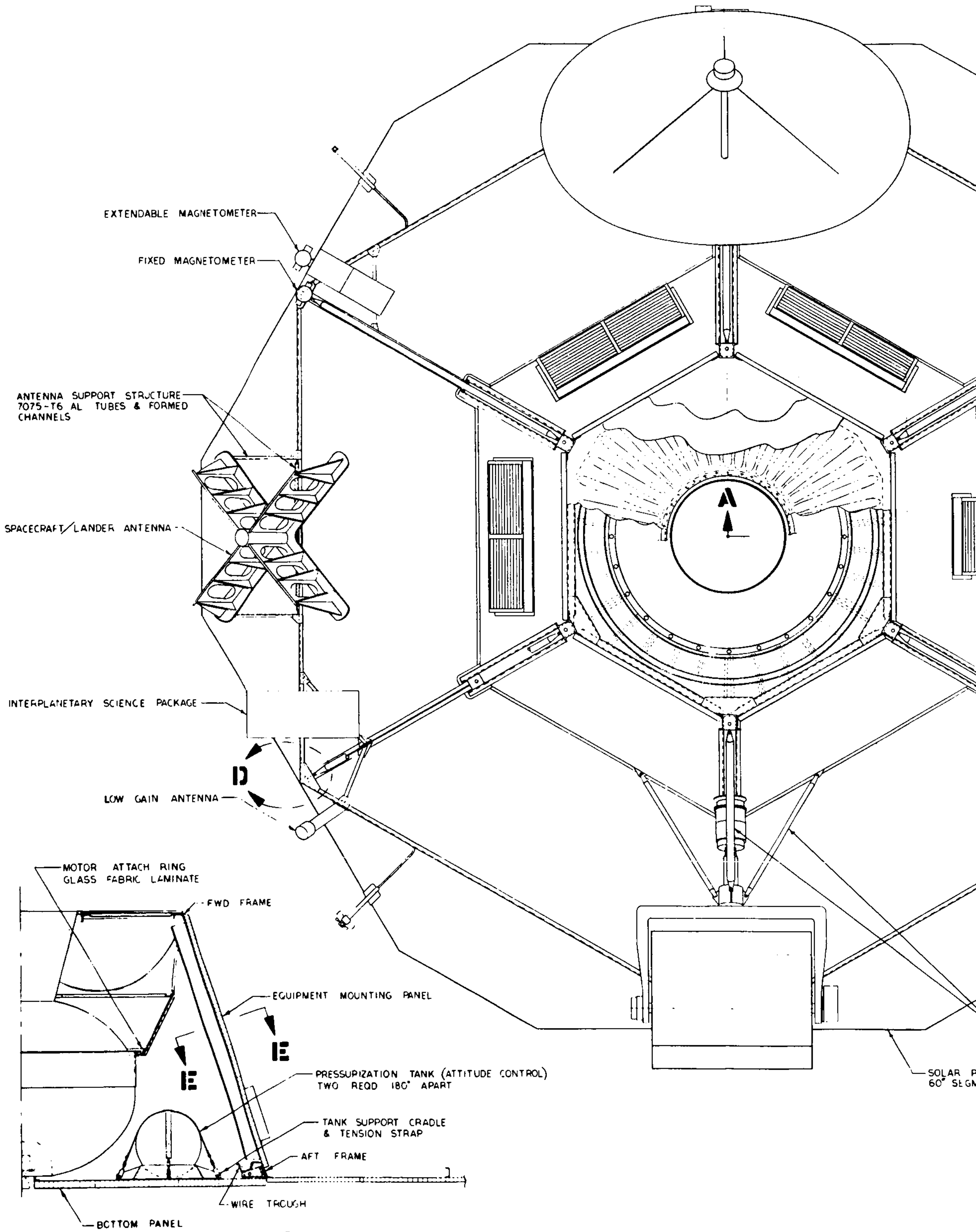
The basic bus module is in the shape of a truncated hexagonal pyramid tapering from 120 inches at spacecraft station 0 to 80 inches at spacecraft station 59. It is composed of two principal components. The outer shell, which is the mounting surface for the electronic subassemblies, serves as the main load-carrying structure. The inner structure provides support for the retropropulsion subsystem.

3.3.1 Bus Module Outer Shell

The outer shell consists of a hexagonal frame structure with panels on the sides and on the forward end of the spacecraft.

a) Frame Structure

Six π-section 7075-T73 machined longerons run down the corners of the hexagon and are the principal axial load-carrying members.


EXTENDABLE MAGNETOMETER
FIXED MAGNETOMETER
ANTENNA SUPPORT STRUCTURE 7075-T6 AL TUBES & FORMED CHANNELS
SPACECRAFT/LANDER ANTENNA
INTERPLANETARY SCIENCE PACKAGE
D
LOW GAIN ANTENNA
A
SOLAR P
60° SEGM
MOTOR ATTACH RING GLASS FABRIC LAMINATE
FWD FRAME
EQUIPMENT MOUNTING PANEL
E
E
PRESSURIZATION TANK (ATTITUDE CONTROL) TWO REQD 180° APART
TANK SUPPORT CRADLE & TENSION STRAP
AFT FRAME
WIRE TROUGH
BOTTOM PANEL

SECTION A - A

0

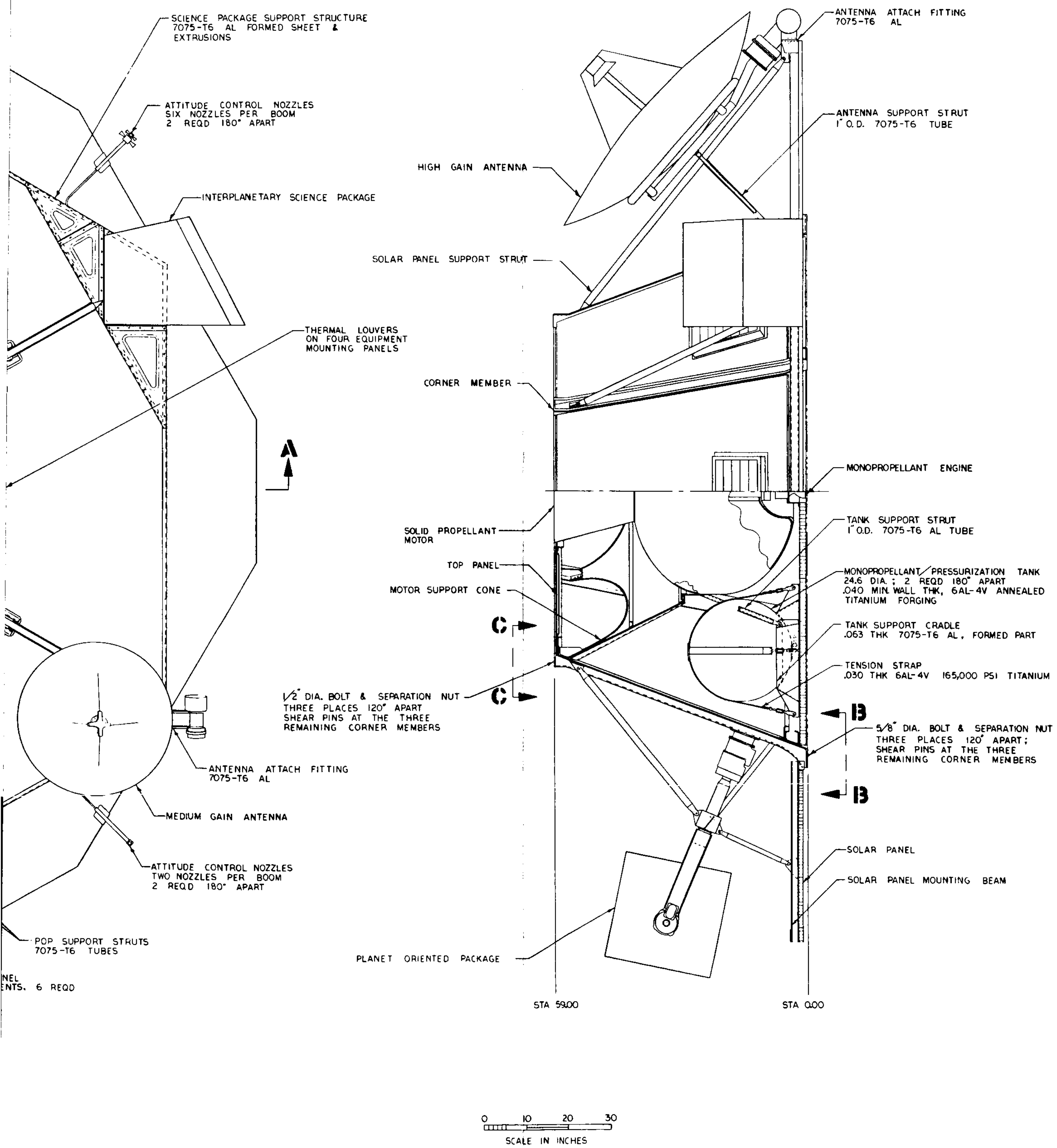
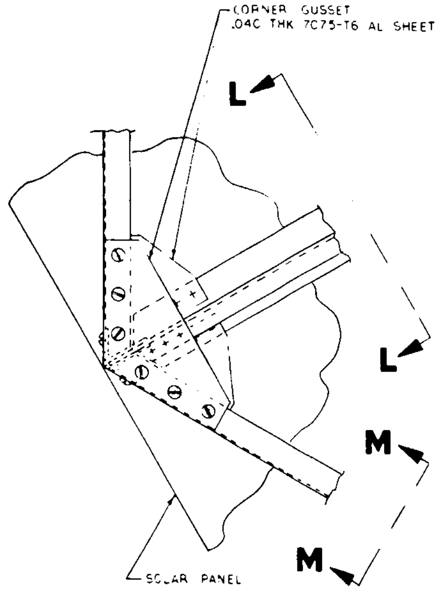
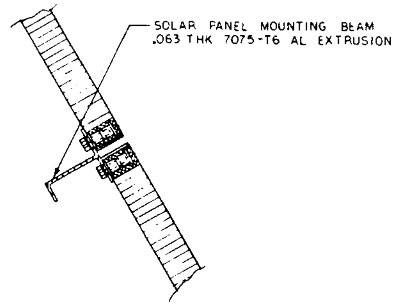


Figure 1. Spacecraft Structural Arrangement

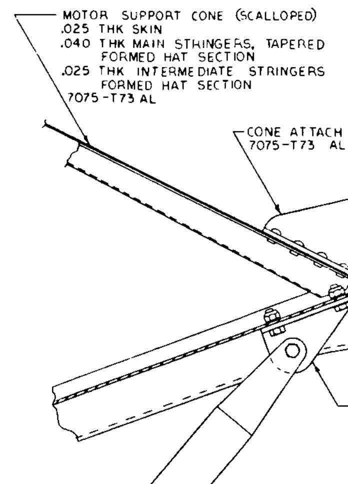
6



VIEW D



SECTION L - L



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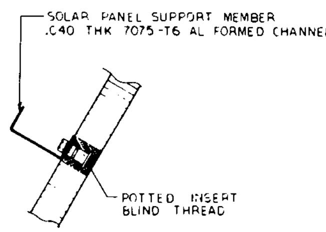
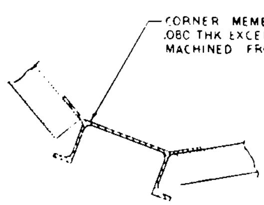
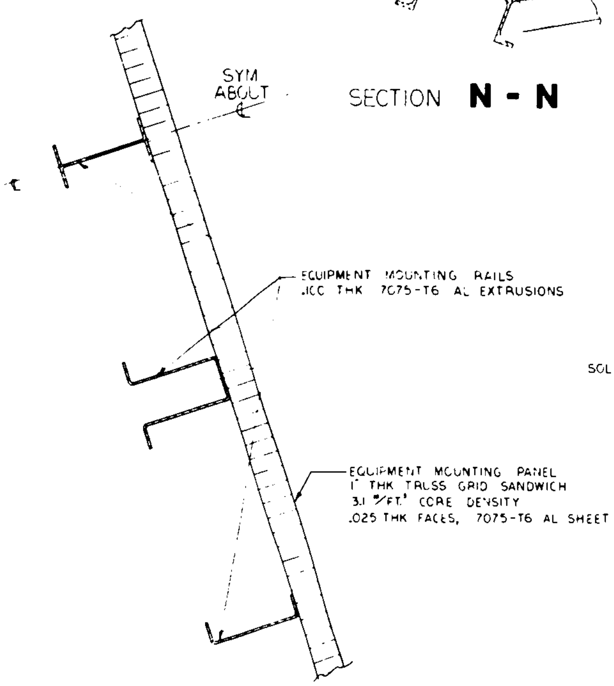


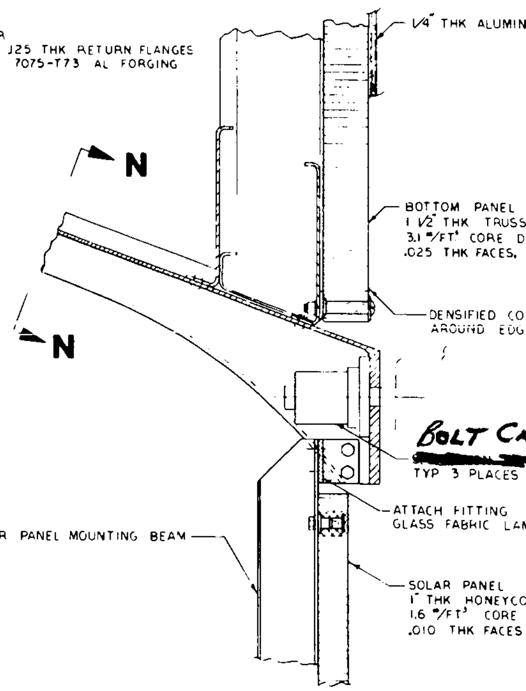
SECTION M - M



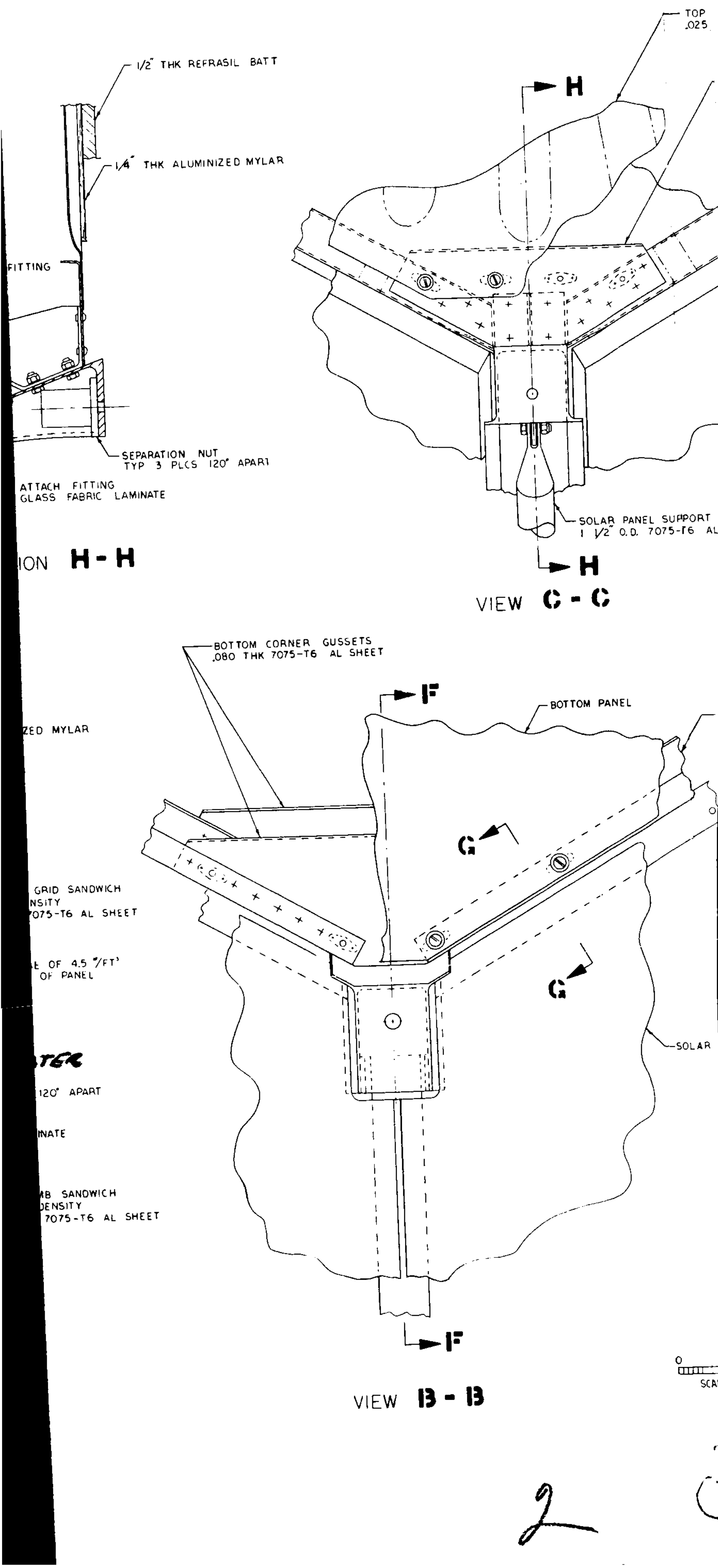
SECTION N - N



SECTION E - E



SECTION F - F



TOP
.025
1/2" THK REFRASIL BATT
1/4" THK ALUMINIZED MYLAR
FITTING
SEPARATION NUT
TYP 3 PLCS 120° APART
ATTACH FITTING
GLASS FABRIC LAMINATE
ION H - H
H
H
SOLAR PANEL SUPPORT
1 1/2" O.D. 7075-T6 AL
VIEW C - C
BOTTOM CORNER GUSSETS
.080 THK 7075-T6 AL SHEET
F
BOTTOM PANEL
ZED MYLAR
G
G
GRID SANDWICH
NSITY
7075-T6 AL SHEET
E OF 4.5 #/FT³
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TER
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VIEW B - B
2

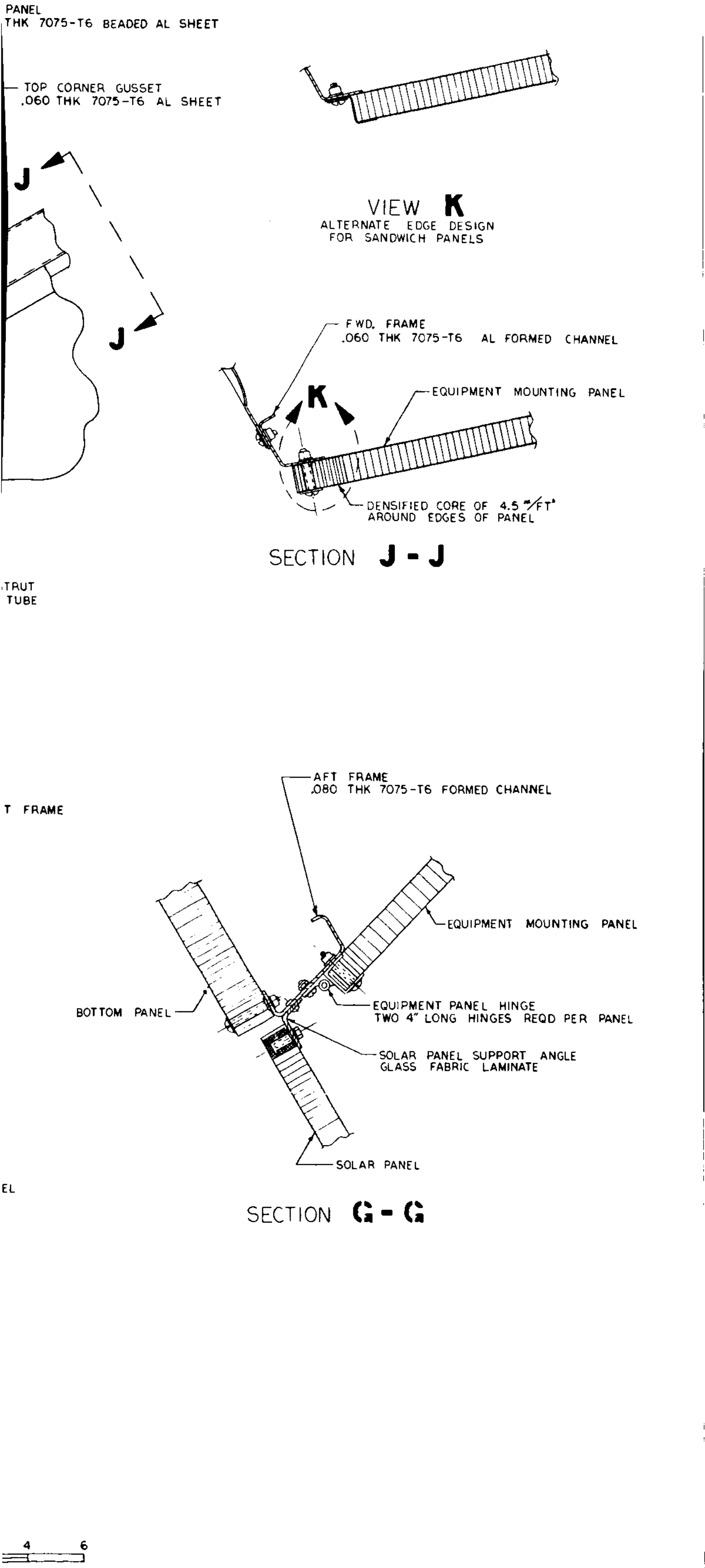


Figure 1. Spacecraft Structural Arrangement (Continued)

These longerons go from stations 0 to 59 and have integrally machined end fittings which serve as the attach points for the flight capsule at the forward end and the Centaur interstage at the other. Attachment to both the flight capsule and the Centaur consists of bolts with separation nuts and bolt catchers at three of the six fittings, and with shear pins at the other three fittings. The six fittings are joined by 7075-T6 aluminum transverse channel frames at spacecraft stations 2 and 58. They are gussetted to the six longeron members to provide structural rigidity during fabrication and assembly.

b) Side Panels

The side panels, which run from spacecraft station 4 to 59, serve as mounting panels for spacecraft and science electronic equipment. In addition, they provide meteoroid protection, serve as a heat sink for the thermal control subsystem, and are the main shear carrying members of the spacecraft. The thermal louver assemblies mount to the outer face of four of the six panels. The sandwich panels are made of 1-inch-thick aluminum truss grid-core with 0.025-inch 7075-T6 aluminum face sheets. Extruded rails are attached to the inner side of the panels and serve as the mounting members for all electronic equipment. The panels are bolted to the frame along all four sides and have a hinge along the lower edge of each panel to facilitate equipment servicing.

c) Forward Panel

The forward surface of the bus is covered with a hexagonal beaded single-skin panel. It serves as a shear web and provides meteoroid protection in conjunction with the thrust cone after capsule separation.

