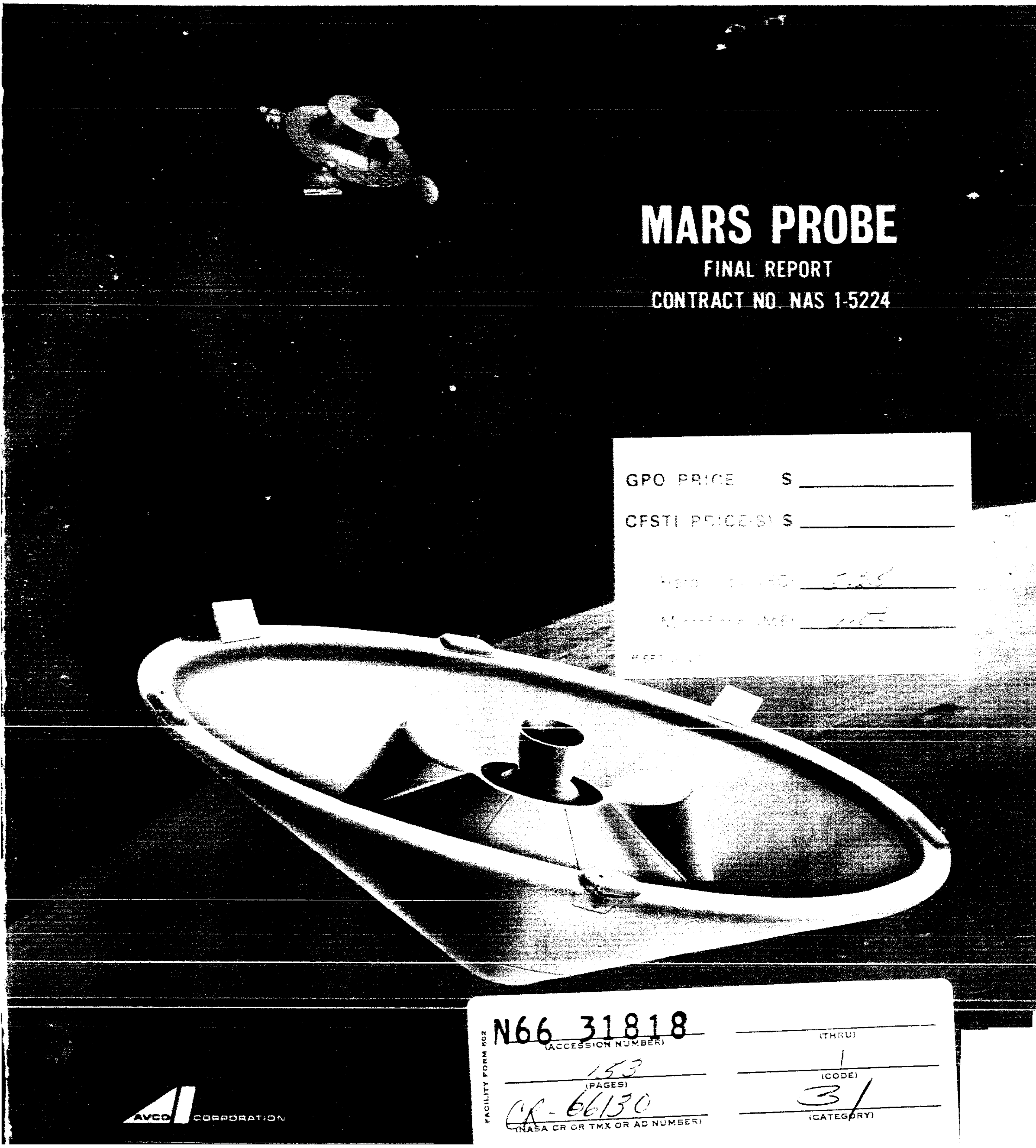
NASA CR - 66130

VOLUME II PROBE/LANDER, ENTRY FROM THE APPROACH TRAJECTORY

BOOK 2

MISSION and SYSTEM SPECIFICATIONS



MARS PROBE

FINAL REPORT

CONTRACT NO. NAS 1-5224

AVCO CORPORATION

BOOK INDEX

NASA CR-66130

COMPARATIVE STUDIES OF CONCEPTUAL DESIGN AND QUALIFICATION PROCEDURES FOR A MARS PROBE/LANDER

FINAL REPORT

VOLUME II PROBE/LANDER, ENTRY FROM THE APPROACH TRAJECTORY

Book 2 - MISSION AND SYSTEM SPECIFICATIONS

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PREFACE

The results of Mars Probe/Lander studies, conducted over a 10-month period for Langley Research Center, NASA, are presented in detail in this report. Under the original contract work statement, studies were directed toward a direct entry mission concept, consistent with the use of the Saturn IB-Centaur Launch Vehicle, wherein the landing capsule is separated from the spacecraft on the interplanetary approach trajectory, some 10 to 12 days before planet encounter. The primary objectives of this mission were atmospheric sampling by the probe/lander during entry and terrain and atmosphere physical composition measurement for a period of about 1 day after landing.

Studies for this mission were predicated on the assumption that the atmosphere of Mars could be described as being within the range specified by, NASA Mars Model Atmospheres 1, 2, 3 and a Terminal Descent Atmosphere of the document NASA TM-D2525. These models describe the surface pressure as being between 10 and 40 mb. For this surface pressure range a payload of moderate size can be landed on the planet's surface if the entry angle is restricted to be less than about 45 degrees.

Midway during the course of the study, it was discovered by Mariner IV that the pressure at the surface of the planet is in the 4 to 10 mb range, a range much lower than previously thought to be the case. The results of the study were re-examined at this point. It was found that retention of the direct entry mission mode would require much shallower entry angles to achieve the same payloads previously attained at the higher entry angles of the higher surface pressure model atmospheres. The achievement of shallow entry angles (on the order of 20 degrees), in turn, required sophisticated capsule terminal guidance, and a sizeable capsule propulsion system to apply a velocity correction close to the planet, after the final terminal navigation measurements.

Faced with these facts, NASA/LRC decided that the direct entry from the approach trajectory mission mode should be compared with the entry from orbit mode under the assumption that the Saturn 5 Launch Vehicle would be available. Entry of the flight capsule from orbit allows the shallow angle entry (together with low entry velocity) necessary to permit higher values of M/C_DA, and hence entry weight in the attenuated atmosphere.

It was also decided by LRC to eliminate the landing portion of the mission in favor of a descent payload having greater data-gathering capacity, including television and penetrometers. In both the direct entry and the entry from orbit cases, ballistic atmospheric retardation was the only retardation means considered as apecifically required by the contract work statement.

Four months had elapsed at the time the study ground rules were changed. After this point the study continued for an additional five months, during which period a new design for the substantially changed conditions was evolved. For this design, qualification test programs for selected subsystems were studied. Sterilization studies were included in the program from the start and, based on the development of a fundamental approach to the sterilization problem, these efforts were expanded in the second half of the study.

The organization of this report reflects the circumstance that two essentially different mission modes were studied -- the first being the entry from the approach trajectory mission mode and the other being the entry from orbit mission mode -- from which two designs were evolved. The report organization is as follows:

Volume I, Summary, summarizes the entire study for both mission modes.

Volume II reports on the results of the first part of the study. This volume is titled Probe/Lander, Entry from the Approach Trajectory. It is divided into two books, Book 1 and Book 2. Book 1 is titled System Design and presents a discursive summary of the entry from the approach trajectory system as it had evolved up to the point where the mission mode was changed. Book 2, titled Mission and System Specifications, presents, in formal fashion, specifications for the system. It should be understood, however, that the study for this mission mode was not carried through to completion and many of the design selections are subject to further tradeoff analysis.

Volume III is composed of three books which summarize the results of the entry from orbit studies. Books 1 and 2 are organized in the same fashion as the books of Volume II, except that Book 2 of Volume III presents component specifications as well. Book 3 is titled Development Test Programs and presents, for selected subsystems, a discussion of technology status, test requirements and plans. This Book is intended to satisfy the study and reporting requirements concerning qualification studies, but the selected title is believed to describe more accurately the study emphasis desired by LRC.

Volume IV presents Sterilization results. This information is presented separately because of its potential utilization as a more fundamental reference document.

Volume V presents, in six separate books, Subsystem and Technical Analyses. In order (from Book 1 to Book 6) they are:

Trajectory Analysis
Aeromechanics and Thermal Control
Telecommunications, Radar Systems and Power
Instrumentation
Attitude Control and Propulsion
Mechanical Subsystems

Most of the books of Volume V are divided into separate discussions of the two mission modes. Table of Contents for each book clearly shows its organization.