3.3.2 Bus Module Inner Structure

The solid propellant motor support structure is a permanent part of the bus module. It is a semimonocoque truncated cone. Construction is 7075-T6 aluminum skin and hat section stringer design. A fiberglass attach angle is located at the lower end of the cone for attachment of the motor. Fiberglass is used to reduce conduction of heat to the spacecraft during and immediately after solid propellant engine firing. Six of the stringers carry the thrust loads into the six axial load carrying members of the basic bus structure.

3.4 Midcourse Propulsion Module Structure

The midcourse propulsion module structure consists of a transverse panel at station 2, which closes the main structure, along with tank support structure.

3.4.1 Panel

The panel is a hexagon measuring 120 inches across the vertices that attaches to the aft end of the bus frame with an interchangeable bolt pattern. Sandwich construction is used for the panel. It has a 1.5-inch-thick aluminum truss grid-core with 0.025-inch 7075-T6 skins. In addition to serving as a structural member, it provides meteoroid protection to the aft end of the spacecraft.

3.4.2 Support Structure

Support structure is provided for the two pressurant/propellant tanks and the two stabilization and control gas tanks which mount on the panel. The tanks will rest on a horizontal coneshaped flange on the support frame. Two straps at 90 degrees to each other go across the top of a tank and attach with tensioning devices to the base of the tank support structure. Thid midcourse engine mounts to a support structure located on the centerline of the panel.

3.5 Solar Panel Structure

The solar array structure consists of the basic panels and their supporting structure. The array is in the form of a fixed flat panel with a 12-sided outer edge and a 6-sided inner edge that matches the shape of the basic bus. The total panel area is 185 ft^2. The array is divided into six identical panel units. These units are constructed of 1-inch-thick aluminum honeycomb core with 0.010-inch 7075-T6 aluminum skins. The inner edge of the panels mounts to the bus with a series of fiberglass clips. Fiberglass is used to reduce heat flow from the panels to the spacecraft. Six transverse radial beams run from the corners of the bus to provide support for the radial edges of the panels. Additional support members run from the end of each beam to the end of each adjacent beam forming a hexagonal support frame for the outer edge of the panels. Vertical loads are carried by tubular struts running from the ends of the transverse support members to the corners of the bus at spacecraft station 59.

3.6 High Gain Antenna Support

The gimbal point for the high gain antenna is located at the outer edge of the solar array at the radial support member for corner E of the spacecraft (spacecraft corner designations are given in VS-3-110). The antenna support fitting attaches to the array support member. A tubular strut which is part of the solar panel support structure runs from this fitting to the bus at station 59 to carry all vertical loads and to provide a support and latch point for the antenna during boost. A second strut runs from the antenna latch point back to the bus at spacecraft station 2.

3.7 Medium Gain Antenna Support

The gimbal point for the medium gain antenna is located just outboard of the array on the radial solar panel support member at spacecraft corner C. A tubular strut that is part of the solar panel support structure carries all vertical loads to the bus at station 59. A transverse support member runs between the radial and peripheral solar panel support members to provide for antenna latching during launch.

3.8 Science Support Structure

3.8.1 Planet Oriented Package

Support for the planet-oriented package is integrated into the solar panel support structure. A yoke support fitting is incorporated into the support tube at spacecraft station 24. Two diagonal struts run from this point to the bus frame at spacecraft station 2. Support for the bearing at the inboard end of the yoke shaft is incorporated into the bus corner member.

3.8.2 Magnetometers

A fixed magnetometer on a short boom and an extendable one are located together near the outer edge of the solar array. They are attached to the peripheral array support members as shown in Figure 1.

3.8.3 Interplanetary Science Packages

Two interplanetary science packages are installed at opposite sides of the spacecraft at the outer edge of the solar array. Each unit is mounted to both a radial and a peripheral solar panel support member.

4. INTERFACE DEFINITION

4.1 Thermal Control

The structure provides heat conduction paths having conductance of no less than 5 Btu/ft^2-hr-^oF from the power dissipating equipment to the radiating areas.

The structure conductively isolates the radiating areas from items of widely varying temperature such as the rocket engine and the solar array, in keeping with VS-4-510.

Equipment panels provide an interchangeable mounting pattern for the louver installation.

The structure surface finishes and provisions for installation are in accordance with VS-4-510.

4.2 Propulsion

The midcourse propulsion subsystem support structure is designed so that the system can be assembled, tested, and installed on the spacecraft as a complete, self-contained module.

The mounting structure for the retropropulsion subsystem (including the thrust-vector control system) is designed so that the subsystem can be installed as a complete, self-contained module. Support structure for the solid propellant motor is designed to minimize the introduction of concentrated loads into the motor case. The solid propellant motor is installed from the top of the spacecraft.

4.3 Electronic Packaging

Relatively heavy components are located so as to minimize dynamic loading of the supporting structure and are secured in a manner that will provide added rigidity.

Equipment mounting rails are utilized on the structural panels for installation of all electronic packages in accordance with VS-4-550. Assembly harnesses are incorporated into the panel mounting rails. The system ring harnesses are incorporated into the bus frame at station 2. Hinges are provided along the bottom edge of all equipment panels to facilitate access.

4.4 Stabilization and Control System

The plane of the mounting surfaces for the midcourse propulsion module is to be parallel to the flight spacecraft X-Y plane within the required tolerances.

The structure does not obstruct the field of view of any of the stabilization and control sensors.

Mounting planes for the stabilization and control system components are to be aligned within the required tolerances.

4.5 Science

The structure is designed to meet the mounting and alignment requirements given in VS-4-210 and VS-4-571.

4.6 Operational Support Equipment

Operational support equipment is not to impose loads on the spacecraft greater than the flight loads. Attach points are provided for assembly and mating fixtures.

5. PERFORMANCE PARAMETERS

Performance for the structure subsystem is indicated by how well it meets the static, dynamic, and acoustic load requirements with a minimum of weight. This is presented in the form of a structure analysis in Volume 5, Section 5.1.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Design Criteria

The structure is in accordance with the structural design criteria of VS-4-521.

6.2 Configuration and Packaging

The structure is compatible with the spacecraft layout and configuration of VS-3-110 and the packaging considerations in VS-4-550.

6.3 Interfaces

The interface and constraints impose upon the structure by the launch vehicle and the flight capsule are contained in VS-3-130 and VS-3-140, respectively.

6.4 Weight

The structural weights are given in VS-3-111.

STRUCTURAL DESIGN CRITERIA

VS-4-521

CONTENTS

1. SCOPE

This document presents the basic requirements and establishes the criteria governing the structural design and testing of the Voyager spacecraft for the 1971 mission. These criteria also define the design conditions and environments, factors of safety, structural design considerations, and requirements necessary to assure structural integrity and a high probability for successful completion of the prescribed mission.

The criteria account for the structure functional requirements:

a) Maintain adequate alignment between components

b) Provide acceptable static and dynamic load environment

c) Support spacecraft components and assemblies and the flight capsule.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-3-110	Configuration
VS-3-112	Weights and Mass Properties

JPL

V-MA-004-001-14-03	Project Document No. 45 Preliminary Voyager 1971 Mission Specifications
30209B	Process Specification, Flight Equipment, Titanium Alloy 6Al-4V Compressed Gas Vessels (Pressurized in the Vicinity of Personnel)
30265	General Specification, Spacecraft Flight Equipment Pressure System, Safety Requirements

Drawings

Phase IA, Vol. V, Section IV, Mechanical Subsystems	Mars Voyager Structural Layouts
Phase IB Phase II	Mars Voyager Current Engineering Structural Layouts and Assembly Drawings

Government Publications

MIL-HDBK-5 — "Metallic Materials and Elements for Flight Vehicle Structures," August 1962

MIL-HDBK-17 — "Plastics for Flight Vehicles," Part I Reinforced Plastics, November 1959

3. REQUIREMENTS

3.1 Basic Structural Requirements

3.1.1 Strength and Rigidity

The structure will possess sufficient strength, rigidity, and other necessary physical characteristics to survive the design conditions and environments given in this document. In addition, the structural subsystem will be sufficiently rigid to minimize response to vibration inputs and to provide solid mounting supports for the electronic and power subsystems and attitude sensors.

3.1.2 Design Approach

Design simplicity, conservatism, and testing will characterize the structural design to increase reliability. Ranger, Mariner R, and Mariner C experience will be utilized wherever possible.

3.2 Strength and Rigidity Requirements

3.2.1 Limit Loads

Limit loads are the maximum loads that the structure is expected to experience in service for the design conditions under consideration. All loads and load factors specified in this document refer to limit loads unless otherwise noted.

3.2.2 Design Load

A design load is the limit load multiplied by the appropriate hazard factor of safety. The structure will be designed for sufficient strength to withstand simultaneously the design loads and other accompanying environmental phenomena for each design condition without experiencing excessive elastic or plastic deformation. Excessive deformations are those which result in structural deflections of sufficient magnitude to cause interference between spacecraft components or violation of the spacecraft envelope by any portion of the spacecraft or otherwise reduce the probability of successful completion of the mission.

3.2.3 Ultimate Design Load

The ultimate design load is the design load multiplied by the ultimate factor of safety. The ultimate factors of safety to be used in the design of the spacecraft or any component are presented in paragraph 3.2.7. No factor of safety is applied to any environmental phenomena except loads. The structure will be designed to withstand simultaneously the ultimate loads and other accompanying environmental phenomena without failure. Structural deformation or deflections will not precipitate structural failure during any design condition and environment at loads less than or equal to ultimate loads.

3.2.4 Failure

Failure is the condition under which the structure or any component can no longer perform its intended mission. Such condition can be caused by instability, excessive deflection, general yielding, or rupture.

3.2.5 Allowable Load

The allowable load for the structure or any component is to be that load at which failure is imminent. The allowable load may be applicable for either design or ultimate load conditions.

3.2.6 Proof Factors

The proof factor is the ratio between the design load and limit load or the design pressure and the working pressure. Items such as pressure vessels, pressure-carrying components, primary or secondary structures of glass reinforced plastic, and bonded joints may require an acceptance proof test as evidence of satisfactory workmanship and material quality. The item is to sustain the proof load or proof pressure without excessive elastic or plastic deformation, as required in paragraph 3.2.2. The proof factor will include a material correction for elevated or cryogenic temperatures occurring in service, if required. The proof factors for pressure vessels and pressurized components are identical to the hazard factors and are specified in paragraphs 3.2.7. All other proof factors are to be 1.0 unless temperature correction is required.

3.2.7 Factors of Safety

Limit loads are multiplied by a hazard factor to obtain design loads. Design loads when multiplied by an ultimate factor of safety become ultimate design loads. The following factors of safety are the minimum values to be applied.

a. General Structure

Hazard Factor = 1.0

Ultimate Factor of Safety = 1.25

b. Pressurized Systems

The ultimate factor of safety to be used for the design of pressure vessels and pressure-carrying components is denoted as the burst factor. The burst factor is the ratio of the burst pressure and the operating pressure. The factors to be used for the design of pressure vessels, pressurized systems, and components are presented in Table 1. These factors will be demonstrated by analysis and test using minimum thickness and material properties.

Table 1. Hazard/Proof and Burst Factors

Item	Hazard/Proof Factor	Burst Factor
Small diameter lines, fittings, valves, and hoses	2.0	4.0
Pressure vessels (including reservoirs, accumulators, and regulators)		
a) No hazard to personnel	1.0	1.25
b) With hazard to personnel	1.76	2.2
Rocket motor cases	1.15	1.25

3.2.8 Coordinate Axes

The coordinate axes system to be used for orientation of components and analysis purposes is given in VS-3-110.

3.2.9 Weight

Weight and mass properties data of the 1971 Voyager spacecraft and of individual components is summarized in VS-3-112.

3.2.10 Margins of Safety

Margins of safety are to be computed at both yield and ultimate load levels. All structures will have a positive margin of safety, i.e., M.S > 0, which will be computed in accordance with MIL-HDBK-5 procedures.

3.3 Design Conditions and Environments

3.3.1 Objectives

The design objectives of the structure subsystem will provide sufficient strength, rigidity, and other physical characteristics necessary to maintain adequate alignment between components, provide acceptable static and dynamic load environments, and support spacecraft components, assemblies and the flight capsule during the mission preflight and flight environments. Other objectives emphasize ease of maintainability, access, and adaptability to accept design improvements and changes in other sub-systems with minimum redesign.

3.3.2 External Loads

All quasi-static loads are to be considered. They include booster thrust, flight, maneuver, and separation loads. Vibratory accelerations occurring in the various phases of flight are to be considered acting simultaneously with quasi-static accelerations.

3.3.3 Combined Loads

The loads due to design constraints caused by thermal requirements and pressures as well as those resulting from preload are to be considered acting separately or superimposed upon the loads.

3.3.4 Dynamic Loads

The analytical model of the spacecraft structure will be of sufficient detail so that frequencies, mode shapes, deflections, separation dynamics, and critical stresses can be approximated. Account will be taken of the dynamic loading induced by elastic response of the spacecraft structure to excitation in addition to the rigid body response. All structural members and appendages will have a natural frequency greater than 5 cps for the boost configuration of the spacecraft.

a. Transient Loads

All loads of a transient nature will be considered in the structural design and analysis of the spacecraft. They include the effect of suddenly applied forces, acoustics, and thrust variations during engine starting, operation and cutoff.

b. Vibration Loads

The applicable vibration environment and requirements will be considered in the design and testing of the entire spacecraft and its components.

3.3.5 Separation Joint Preload

The structural design and installation procedures for tension joints, such as separation joints and field joints, will provide sufficient axial preload to prevent separation of the mating surfaces during any ultimate load condition.

3.3.6 Thermal Effects

The influence of transient and quasi-static variations in temperature will be incorporated in the appropriate design conditions, both for influencing material properties and causing thermal stresses.

3.3.7 Materials and Processes

a. Selection

All materials and processes used in the spacecraft will be selected on the basis of suitability for the intended application and for reliable performance, and will conform to specifications and standards selected from released lists of NASA, military, industry, and company specifications and standards, in that order of preference if there is conflict. If the requirements of specifications and standards selected are less stringent than those imposed by the application and the reliability goal specified for the part, material, or processes covered, the more stringent requirement will apply. Materials and processes selected will be subject to the approval of JPL.

b. Corrosion

All metals used in the spacecraft will be corrosion resistant under the environmental conditions to which they will be exposed. Material selection, fabrication techniques, and design will include as a requirement the prevention of galvanic and concentration cell corrosion and stress corrosion cracking.

c. Pressure Vessels

The design of pressure vessels and methods of mounting will avoid abrupt changes in cross-sections and restraints that could induce high stress concentrations during pressurization. Designs with welded ports and mounting bosses and wall thickness-to-diameter ratios smaller than 1/1000 are to be avoided.

Pressure vessels which may be hazardous to personnel are to be fabricated from Ti-6AL-4V titanium alloy in the annealed condition. For small, low-pressure vessels, 6061-T6 aluminum is acceptable. Vessels which will not be pressurized in the vicinity of personnel may be fabricated of Ti-6AL-4V heat treated to 165,000 psi maximum ultimate strength.

d. Magnetic Requirements

Nonmagnetic materials are to be used wherever possible. All materials, including bulk and raw materials, on the spacecraft will be magnetically evaluated in an ambient magnetic field of less than 100 gamma. Measurements will be made after exposure to a magnetic field of 100 oersteds and after demagnetization in an alternating magnetic field of initial amplitude of not less than 100 oersteds.

3.3.8 Fatigue

The effect of repeated loads will be considered in the structural design. Special consideration will be given to the design of components subject to fatigue failure, i.e., vessels subjected to repeated pressure cycles, hardware adjacent to severe vibration sources, etc. Materials selected for design are to exhibit satisfactory fatigue characteristics. Conditions of poor surface finish, stress concentrations, or unfavorable residual stresses are to be avoided.

3.3.9 Pressure Changes

Structural components will be vented to account for ambient pressure changes during ground and air transportation and the ascent phase of flight. The assumed environment is specified in the JPL Preliminary 1971 Voyager Mission Specification.

3.3.10 Energy Dissipating Mechanisms

All energy-dissipating mechanisms will have, within prescribed tolerance limits, linear force-velocity relationships over a wide range of frequencies and loads and will be relatively insensitive to the temperature environment. Mechanical backlash will be kept to a minimum in all mechanical connections.

3.3.11 Deployment

The design of articulated appendages and associated supporting structure will account for the loads induced by unlatching, latching, and deployment or pointing maneuvers. Vibration sensitivity will be considered for all appendage positions.

3.3.12 Meteoroid

A major design consideration will be the protection of the spacecraft and its components from the hazards of micrometeoroids. The definition of the micrometeoroid environment applied during Phase IA is provided in paragraph 3.3.17.f.

Rational procedures and state-of-the-art equations for micrometeoroid penetration will be used in determining the necessary protection to assure high probability of mission success. Individual protection of critical packages versus overall spacecraft protection will be subject to tradeoff studies. Data on micrometeoroid environment and penetration formulas used in design and analysis will be reviewed and upgraded throughout the development period.

3.3.13 Radiation

The influence of radiation will be considered in the selection of materials and in the protection requirements of sensitive spacecraft components and equipment.

3.3.14 Interface Requirement

Consideration will be given to the structural compatibility at the interfaces of all subsystems with the structures subsystem.

3.3.15 Prelaunch Conditions

Provisions will be made in the design of ground handling equipment to ensure that loads encountered during assembly, transportation, and handling do not control the design of the spacecraft or of any component, except where particular requirements are specified.

3.3.16 Launch, Exit and Separation (Phase IB and II)

The launch phase is defined to include liftoff, flight through the atmosphere, S-IB burnout, S-IVB burnout, Centaur first burn, and Centaur second burn. The dynamic structural interaction of the launch vehicle and the planetary vehicle will be considered in the launch phase analysis of the composite spacecraft design loads. These loads will be refined by an iterative approach throughout the development period. The loads during the launch phase will be based on analyses for the appropriate conditions defined below.

a. Liftoff Through S-IB Burnout

The acceleration builds up to a maximum value at S-IB burnout, at which time the maximum axial acceleration in the negative thrust direction is 4 g's and the lateral acceleration is less than 2 g's.

b. Saturn S-IVB Operation

The acceleration builds up throughout S-IVB operation. At burnout, the maximum axial acceleration in the negative thrust direction is 2.6 g's and the lateral acceleration is less than 2 g's.

c. Centaur First Burn

The maximum acceleration during the Centaur first burn occurs at burnout. The maximum axial acceleration in the negative thrust direction is 1 g and the lateral acceleration is less than 2 g's.

d. Centaur Second Burn

The maximum acceleration during the Centaur second burn occurs at burnout. The maximum axial acceleration in the negative thrust direction is 2.2 g's. The lateral static acceleration is less than 2 g's.

e. Low Frequency Vibration

The low frequency flight vibration, covering all events from liftoff to spacecraft injection, is estimated to be a sinusoidal input as follows:

Lateral	0.6 g rms	5 to 200 cps
Axial	1.2 g rms	5 to 200 cps

f. Random Vibration

The liftoff and transonic vibration environment, with the exception of low frequency, is assumed to be the following omnidirectional input to the spacecraft separation plane: power spectral density peaks of 0.07 g^2/cps ranging from 100 to 1500 cps with a 6 db/octave rolloff in the envelope defining peaks below and above these frequencies. Maximum total time is 60 seconds. Random vibration at other mission times is predicted to be insignificant by comparison.

g. Shock

Transient accelerations at shroud and spacecraft separation will be investigated. Depending on the launch vehicle characteristics, shock response during other mission events is predicted to be insignificant by comparison. The shock response to these environments may be approximated by an input consisting of a 200 g terminal peak sawtooth of 0.7 to 1.0 millisecond rise time.

h. Acoustic Sound Field

The maximum acoustic field, for either liftoff or transonic, is assumed to be a reverberant field as follows: Overall sound pressure level (SPL) is approximately 142 db (re 2×10^{-5} N/m^2). SPL of 133.5 db/third octave from 85 to 250 cps. Rolloff at 11 db/octave. Rolloff at 5 db/octave above 250 cps. Total duration is about 2 minutes.

3.3.17 Space Flight

a. Dynamic Loads

The dynamic interaction of the flight spacecraft structure with the stabilization and control subsystem is to be considered in the midcourse, capsule separation, and retropropulsion sequence analysis.

b. Staging Loads

Loads and the shock and vibration environment that result from the Centaur-planetary vehicle and flight capsule-spacecraft separation events are to be considered.

c. Attitude Control Load

Loads generated about the three control axes as a result of stabilization and control during orientation maneuvers will be investigated.

d. Midcourse Maneuvering

The loads and vibration environment generated during firing of the midcourse propulsion subsystem will be investigated.

e. Retromotor Firing

Loads and vibration environment generated during the retro-propulsion maneuver will be considered.

f. Meteoroid Environment

Near Earth. The particle flux is taken as:

$$\log N = -17.0 - 1.70 \log M$$

where N = the number of particles/(m^2 sec) of mass M and greater. The relative velocity of the particles is taken as varying from 0 to 10 km/sec and the average density as 0.4 gm/cm^3.

Interplanetary. The particle flux is assumed to be:

$$\log N_{\text{near earth}} = -13.80 - \log M + 2 \log \left(\frac{0.44}{\rho}\right)$$

$$\text{to} \quad -14.48 - 1.34 \log M + 2.68 \log \left(\frac{0.44}{\rho}\right)$$

and

$$\log N_{\text{near Mars}} \geq -13.30 - \log M + 2 \log \left(\frac{0.44}{\rho}\right)$$

$$\text{to} \quad -13.98 - 1.34 \log M + 2.68 \log \left(\frac{0.44}{\rho}\right)$$

where N = the number of particles/m^2sec of mass M and greater and of density ρ. The velocity of the particles is assumed to range from 10 to 70 km/sec with an average of 40 km/sec. The average density of the particles is 0.4 gm/cm^3. An extreme upper limit for $N_{near\ Mars}$, on the basis of Kessler's equation for asteroidal debris, is:

$$\log N_{AM} = -12.83 - \log M$$

where N_{AM} = the number of micrometeoroid particles/(m^2sec) in the vicinity of Mars of mass M and greater. The velocity of the particles is taken as 20 to 40 km/sec and the average density as 4.37 gm/cm^3.

<u>Near Mars.</u> For the flyby mission the particle flux near Mars is assumed to be:

$$\log N = -13.30 - \log M + 2 \log \left(\frac{0.44}{\rho}\right)$$

where N = the number of particles/(m^2sec) of mass M and greater. The velocity of particles is taken as 10 to 70 km/sec, with an average of 40 km/sec, and the average density as 0.4 gm/cm^3. The upper limit of particle flux is:

$$\log N_A = -12.83 - \log M$$

where N_A = the number of micrometeoroid particles/m^2sec of mass M and greater. The velocity of particles is 20 to 40 km/sec, and the average density is 4.37 gm/cm^3.

In the circular orbit about Mars the particle flux is taken as:

$$\log N \geq -17.50 - 1.70 \log M$$

where N = the number of particles/(m^2sec) of mass M and greater. The relative velocity of particles is assumed as 0 to 5 km/sec and the average density as 0.4 gm/cm^3.

The upper limit is:

$$\log N_A = -9.83 - 1.70 \log M$$

where N_A = the number of micrometeoroid particles/m^2sec of mass M and greater. The relative velocity of the particles is assumed as 0 to 5 km/sec and the average density as 4.37 gm/cm^3.

For the elliptical orbit mission the particle flux is taken as:

$$\log N \gtrapprox - 17.20 - 1.70 \log M$$

where N = the number of particles/(m^2sec) of mass M and greater. The assumed particle velocity is 0 to 5 km/sec and the average particle density is 0.4 gm/cm^3. The upper limit is:

$$\log N_A \approx - 10.13 - 1.70 \log M$$

where N_A = the number of micrometeoroid particles/m^2sec of mass M and greater. The particle velocity is assumed to be 0 to 5 km/sec and the average particle density is 4.37 gm/cm^3.

3.3.18 Design Loads (Phase IA)

a. Static Accelerations

For Phase IA, the arbitrary design loads in Table 2 will be used. The combined longitudinal and lateral static loads will be combined with vibration in the most critical direction. All lateral loads will be considered in the most critical lateral direction. Table 2 also defines the zero-to-peak acceleration input levels for each loading condition. These loads are considered to act at the spacecraft-launch vehicle adapter mechanical interface.

Table 2. Phase IA Static and Vibration Loads

	Static		Vibration		
Condition	Longitudinal	Lateral	Longitudinal O-peak	Lateral O-peak	Torsion O-peak
	g	g	g	g	Rad/sec^2
(1)	6	1	0.8	0.5	0
(2)	2	1	1.2	0.75	60
(3)	1	0	1.6	1.0	60
(4)	0	0	1.6	0	0

b. Low Frequency Vibration

For Phase IA, the vibration acceleration level specified in paragraph 3.3.18a will be used. All vibration inputs will be considered to be discrete transients which may occur at any frequency for the duration shown in Table 3.

Table 3. Length of Vibration Transient

Direction	Vibration Frequency Range (cps) 2.5 - 10	10 - 40	40 - 160
Axial	40 cycles	30 cycles	0.5 sec
Lateral	40 cycles	30 cycles	0.5 sec
Torsion	20 cycles	20 cycles	0.25 sec

c. Random Vibration

For Phase IA, environmental requirements pertaining to shock and random vibration are to be considered when applicable.

4. STRUCTURAL ADEQUACY

Structural adequacy in strength, rigidity, and other desired physical characteristics will be demonstrated by analyses and tests.

4.1 Analyses

4.1.1 Load Analyses

Loads analyses will be performed for all flight and nonflight design conditions.

4.1.2 Stress Analyses

Stress analyses of all structural components will be performed for all critical flight and nonflight conditions. Internal load distributions, structural allowables, margins of safety, and critical deformations will be indicated.

4.1.3 Dynamic Analyses

Dynamic analyses will be performed to determine vibrational modes, frequencies, and structural response of major structural components and the overall spacecraft for all critical flight and nonflight conditions.

4.2 Tests

4.2.1 Structural Load Tests

a. Development Tests

Structural development tests on components and assemblies will be performed to support analysis or to substantiate design adequacy.

b. Proof Tests

Items requiring an acceptance proof test as evidence of satisfactory workmanship and material quality will be tested in accordance with the requirement of paragraph 3.2.6.

c. Type Approval Test

The ability of the spacecraft structure, its components and assemblies, to sustain all critical ultimate design loads and environmental conditions in the manner required will be demonstrated by structural type approval tests utilizing the structural test model.

4.2.2 Vibration Tests

a. Development Tests

Development vibration tests will be conducted on components, assemblies of components, and spacecraft test models as required to support analysis or substantiate design adequacy.

b. Flight Acceptance Vibration Tests

Flight acceptance vibration tests are required to qualify all flight spacecraft prior to launch. These tests will be performed on the PTM and all flight spacecraft. Loads generated during vibration tests to flight acceptance levels are designated as "design loads." Satisfactory performance under flight acceptance vibration shall conform to the requirements of paragraph 3.2.2.

c. Type Approval Vibration Tests

Type approval vibration tests are required to demonstrate the structural adequacy of the spacecraft. Type approval as well as flight acceptance vibration tests are performed on the proof test model. Loads generated during vibration test to type approval levels are designated as "ultimate design loads." Satisfactory performance under type approval vibration testing will conform to the requirements of paragraph 3.2.3, i.e., there be no failure of the structure. Permanent or elastic deformation is not to be taken as evidence of unsatisfactory structural performance unless such deformation contributes directly to equipment malfunction or another subsystem failure. Failure of qualified structure occurring during any subsequent requalification testing of other subsystems is not to be considered evidence of unsatisfactory structural performance.

PYROTECHNICS

VS-4-530

CONTENTS

1. SCOPE

This document presents the design for pyrotechnic devices in the Voyager flight spacecraft. These pyrotechnic devices are utilized for various actuation, disconnection and ignition during the Voyager mission. The subsystem interfaces, design constriants, and safety requirements are discussed for each pyrotechnic function.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

JPL

V-MA-004-001-14-03	Preliminary 1971 Voyager Mission Specification

TRW 1971 Voyager Mission Design Documents

VS-2-110	Design Characteristics and Restraints
VS-3-111	Component Design Parameters
VS-3-130	Spacecraft-Launch Vehicle Interface
VS-4-470	Electrical Distribution
VS-4-570	Launch Vehicle-Spacecraft Separation
VS-4-571	Planet-Oriented Package
VS-4-573	Capsule Adapter and Canister Jettison
VS-4-610	Midcourse Propulsion
VS-4-611	Retropropulsion
VS-4-612	Evasive Maneuver Propulsion

3. FUNCTIONAL DESCRIPTION

3.1 General Description

Each of the spacecraft pyrotechnic subsystems elements contains one or more electroexplosive devices (EED) which are initiated by electrical impulses from the electrical distribution subsystem. The EED produces a pressure impulse, an ignition flame, or a detonating shock wave depending upon the function of the initiated device. It is desirable that initiator subassemblies of all EED be manufactured as a single large production lot to facilitate reliability testing. The initiator subassemblies are to be devided into three lots and assembled with output charges as follow:

a) Low explosive, gas producing output charges to operate pin puller and piston actuated devices

b) Deflagrating output charges to initiate flame producing igniters for solid propellant rockets and gas generators

c) High explosive output charges to initiate high order detonation waves in high explosive bolts or confined detonating fuze.

3.2 Pyrotechnic Operations

3.2.1 Spacecraft-Launch Vehicle Electrical Umbilical Disconnect

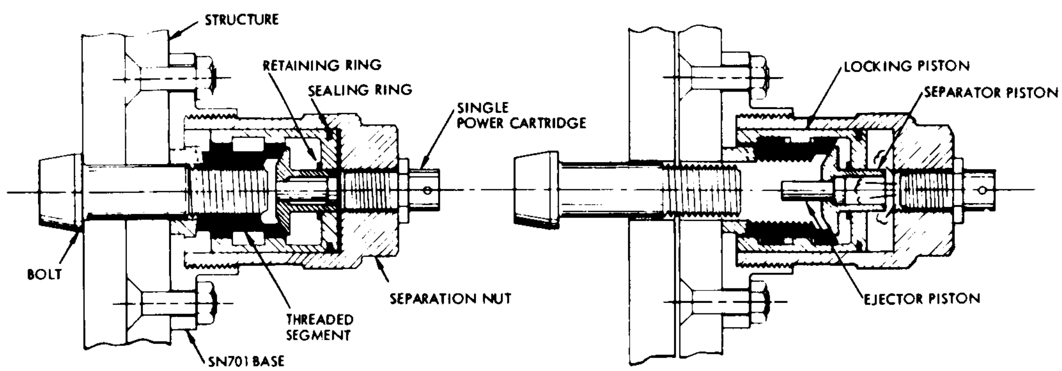
The electrical umbilical connector is rotated to separate it from the spacecraft by action of two explosively actuated pistons. The actuation of the pistons alone or together will disconnect the connector. The simultaneous operation of both pistons will not damage the connector or surrounding structure. The spacecraft-launch vehicle umbilical is disconnected and stowed in keeping with 3.5.1 of VS-3-130 as a part of the prelaunch countdown sequence. This separation is verified prior to liftoff.

3.2.2 Spacecraft-Launch Vehicle Separation

The three nuts which hold the spacecraft on the launch vehicle during boost and injection will be released by action of two electroexplosive devices per nut. The separate initiation of either EED will remove the release nut. Both the nuts and bolts are restrained from flying free. The simultaneous initiation of both EED will not damage adjacent structure. The design of the bolt and release nut device is shown in Figure 1. Information on the spacecraft-launch vehicle separation is presented in VS-4-570.

3.2.3 High Gain and Medium Gain Antenna Deployment

The pin which holds the high gain antenna in the stowed position during boost and injection will be retracted to allow antenna deployment. This retraction is accomplished by the initiation of either or both of two pressure type electroexplosive devices. The deployment release of the medium gain antenna will be done in an identical manner.



INSTALLED **SEPARATED**

1. LOCKING PISTON MOVES FORWARD TO UNLOCK THREADED SEGMENTS
2. SEGMENTS DISPLACE RADIALLY AWAY FROM BOLT
3. SEPARATOR PISTON LOCKS SEGMENTS IN OPEN POSITION
4. EJECTOR PISTON THRUSTS BOLT OUT OF STRUCTURE JOINT

DESIGN FEATURES

1. GAS RETAINING
2. LOW REACTION FORCE TRANSMITTED TO STRUCTURE
3. ULTRA FAST RELEASE TIME DUE TO MINIMUM MOMENT OF PARTS 1.6 MILLISECONDS HAS BEEN OBTAINED
4. POSITIVE BOLT EJECTION
5. NO FRAGMENTATION. ALL PARTS ARE RETAINED AND LOCKED IN PLACE

Figure 1. Bolt and Release Nut Device

3.2.4 Planet-Oriented Package Gimbal Uncaging

The two gimbals supporting the planet-oriented package during boost and injection will be uncaged by the retraction of a restraining pin on each gimbal. Each pin is retracted by the initiation of either or both of two pressure type EED. The planet orientation package is discussed in VS-4-571.

3.2.5 Midcourse Correction Motor Control

The monopropellant midcourse correction motor will be started by the initiation of a normally closed explosively actuated valve. A normally open explosively actuated valve will be initiated to stop the motor. Four parallel normally closed valves and three parallel normally open valves allow three motor burn sequences. A solenoid operated valve provides backup for additional sequences if required. The midcourse propulsion subsystem is described in VS-4-610.

3.2.6 Spacecraft Evasive Maneuver Jet Actuation

The spacecraft will be displaced out of the path of the capsule vehicle by action of a cold gas impulse subsystem after capsule separation. The evasive maneuver will be performed by initiation of a normally-closed explosively actuated valve. This valve is opened by the firing of one or both of two pressure output type EED. The spacecraft evasive maneuver subsystem is described in VS-4-612.

3.2.7 Jettison of Capsule Adapter and Canister

The capsule adapter and canister remaining after capsule vehicle separation will be jettisoned prior to retropropulsion engine ignition. The base will be removed by the action of an explosive charge on each of the three release nuts holding the capsule adapter to the spacecraft. These bolts and release nuts are the same design as those utilized for spacecraft-launch vehicle separation. The explosive charges are fired by confined detonating fuse (CDF). The EED for activating the CDF will be installed in the bottom plane of the spacecraft to provide access for arming after the mating of the planetary vehicle and Centaur. The capsule adapter separation system is discussed in VS-4-573.

3.2.8 Retropropulsion Liquid Injection Thrust Vector Control Gas Generator Ignition

A gas generator will provide pressure for the injection of fluid to achieve thrust vector control of the retropropulsion engine. This gas generator will be started by either of two flame producing EED. These EED will be initiated just before retropropulsion engine ignition. Design of the retropropulsion thrust vector control is given in VS-4-611.

3.2.9 Spacecraft Retropropulsion Engine Ignition

The solid rocket engine for spacecraft retropropulsion will be ignited by either of two flame producing electroexplosive devices. In keeping with safety requirements, the igniter electroexplosive devices will be mechanically and electrically displaced from the firing position during spacecraft final preparations. The igniter will be armed by an electrical command before launch. Design of the retropropulsion motor is given in VS-4-611.

3.3 Summary of Pyrotechnic Devices

The pyrotechnic devices used by the spacecraft are summarized below.

Item	Number Required
Spacecraft Launch Vehicle Umbilical Separation	
Electrical disconnect	1
EED (pressure)	2
Spacecraft Launch Vehicle Separation	
Separation nuts	3
EED (pressure)	6
High Gain and Medium Gain Antenna Deployment	
Pin pullers	2
EED (pressure)	4
Planet-Oriented Package Uncaging	
Pin pullers	2
EED (pressure)	4
Midcourse Propulsion Control	
Normally closed valves	4
Normally open valves	3
EED (pressure)	7
Spacecraft Evasive Maneuver Jet Actuation	
Normally closed valve	1
EED (pressure)	2
Flight Capsule Remaining Adapter and Cannister Jettison	
Separation nuts	3
EED (detonating)	6
CDF leads	6
Retropropulsion LITVC Gas Generator Igniter	
EED (deflagrating)	2
Spacecraft Retropropulsion Motor Igniter	
EED (deflagrating)	2

4. INTERFACES

4.1 Input Interfaces

The firing circuitry and impulse interface with the electrical distribution subsystem is given in VS-4-470.

4.2 Interface with Actuated Elements

All EED are to produce a pressure, ignition, or detonation output as specified. Confined detonating fuses (CDF) produce an explosive output from a donor charge that is integral with the enclosed CDF assembly.

5. PERFORMANCE PARAMETERS

5.1 Pyrotechnic Device Output

Detonating EED posses outputs of sufficient brisance to initiate the next explosive in the train. Therefore, a definite relationship must exist between the donor charge and the acceptor charge weight, density, and standoff to ensure a reliable transfer.

Igniter squib EED are properly sized and tested in igniters to ensure their proper functioning in the motor ignition system.

Pressure-producing EED must possess sufficient charge of a low-shock-producing type to operate specific devices.

5.2 Normal Operating Modes

5.2.1 Pressure Cartridge (EED)

When 5 amperes is passed through the bridgewire, the resulting reaction produces gas at high pressure to pull pins, release bolts, or operate valves.

5.2.2 Igniter Cartridge (EED)

When 5 amperes is passed through the bridgewire, the resulting reation produces flame and hot metal particles to initiate the igniter.

5.2.3 Detonator (EED)

When 5 amperes is passed through the bridgewire, the resulting reaction produces a high-order detonation. This output is used to initiate CDF, or other leads.

5.2.4 CDF

CDF is used to transfer explosive energy from a detonator to release bolts.

5.3 Functional Reliability

All electroexplosive devices require a qualification test of each lot. Part of the qualification consists of an all-fire Bruceton analysis of 60 initiators fired under specific conditions and a no-fire Bruceton analysis of 60 initiators. Test data should be sufficient to show, by statistical analysis with 90 per cent confidence, that at least 99.9 per cent of all initiators have a most pessimistic all-fire current equal to or less than 3.5 amperes per bridgewire, and a most pessimistic no-fire current and power equal to or greater than 1 ampere and 1 watt per bridgewire.

6. CHARACTERISTICS AND CONSTRAINTS

6.1 Design Criteria

Pyrotechnic devices are designed in keeping with 5.5 of VS-2-110.

6.2 Configuration

All EED have a standard squib envelope and match head, and contain two bridgewires.

EED bodies are of one-piece construction with an integral receptacle. They are designed to provide continuous circumferential shielding between the cable and device, and to ensure that the shield circuit is completed before contact is made with the bridge pins.