CONTENTS

CONTENTS (Cont'd)

CONTENTS (Cont'd)

CONTENTS (Cont'd)

CONTENTS (Concl'd)

ILLUSTRATIONS

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TABLES

ACKNOWLEDGEMENT

The conduct of the study and technical preparation of this report involved the participation and close coordination of many people, all of whose contributions were important to the end results.

It is impractical to single out each individual, but the major contributors to the study, reflected in the material presented in this Book, are as follows:

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INTRODUCTION

This Mission and System Specification has been prepared as a typical baseline document representative of a Flight Capsule mission of limited life on the Mars surface and a mission mode based upon entry of the capsule from the interplanetary approach trajectory. This document provides the Capsule System ground rules, guide lines, and definitions of the general design philosophy, including listing of design criteria for the Flight Capsule and Operational Support Equipment for the 1971 Mars opportunity. Primary functional areas are identified and functionally described.

Since study efforts on this mission-mode were terminated before a conceptual design had been selected, this document is necessarily tentative and incomplete; the design presented should, therefore, not be interpreted as a recommended approach. Many analyses and tradeoffs remain to be undertaken before firm selection of system and subsystem design parameters can be made.

An asterisk following any paragraph in this specification indicates that the information was not available at the time the concept herein specified was synthesized or that the information was not critical to the synthesis.

1.0 MISSION DESCRIPTION

The Flight Capsule mission objectives are defined related to the overall Mars exploratory mission, including a nominal flight profile.

1.1 FLIGHT CAPSULE MISSION OBJECTIVE

1.1.1 Primary

The primary objective of the Flight Capsule associated with the 1971 mission is to:

1. Place a significant payload on Mars
2. Conduct experiments on Mars surface
3. Transmit results to Earth.

The events required to accomplish this objective are ordered in the following manner:

1. Perform a successful launch and injection of the Planetary Vehicle* (including Flight Capsule) into a prescribed transfer orbit.
2. Perform a successful Flight Spacecraft - Flight Capsule separation maneuver at a preselected time and location.
3. Place the Separated Vehicle on a preselected impact trajectory to Mars.
4. Enter the Martian atmosphere and obtain data on the lower Martian atmosphere from the Flight Capsule science instruments and communicate data to the Flight Spacecraft and/or Deep Space Network.
5. Land the Landed Capsule, establish communications to Earth, and return entry, landing, and system status data to Earth.
6. Perform necessary landed operations to obtain data with the science instrumentation over at least one Martian diurnal cycle and return data to Earth.

The primary objective can be expanded in the area of atmospheric and surface data requirements. The atmospheric data obtained during entry and terminal descent are to be considered prime data. Redundant instrumentation and/or techniques should be used as necessary to obtain this data. The following data are to be obtained from Entry Vehicle entry to parachute deployment:

*See section 1.3.2 for terminology and definitions.

1. Atmospheric density

2. Atmospheric pressure

3. Atmospheric temperature

4. Atmospheric composition

5. Trapped radiation within atmosphere

6. Ionosphere characteristics.

The following data are to be obtained while the Suspended Capsule is on the parachute, until impact on the Martian surface:

1. Chute deployment conditions

2. Atmospheric density

3. Atmospheric pressure

4. Atmospheric temperature

5. Atmospheric composition

6. Impact g levels

7. Trapped radiation within atmosphere

8. Surface wind velocities

9. Altitude determination

10. Surface roughness.

The following data are to be obtained while on the Martian surface in the immediate vicinity of the landing site:

1. Atmospheric density

2. Atmospheric temperature

3. Atmospheric composition

4. Surface winds velocities

5. Surface hardness

6. Soil composition

7. Dust particle concentration

8. Solar constants

9. Water vapor concentration within atmosphere and on the surface.

1.1.2 Secondary

A secondary objective is to provide experience with both the flight and ground systems required to deliver and operate the Flight Capsule science instrumentation, to ferry and separate the Flight Capsule and to deliver and operate the Flight Capsule science instrumentation.

1.1.3 Tertiary

A tertiary objective is to provide flight and ground equipment designs compatible with subsequent missions.

1.2 MISSION DEFINITION

The 1971 Mars mission will include all preflight, flight, and postflight activities that are required to provide two totally integrated space vehicles for performance of the required operations during the 1971 Mars launch opportunity. The mission, for each system, is considered to begin when that system has completed mission acceptance review. The mission is completed when all of the scientific and engineering data has been returned to Earth, reduced to acceptable form and disseminated to all cognizant organizations.

1.3 MISSION ELEMENT DEFINITION

1.3.1 General Mission

Two operational Planetary Vehicles will be launched on separate Saturn IB/Centaur Launch Vehicles for the 1971 launch opportunity. A minimum separation of 10 days is required between arrival dates at Mars of the two Flight Capsules. Two launch pads will be utilized for the 1971 launch opportunity. A launch period of 30 to 60 days will be provided, allowing for a minimum daily firing launch window of two hours.

AFETR facilities at Cape Kennedy, Florida will be utilized for prelaunch and launch activities. Prelaunch assembly, checkout and test will be conducted in the Flight Capsule portion of the Flight Spacecraft facility. The Explosive Safe Facility will be furnished for all operations involving the Flight Capsule that are considered of a hazardous nature, such as terminal sterilization, propellant loading and hazardous component activation.

AFETR tracking and telemetry facilities will be required during the launch through injection phase of the mission to accommodate the Space Vehicle instrumentation and assist in the Deep Space Network acquisition.

The Capsule System is that portion of the overall project which accomplishes the planet atmospheric and surface experiments with an atmospheric Entry Vehicle and an operating Landed Capsule.

The integration and further definition of the project systems to the Capsule System is discussed in paragraph 1.3.2.

1.3.2 Capsule System

The relation of the Capsule System to the rest of the project is shown in Figure 1. Each of the system level elements shown contains all flight hardware, developmental hardware, qualification hardware, models, spares, operational support equipment, test equipment, software, associated manpower and facilities to accomplish the assigned integrated mission. The detailed definition of all of the Flight Capsule subsystems is presented in the appropriate end item functional specification presented in Section 6.0. A general definition for all of the other primary project elements follows:

1.3.2.1 Capsule System

The Capsule System includes the Flight Capsule (FC), as the flight hardware; plus flight hardware spare parts (or spare Flight Capsules depending on the time of spares replacement), development and sterilization assay models, control documentation and associated software, operational support equipment, and the management and engineering teams.

1.3.2.2 Spacecraft System

The Spacecraft System includes the Flight Spacecraft (FS), as its flight hardware; plus flight hardware spare parts, development models, associated operational support equipment (hardware and software) and the management and engineering teams.

1.3.2.3 Planetary Vehicle (PV)

The Planetary Vehicle is defined as the composite Flight Spacecraft and Flight Capsule integrally attached and operated up to separation in the vicinity of the selected planet. The Flight Spacecraft to Flight Capsule adapter includes the inflight separation joint between the Flight Capsule and Flight Spacecraft, the load path for mounting the Flight Capsule to the Flight Spacecraft and a mounting bulkhead for the umbilical connections between the Flight Capsule and Flight Spacecraft.