All fragments and gas produced by EED must be retained in the part in which the EED is installed.

6.3 Firing Circuitry

Redundant firing circuitry is provided. All EED leads are normally shorted together and to ground. Firing circuit conductors and EED leads are twisted to maintain electrical balance and reduce induction. Shielding provides a minimum attenuation of 40 db within the frequency range of 150 kc to 10,000 mc. The shield must be continuous and uninterrupted from the EED to the point at which the leads are shorted together and to ground or, preferable, back to the power supply. The shield must be grounded at each end.

When EED and firing circuit are being joined, the shield circuit must be completed before contact is made with the bridge pins.

The EED must not fire or dud when exposed to the electromagnetic environment specified in the JPL 1971 Voyager Mission Specification.

All EED are capable of withstanding static discharge of 25 kv from a 500-picofarad capacitor applied between the pins, or between pins and case.

6.4 Firing Impulse

To fire EED, a maximum of 22 amperes is permitted to reach a single bridgewire, and a minimum of 5 amperes is applied to the EED for a minimum of 50 msec.

EED are to have the ability to withstand a current of 1 ampere and 1 watt power for 5 minutes on each of the two bridgewires simultaneously without firing.

6.5 Operational Constraints

The primary operating constraint is safety for personnel and mission. Test plugs which simulate EED are installed to enable circuit checkout.

All Category A EED are installed as late as possible in the count-down. This requires installation of all Category A explosive components at the explosive safe facility or on the launch pad. Arming of such devices is accomplished after the spacecraft is on the pad. Access for the arming of Category A devices must be provided at the bottom of the launch vehicle-spacecraft interface and a seal is provided around each arming block.

6.6 Environment

The pyrotechnic devices are to be compatible with the environment of 4.2 of VS-2-110. Allowable temperature limits are given in VS-3-111.

6.7 Weight and Power

Weight and power data is given in VS-3-111.

7. SAFETY CONSIDERATIONS

All testing and handling of EED is to be performed by personnel specifically designated and indoctrinated for such work. The bridgewires (resistance elements) of the electroexplosive devices will be shorted together except during checks. Resistance checks of EED must be made with special approved low-current ohmmeters. The associated circuitry must be checked to verify freedom from AC or DC voltage before connection of EED. Personnel must wear approved safety glasses or face shields when handling or inspecting electroexplosive devices. Personnel and test equipment must be properly grounded to spacecraft during operations involving EED.

ELECTRONIC EQUIPMENT PACKAGING

VS-4-550

CONTENTS

1. SCOPE

This document presents the packaging approach for the 1971 Voyager spacecraft. It gives detailed layout data for the spacecraft equipment panels.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW Voyager Spacecraft Design Documents

VS-2-110	Design Characteristics and Restraints
VS-4-470	Electrical Distribution

3. DEFINITIONS

3.1 Electronic Equipment Packaging

Electronic equipment packaging covers mechanical design of electronic and electromechanical devices and includes application of components, selection of materials and processes, equipment thermal and structural design, and fabrication techniques.

3.2 Subsystem

A subsystem is a functional portion of the system, such as the communication subsystem or power subsystem.

3.3 Assembly

A functional portion of a subsystem which is physically removable from the spacecraft as an single integral item is an assembly. Electrical connections within the assembly are internal to the assembly and do not, in general, use external jumpers and cabling.

3.4 Subassembly

Functionally and physically separable portions of an assembly which can be individually tested are subassemblies.

3.5 Modules

Modules are a series of components physically combined to form a single circuit function, such as a flip-flop, gate, etc. These are generally encased for protection with only input and output connections exposed.

3.6 Parts

A part corresponds to an item such as a resistor, capacitor, transistor, etc.

3.7 Interconnections

All electrical interconnections from the part level to the system harness come under the term "interconnections." This includes bonding techniques for component leads, printed circuit boards, connectors, etc.

4. DESIGN RESTRAINTS

4.1 Standard Package Form Factor

Panel-mounted electronic equipment is packaged so that external dimensions and mounting provisions correspond to the outline and mounting drawing shown in Figure 1. The standard dimensions are 6 inches wide by 6 inches high, while length may vary up to a maximum of 50 inches. The width dimension is fixed by spacing between the mounting rails and can only be varied by eliminating a rail and spanning across two rows. The width in this case is 15 inches for two side rows, or 13.5 inches for two center rows. The height dimension is restricted to a maximum of 6 inches near the engine but at some 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 the gyro assembly, need not conform to the standard package size.

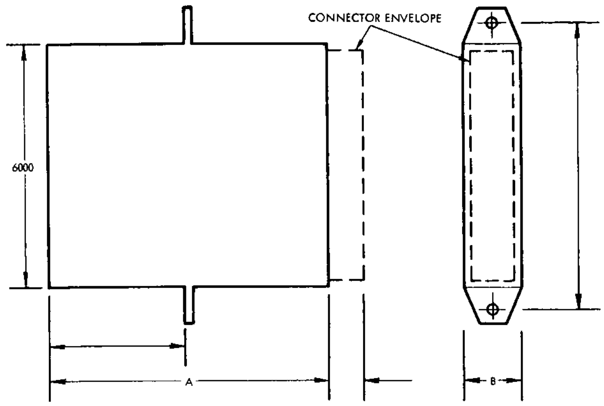


Figure 1. Outline and Mounting

4.2 Spacecraft Arrangement

Most of the electronic equipment assemblies are mounted to panels which form the six facets of the hexagonal spacecraft. The following constraints are considered when installing this equipment in the spacecraft:

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

b) Placement of heat dissipating equipment for as uniform as possible a power density distribution

c) Placement of equipment for proper spacecraft mass distribution

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

Equipment panels are hinged along the aft edge to facilitate access to installed equipment. Cable loops are provided between the system ring harness and the individual assembly harnesses, to permit retention of the panels about the hinges without undue strain and flexing of the harness.

4.3 Structural

Equipment mounting panels are stiffened by rails as shown in Figure 2. These rails incorporate positive mounting provisions for the assemblies. The assemblies are 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 lowest mode vibration resonance of the panels with all equipment installed is estimated to be 145 cps, and the lowest primary vibration resonance within an assembly is designed to be above 400 cps.

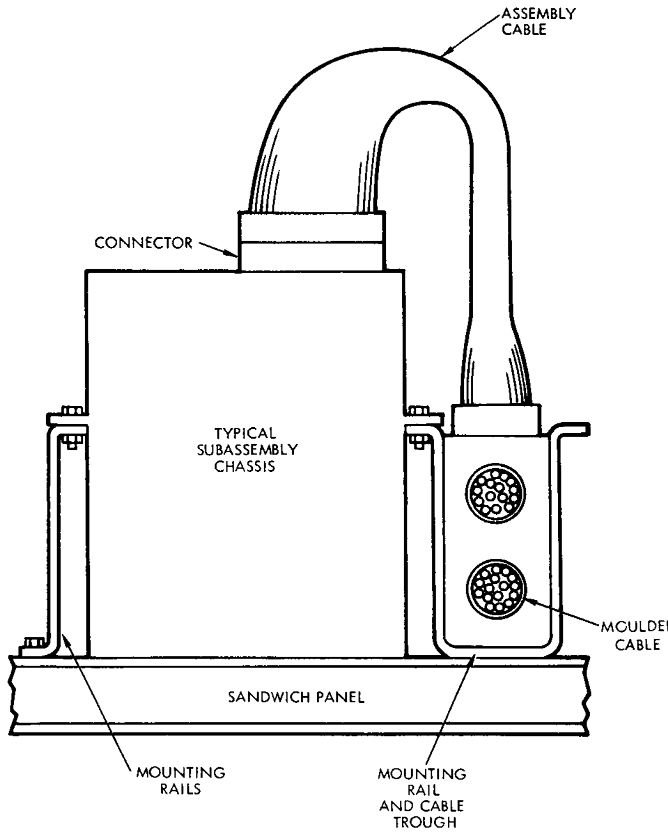


Figure 2. Mounting Equipment

4.4 Thermal

Assemblies are mounted to the rails with positive dimensional tolerance allowed to assure intimate contact between the base of the assembly and the panel.

A thin coating of an elastomeric filler material (such as RTV-11) is used for equipment possessing a base area average power density greater than 0.2 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 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 an assembly is employed only on gyros and other devices where temperatures must be maintained within a very close tolerance.

Part temperatures do not exceed derated temperatures according to 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.

4.5 Weights

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

4.6 Electrical

Design criteria for electrical interface grounding and cabling are presented in VS-2-110 and VS-4-470.

4.6.1 RF Shielding

Packages containing RF circuitry are 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.

4.6.2 Magnetic Interference

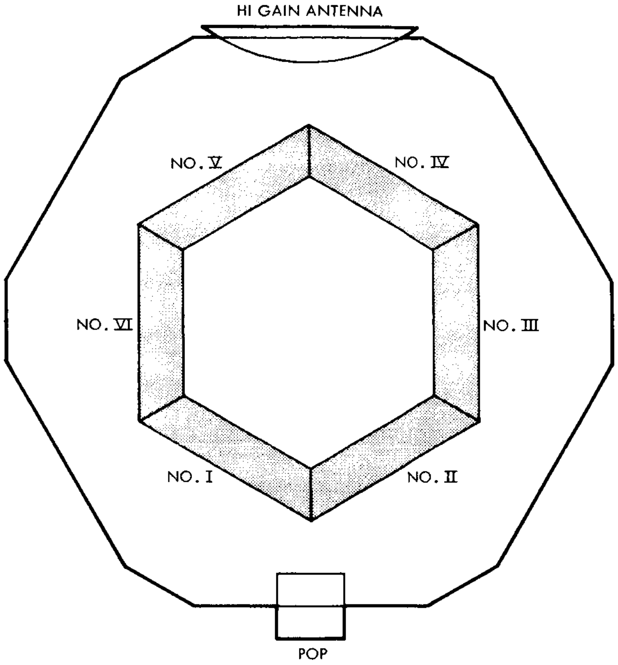
Shielded or twisted wires are used to satisfy spacecraft magnetic field requirements.

4.7 Access for Checkout

The ability to service assemblies during systems checkout is provided.

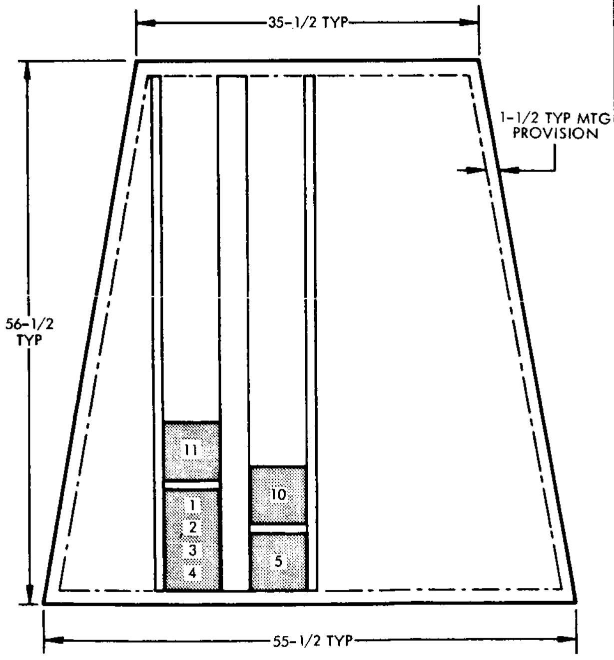
5. EQUIPMENT PANEL LAYOUTS AND TABULATIONS

Figures 3 through 7 identify equipment and related parameters and show equipment panel layouts.



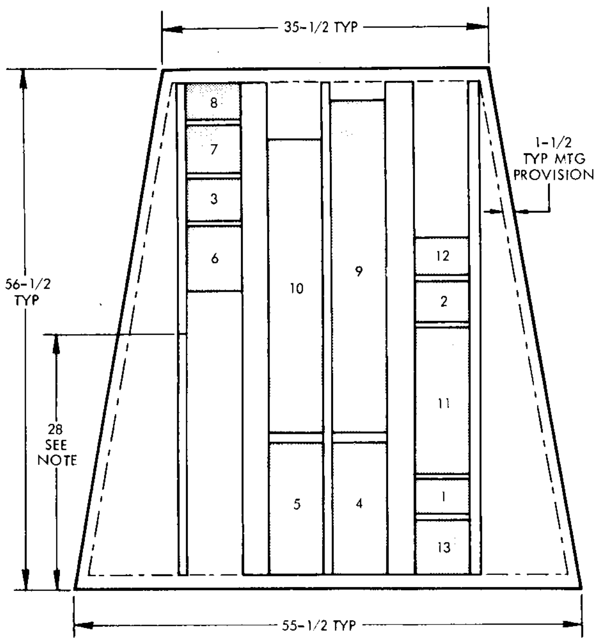
PANEL NO.	TITLE	EQUIPMENT WEIGHT	FIGURE NO.
I	BLANK	—	
II	BLANK	—	
III	STABILIZATION AND CONTROL	40.9 LB	1
IV	BUS EXPERIMENTS	123.0	2
V	POWER	124.1	3
VI	COMMUNICATION	102.2	4
	TOTAL	390.2 LB	

Figure 3. Voyager Flight Spacecraft Panel Arrangement



	ITEM	QUANTITY	SIZE EACH	WEIGHT/UNIT	TOTAL WEIGHT
COMMAND SYSTEM	1. INPUT DECODER	2	6x6x10.56	1.0 LB	26.6
	2. COMMAND DECODER	2		1.0	
	3. SEQUENCER	2		7.5	
	4. POWER	2		3.8	
	5. J-BOX	1	6x6x6		
	6.				
	7.				
	8.				
	9.				
ATTITUDE CONTROL	10. CONTROL ELECTRONIC	1	6x6x6	9.0	9.0
	11. DRIVE ELECTRONICS ANTENNAS AND POP	1	6x6x4	5.3	5.3
				TOTAL	40.9 LB

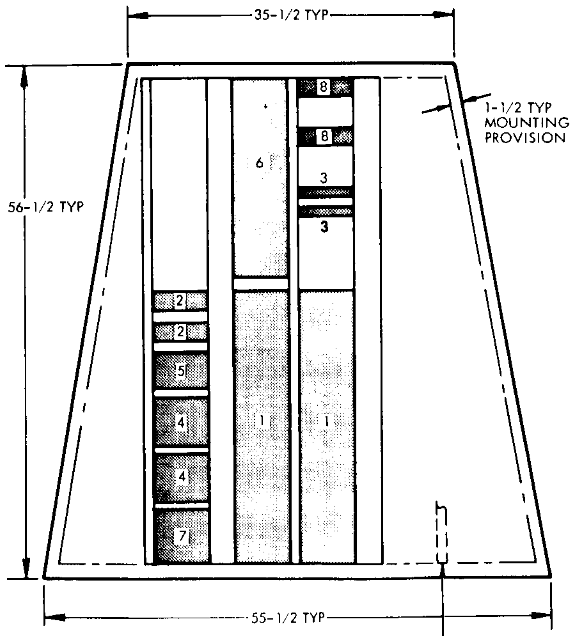
Figure 4. Stabilization and Control and Central Sequencing and Command



NOTE: RAIL CUT AWAY TO CLEAR FUEL BOTTLE.
DOUBLER ADDED TO RAIL TO RETAIN STIFFNESS

ITEM	QUANTITY	SIZE EACH	WEIGHT/ UNIT	TOTAL WEIGHT
1. METEOROID IMPACT (4 SENSORS)	1	6x6x4	10 LB	10 LB
2. MAGNETOMETER	1	6x6x4.47	5	5
3. PLASMA (2 SENSORS)	1	6x6x4.47	5.5	5.5
4. COSMIC RAY (4 SENSORS)	1	6x6x14.25	5	5
5. TRAPPED RADIATION	1	6x6x14.25	12.5	12.5
6. IONOSPHERE EXPERIMENT	1	6x6x7	6	6
7. METEOROID FLASH	1	6x6x5.34	5	5
8. IR SPECTROMETER	1	6x6x4	2	2
9. DATA AUTOMATION EQUIPMENT	1	6x6x36	57	57
10. TV EXPERIMENT	1	6x6x32	16	16
11. UV SPECTROMETER	1	6x6x16	7	7
12. SCAN RADIOMETER	1	6x6x4	2	2
13. J-BOX	1	6x6x6		
			TOTAL	123.0 LB

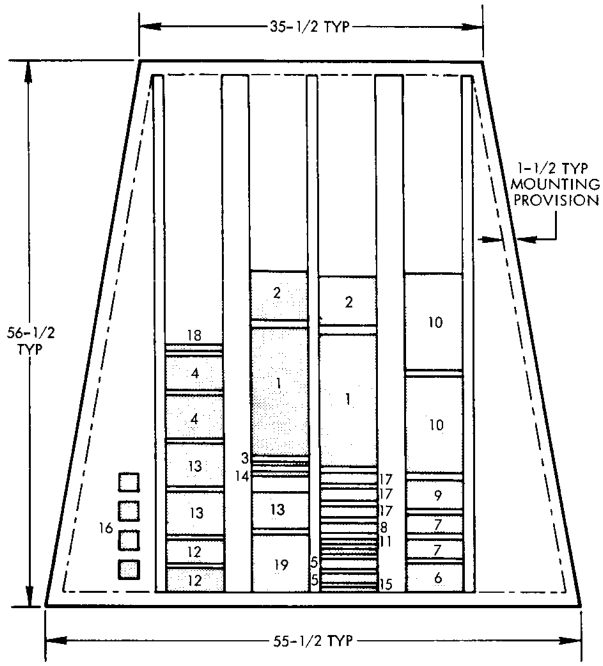
Figure 5. Science



NOTE: RAIL ELIMINATED TO CLEAR FUEL BOTTLE

ITEM	QUANTITY	SIZE EACH	WEIGHT/ UNIT	TOTAL WEIGHT
1. BATTERIES	2	6x4x30	40.2 LB	80.4 LB
2. INVERTER 300W 4.1KC	2	6x6x2	3.5	7
3. INVERTER 20W 820 CYCLE	2	6x6x1	2	4
4. BATTERY REGULATOR	2	6x6x5.34	5.2	10.4
5. PCU	1	6x6x4.34	6.3	6.3
6. SHUNT ELEMENT ASSEMBLY	1	6x6x20	10	10
7. J-BOX	1	6x6x6		
8. INVERTER 400W	2	6x6x2	3	6
			TOTAL	124.1 LB

Figure 6. Electrical Power



ITEM	QUANTITY	SIZE EACH	WEIGHT/ UNIT	TOTAL WEIGHT
1. POWER AMPLIFIER 20W	2	6x6x14	2 LB	4 LB
2. POWER SUPPLY 20W	2	6x6x6	5.5	11
3. PREAMPLIFIER (VHF)	1	6x6x0.5	0.4	0.4
4. VHF RECEIVER	2	6x6x3.5	2	4
5. COMMAND DEMODULATOR	2	6x6x1	2	4
6. LOW POWER TRANSMITTER 1 WATT	1	6x6x3	2	4
7. DIGITAL T/M UNIT AND COMBINER MODULATOR	2	6x6x2	3	6
8. SIGNAL CONDITIONER	1	6x6x1	2	2
9. BUFFER (T/M)	1	6x6x2.84	4	4
10. BULK STORAGE UNIT	2	6x9x9	15	30
11. CAPSULE DEMODULATOR	2	6x6x0.5	0.4	0.8
12. MODULATOR EXCITOR	2	6x6x2.5	3	6
13. S-BAND RECEIVER	3	6x6x4.5	5	15
14. RECEIVER SELECTOR	1	6x6x0.5	0.8	0.8
15. TRANSMITTER SELECTOR	1	6x6x0.5	0.8	0.8
16. CIRCULATOR SWITCH	4	2x2x0.6	1.6	6.4
17. DIPLEXER	3	6x6x1.13	0.8	2.4
18. 4 PORT POWER DIVIDER	1	6x6x0.5	0.6	0.6
19. J-BOX	1	6x6x6		
			TOTAL	102.2

Figure 7. Telecommunications

LAUNCH VEHICLE-SPACECRAFT SEPARATION

VS-4-570

CONTENTS

1. SCOPE

This document presents the design for the launch vehicle-spacecraft separation mechanism. This equipment releases the spacecraft from the launch vehicle upon receipt of an appropriate signal.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-2-110	Design Characteristics and Restraints
VS-3-111	Components Design Parameters
VS-3-130	Spacecraft-Launch Vehicle Interface
VS-4-520	Structure
VS-4-530	Pyrotechnics

3. FUNCTIONAL DESCRIPTION

The launch vehicle-spacecraft separation utilizes three 1/2-inch bolts with release nuts whose design is presented in VS-4-530. The spacecraft and launch vehicle are attached by these bolts at the interface plane as shown in Figure 1. This attachment is accomplished on the launch pad in the manner of a field joint. The joint supports the imposed structural loads with the separation bolts taking tension as required. Shear loads across the joint are taken by three shear pins, with the separation bolts in oversized holes to isolate them from such loads.