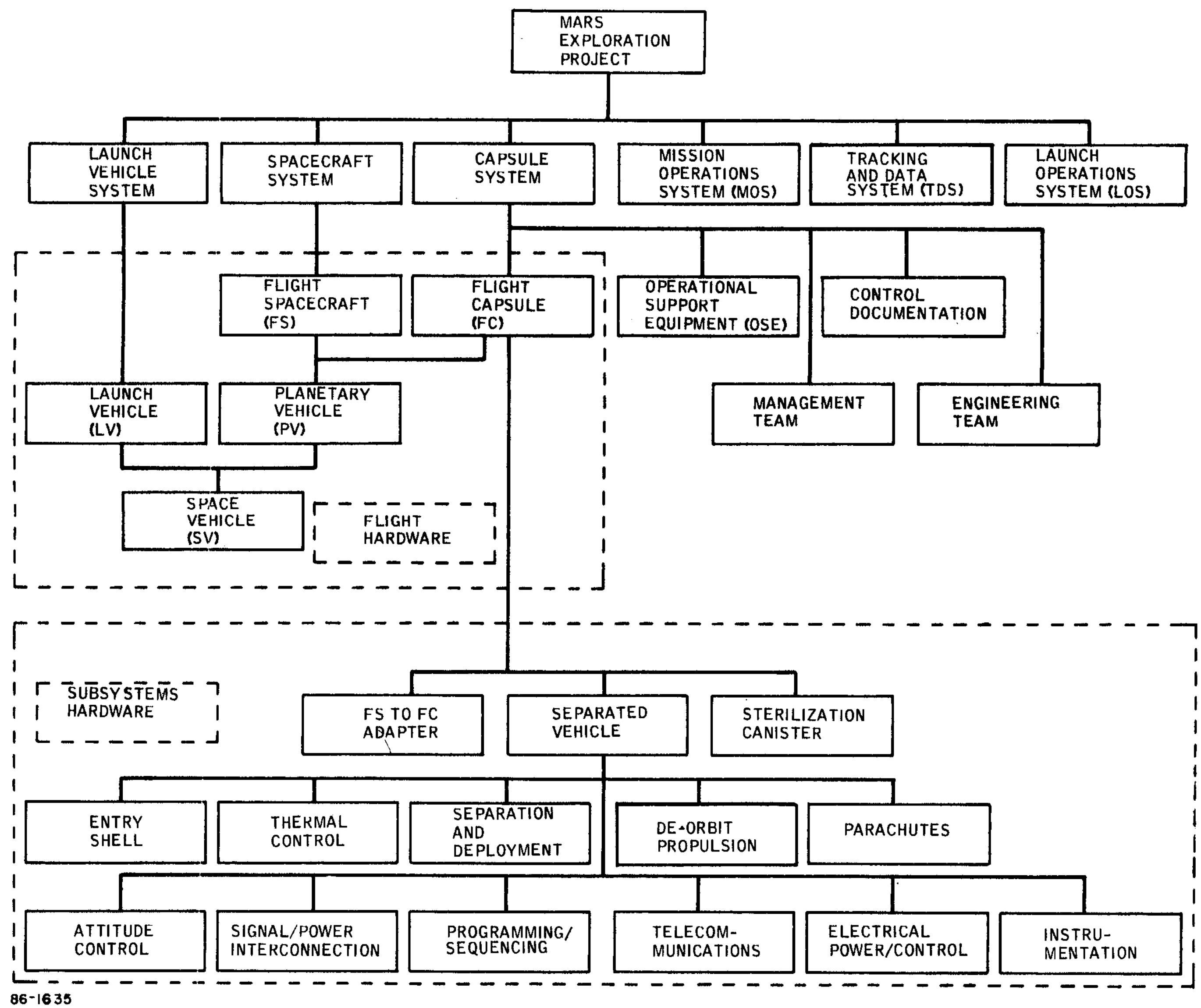

Figure 1 CAPSULE SYSTEM FUNCTIONAL DIAGRAM AND TERMINOLOGY

1.3.2.4 Launch Vehicle System

The Launch Vehicle System includes the three stages of the Saturn IB/ Centaur, with its guidance subsystems and the ascent fairing which shrouds the Planetary Vehicle, to make up the Launch Vehicle (LV) as the flight hardware; plus the supporting ground equipment, software and associated manpower.

1.3.2.5 Space Vehicle (SV)

The Space Vehicle is the combined Launch Vehicle and Planetary Vehicle which physically leave the launch pad in conduct of the mission.

1.3.2.6 Mission Operations System (MOS)

The MOS includes 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 Planetary Vehicle on its trajectory, the Mission Dependent Equipment required at the Deep Space Network, and the operational teams.

1.3.2.7 Deep Space Network (DSN)

The DSN is comprised of the Deep Space Instrumentation Facility (DSIF), the Space Flight Operations Facility (SFOF), the Ground Communications System (GCS) connecting the two facilities, and the personnel who regularly operate these facilities. These facilities are defined in the following documents:

1. JPL TM No. 33-83, Revision 1

2. Deep Space Network - Space Flight Operations Facility Capability

3. Deep Space Network Ground Communications System Development Plan.

1.3.2.8 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.

1.3.2.9 Operational Support Equipment (OSE)

The OSE includes the equipment and facilities required for the assembly, servicing, checkout, sterilization, and testing of the subsystems of the Flight Capsule; described as follows:

1. Assembly, Handling and Shipping Equipment (AHSE) -- This equipment includes all lifting, holding, and positioning fixtures and other items required in the assembly and test of the Flight Capsule and its OSE, and for moving the Flight Capsule from place to place.

2. System Test Complex (STC) -- The STC is the basic set of test equipment used in tests and checkout to verify the adequacy of the Capsule System design, fabrication, assembly, and flight readiness. It is employed to monitor and record the performance of the total Capsule System and of its subsystems and subassemblies, while providing power, command, external stimuli, and simulation of the associated systems.

3. Launch Complex Equipment (LCE) -- The LCE is used to power and command the Flight Capsule and to monitor and record its functions (both by hardline and by telemetry) during prelaunch checkout while the Planetary Vehicle is installed in the SV on the launch pad. It includes all equipment on the launch pad and in the blockhouse. It is also used to support operations in the Explosive Safe Facility (ESF).

4. Mission Dependent Equipment (MDE) -- The MDE includes all items required at the site of the DSN to meet the functional requirements of a particular project which are not required on any other project. Items may be categorized as either software or hardware.

5. Maintenance Support Equipment (MSE) -- The MSE is the equipment required to support the Flight Capsule and its OSE such that the Capsule System maintains its capability of performing its mission.

6. Factory Support Equipment (FSE) -- The FSE is the equipment required to fabricate, assemble, and checkout the Flight Capsule and its support equipment. These equipments are limited to in-plant operations, i.e., tooling, checkout gear, etc.

7. Facilities -- The buildings that house the engineering and administrative personnel, test areas and chambers, manufacturing, assembly, training and storage.

1.3.2.10 Control Documentation

The software required to define, integrate and test the Capsule System with the rest of the Project.

Typical documents in this category include, but are not limited to the following:

Model Specification

Electrical Interface Drawing

Mechanical Interface Drawing

Interference Control Specification

Interconnecting Cabling Diagram

Signal and Power Flow Diagram

Schematic Diagram

Storage and Handling Procedures

Integrated Test Plans and Procedures

1.4 MISSION PROFILE

The nominal mission for each Space Vehicle and/or its operating elements shall be composed of the following phases:

1. Space Vehicle prelaunch operations	All final assembly, integration, checkout, and test procedures and activities resulting in a commitment to launch.
2. Space Vehicle launch and Planetary Vehicle injection	Final Space Vehicle countdown, launch, parking orbit execution, insertion into the transit trajectory, and separation of the Planetary Vehicle from the Launch Vehicle.
3. Planetary Vehicle acquisition	Acquisition by the Planetary Vehicle of external attitude references and execution of all sequences leading to cruise status.
4. Planetary Vehicle interplanetary cruise	All events and sequences during the transit flight to Mars when the Planetary Vehicle is on external references.
5. Planetary Vehicle interplanetary trajectory corrections	All events and sequences used to alter the transit trajectory of the Planetary Vehicle until the return to cruise status.

6. Flight Spacecraft /Flight-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.
7. Flight Capsule trajectory deflection	All events and sequences used to place the Separated Vehicle on a selected Mars impact trajectory.
8. Flight Spacecraft cruise	All Flight Spacecraft events and sequences between Flight Spacecraft/Flight Capsule separation and the Flight Spacecraft orbital insertion phase.
9. Separated Vehicle cruise	All Flight Capsule events and sequences between Flight Spacecraft/ Flight Capsule separation and Entry Vehicle entry.
10. Entry Vehicle entry	All Flight Capsule events and sequences from the time and Entry Vehicle senses $0.1G_e$ until the beginning of terminal velocity descent at $10G_e$ descending.
11. Suspended Capsule descent and landing	All Flight Capsule events and sequences from the beginning of terminal velocity descent until the Landed Capsule comes to rest on the surface.
12. Landed Capsule landed operations	All Flight Capsule operations from the time the Landed Capsule comes to rest on the planet surface until the last time the communications signal is lost due to power supply depletion or the surface mission is declared to have ended.
13. Flight Spacecraft orbit insertion	All events and sequences used to insert the Flight Spacecraft into orbit about Mars.

14. Flight Spacecraft orbital operations	All Flight Spacecraft operations from the t me external references are reacquired after orbit insertion until the communications signal is finally lost or the orbiter mission is declared to have ended.

2.0 MISSION CONSTRAINTS AND COMPETING CHARACTERISTICS

2.1 MISSION CONSTAINTS

The following items are the major constraints placed on the Capsule System by the mission and are to be considered fixed.

2.1.1 Mission Schedule

Since the mission objectives involve the 1971 Mars launch opportunity, all designs, techniques, and components must be compatible with that schedule. On this basis, the technology cut-off date for the 1971 mission shall be September 1966.

2.1.2 Planetary Cuarantine

The probability of Mars contamination due to each Flight Capsule, shall be less than 10^{-4} for any one mission.

2.2 COMPETING CHARACTERISTICS

The order of the following list of characteristics shall be utilized to govern the priority of conflicting technical requirements:

1. Planetary sterility requirement
2. Telecommunication functions and operations
3. Flight Capsule temperature control
4. Flight Capsule electrical power
5. Flight Capsule - Flight spacecraft separation
6. Separated Vehicle injection into proper Mars capture trajectory
7. Operation of instrumentation while in Martian atmosphere
8. Entry Vehicle entry and Landed Capsule landing
9. Operations of instrumentation while in Martian atmosphere
10. Application of design philosophy, and flight and ground hardware to the 1973 mission.

3.0 MISSION REQUIREMENTS

3.1 GENERAL

The general requirements of this section apply to all systems required to perform the mission with particular emphasis on the mission requirements of the Capsule System.

3.1.1 Commitment to Launch

The Capsule System shall utilize instrumentation necessary to perform monitoring during countdown to detect malfunctions and nonstandard system operation. In the event of nonstandard performance, the system shall be capable of evaluating its effect on mission performance and shall affirm commitment to launch or request hold for maintenance as prescribed by the mission launch and hold criteria in effect at the time of launch.

3.1.2 Mission Success

All Capsule System equipment shall utilize design, test, and operational procedures that will maximize the probability of mission success. To obtain this objective, consideration shall be given but not be limited to the following:

1. Failure-mode analysis and partial mission success in the event of failure.
2. Adequate design margins
3. Block and functional redundancy techniques to minimize effect of failure.

The philosophy of design shall emphasize simplicity and conservatism and shall include a complete and integrated test program for components, subsystems, and system to assure the highest degree of mission success. Wherever possible, previous experience gained on other programs related to hardware and development techniques should be applied.

3.1.3 Trajectories

3.1.3.1 Transfer Trajectories

The Planetary Vehicle shall be placed on a Type 1 transfer trajectory for the 1971 Mars opportunity. A maximum value of C_3 is 18 km^2/sec^2, compatible with a Flight-Capsule weight of 2500 pounds and a Planetary Vehicle weight of 7500 pounds. The hyperbolic excess velocity at Mars encounter shall not exceed a value of 5 km/sec. The declination of the departure asymptote (DLA) will be equal to or less than 33 degrees.

3.1.3.2 Flight Capsule Separation

Flight Capsule separation from the Flight Spacecraft shall occur over a period of from 2 to 20 days prior to Flight-Spacecraft orbital injection. The Flight Capsule shall be capable of orbit change after separation to achieve a Mars impact trajectory.

3.1.3.3 Entry Over Goldstone

Entry Vehicle atmospheric entry, Suspended Capsule, and Landed Capsule surface impact will occur in view of the DSIF station at Goldstone. If this requirement should place undue constraints on separation time, landing-site selection or prime mission objectives, the requirement may be negotiated.

3.1.3.4 Landing Site

The Landed Capsule shall be landed (including dispersion) at a Mars latitude which is within ±30 degrees of the sub-Earth point for preliminary consideration. Nominally, landing shall occur to allow direct-link communications in an area of 500 km radius (3 sigma).

3.1.3.5 Entry Conditions

The Entry Vehicle shall be designed to withstand entry angles of -20 to -90 degrees and entry velocities of 18,000 to 23,800 ft/sec at 800,000 feet Martian altitude.

3.1.4 <u>Accuracy</u>*

3.1.5 <u>Weight Allocation</u>

The Planetary Vehicle weight shall not exceed 7500 pounds.

3.1.6 <u>Environment</u>

All Capsule Systems equipment shall be required to operate and/or survive, as appropriate, in the natural or induced environments shown in Figure 2. Additionally, the Flight Capsule shall be required to operate and perform its required functions over the entire range of postulated Martian atmospheres as shown in Table I.

3.1.7 <u>Test</u>*

3.1.8 <u>Reliability</u>

The estimated cumulative probability of success that has been allocated at each significant mission phase is as follows:

Phase	Function	Duration	Temperature (°F)	Pressure (mm Hg)	Relative Humidity	Sand and Dust	Precipitation	Corrosive Atmosphere	Fungus	Atmospheric Wind	Solar Wind	Meteoroids and Dust
	Mission Description		Environmental Conditions									
1. Manufacturing and Assembly	Manufacturing, assembly, and test of subsystems into FC for acceptance test; seal in container, sterilize, checkout, and prepare/for shipment; all operations in clean room.	Up to 26 weeks	60 to 80°F during assembly and test. 275°F for 24 hours during sterilization.	Sea level ambient 760	40 to 50%	Negligible	Not applicable	Ethylene oxide flush during sterilization cycles.	Non-fungus nutrients to be used for all equipments. Sterilization further reduces any fungus probabilities internal to sterile barrier. External to barrier assumed per paragraph 4.8.1 of MIL-E-5272.	Not applicable	Not applicable	Not applicable
2. Storage and Transportation	OSE to be provided to maintain clean room conditions for FC/container during any operation not conducted in clean room.	Up to 13 weeks storage. Up to 12 hours air transport.	-35 to 150, includes 25° solar radiation rise.	760 to 87	0 to 100%	10^{-4} to 0.3 mm diameter blown by 60-mph winds	Same as launch pad operations.	Salt fog of 20% salt/80% water at specific gravity, of 1.126 to 1.157 and a pH of 6.5 to 7.2 at 95°F and 85% relative humidity.		0 to 60 mph		
3. Field Operations - Spacecraft Integration	FC integration and checkout with FS; all operations in clean room.	Up to 9 weeks	60 to 80	760	40 to 50%	Negligible	Not applicable	None		Not applicable		
4. Launch Pad Operations	FS/FC integration and check with launch vehicle. Clean room facilities on gantry prior to fairing installation.	Up to 4 weeks	30 to 150, includes 25° solar radiation rise.	760	10 to 100%	10^{-4} to 0.3 mm diameter blown by 60-mph wind	Rain: maximum rate of 10 cm/hr blown by 60 mph wind. Ice: Up to a cm frozen on exposed surfaces. Snow: 1 to 3 mm diameter crystals at maximum rate of 7.5 cm/hr blown by 60-mph wind. Hail: 0.5 to 5 mm diameter frozen water pellets at maximum rate of 10 cm/hr blown by 50-mph wind for 15 minutes at above freezing temperatures. Sleet: 0.75 to 1.5 mm diameter ice pellets at maximum rate of 10 cm/hr blown by 60 mph wind for 30 minutes at below freezing temperature.	Salt sea atmosphere of 5% salt/95% water.		0 to 60 mph		
5. Launch through Booster Separation	PV is injected into interplanetary transfer orbit. Ascent fairing separated after atmospheric boost.	Up to 1 hour	Temperature of capsule system and subsystems to result from atmospheric ascent heating, solar radiation, internal equipment operations, space shadow effects, materials absorptivity/emmisivity characteristics and related configuration. Analysis required for each critical area assuming more than + 300° at an AU, to less than -300° in shadow at 1.6 AU. Maximum aerodynamic heating rate = 24.2 watts/ft²	760 to 10^{-6}	10 to 100% through none.	10^{-4} to 0.3 mm diameter blown by 60-mph winds to negligible		From salt sea atmosphere to none.		Velocity versus altitude per ARDC handbook		Near Earth model - $N = 10^{-12} m^{-3/4}$ where N = impacts/meter²/sec of particles of mass, m (grams) or greater; at an average velocity of 30 km/sec. and a density of 8 gm/cm³ - 12%, 3.5 gms/cm³ - 10%, 0.44 gms/cm³ - 88%. Near Mars model to be updated from Mariner 4 data.
6. Interplanetary transit. (PV cruise)	PV in unpowered trajectory except for attitude control maneuvers during acquisition and maintenance of lock on to Sun and Canopus, plus midcourse corrections with FS propulsion; operating of diagnostic subsystems.	Up to 384 days		10^{-6} to 10^{-12}	None	None	None	None	None	None	At an AU-(dynes/cm²) 10^{-7} quite sun 10^{-5} average 10^{-3} major activity	
7. FC from FS separation and FC trajectory maneuver.	The FC is separated from FS approximately 5 x 10^6 from Mars; ACS maneuver and ΔV propulsion firing.	Up to 1 hour										
8. Capsule Cruise	Communications relay of FC status to FS to DSIF during cruise.	Up to 4 days										
9. Capsule Entry	Store entry scientific measurements from 0.1 G_e rising to approximately Mach 10.0; Radiometer operation from $10G_e$ rising to $10G_e$ falling.	Up to 10 minutes	To be determined	10^{-12} to 30		Negligible		Improbable		To 100 Ft/sec with 50 ft/sec gusts.	Not applicable	Not applicable
10. Atmospheric Descent	Playout entry and descent scientific data.	Up to 10 minutes	-300 to +80	8 to 30	Unknown	Unknown quantities of sand and dust blown by 100 ft/sec winds	H_2O and CO_2 crystals blown by 100 ft/sec winds		Unknown			
11. Payload Landing	Capsule impacts planet surface. Scientific payload is protected by impact attenuator.	Less than 1 second	-135 to +80									
12. Mars Surface Operations	Scientific instruments are deployed. Data outputs are transmitted direct and/or relayed to bus to DSIF.	More than 24 hours										