At the appropriate time in flight the launch vehicle initiates the separation process as described in 3.12 of VS-3-130. An electrical signal from the launch vehicle initiates the electroexplosive devices (EED) which in turn detach the release nuts. Physical separation of the spacecraft and the launch vehicle is caused by firing retrorockets located on the launch vehicle. The bolts and detached release nuts are contained so as not to fly free.

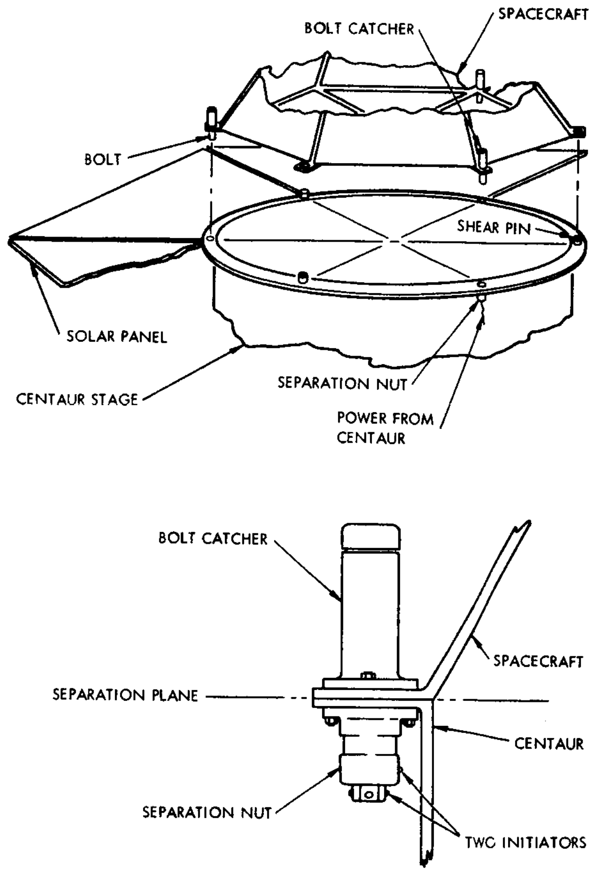


Figure 1. Launch Vehicle-Spacecraft Separation Arrangement

4. INTERFACE ELEMENTS

4.1 Mechanical

The mating surfaces of the Centaur stage and the spacecraft stage are mechanically attached with separation bolts and release nuts as shown in Figure 1 of VS-4-520.

4.2 Electrical

Input separation signals and power come from the Centaur. There is a separate electrical interface in the form of an umbilical between the spacecraft and the Centaur. This utilizes an electroexplosive actuated connector that is disconnected prior to launch and hence is not part of the separation arrangement. Electrical connections to the separation release devices are all on the Centaur side of the separation plane and remain with the Centaur stage after separation. Two electrical inputs are provided for redundant ignition of each of the explosive actuated release nuts.

5. PERFORMANCE PARAMETERS

The separation bolts serve as attachment devices and must hold the spacecraft to the launch vehicle in such a manner as to ensure that the spacecraft survives the launch environment, shroud deployment, and eventual separation.

The rate of separation and angle rate are controlled by the Centaur stage. The release points are made to detach within 10 milliseconds of one another.

Heat output of the release devices is to be in the order of 500 calories for each device.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Design

The separation mechanism is designed to achieve high reliability. The pyrotechnic elements are designed in keeping with 5.5 of VS-2-110 and VS-4-530.

6.2 Environment

The separation mechanism is designed to be compatible with the environment of 4.2 of VS-2-110. The thermal limits are given in VS-3-111.

6.3 Reaction Products

Electroexplosive reaction products are contained within the pyrotechnic devices.

6.4 RF Requirements

The EED must not fire or dud when exposed to the specified electromagnetic environment. All EED leads are normally shorted together and to ground. Firing circuit conductors and EED leads are twisted to maintain electrical balance and reduce induction. Shielding is used to provide a minimum attenuation of 40 db within the frequency range of 150 kc to 10,000 MC. The shield is continuous and uninterruped from the EED to the point at which the leads are shorted together and to ground or, preferably, back to the power supply. The shield is grounded at each end.

6.5 Operations

The spacecraft is encapsulated in the shroud with a barrier across the bottom of the spacecraft when installed on the Centaur at the launch pad. Provision is made to have the attach bolts extend through the barrier. They are prevented from being pushed upward into the bolt catchers. In addition, means are provided for preventing the bolts from turning when nuts are turned. All work of attaching the spacecraft is to be done on the launch vehicle side of the separation plane.

6.6 Weight

Weights data is given in VS-3-111.

7. SAFETY CONSIDERATIONS

The separation design is compatible with 6.9 of VS-1-110.

PLANET-ORIENTED PACKAGE

VS-4-571

CONTENTS

1. SCOPE

This document presents the design of the planet-oriented package. This unit provides a means of controlled pointing for science instruments that require articulation with respect to the spacecraft.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-2-110	Design Characteristics and Restraints
VS-3-111	Components Design Parameters
VS-3-120	Reliability Design Objectives
VS-4-460	Electrical Power
VS-4-510	Thermal Control
VS-4-520	Structure

3. FUNCTIONAL DESCRIPTION

3.1 General

The planet-oriented package (POP) is mounted on an electrically driven two-axis gimbal system. The gimbal system is mounted to the spacecraft as illustrated in Figure 1, and requires no deployment mechanism. Pointing of the package is accomplished by programmed commands from the CS&C based on pointing data received via ground command. Control and sequencing of the science instruments is accomplished by the DAE. All high speed components of the drive mechanisms are sealed and pressurized.

The POP provides a means for mounting science experiment equipment (SEE) in a controlled environment and for pointing the viewing axis of this package as required. The design of the POP provides precision pointing in both a zero g and one g environment without realignment or compensation. The drives prevent platform rotation in a 3-g field with an equipment weight of 100 pounds.

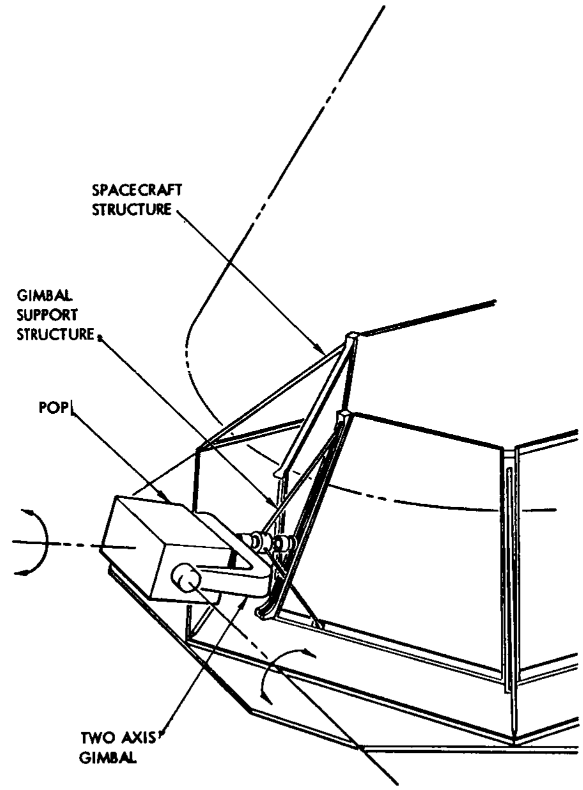


Figure 1. Gimbal System Mounted to Spacecraft

3.2 Modes of Operation

The planet-oriented package provides three modes of pointing control.

- Mode 1 - Strip Mapping. In the strip mapping mode, the POP is pointed by commanding the required gimbal angles. The surface velocity resulting from orbital motion provides picture separation. The camera shutter sequence is set to correspond with the field of view and image velocity.

- Mode 2 - Closed-loop pointing. The closed-loop pointing mode uses horizon scanners to provide POP error signals. This mode is used for strip mapping and for general pointing of instruments other than the cameras. To enable pointing at other than the center of Mars, a variable bias is added to the outputs of the horizon scanners.

- Mode 3 - Semirandom photography mode. In this mode strips of three of more pictures may be taken at up to four arbitrary points on the visible surface. This is accomplished by using the strip mapping mode with command angles and shutter times adjusted to allow sufficient slew times to arrive at the successive pointing angles.

3.3 Drive Electronics

The drive electronics for the POP is dictated by the accuracy requirements and the type of shaft encoder. A single channel electronics block diagram is shown in Figure 2. Commands from the spacecraft sequencer are fed serially into a register which indicates the commanded angle relative to the spacecraft. A similar register holds the actual gimbal angle based on pulses from a digital shaft position encoder. The logic compares these registers and generates a DC error signal to drive

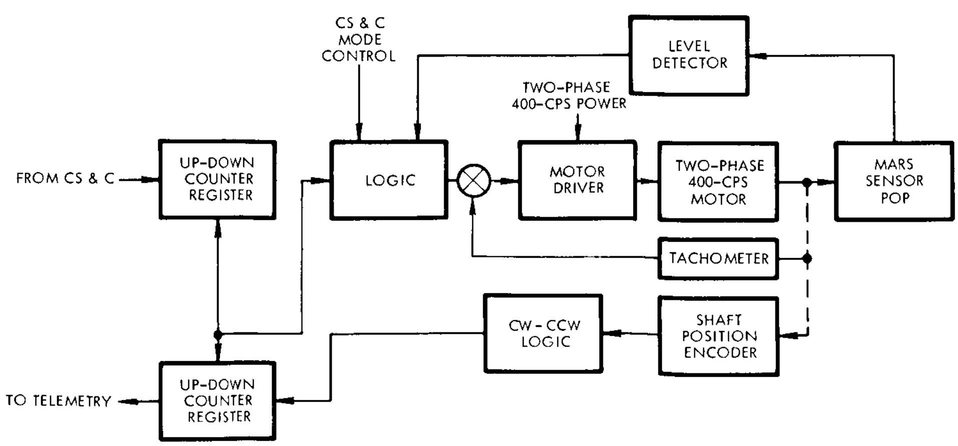


Figure 2. Planet-Oriented Package Drive Electronics Block Diagram

the motor. The logic also performs the mode switching to enable use of the horizon scanners. In the closed-loop pointing mode the offset bias is placed in the command register and added to the scanner output. In this case the logic ignores the angle register.

3.4 Gimbal Caging Mechanism

To prevent damage to the POP during launch, the gimbal is caged until the shroud has been separated. The caging mechanism consists of a pin inserted into a slot located close to the POP trunnions. To uncage the gimbal, the pin is extracted by means of a squib-operated actuator. The pin and actuator are integrated into a single assembly equipped with dual charges. The assembly is sealed to prevent the escape of contaminating gases.

4. INTERFACE DEFINITION

4.1 Mounting

The POP is mounted and supported by the structure as shown by Figure 1 of VS-4-520. The structure for mounting the POP drive shaft allows a limited amount of adjustment both at the bearing mount and the drive mount. This insures proper alignment of the POP electrical zero with the mechanical zero.

4.2 Electrical Power

Electrical power of 410 cps is provided by the power subsystem as described in VS-4-460.

4.3 Data Automation Equipment

The DAE commands television shutter settings and exposure times and controls operations. It sends calibration and operation signals to the POP mounted science instruments. Table 1 lists these commands.

4.4 Central Sequencer and Command Interfaces

The CS&C sends gimbal angle commands to the POP gimbal drives and provies mode control discretes.

4.5 Science Equipment Interfaces

The POP configuration with a reference set of science instruments installed is shown in Figure 3. The structure provides rigid support for the installed science equipment. All instruments are installed on a honeycomb center shelf using top and bottom mounting. Access covers are provided in the top, bottom, and rear of the compartments. To avoid feed-through bolts which could interfere with equipment on the opposite face of the shelf, the instruments are mounted to inserts bonded into the honeycomb. The shelf method of mounting provides a means of attaching the science payload with a maximum frontal area for experiment sensors. Electronics components that do not require frontal area are mounted in the rear sections of either the upper or lower compartments.

Table 1. DAE Commands to the POP

Functions	Number of Separate Functions	Remarks
Science Power Commands	6 to 10	Power commands required to turn experiments on and off
Science Settings and Calibration Commands	20 to 25	Required for calibration and mode control
Camera Shutter Actuation	1	
Magnification	2	
Filter Selection	4	
Image Motion Compensation	2	
Exposure Time	3	
Timing Pulses	2 to 3	Required to function experiment operation such as automatic calibrates, etc.
Shift Pulses		Required to shift data out of science shift registers

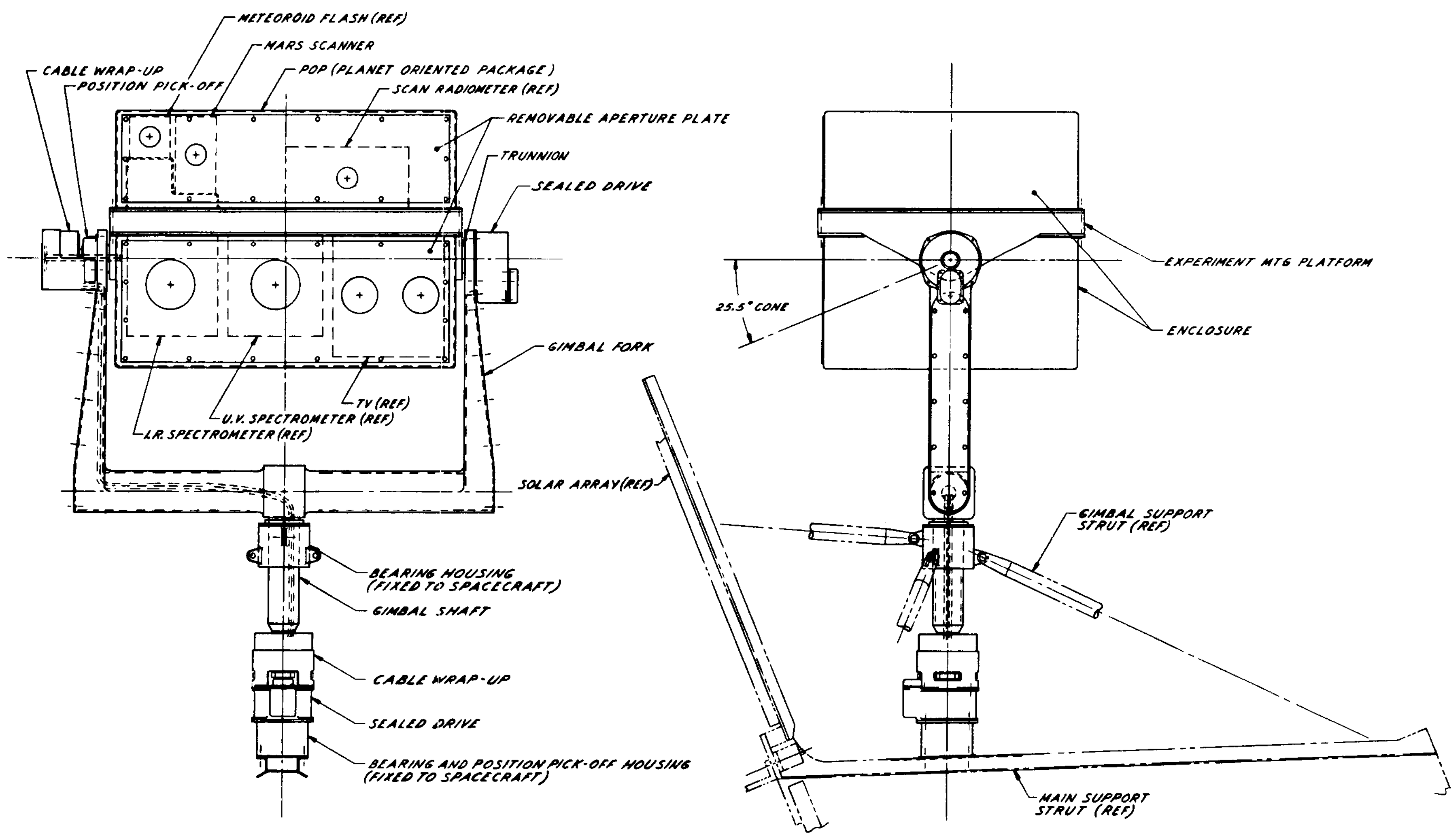


Figure 3. Planet-Oriented Package Structure and Equipment Layout

5. PERFORMANCE PARAMETERS

5.1 Gimbal Drive Parameters

POP gimbal drive design requirements are as follows:

Pointing accuracy relative to spacecraft, ± 0.25 degree

Angular tracking rate, 3 mr/sec

Slew rate, 10 mr/sec

Angular acceleration, 0.6 mr/sec^2

Angular freedom:

Shaft axis, ± 180 degrees

Package axis, ± 135 degrees

Allowable mass unbalance - 8 ft-lb without tumbling

Stall conditions: the drive must withstand stalled conditions without internal damage.

5.2 Orientation Performance

5.2.1 Pointing Accuracy

The POP pointing requirement is ± 0.5 degree, 3σ (each axis). Including attitude control system errors, the primary error sources and their magnitudes for open-loop pointing are

Attitude reference accuracy	± 0.1 degree, 3σ
Limit cycle error	± 0.249 degree, 3σ
Drive accuracy	± 0.25 degree, 3σ
POP pointing sensing error	± 0.2 degree, 3σ
Camera boresight error	± 0.05 degree, 3σ
POP alignment to spacecraft	± 0.25 degree, 3σ

The RSS total is ± 0.49 degree, 3σ.

5.2.2 Stability

The dynamic stability of the POP pointing relative to inertial space is 10^{-4} rad/sec.

5.3 Reliability

Reliability data for the POP is given in VS-3-120.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

The planet-oriented package consist of a science payload container, gimbal drive and associated electronics, and a two-axis Mars horizon scanner. These elements are described briefly below.

6.1 Payload Container

The payload container provides a means of mounting the science instruments that require articulation with respect to the spacecraft. The container provides 8 cubic feet for mounting the science payload with a 6 square foot viewing area perpendicular to the package viewing axis. The structure provides a rigid means of mounting the instruments to insure alignment integrity. The basic structure contains a honeycomb center shelf to provide support for all instruments using both top and bottom mounting. Access covers are provided in the top, bottom, and rear compartment. The instruments are mounted using inserts bonded into the central shelf honeycomb. See Figure 4 for the structure and equipment layout.

6.2 Gimbal Drives

The gimbal arrangement consists of a fork mounted to a rotating shaft. The POP is located within the fork and is supported by bearings contained in the end of the fork prongs. The shaft is supported by bearings contained in two housings which are fixed to the spacecraft structure. The gimbal provides two axes of rotation for the planet-oriented package.

The POP gimbal is operated by sealed wobble gear drives similar to those developed for the OGO solar array drives. All rolling and sliding surfaces of the motor, gearhead, associated gears, and ball

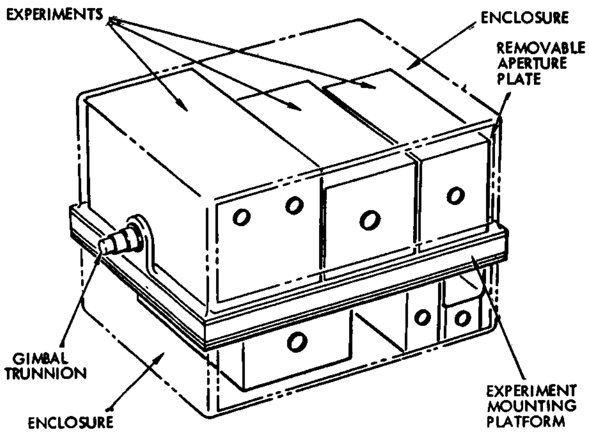


Figure 4. Equipment Layout

bearings are hermetically sealed in pressurized inert gas for protection against the effects of space vacuum environment, except for one pair of gears and one ball bearing pair specifically designed to operate at low speed in vacuum. These parts are fabricated of special materials and employ solid lubricate coatings not subject to cold-welding or sublimation. The sealed lubricants are radiation-resistant.