Phase	Electro Magnetic Interference: Generated	Solar	Particle	Magnetic Fields	Vibration	Shock	Sustained Acceleration	Acoustic Noise
1. Manufacturing and Assembly	Various forms of electro magnetic fields may be generated by capsule, bus, launch vehicle systems and/or OSE. Refer to MIL-I-26600 or RAD-E-59064.	Negligible	Negligible	Earth field - 0.173 gauss. Vibration exciters up to 80 oersteds. Tools up to 100 oersteds.	Unpackaged (accel. (RMS) (Freq. cps) ±3.5 G_e 2 to 50; ±1.5 G_e 5 to 300. Packaged ±1.3 G_e 2 to 26; 0.036 in. DA 26 to 52; ±5.0 G_e 52 to 300	Unpackaged components: $100G_e$ for 6 milliseconds. Unpackaged major assemblies: $10G_e$ for 11 milliseconds applied along 3 axes or 1 inch flat drops and 4 inch pivot drops.	Earth gravity, G_e = 32.17 ft/sec²	Negligible
2. Storage and Transportation		Intensity varies sinusoidally from 0 to 0.140 watts/cm² in 24 hour period; spectrum percentage - 8% ultraviolet, 41% visible, 51% infrared.		Earth field - 0.173 gauss	Unpackaged - no longer applicable; packaged - same as manufacture and assembly.	Unpackaged assemblies and components: not applicable. Packaged assembly with OSE: MIL-P-7936		Related to transportation equipment and OSE attenuation.
3. Field Operations - Spacecraft Integration						Unpackaged: not applicable. Packaged: same as manufacturing and assembly.		Negligible
4. Launch Pad Operations								
5. Launch through Booster Separation		Maximum 0.140 watts/cm² at 1 AU.	Radiation dosages in space produced by atomic particles vary due to Earth "Van Allen" radiation belts, cosmic rays, solar flare activities and Mars radiation belts (if any). Maximum energy and ionization field of protons, electrons and photons dependent on spacecraft trajectory versus time.	from 0.173 to less than 0.0013 gauss.	Omnidirectional to the PV separation plane - PSD peaks of 0.07 G_e^2/CPS ranging from 100 to 1500 CPS with a 6 db/octave roll off in the envelope defining peaks below and above these frequencies for a maximum time of 60 seconds.	200 G_e peak saw tooth of 0.7 to 1.0 millisecs rise time	Maximum additional loads: axial, 4.5G_e: lateral, ± 1 G_e.	142 DB maximum from launch vehicle engines, for 2 minutes
6. Interplanetary transit. (PV cruise)		From 0.140 watts/cm² at 1.0 AU to 0.045 watts/cm² at 1.6 AU. Albedo factor: Earth 0.40, Mars 0.14. Light Pressure: 2 x 10^{-7} lb/ft² at 1.0 AU, 0.8 x 10^{-7} lb/ft² at 1.6 AU. Particle wavelength fractions of total energy vary throughout entire spectrum.		Negligible	Negligible - to be determined from analysis of the selected midcourse correction propulsion subsystem on the FS versus PV dynamics.			
7. FC from FS separation and FC trajectory maneuver.					Negligible - to be determined from analysis of the selected capsule spin up and trajectory change propulsion subsystems on the capsule versus capsule dynamics.			
8. Capsule Cruise					None	None	None	None
9. Capsule Entry				to be determined - assumed to be 0.5 Earth	To be determined from analysis of entry dynamics, capsule configuration, possible deceleration device (parachute) mode of operations, atmospheric conditions etc.			To be determined
10. Atmospheric Descent		Some attenuation of short wavelengths in the atmosphere	Unknown, probably negligible					
11. Payload Landing					Negligible	Impact up to $500G_e$	Mars gravity. Gm = 12.3 ft/sec²	Negligible
12. Mars Surface Operations						Negligible		

Figure 2 SUMMARY OF FLIGHT CAPSULE MISSION PROFILE AND ENVIRONMENTAL CONDITIONS

14-2

14-3

TABLE I

SUMMARY OF STANDARD MODEL ATMOSPHERE PARAMETERS FOR MARS

Parameter	Units	Maximum (Model 1)	Mean (Model 2)	Minimum (Model 3)
Surface Pressure	mb	40	25	10
Pressure	lb/in^2	0.58	0.363	0.145
Composition	CO percent by Mass	7.5	16	60
	N_2 percent by Mass	92.5	84	40
	CO_2 percent by volume	4.9	10.8	48.8
	N_2 percent by Volume	95.1	89.2	51.2
Molecular Weight	--	28.8	29.7	35.85
Acceleration of gravity at surface	cm/sec^2	375	375	375
	ft/sec^2	12.3	12.3	12.3
Surface Temperature	°K	300	250	200
	°R	540	450	360
Troposphere lapse rate	°K/km	-3.636	-3.89	-4.55
	$°R/10^3 ft$	-1.995	-2.134	-2.496
Tropopause altitude	km	11	18	22
	ft	36,100	59,100	72,200
Stratosphere Temp.	°K	260	180	100
	°R	468	324	180
Top of Stratosphere	km	150	150	150
	ft	492,100	492,100	492,100
Thermosphere lapse rate	°K/km	2	2	--
	$°R/10^3$ ft	1.097	1.097	--
Surface Density	gm/cm^3	4.62×10^{-5}	3.57×10^{-5}	2.16×10^{-5}
	$slugs/ft^3$	8.97×10^{-5}	6.94×10^{-5}	4.19×10^{-5}

1. Launch and inject Planetary Vehicle into a predetermined transfer trajectory - 90 percent probability of success
2. Separate Flight Capsule from Flight Spacecraft at a predetermined spatial location - 80 percent probability of success
3. Place the Separated Vehicle on Martian impact trajectory - 75 percent probability of success
4. Enter Martian atmosphere and obtain atmospheric data - 65 percent probability of success
5. Land the Landed Capsule and transmit entry, landing and subsystem status data - 45 percent probability of success
6. Deploy scientific instrumentation as required, obtain data over on diurnal cycle and transmit data - 35 percent probability of success.

3.2 CAPSULE SYSTEM REQUIREMENTS

3.2.1 General

A total of four flight-qualified Flight Capsules shall be supplied to support the two proposed flights. An adequate number of spares shall be supplied to augment the four flight-qualified vehicles. Test hardware to support the Flight-Capsule test program shall be supplied as specified.

3.2.2 Functional

The Flight Capsule shall be capable of independent space flight from the point of separation from the Flight Spacecraft to entry into the Martian atmosphere. Operating subsystems shall be capable of surviving entry and landing on the Martian surface and will provide all power, thermal control and communications requirements of the capsule science instrumentation.

To accomplish the above functions, the Flight Capsule shall contain but not be limited to, the following:

1. A propulsion subsystem capable of altering the Separated Vehicle trajectory from a flyby trajectory to a planetary impact trajectory and accelerating the Separated Vehicle to provide communications lead time.
2. An active attitude control subsystem for thrust alignment, thrust vector control, and entry angle of attack maneuvers.
3. Temperature control subsystems
4. Storage batteries and power conversion subsystems

5. Communications subsystems capable of relaying data to the Flight Spacecraft and/or direct communication with the DSIF on Earth.

6. A parachute-descent subsystem

7. An impact-attenuation subsystem

8. Sequencing and command-reception subsystems

9. Atmospheric and surface scientific instrumentation

10. Diagnostic instrumentation

11. Data handling and storage subsystems

3.2.3 Performance

The Capsule System shall be capable of accomplishment of the mission objectives of paragraph 1.1 according to the performance and design criteria in Section 4.0.

3.2.4 Sterilization

All flight hardware placed on a Mars impact trajectory shall be biologically clean per NASA planetary quarantine requirements. A sterilization canister shall be designed and built as part of the Flight Capsule. The Separated Vehicle shall be placed in the canister after final assembly and prior to terminal sterilization. The canister shall not be opened until it is considered biologically safe to do so sometime after launch.

3.2.5 Flight Spacecraft Requirements on the Flight Capsule

3.2.5.1. Center of Gravity

The Flight Capsule center of gravity shall be within those limits specified in paragraph 4.4. Figure 3 shows the geometric constraints placed on the Flight Capsule by the Launch Vehicle ascent fairing and Flight Spacecraft.

3.2.5.2 Flight Capsule Separation

At separation, the Flight Capsule shall not impart any motion to the Flight Spacecraft that will cause it to lose attitude-control reference.

3.2.5.3. Thermal Control

The design of the Flight Capsule shall minimize the heat-transfer effects to the Flight Spacecraft when in the Planetary-Vehicle configuration.