Sealing is accomplished by two bellows which are installed between the nonmoving parts of the mechanism and the driving gear. The unique feature of the drive mechanism is the use of a pair of specially cut wobble gears for the output stage. Action of the bearing carrier and a titled bearing internal to the unit produces a nonrotating conical nutation (or wobble) motion of the driving gear at the end of the main bellows. This motion causes rotation of the output gear and shaft by sequential engagement of a limited number of gear teeth.

6.3 Cable Wrap-Up

Two cable wrap-up assemblies are required to accommodate wires from the POP. Each assembly consists of approximately 100 wires grouped together in flat ribbons and curled between two discs. A cover is provided to shield the wires from direct exposure to space environment.

6.4 Configuration and Coverage

The POP shaft permits ± 180 degrees of rotation and the fork permits ± 135 degrees to provide coverage of somewhat more than a hemisphere, as limited by the spacecraft body or solar panels. The shaft is supported by bearings contained in two housings which are fixed to the spacecraft structure. The shaft angle is set a 22.5 degrees relative to the solar panel which places the shaft nearly normal to the orbit plane when the spacecraft is in the sun-Canopus attitude. This arrangement permits viewing any point on the visible surface of Mars from any point of the orbit. For most viewing conditions the field of view is perpendicular to the shaft axis, thus requiring near zero rotation about the fork mounted axis.

6.5 Instrument Mounting

The instruments with the largest fields of view are to be mounted on the side of the compartment away from the solar array. This allows the largest possible gimbal rotation before obstruction by the solar panel. Also, it is desirable to locate the television camera closest to the array so as to maximize aperture shading. This is desirable as the TV is not expected to be used when the POP is pointing in the general direction of the sun.

6.6 Environment

The POP is compatible with the environment of 4.2 in VS-2-110. Thermal control is accomplished as described in VS-4-510. Allowable temperature limits are given in VS-3-111.

6.7 Weight and Power

Weight and power data are given in VS-3-111.

7. SAFETY CONSIDERATIONS

The POP subsystem is to be designed with factors of safety so that high voltage circuits, such as those required for the Mars horizon scanner and the science payload, will be shielded to provide no hazard to personnel.

MAGNETOMETER DEPLOYMENT BOOMS

VS-4-572

CONTENTS

1. SCOPE

This document presents the design for magnetometer deployment booms which position magnetometers so as to reduce spacecraft magnetic effects at the instruments as required.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-2-110	Design Characteristics and Restraints
VS-3-110	Configuration
VS-3-111	Components Design Parameters
VS-3-120	Reliability Design Objectives

3. FUNCTIONAL DESCRIPTION

3.1 Functions

The magnetometer booms provide the following functions:

a) Hold the magnetometer sensors in the proper spatial relationship to the spacecraft during the mission as required.

b) Provide the necessary support functions required by the magnetometers.

3.2 Description

An extendible boom is provided to position the magnetometer used to obtain measurements during the transit phase, when the spacecraft magnetic effects at the magnetometer must be held to a minimum. A fixed boom is used for measurements in orbit about Mars when greater spacecraft magnetic effects can be tolerated at the sensor. The boom configuration is shown in Figure 1.

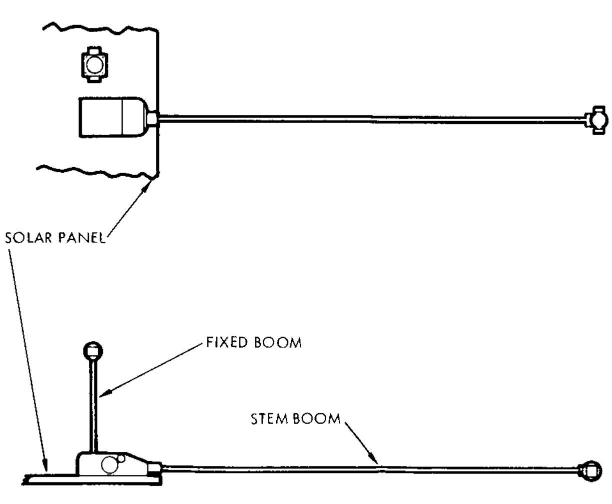


Figure 1. Magnetometer Boom Configuration

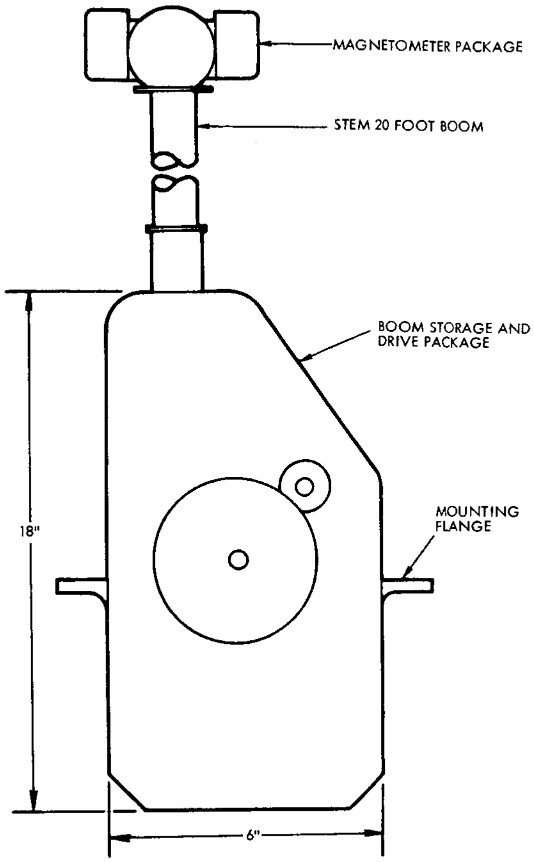


Figure 2. Dehavilland Stem Type Boom

3.2.1 Extendible Boom

As shown in Figure 2, a model A-32 storable tubular extendible member (STEM) unit or equivalent is utilized. This unit is manufactured by the Dehavilland Aircraft Company of Canada specifically for a magnetometer boom application. The boom is a 1-inch-diameter tubular element, formed from several layers of beryllium copper strips of various lengths. When retracted, the element is stored in a strained, flattened condition by winding it onto a drum. As the circular element is retracted, it is smoothly transformed into the flattened condition by passing it through a suitable guiding system. The element can be extended to any length up to 20 feet or restricted by rotating the drum in the appropriate direction. The unit is powered by an electric motor and extension or retraction is achieved by controlling the time during which power is applied. The lead wires to the sensor are deployed through the tube by a system which is an integral part of the STEM unit.

The STEM unit is mounted on the edge of the spacecraft solar array. At the start of cruise, a command is given to extend the boom. The extended boom is capable of surviving the retropropulsion maneuver environment and can be left in that configuration for orbital operations. Alternatively, it can be retracted to achieve improved prediction of the magnetometer orientation relative to the spacecraft.

3.2.2 Fixed Boom

A second magnetometer is mounted on an 18-inch-long boom. This boom is permanently mounted as near the outer edge of the solar array as possible, with the axis parallel to the spacecraft roll axis. The sensor lead wires pass through the centerline of the bottom.

4. INTERFACE DEFINITION

4.1 Mechanical

Installation of the booms on the spacecraft is shown in Figure 3 of VS-3-110.

4.2 Electrical

Electrical cabling and connectors are to be provided for the magnetometer as required. The preliminary design calls for twelve 22-gage wires. Extension of the STEM unit requires power under command of the CS&C.

5. PERFORMANCE

5.1 Deployment Rate

The maximum time for the STEM to deploy will not exceed 2 minutes.

5.2 Dynamic Amplification

The dynamic amplification factor on either boom prior to the deployment will not exceed 2.0.

5.3 Reliability

Reliability data is given in VS-3-120.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Sensor Characteristics

The booms allow for the following sensor characteristics:

Size	6" x 3" diameter
Weight	1.3 pound
Power dissipated	1.8 watt

6.2 Sensor Orientation

The orientation relative to the spacecraft of the magnetometer or the extended boom is predictable within 3 degrees. The corresponding number for the fixed boom is 1 degree.

6.3 Environment

The booms are compatible with the environment of 4.2 in VS-2-110. The temperature limits are given in VS-3-111. The extendible boom is designed to withstand the retropropulsion maneuver in the extended configuration.

6.4 Life

The STEM subsystem is designed for a minimum life of 75 cycles without degradation of performance.

6.5 Magnetic Materials

Selection of materials is compatible with the allowable magnetic field at the magnetometer.

6.6 Electrical Power

The maximum power required to extend the STEM does not exceed 5 watts. The maximum power required to retract the STEM does not exceed 50 watts.

6.7 Weight

Weight data is given in VS-3-111.

CAPSULE ADAPTER AND CANISTER JETTISON

VS-4-573

CONTENTS

1. SCOPE

This document presents the design for the capsule adapter and canister jettison. At an appropriate time this equipment serves to release and separate the capsule hardware that remains attached to the spacecraft bus after separation of the capsule vehicle.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-1-110	Mission Objectives and Design Criteria
VS-2-110	Design Characteristics and Restraints
VS-3-102	Maneuver and Accuracy Data
VS-3-111	Components Design Parameters
VS-4-520	Structure
VS-4-530	Pyrotechnics

3. FUNCTIONAL DESCRIPTION

The capsule adapter and canister jettison utilizes mechanical springs and three bolts with release nuts whose design is described in VS-4-530. The spacecraft and flight capsule are attached by these separation bolts at the interface plane as shown in Figure 1. This attachment is accomplished during prelaunch activities in the manner of a field joint. The joint supports the imposed structural loads, with the separation bolts taking tension as required. Shear loads across the joint are taken by three shear pins, with the separation bolts in over-sized holes to isolate them from such loads.

At the appropriate time in flight subsequent to separation of the capsule vehicle and the spacecraft evasive maneuver as described in VS-3-102 and prior to firing of the solid propellant motor, it is necessary to jettison the capsule adapter and canister remaining with the spacecraft. Upon electrical signal from central sequencing and command, the necessary electroexplosive devices (EED) are initiated. This results in disconnecting an electrical connector as shown in Figure 1 and the detachment of the three release nuts. Physical separation occurs by means of mechanical springs. The bolts and detached release nuts are contained so as not to fly free.

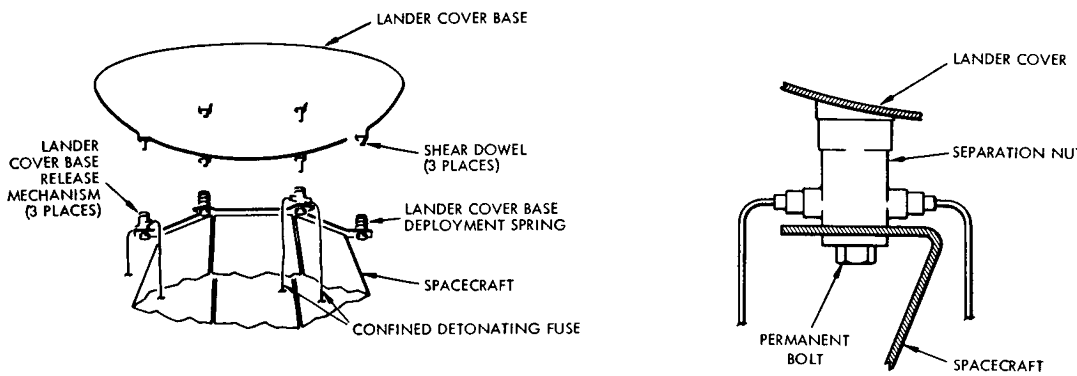


Figure 1. Arrangement for Capsule Adapter and Canister Jettison

4. INTERFACE ELEMENTS

4.1 Mechanical

The mating surfaces of the flight capsule adapter and the spacecraft are mechanically attached with separation bolts and release nuts as shown in Figure 1 of VS-4-520.

4.2 Electrical

The separation signal comes from the spacecraft on command from the CS&C. The in-flight disconnect is an EED actuated connector.

Electrical connections to the separation release devices and the in-flight disconnect are on the spacecraft side of the separation plane and remain with the spacecraft after separation.

5. PERFORMANCE PARAMETERS

The attachment devices hold the spacecraft to the flight capsule in such a manner as to ensure that both units survive the launch environment, shroud deployment, launch vehicle separation, cruise, and capsule separation.

All attach fittings must function with a high degree of simultaneity. The rate of separation and angle rate are controlled by springs.

Heat output of the release devices is of the order to 500 calories for each device.

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Design

The capsule adapter and canister jettison is designed to achieve high reliability. In the event that the capsule vehicle fails to separate, the spacecraft has the capability to jettison the entire flight capsule so as to proceed with its orbital mission. The pyrotechnic elements are designed in accordance with 5.5 of VS-2-110 and VS-4-530.

6.2 Environment

Design is compatible with the environment of 4.2 of VS-2-110. The thermal limits are given in VS-3-111. Electroexplosive reaction products are contained within the pyrotechnic devices.

6.3 Requirements

The EED must not fire or dud when exposed to the specified electromagnetic environment.

All EED leads are to be normally shorted together and to ground. Firing circuit conductors and EED leads are to be twisted to maintain electrical balance and reduce induction. Shielding provides a minimum attenuation of 40 db within the frequency range of 150 kc to 10,000 Mc.

The shield is continuous and uninterrupted from the EED to the point at which the leads are shorted together and to ground or, preferably, back to the power supply. The shield is grounded at each end.

6.4 Operations

The planetary vehicle is to be assembled prior to installation on the launch vehicle. It is placed on the Centaur while encapsulated in the shroud and with a barrier across the bottom of the spacecraft. Provision is to be made to install the capsule adapter and canister jettison initiators in the bottom plane of the spacecraft and to transfer the output by confined detonating fuses to the separation devices at the spacecraft-capsule interface plane. All arming of the spacecraft is to be done on the launch vehicle side of the spacecraft-launch vehicle separation plane.

7. SAFETY CONSIDERATION

The capsule adapter and canister jettison design is compatible with 6.9 of VS-1-110 and with the safety plan described in Volume 3.

MIDCOURSE PROPULSION SUBSYSTEM

VS-4-610

CONTENTS

1. SCOPE

This document presents the design for the midcourse propulsion subsystem for the Voyager spacecraft in the 1971 mission. This subsystem is used to accomplish velocity adjustments at selected times during earth-Mars transit and to effect an orbital velocity correction if required.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-1-110	Mission Objectives and Design Criteria
VS-2-110	Design Characteristics and Restraints
VS-3-111	Components Design Parameters
VS-3-120	Reliability Design Objectives
VS-4-530	Pyrotechnics

3. FUNCTIONAL DESCRIPTION

3.1 Function

The primary function of the midcourse propulsion subsystem (MPS) is to remove or reduce injection dispersion errors so as to achieve the required approach trajectory to Mars. This function is performed during midcourse maneuvers, and the MPS has the capability to carry out any number of such maneuvers. During these maneuvers, the spacecraft is oriented by the stabilization and control subsystem to a prescribed attitude prior to the thrusting period for generating the desired velocity increment. The secondary function of the midcourse propulsion subsystem is to provide orbit trim capability to the orbiting spacecraft subsequent to the retro maneuver. This capability exists to the extent that the propellants are available after accomplishment of the midcourse maneuvers.

3.2 Design Philosophy

The design philosophy for the system is directed toward achieving maximum simplicity at the possible expense of optimum performance. The system attempts to avoid the technique of backing up relatively unreliable components by various redundant arrangements. The consequence of this philosophy is a somewhat heavier but nearly foolproof system.

3.3 System Description

The midcourse propulsion subsystem incorporates a liquid propellant rocket engine and associated feed system as shown schematically in Figure 1. The system consists of two combination gas storage and propellant tanks, propellant flow control valves, and a monopropellant rocket thrust chamber assembly. The propellant is anhydrous hydrazine. The thrust chamber contains a catalyst which is capable of initiating spontaneous decomposition of the hydrazine. Two pressurized propellant tanks are used, one located on each side of the vehicle at a principal diameter. The tanks are 24 inches in diameter and are fabricated from two forged hemispheres of annealed 6Al-4V titanium with all flanges and bosses integrally machined into the tank. The tank halves are joined by welding at a major diameter. The tanks are designed for a 380-psi maximum operating pressure.

The system is designed to provide essentially unlimited space storage life consistent with the environmental constraints imposed by the Voyager missions. Multiple start capability is achieved for the nominal mission by ganged explosive valves. Unlimited start capability is provided as shown in Figure 1 by a solenoid-operated back-up valve which takes over after the explosive valves have been used. The engine is inherently capable of unlimited starts because no external devices are required to initiate decomposition of the hydrazine.

The design philosophy of the system is directed toward minimizing the number of parts and interactions with other subsystems. The operational philosophy of the system is directed toward minimizing preflight handling and spacecraft interactions.

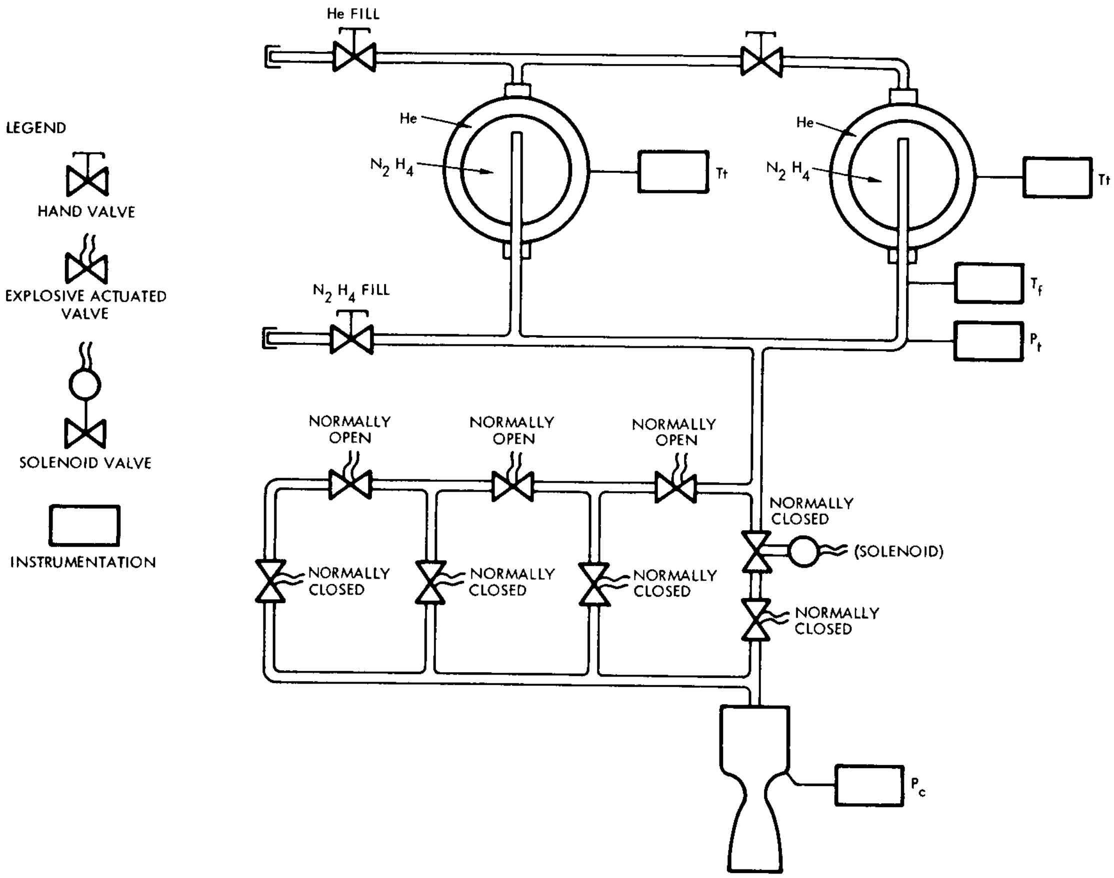


Figure 1. Liquid Propellant Rocket Engine and Associated Feed System

The system is of modular construction capable of being stored in a flight-ready condition and installed in the spacecraft as a self-contained unit (exclusive of the electrical interface). The subsystem in the pressurized and fueled condition meets the factor of safety requirements as in Paragraph 7. 2. No spacecraft umbilicals are required to maintain the subsystem in the "ready" condition.

A preliminary engineering drawing of the rocket engine is given in Figure 2.

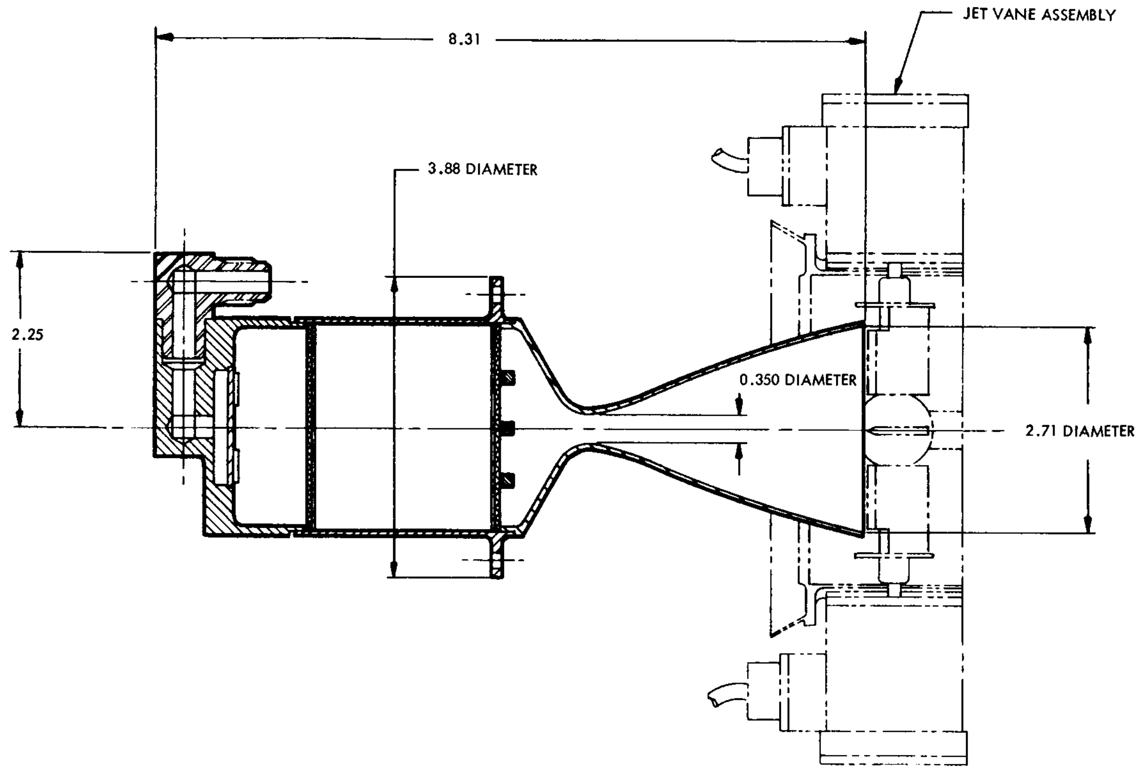


Figure 2. Midcourse Preliminary Rocket Engine

3.4 Flight Operation

The firing of the midcourse propulsion subsystem is controlled by central sequencing and command CS&C which receives time, direction, and duration of the firing through the ground to spacecraft communication link. After the spacecraft attitude has been verified as correct, the system is ready for firing. Ignition of the engine occurs spontaneously following actuation of the appropriate engine normally closed flow control valve. Thrust termination is achieved after operation for a predetermined time increment by an electrical signal from the CS&C, which fires the appropriate normally open propellant valve to the closed position. During the rocket firing, thrust vector control is accomplished by jet vanes which are positioned by actuators in response to stabilization and control commands.

The three sets of explosive valves are adequate to perform an anticipated nominal mission of two midcourse corrections plus one orbit trim correction. The solenoid valve provides an additional capability in the instance that any of the explosive valves fail to actuate or that the mission requires more than the nominal number of velocity corrections.

4. INTERFACE DEFINITION

4.1 Mechanical

Installation of the subsystem and its allowable envelope are shown in Figure 3.

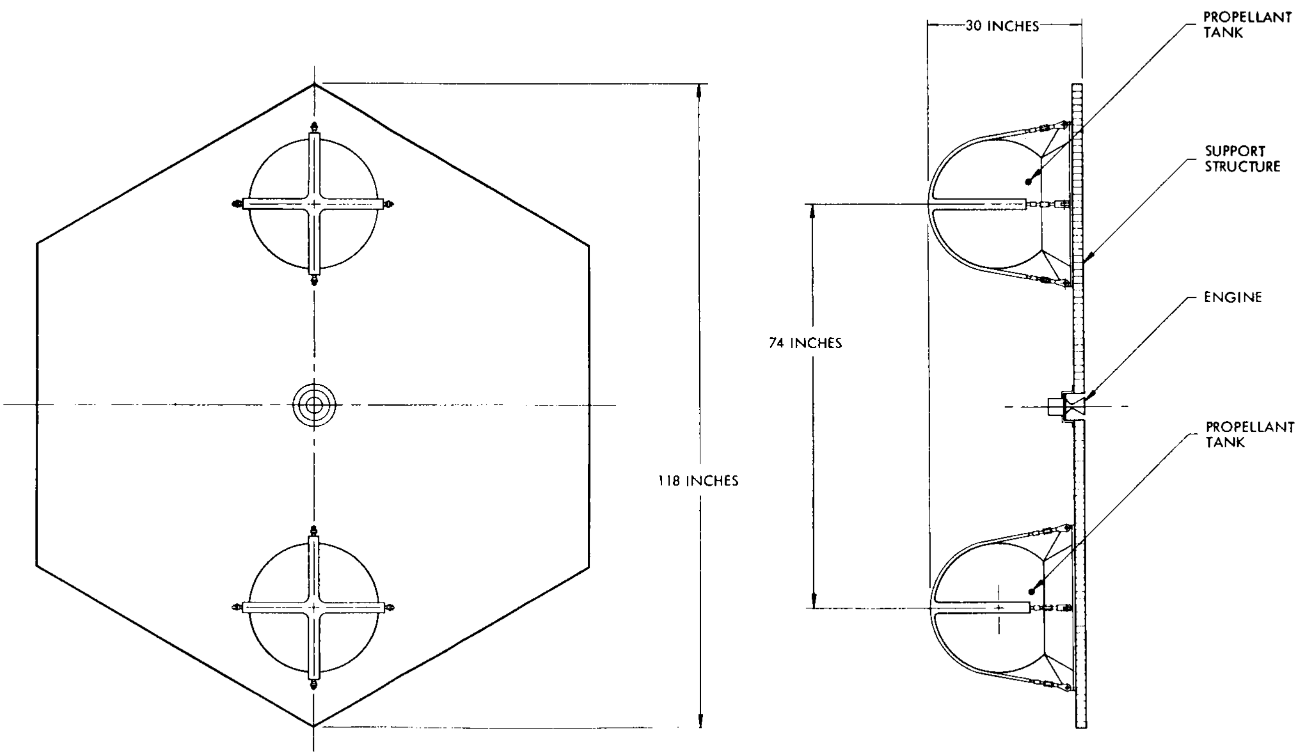


Figure 3. Subsystem Installation and Envelope

4.2 Electrical

4.2.1 Central Sequencing and Command Subsystem

Two electrical inputs are provided for redundant ignition of each of the explosive actuated valves, and one electrical input is required for each activation of the solenoid valves. Hence, for the nominal mission, two midcourse corrections and one velocity trim, twelve separate inputs are required. Each additional velocity correction requires two additional electrical inputs.

Electrical inputs for the jet-vane actuators are provided by the stabilization and control subsystem. A feedback signal is required from the jet-vane actuator.

4.2.2 Telemetry Measurements

The subsystem outputs consist of the following telemetry measurements:

- Propellant tank pressure (0 to 500 psia)
- Thrust Chamber pressure (0 to 500 psia)
- Propellant temperature (+25 to +125°F)
- Valve actuation event blips (nominally 14).

5. PARAMETERS

System pressures are shown in Table 1 and nominal engine performance characteristics are presented in Table 2.

Table 1. Nominal System Pressures and Temperatures

	Nominal Pressure, psia	Nominal Temperature °F
Propellant Tank, Initial	350	70
Thrust Chamber	275	1800

Table 2. Nominal Engine Performance Summary (without jet vanes)

Vacuum Thrust, F_{vac} at P_t = 350 psia	50 lb_f
Vacuum Specific Impulse, $I_{sp_{vac}}$	230 sec
Vacuum Thrust Coefficient, $C_{F_{vac}}$	1. 760
Propellant Flow Rate, w at P_t = 350 psia	0. 217 lb/sec
Throat Area, A_t	0. 1034 in^2
Chamber Pressure, P_c at P_t = 350 psia	275 psia
Characteristic Velocity, C*	4300
Engine Expansion Ratio, ϵ	50:1

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6. 1 Design

The design is constrained to achieve maximum simplicity and reliability.

6. 2 Total Impulse

The subsystem has a capability of imparting a velocity increment of 75 meters/sec to a 7800-pound spacecraft.

6. 3 Impulse Accuracy

The subsystem has the capability of delivering a minimum mid-course correction velocity increment of 0. 1 meters/sec to a 7800-pound spacecraft with a 3σ nonproportional error of 0. 01 meters/sec.

The subsystem has the capability of providing variable total impulse for velocity increments of 0.1 to 75 meters/sec. For velocity increments greater than 1 meter/sec, the 3σ error in velocity increment does not exceed 3 per cent.

6.4 Thrust

The rocket engine steady-state vacuum thrust, decreases predictably as propellant is used, reducing tank pressure. The intial throat level is 50 pounds and this is reduced to 25 pounds at propellant depletion.

6.5 Specific Impulse

The vacuum specific impulse upon which propellant load and velocity increment capability are determined is 230 seconds.

6.6 Environment

6.6.1 General

The midcourse correction subsystem is designed in accordance with the environmental conditions of 4.2 of VS-2-110.

6.6.2 Operation

The subsystem will ignite and operate in a vacuum at any attitude in an initial free-fall condition over an ambient temperature range of +35 to +110°F. The subsystem is capable of vacuum environment storage in excess of 250 days.

6.7 Thrust Vector Control

Thrust vector control is provided by four jet vanes, each capable of producing approximately 2 pounds lift in the fully deflected position. The radial distance from the engine centerline to the center of pressure of each jet vane is approximately 1 inch.

6.8 Alignment

The nominal thrust vector is located parallel with the spacecraft roll axis and nominally passes through the center of mass of the planetary vehicle. The thrust vector of the engine is adjustable within a 1-inch-diameter circle and $\pm$ 0.2-degree of the geometric engine centerline measured at a plane passing through the throat of the engine.

6.9 Pyrotechnics

Design of the valve explosive actuators in compatible with VS-4-530.

6.10 Reliability

Reliability data is presented in VS-3-120.

6.11 Weight and Power

Weight and power values are given in VS-3-111.

6.12 Prelaunch Considerations

The system can be loaded, pressurized, and monitored for several weeks prior to emplacement within the spacecraft.

No spacecraft umbilical or hardlines are required to maintain the subsystem in the "ready" condition.

7. SAFETY CONSIDERATIONS

7.1 General

The midcourse propulsion subsystem is compatible with the safety plan described in Volume 3, Appendix C.

7.2 Temperature

Propellant tanks satisfy a factor of safety of 2.2 up to a temperature of 110°F as per 6.9.3 of VS-1-110.

RETROPROPULSION SUBSYSTEM

VS-4-611

CONTENTS

1. SCOPE

This document presents the design of the retropropulsion subsystem which is used for Martian orbital injection of the Voyager spacecraft in the 1971 mission.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-1-110	Mission Objectives and Design Criteria
VS-2-110	Design Characteristics and Restraints
VS-3-111	Components Design Parameters
VS-3-120	Reliability Design Objectives
VS-4-410	Stabilization and Control System
VS-4-470	Electrical Distribution
VS-4-530	Pyrotechnics

3. FUNCTIONAL DESCRIPTION

3.1 Function

The function of the Voyager retropropulsion subsystem is to provide thrust at Mars arrival upon receipt of a suitable signal. The resulting propulsive impulse is to reduce the velocity of the flight spacecraft sufficiently so as to achieve insertion into a Martian orbit with suitable characteristics. This function is performed after the separation of the capsule vehicle and the jettisoning of the remaining capsule adapter and canister.

3.2 Subsystem Description

The retropropulsion subsystem is a solid propellant rocket motor equipped with liquid injection thrust vector control (LITVC). A preliminary design rocket motor configuration is shown in Figure 1.

The motor consists of a solid propellant grain contained in a fiberglass pressure case, a filled rubber internal insulation, an ablative exhaust nozzle, a refractory throat insert, a nozzle seal, an igniter with safety and arm unit, and a LITVC.

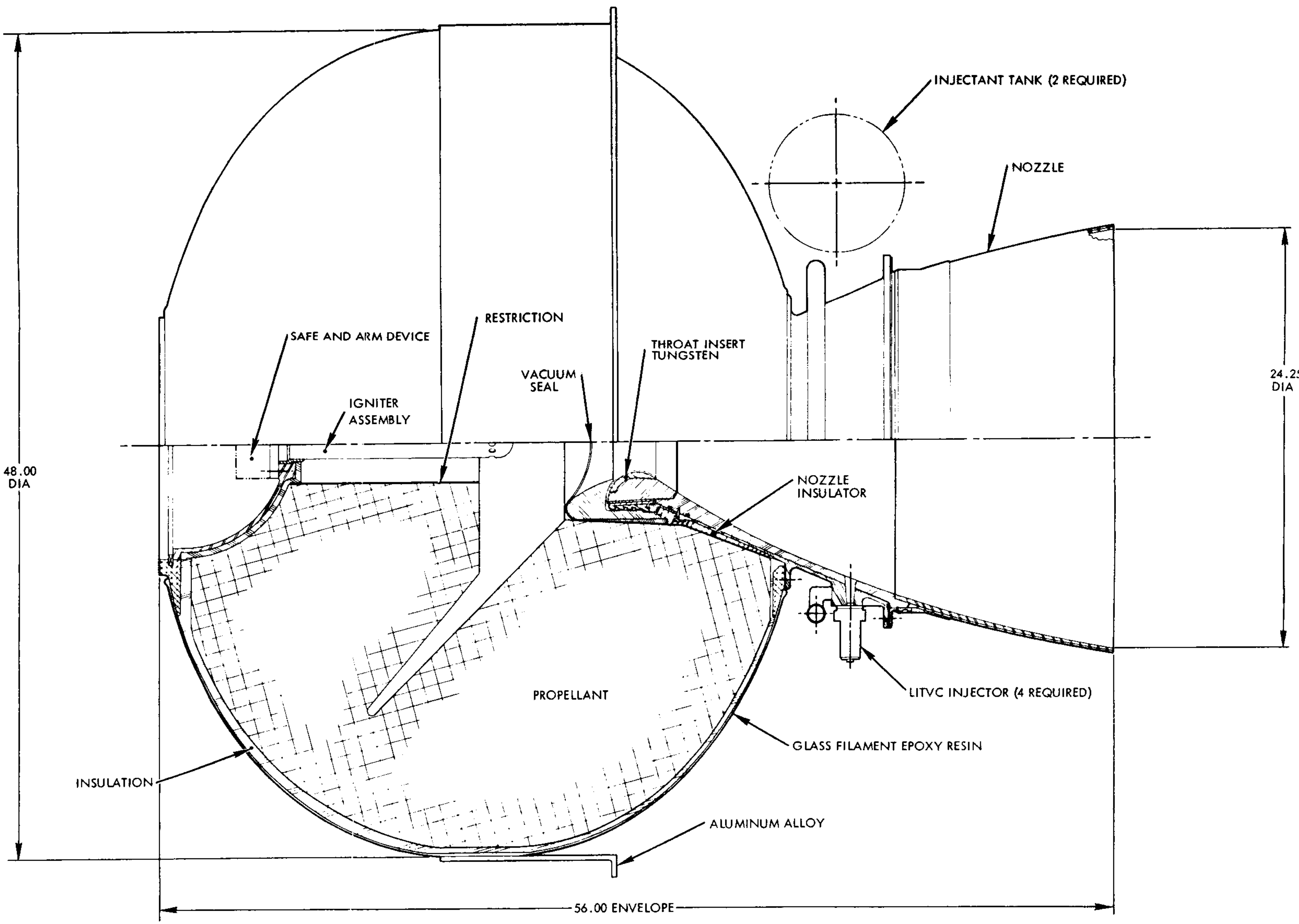


Figure 1. Preliminary Retropropulsion Motor Configuration

The LITVC system consists of four electrically controlled modulating injector valves, an injectant tank and pressurization system, sufficient Freon 114B2 for the mission, and associated electronics.

3.3 Design Philosophy

To ensure satisfactory operation after long-term storage as required for the Voyager mission, a nozzle seal is installed in the nozzle throat to prevent exposure of the propellant grain to the space vacuum. In addition, welded fittings are used throughout the LITVC system and the valve ports and gas generator outlet on the LITVC system are sealed with metallic burst diaphragms to minimize leakage and sensitivity to the effects of the space environment.

3.4 Flight Operation

Firing of the solid propellant motor is initiated by an electrical signal from the spacecraft central sequencing and command subsystem (CS&C) which receives the time of firing through the ground-to-spacecraft communication link. The motor operates until the depletion of propellant causes shutdown and thrust termination. During motor firing, spacecraft pitch and yaw attitude control torque are supplied for the stabilization and control subsystem by the LITVC system. Sequence of events during operation of the rocket motor is as follows (see Figure 2).

a) The command signal No. 1 from the CS&C causes the gas generator igniters to fire, igniting the solid propellant gas generator grain and pressurizing the Freon 114B2 for the LITVC. Pressurizing the Freon causes the burst diaphragms in the LITVC modulating valve to burst, permitting liquid flow through the injector valves of the LITVC system. The pressure level in the Freon tank is controlled by a preset relief valve with overboard dump.

b) A command signal No. 2 from the CS&C causes the igniter to fire, igniting the solid propellant grain. The motor attains steady state thrust within 0.75-second after signal No. 2. Command signal No. 1 may precede signal No. 2 or the signals may be simultaneous, depending upon the dynamics of the system.

c) The solid propellant motor continues to fire until the propellant is consumed. During the firing period the stabilization and control subsystem controls the flow of fluid through the LITVC modulating valves to provide yaw and pitch control

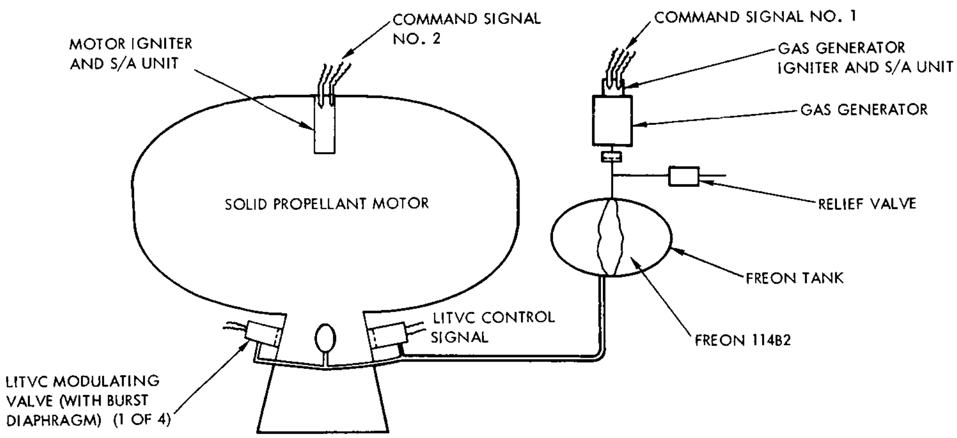


Figure 2. Retropropulsion Motor Sequence of Operations

4. INTERFACE DEFINITION

4.1 Mechanical

Installation of the motor and its allowable envelope are shown in Figure 3.

4.2 Electrical

4.2.1 Central Sequencing and Command Subsystem

Commands are required for ignition of the LITVC gas generator and the rocket motor igniter as described in 3.4.

4.2.2 Electrical Distribution

Firing circuitry to implement the commands of 4.2.1 will be implemented in accordance with 3.3.5 of VS-4-470.

4.2.3 Command Signals to LITVC

Command signals to the LITVC are supplied by the stabilization and control subsystem.