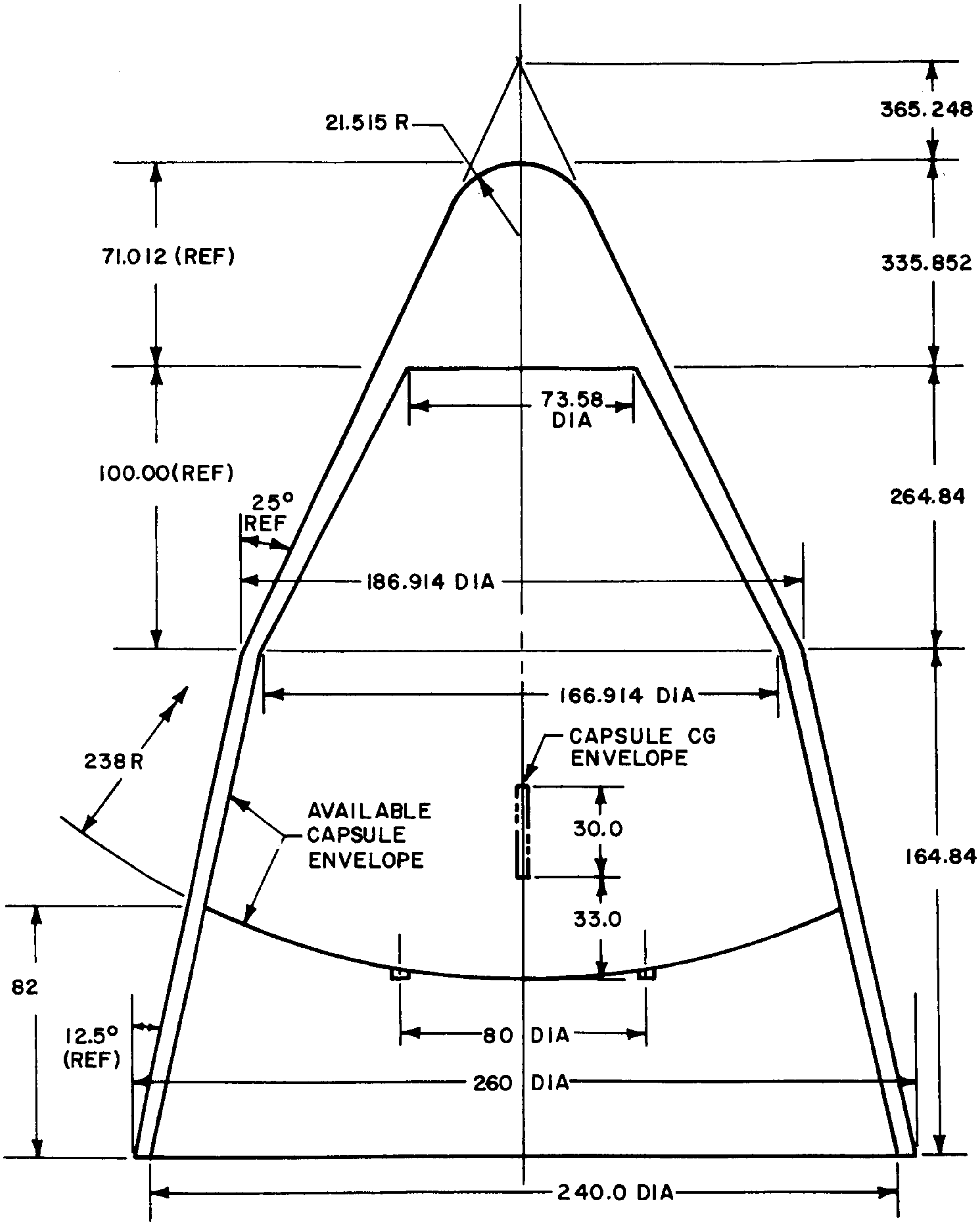

Figure 3 FLIGHT CAPSULE ENVELOPE

3.2.6 Flight Capsule Requirements on the Launch Vehicle

During the launch, parking-orbit and planetary transfer-orbit injection phases, the Launch Vehicle is required to functionally support the Planetary Vehicle. The Flight Capsule shall not require any unique capabilities over and above those needed for the Planetary Vehicle.

3.2.7 Flight Capsule Requirements on the Flight Spacecraft

The Flight Spacecraft shall supply all power, thermal control, and communication requirements of the Flight Capsule during the time period after planetary transfer - orbit injection and prior to Flight Capsule - Flight Spacecraft Separation. After separation, the Flight Spacecraft will relay all required Flight-Capsule telemetry data to Earth.

3.2.8 Flight Capsule Requirements on the Mission Operations System

The Mission Operations System is required to perform the following functions in direct support of the Flight Capsule mission:

1. Determine Flight Capsule separation location and time as a function of actual orbital parameters.

2. Determine Flight Capsule postseparation ACS and ΔV propulsion maneuvers.

3. Determine Flight-Capsule time sequence of events to provide initiation commands for those entry events which require updating as the mission progresses.

3.2.9 Flight Capsule Requirements on the Tracking and Data Systems

Data acquired by the FC shall be received at the Space Flight Operations Facility for data-processing and display. The communications, data processing and display facilities provided by the SFOF may require augmentation by Mission Dependent Equipment specifically implemented to the requirements of the FC data output. These data shall be analyzed by the mission operations personnel and published in a form suitable for presentation to members of the engineering and scientific teams to determine the success of the Capsule System mission.

4.0 FLIGHT CAPSULE PERFORMANCE AND DESIGN REQUIREMENTS

4.1 RELIABILITY

4.1.1 Reliability Criteria

Where operational lifetime of equipment is of primary concern and existing equipment is compatible with the basic Capsule System mission requirements, existing equipment should be utilized. Changes warranted on the basis of actual test experience, for incorporation of redundancy, or for other similar demonstratable reliability improvements are permitted.

Where operational lifetime of equipment is not of primary concern, new designs may be used, provided these designs can be adequately tested and provide a significant improvement in system capabilities.

Wherever the weight penalty is not prohibitive, functional redundancy shall be provided, (degraded capability is acceptable) particularly in the following areas:

1. Separation of Flight Capsule from Flight Spacecraft
2. Space flight stabilization
3. Atmospheric data acquisition
4. Flight Capsule communication

4.1.2 Reliability Design Goals

The Flight Capsule mission reliability goal shown in Table II has been apportioned to the capsule subsystems. The subsystem goals are tentative in that they are based on judgement factors of mission time, complexity, environmental hazards, and state-of-the-art. These goals are preliminary and are presented to indicate relative magnitudes for planning purposes.

4.2 STERILIZATION

The Flight Capsule shall be designed to withstand the following criteria as specified by the program sterilization requirements. (JPL Spec Vol-50503-ETS - Environmental Specification Voyager Capsule Flight Equipment - Heat Sterilization and Ethylene Oxide Decontamination Environments).

TABLE II

SUBSYSTEM/SUBASSEMBLY RELIABILITY DESIGN GOALS

Subsystem/Subassembly	Reliability Design Goals	
1. Sterilization Canister Assembly	0.982	
a. Lid assembly		0.996
b. Base assembly		0.995
c. Pressurization assembly		0.994
d. FC/FS separation		0.996
2. Entry Shell Assembly	0.978	
a. Entry shell body		0.987
b. Entry shell sensors		0.991
3. Capsule Bus Assembly	0.952	
a. External payload hardware		0.992
b. External science payload		0.990
c. Propulsion		0.990
d. Attitude control		0.987
e. Parachute		0.992
4. Landed Capsule Assembly	0.903	
a. Impact attenuation		0.965
b. Landed structure		0.964
c. Landed payload hardware		0.970
5. Electronic Assemblies	0.687	
a. Power		0.955
b. Thermal control		0.958
c. Central computer and sequencer		0.939
d. Telemetry		0.942
e. Data handling		0.939
f. Communications		0.940
g. Antenna		0.959
6. Internal Payload Assembly	0.900	
a. Instruments		0.949
b. Sensors		0.949
OVER-ALL FLIGHT CAPSULE DESIGN RELIABILITY GOAL*	0.51	

*The design goal, 0.51, was obtained by scaling the NASA/Langley mission goal upwards to account for expected reliability degradation between design and use.

1. Qualification testing of 3 cycles of a dry-heat atmosphere at 145°C for 36 hours per cycle with sufficient time between cycles to return to ambient temperature.

2. Terminal sterilization of the Flight Capsule shall consist of heating for 24 hours at 135°C in a dry nitrogen atmosphere.

3. The total microbial content within the Flight Capsule immediately prior to terminal heat sterilization shall be less than 10^8 viable organisms.

4.3 ENVIRONMENT

Figure 2 presents a summary of the Flight Capsule mission profile versus the environmental conditions expected to be encountered. These environments are the basis for the design criteria used for Flight Capsule subsystem analysis, the primary functional area definition, and the operational mission sequences and planning.