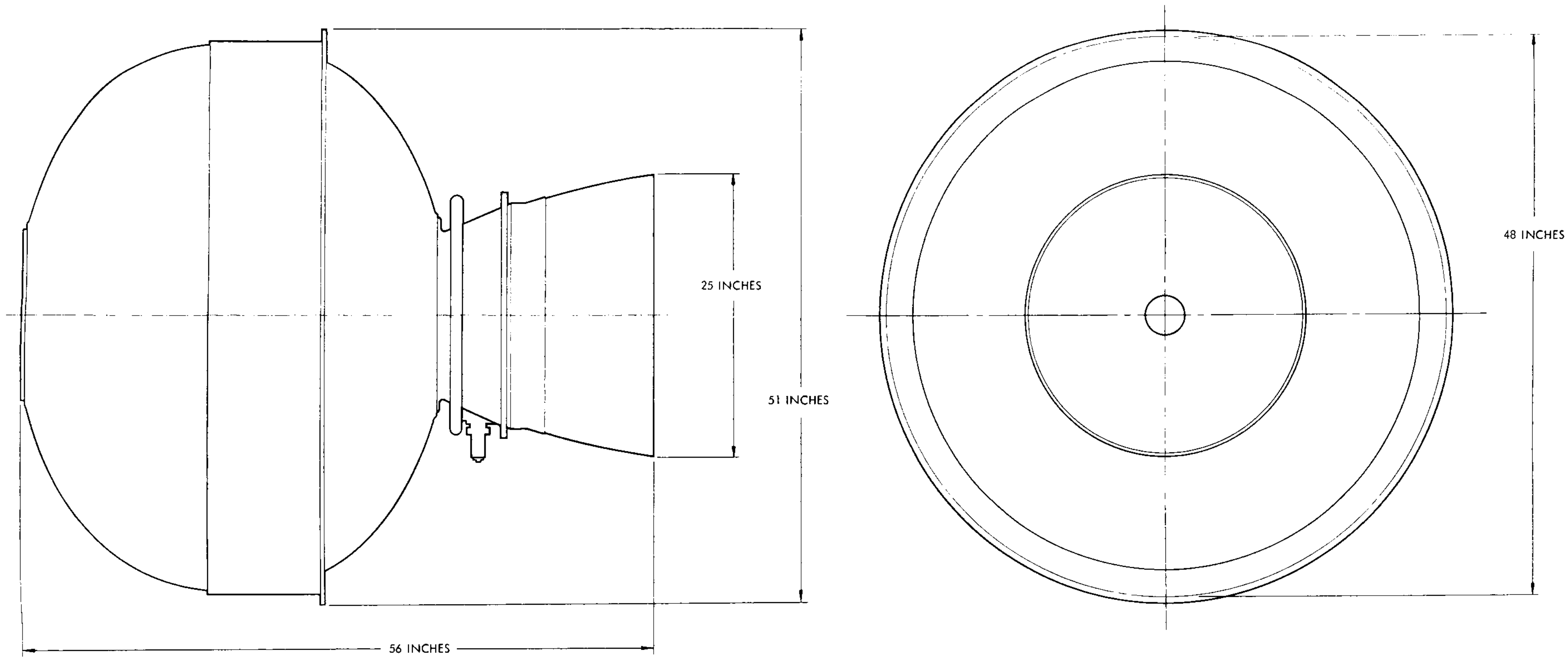


Figure 3. Motor Installation and Allowable Envelope

4.2.4 Telemetry Measurements

The following measurement signals are to be provided to the telemetry subsystem:

a) Freon tank pressure (0 to 500 psia)
b) Thrust chamber pressure (0 to 800 psia)
c) Engine case temperature (2) (+25°F to 300°F)
d) Two event blips:

-Start LITVC current
-Start propulsion current

4.3 Exhaust Plume

For preliminary design purposes, plume characteristics are shown in Figure 4.

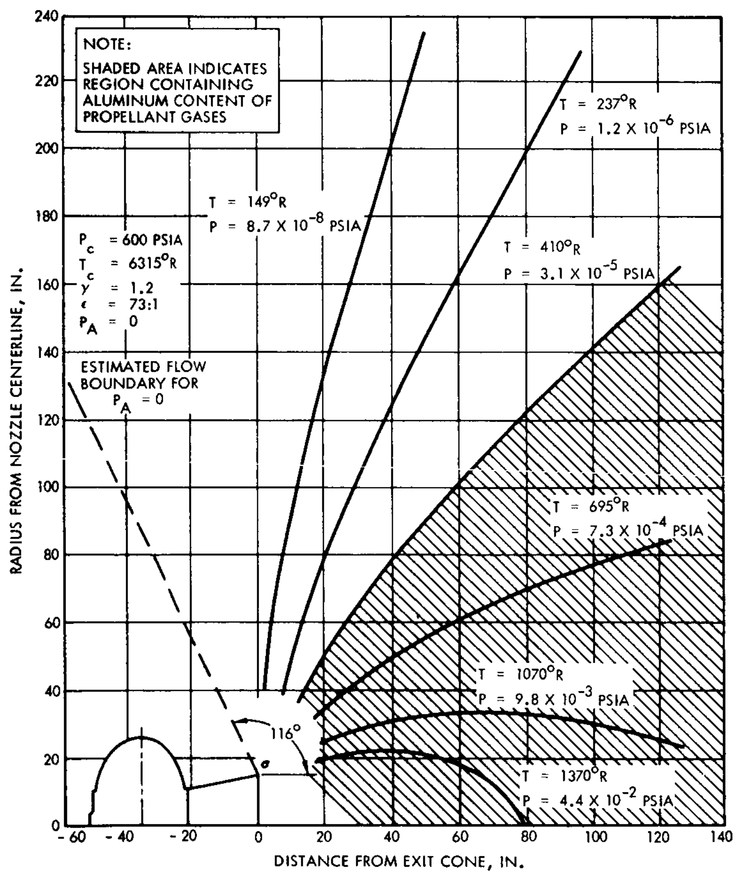


Figure 4. Calculated Exhaust Plume Profile for Retropropulsion Motor

5. PERFORMANCE

5.1 Motor Performance

A summary of nominal motor performance is presented in Table 1.

5.2 Velocity Increment

The rocket motor can impart a nominal velocity increment of 7000 ft/sec to an inert weight of 2300 pounds.

5.3 Thrust

A typical thrust versus time curve is shown in Figure 5.

5.4 LITVC

Predicted LITVC performance characteristics are depicted in Figure 6. The average commanded thrust deflection is ± 0.5 degree.

Table 1. Performance Characteristics

Standard $I_{sp_{1000}}$, sec	249
Effective Vacuum I_{sp}, sec	293
Mass Fraction (Weight of Propellant/Total Weight)	0.87
Mass Fraction (Total Expended/Total Weight)	0.90
Maximum Thrust, lb	15,000
Average Thrust, lb	8,500
Maximum Chamber Pressure, psia	700
Nozzle Expansion Ratio	50:1
Burn Time, /sec	90 - 100
Propellant Properties	
Density, lb/in.^{3}	0.064
Burning Rate, in/sec	0.21 - 0.25

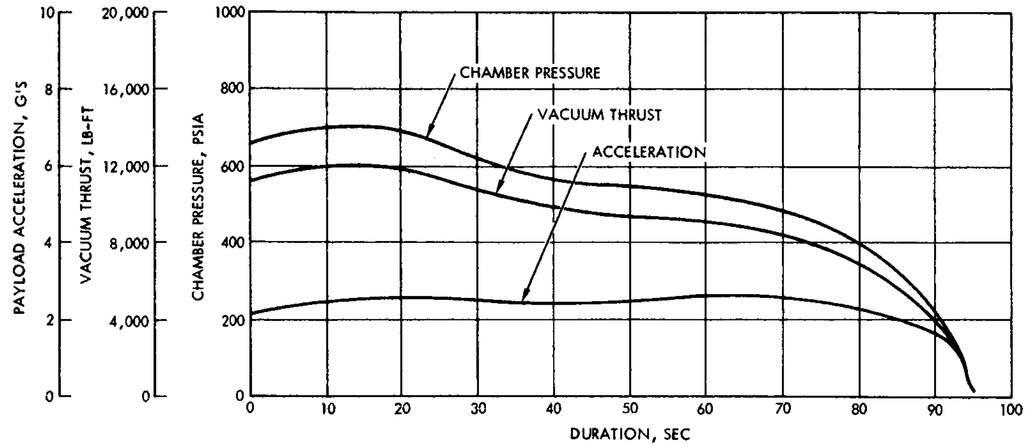


Figure 5. Voyager Retropropulsion Motor Performance

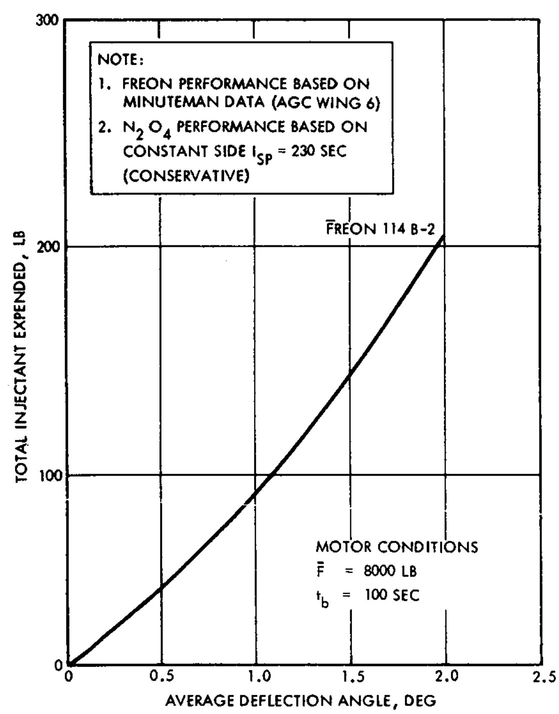


Figure 6. Voyager Secondary Injection TVC Performance

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Design

The design of the motor is constrained to use flight proven hardware and propellants and to minimize spacecraft interactions.

6.2 Thrust Level

The rocket motor vacuum thrust is limited at any time to a value less than that producing a spacecraft acceleration of 96 ft/sec^2. For preliminary design, the inert weight under thrust other than the rocket motor is taken to be 2300 pounds.

6.3 Impulse Accuracy

The 1 sigma accuracy with which the total impulse of 850,000 lb/sec is delivered is estimated to be 0.25 per cent.

6.4 Nozzle Erosion

Nozzle erosion is expected to be sufficiently uniform to ensure compliance with ballistic performance tolerances and such that the lateral shift in the centroid of the nozzle throat (post burn) will not exceed 0.02 inch.

6.5 Nozzle Seal

A nozzle seal is provided to ensure against loss of propellant volatiles and to promote ignition. The nozzle seal maintains at least 5 psia internal pressure after 9 months, assuming 1 atmosphere at launch, and provides thermal insulation at least equal to the case insulation in the motor aft end.

6.6 Thrust Vector Control

6.6.1 Thrust Deflection

Thrust vector deflection is provided by the LITVC. The LITVC is capable of generating upon command any desired thrust deflection up to 4 degrees and a total control impulse of up to 1 per cent of axial total impulse. The retropropulsion subsystem is not required to provide roll control torque.

6.6.2 Frequency Response

The minimum frequency response for the LITVC is estimated to be 1 cps.

6.7 Alignment

The nozzle axis is parallel to the motor axis within 0.001 radian and coincident at the throat within 0.01 inch laterally. The thrust axis is assumed to be coincident with the nozzle axis when no deflection is commanded.

6.8 Center of Mass

The loaded motor center of mass radial offset from the motor axis is not to exceed 0.05 inch.

6.9 Environment

6.9.1 General

The rocket motor is designed in accordance with the environmental conditions of 4.2 of VS-2-110.

6.9.2 Operation

The motor is capable of ignition and operation in a vacuum environment and under an initial free-fall condition.

6.9.3 Flight Environment

Flight temperature limits are given in VS-3-111. The temperature gradient between any two locations in the solid propellant grain is estimated to be less than $10^{\circ}F$.

The motor is capable of vacuum environment storage in excess of 250 days without causing any deleterious effects to the motor or the spacecraft.

6.9.4 Prelaunch Temperatures

The preflight to launch ambient temperature range of the propulsion system is estimated to be +35 to $+100^{\circ}F$.

6.10 Pyrotechnics

Design of pyrotechnic devices is compatible with VS-4-530.

6.11 Reliability

Reliability data are presented in VS-3-120.

6.12 Weight and Power

Weight and power values are given in VS-3-111.

6.13 Prelaunch Considerations

The LITVC system can be loaded and monitored for several weeks prior to emplacement within the spacecraft.

No spacecraft umbilicals or hardlines are required to maintain the rocket motor in the "ready" condition.

7. SAFETY CONSIDERATION

7.1 Rocket Motor

The motor is safe for personnel to work around at temperatures up to 125°F.

7.2 LITVC

The system in the loaded condition is safe for personnel to work around over the temperature range of +35 to +125°F.

7.3 Pyrotechnics

The safety considerations of 6.9 in VS-1-110 apply. In particular, an electromechanical safe and arm device provides electrical safety between the power supply and the squibs and interposes a mechanical barrier between the squibs and the subsequent pyrotechnic train until such time as the ability to achieve actuation (arming) is desired.

EVASIVE MANEUVER PROPULSION SUBSYSTEM

VS-4-612

CONTENTS

1. SCOPE

This document presents the design of the evasive maneuver propulsion subsystem (EMPS) for the Voyager spacecraft in the 1971 mission. This subsystem provides the propulsive capability for an evasive maneuver after capsule separation to avoid interference with the capsule vehicle.

2. APPLICABLE DOCUMENTS

Supplementary and supporting documents are as follows:

TRW 1971 Voyager Spacecraft Design Documents

VS-1-110	Mission Objectives and Design Criteria
VS-2-110	Design Characteristics and Restraints
VS-3-111	Components Design Parameter
VS-3-120	Reliability Design Objectives
VS-4-530	Pyrotechnics

3. FUNCTIONAL DESCRIPTION

3.1 Function

The function of the EMPS is to provide propulsive impulse to translate the spacecraft bus in such a manner as to ensure that the capsule vehicle does not collide with the bus as a result of the latter's propulsive maneuver. This function is performed after the capsule vehicle has separated from the bus a sufficient distance to avoid interference and to ensure that gas from the EMPS does not contaminate the capsule vehicle.

3.2 System Description

The EMPS consists of a spherical gas storage vessel, an explosive actuated normally closed valve, and a convergent-divergent nozzle. The propellant is N_2 stored at a pressure of 2000 psia.

3.3 Flight Operation

The flight operation consists of a single firing which is initiated by a command signal to fire the explosive actuated normally-closed valve. The gas in the storage vessel exhausts through the nozzle, providing an initial thrust of 0.1 pound. The thrust gradually decays until all of the gas in the bottle has been expended.

4. INTERFACE DEFINITION

4.1 Mechanical

The subsystem is mounted to the spacecraft structure at a suitable location and in a manner such that the thrust vector is directed through the estimated nominal center of mass of the spacecraft.

4.2 Electrical

4.2.1 Firing Signal

The CS&C provides a firing command at the proper time. Electrical distribution provides two firing signals to the dual bridge-wire squib valve.

4.2.2 Telemetry Measurements

A valve actuation event signal is supplied to telemetry.

5. PARAMETERS

The performance characteristics of the system are as follows:

Initial thrust	0.1 pound
Specific impulse, average	50 seconds
Initial pressure	2000 psia

6. PHYSICAL CHARACTERISTICS AND CONSTRAINTS

6.1 Design

The design is constrained to achieve maximum simplicity.

6.2 Total Impulse

The subsystem has a capability of imparting 0.2 ft/sec velocity increment to a 5500-pound spacecraft.

6.3 Impulse Accuracy

Impulse accuracy is not a critical factor. Tolerances consistent with a loading accuracy of 5 to 10 per cent are considered sufficiently accurate.

6.4 Specific Impulse

Propellant, N_2, is loaded on the basis of an average specific impulse of 50 seconds. (The specific impulse will be a function of the average temperature during the expansion of the gases.)

6.5 Environment

6.5.1 General

The EMPS is designed in accordance with the environmental conditions of paragraph 4.2 of VS-2-110.

6.6 Alignment

The thrust vector is aimed through a nominal center of mass of the vehicle. Maximum error to the true center of mass is predicted to be 3 inches. Resultant torque about any axis is estimated not to exceed 0.15 in-lb.

6.7 Pyrotechnics

Design of the valve explosive actuators is compatible with VS-4-530.

6.8 Reliability

Reliability data is presented in VS-3-120.

6.9 Weight and Power

Weight and power values are given in VS-3-111.

6.10 Prelaunch Considerations

The system can be loaded, pressurized, and monitored for several weeks prior to emplacement within the spacecraft. No spacecraft umbilicals are required.

The system will be sterilized in the pressurized condition. (The tanks are stressed to accommodate the resultant pressure rise without exceeding the rated design factor of 2.2.)

7. SAFETY CONSIDERATIONS

The EMPS is compatible with 6.9 of VS-1-110 and the Safety Plan described in Appendix C of Volume 3. The subsystem in the pressurized condition meets the required factor of safety.

AUG 12 1965

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

Volume 1. Summary

Substitute new p. 79 attached.

Volume 2. 1971 Voyager Spacecraft

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

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

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

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

p. 327. Denominator of second term on right hand side of equation should read

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

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

Volume 3. Voyager Program Plan

Substitute new p. 12 attached.

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

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

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

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

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

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

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

p. 167. Figure 5-21. In line 20 change "design revisions" to "design reviews"

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

Volume 5. Alternate Designs: Subsystem Considerations

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

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

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

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

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

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

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

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

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

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

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

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

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

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

Volume 5. <u>Alternate Designs: Subsystem Considerations. Appendix I</u>

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

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

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

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

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

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

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

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

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

p. C-35 In the first equation, the expression "(t in m^2)" in two places should read "(t in cm)" and "A" in two places should read "(A in m^2)"

p. C-38 In Table C-2, all values in inches should be in centimeters. A zero should be inserted immediately following the decimal point, for example: (0.020-inch) = 0.05080, (0.020-inch) = 0.06096, (0.020-inch) = 0.04064, etc.

p. C-40 In Section 1.8.7 Computation of R_i's, the sixth line should read "... than 10^6 are neglected"

p. C-45 In listing under "Values of t Used for Extreme Environment Analysis," under Inch, the first number should read 0.020 instead of 0.202

p. C-52 In 1.10 NOMENCLATURE, "K_2" should be defined as "$K^{-2/3}(4 \pm 2)$" and "B" should be

$$\frac{1000\, \rho_t V^2}{9.806\, H_t}$$

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

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

Volume 5. Alternate Designs: Subsystem Considerations. Appendix II

p. F-23 Lines 7 and 10 change all subscript τ to T

p. F-24 Line 14, change "ME_i" to "mE_i"

p. F-29 Figure F-9 title should be "Reflection Phase Angle ϕ (deg)" and Figure F-10 title should be "Reflection Magnitude R"

p. F-30 Last line, change "0.27" to "0.175"

p. F-31 Lines 14 and 15, change "14,700 ft/sec to 460 ft/sec" to 14,700 ft/sec minus 460 ft/sec" and "14,700 ft/sec to 10,000 ft/sec" to "14,700 ft/sec minus 10,000 ft/sec"

p. F-32 Last line in item 4), change "27 per cent" to "17.5 per cent"

p. F-35 Table F-4, under Assumed Parameter for item 2 insert "$\pm 2 \times 10^{-5}$", for item 3 insert "$\pm 3 \times 10^{-5}$", and for item 4 insert "$\pm 2 \times 10^{-5}$"

p. F-53 Item d. Noise Figure, change "4 db" to "3.5 db"; Gain, change "20 db" to "10 db", last line change "10 db" to "4 db"

p. F-58 Figure F-21. Change 102 kc to 112 kc.

p. F-59 Line 22, change to "M_1 = 21.5 deg or 0.375 radians (rms, peak)"

p. F-60 Line 2, change to

$$"M_2 = \sqrt{(1.1)^2 - (0.375)^2}\ "$$

p. F-60 Line 3, change to "M_2 = 1.03 radians (rms) or 1.46 radians (peak)"

p. G-6 Paragraph 1.4, second line, change "from $E_M = 10' E_o$ to $10^4 E_o$ . . ." to read "from $E_M = 10^{-1} E_o$ to $10^4 E_o$ . . ."

Volume 6. Operational Support Equipment

p. 25 Figure 6. Caption should be "Typical Grounding Scheme"

p. 39 Section 1.3.3, change opening of first sentence to read "Launch pad equipment consists of the ground power and RF consoles and the test flight program power and control equipment . . . "

p. G-31 Figure 1. Lines enclosing Data Format Generator should be solid.

p. G-102 Last line substitute "4500" for "45"

p. G-113 In Section 4.4.2, change "25 per cent" to "250 per cent"

p. G-184 Section 4.5, substitute "6.5 feet" for "six feet"

p. G-311 Fifth line, change "30 per cent" to "20 per cent"

p. G-398 Section 4.2 should begin with "The hoist beam is . . . "

p. G-419 Second line "4 optical alignment targets" instead of 8. Same correction top of p. G-421.

p. G-423 Section 4.9.2, substitute "20 per cent" for "50 per cent"

Volume 7. 1969 Flight Test Spacecraft and OSE

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

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