4.4 ALIGNMENT AND ACCURACIES

Alignment of the many Flight Capsule components will require special techniques and hardware reference pads at critical points of the Flight Capsule structural elements. At this time the detail design implementation of these techniques has not been developed but the expected alignment and accuracy requirements are known within reasonable limits to aid in the definition of the Flight Capsule subsystems. Figure 4 illustrates the X, Y, and Z reference axes of the Flight Capsule.

4.4.1 The Separated Vehicle cg location shall be known within 0.0833 inch (1 sigma) of the Separated Vehicle true centerline, and shall be 0.0833 inch fore or aft of the designated cg location.

4.4.2 To provide the required accuracy during the trajectory change maneuver, the thrust vector of the propulsion subsystem shall lie within 10 milliradians of the X axis through the separated vehicle center of gravity. Errors contributing to this tolerance are allocated between the structures and the propulsion subsystem.

4.4.3 The thrust vector shall be perpendicular to the plane of the propulsion-subsystem mounting pads with ±3.5 milliradians. The maximum offset of the engine centerline from its nominal location relative to the Flight Capsule centerline shall be ±0.026 inch.

4.4.4 The plane of the Flight-Capsule mounting pads for the propulsion subsystem shall be parallel to the plane of the Flight Capsule Y-Z axes within ±0.5 milliradians.

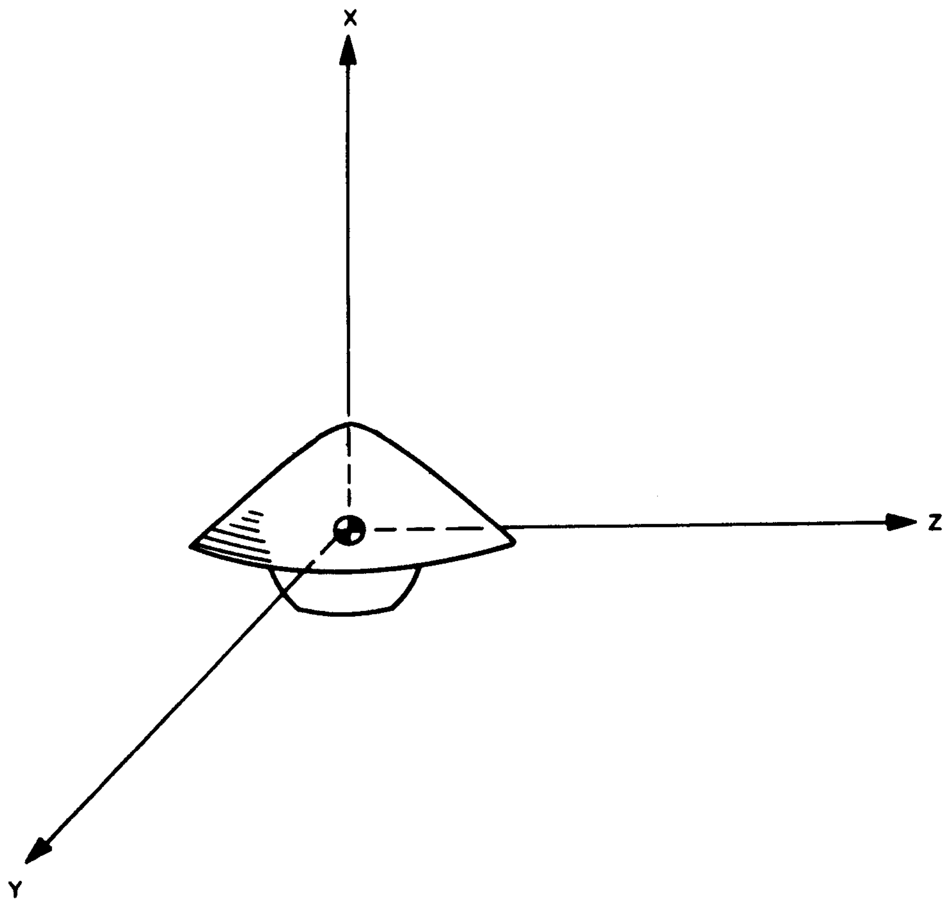

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Figure 4 FLIGHT CAPSULE REFERENCE AXES

4.4.5 The mounting plane of the spin-up rockets shall be parallel to the plane of the Flight Capsule Y-Z axes within ±0.5 milliradians. The angle between the plane of the spin rockets and the plane of the Flight-Capsule mounting pads for the propulsion subsystem shall be 0 degree ±0.5 milliradians.

4.4.6 A reference surface for mounting the spin-up rockets is perpendicular within ±0.5 milliradians to the plane formed by the Y-Z axes of the Flight Capsule.

4.4.7 Alignment of the spin-up rockets shall provide that the centerline of each rocket is parallel to the plane of the Y-Z axes of the Flight Capsule within ± 5.0 milliradians. The centerline of each spin-up rocket shall be 90 degrees ±5.0 milliradians from the perpendicular line from the X-axis to the center of gravity of the spin-up rocket.

4.4.8 Spacing of the spin-up rockets in angular position around the X-axis of the Flight Capsule shall be in equal increments within ±10 milliradians.

4.4.9 The separation plane of the Flight Capsule when mated to the Flight Spacecraft shall be parallel within ±0.5 milliradians to the separation plane of the Flight Spacecraft in order to minimize tip-off rates arising during separation.

4.4.10 The resultant force of the separation springs during separation of the Flight Capsule from the Flight Spacecraft shall be within ±0.10 inch of the centerline of the Flight Capsule. Orientation of the separation thrust vector shall be parallel within ±3.5 milliradians to the centerline of the Flight Capsule.

4.5 DIAGNOSTIC MEASUREMENTS

4.5.1 General

All subsystems and critical components shall be equipped with diagnostic outputs to the telecommunications subsystem which will provide the following:

1. Information to determine the cause of a mission failure.
2. Information to determine the cause of a subsystem failure
3. Information to determine the cause of critical component failure.

4. Information which is necessary for more accurate reduction of engineering data.

5. Information which will aid in determining the performance of subsystems and components.

4.5.2 Output Format

All voltage, current, temperature, pressure, vibration and ablation outputs shall be 0 to 5 volts d c analog. All separation, initiation and continuity outputs shall be in the form of on/off or binary data.

4.5.3 Measurements

The measurements to be made shall be in accordance with telecommunications data-handling requirements presented in paragraph 6.11.2.6.

4.6 SAFETY

4.6.1 General

Factors that present a real or potential hazard to equipment and/or personnel, typically include pyrotechnic devices, solid propellants, high-pressure gas-storage vessels, high voltages, toxic materials, and radioactive materials.

All systems shall consider all these, and any other equipment or personnel hazards, in the design of flight and ground equipment, and in the handling, use and testing thereof. Safety factors for pressure vessels, lines and valves shall be considered, and the rationale for determining the tradeoffs between vehicle performance and personnel safety shall be documented. At the same time, the hazards to the flight equipment itself shall be considered. Consideration shall 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, radio-active sources used, etc. The design of hoisting, handling, and testing fixtures shall give special attention to minimize hazard to both personnel and equipment. The procedures utilized to fill pressure vessels, install and connect squibs, and load rocket propellants shall also consider these safety aspects.

4.6.2 Pyrotechnics

If a pyrotechnic device is utilized, a switch (or switches) shall be incorporated to maintain the equipment in a safe condition until such time as activation of the pyrotechnic will not cause damage to personnel or equipment. Any unlatching device shall incorporate a safety device to protect against spurious signals. Electroexplosive devices, associated wiring and firing circuitry shall conform to AFETRP 80-2, General Range Safety Plan, Volume I, and associated Appendix A.

4.6.3 Pressure Vessels

Pressure vessels for flight hardware shall be considered in two categories; hazardous to personnel and nonhazardous to personnel. Hazardous vessels shall be designed with a safety factor of 2.2 (burst pressure/operating pressure). Nonhazardous vessels will carry the same safety factor as the associated structure. Rocket motor cases shall be designed to a factor of at least 1.15, based on yield strength. Special attention shall be given to the method of mounting pressure vessels to avoid undue high-stress concentrations during pressurization. Designs with minimum welding (integral ports and integral mounting bosses) are preferred. Vessels designed with a wall thickness-to-diameter ratio smaller than 1/1000 shall be avoided.

4.6.4 Sterilization Constraints

The entire flight capsule, complete with ΔV rocket motor, spin rockets, compressed gas ACS subsystem and a large number of electroexplosive devices (EED) used for separation and deployment, will be hermetically sealed inside the sterilization canister.

The assembly of the hazardous propellants, compressed gasses and EED's onto the Flight Capsule; the testing; sterilization; post sterilization testing and storage must be done in an Explosive Safe Facility.

4.7 ELECTRICAL BONDING AND SHIELDING*

4.8 ELECTROMAGNETIC INTERFACE

The Capsule System subsystems shall be designed to be compatible with the requirements of MIL-I-26600 Interference Control Requirements and other applicable NASA specifications.

4.9 MAGNETIC CLEANLINESS*

4.10 MAINTENANCE*

4.11 USEFUL LIFE

After assembly and sterilization, the Flight Capsule is required to survive 6-months storage, 500 hours of test and checkout, 90 minutes of launch and Earth parking orbit, 234 days of interplanetary transit, 30 minutes from planetary entry to impact, and 24 hours of planetary surface operation.

4.12 TRANSPORTATION

All Flight Capsule equipment must be capable of withstanding the transportation environments list in Figure 2.

4.13 DESIGN AND CONSTRUCTION STANDARDS*

4.14 WEIGHT

The weight of the Flight Capsule on board the Flight Spacecraft shall be no more than 2500 pounds for the 1971 launch opportunity.

4.15 SCHEDULE

Since the mission objectives involve the 1971 Mars opportunity, Flight Capsule designs, techniques and components must be compatible with that schedule. On this basis, the Flight Capsule design technology cutoff date for the 1971 mission shall be taken as September 1966.

4.16 TRAJECTORY

4.16.1 Definitions

1. A launch opportunity is a reoccuring duration of time, every 25.6 months, when favorable Earth-Mars spatial positions allow for practical interplanetary transfer trajectories.

2. A launch period is the number of days within the launch opportunity when practical Earth-Mars transfer trajectories are selected depending on mission objectives and Launch Vehicle constraints.

3. A launch window is the duration of time each Earth day when Space-Vehicle launch is practical to achieve desired Planetary Vehicle transfer orbit orientation and characteristics depending on mission objectives and Launch-Vehicle constraints.

4.16.2 Selected Design Conditions

The Flight Capsule design shall be compatible with the following trajectory conditions:

1. Launch opportunity	1971
2. Launch period	10 May 1971 to 29 June 1971
3. Launch window	2 hours minimum

4.	ZAP angle	70 to 76 degrees
5.	Communications range	176 x 10^6km maximum
6.	Approach velocity	2.85 to 3.13 km/sec
7.	Departure velocity	2.80 to 4.24 km/sec
8.	Time of flight	184 to 234 days
9.	Arrival date	30 December 1971

5.0 CAPSULE SYSTEM DEFINITION

5.1 FLIGHT CAPSULE TERMINOLOGY

Figure 5 presents a further breakdown of the Flight Capsule, identifying the terminology at the operational stages of separation and/or deployment. In summary, the Flight Capsule is attached to the Flight Spacecraft by the forward and aft sections of the Flight Capsule to Flight Spacecraft adapter. Operation of the sterilization canister lid separation mechanism followed by the operation of the separation system of the Flight Capsule to Flight Spacecraft adapter, results in the Separated Vehicle. Attitude Control and propulsion maneuvers are performed to place the Separated Vehicle on a preselected planetary impact trajectory. After these maneuvers, the propulsion and ACS electronics assembly is separated and the resultant Entry Vehicle cruises to and enters the Martian atmosphere. After entry, the entry shell (including the ACS reaction subsystem and spin/despin rockets) is separated and the Suspended Capsule descends through the atmosphere suspended on the parachute. At a preselected time during descent, a separation mechanism operation extends the Landed Capsule from the parachute by use of a tether. At impact, the Landed Capsule is separated from the tether for landed operations.

Instrument packages are located in two major groups. One group is externally mounted on the Landed Capsule support structure. The other group is internal to the Landed Capsule. In addition to the selected portions of the science instrumentation mounted externally and internally, appropriate portions of the programming and sequencing, thermal control, telecommunications, electrical power and control, and the signal and power interconnections subsystems hardware are also individually attached and operationally integrated both internally and externally, to accomplish the data acquisition and transmission functions of the Capsule System mission.

5.2 PRIMARY FUNCTIONAL AREA LIST

The Flight Capsule shall contain the following primary functional areas:

1. Sterility Control
2. Separation
3. Programming and Sequencing
4. Trajectory Change Propulsion
5. Attitude Control
6. Descent Retardation Entry

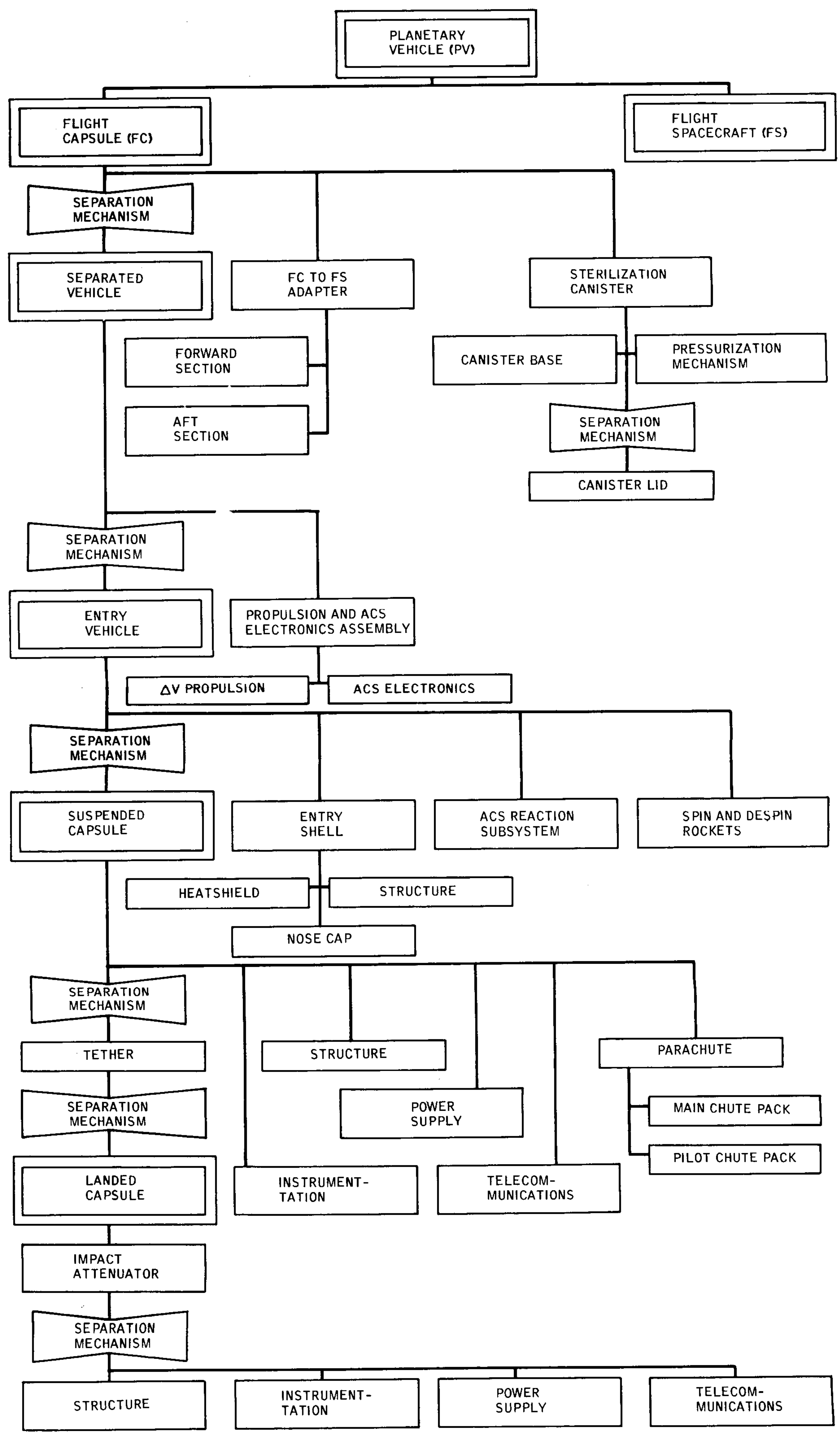

Figure 5 ENTRY FROM APPROACH TRAJECTORY FLIGHT CAPSULE -- OPERATIONAL DIAGRAM AND TERMINOLOGY

7. Thermal Control

8. Descent Retardation - Post Entry

9. Impact Attenuation

10. Data Acquisition - Science Instrumentation

11. Telecommunications

12. Electric Power and Control

13. Signal and Power Interconnection

Section 6.0 of this book contains the functional specification for each of these primary functional areas.

5.3 FUNCTIONAL FLOW DIAGRAM

Figure 6 presents a functional flow diagram of a representative Flight Capsule as defined by the subsystem descriptions of the primary functional areas presented in Section 6.0.

5.4 INTERFACES

5.4.1 General

The interfaces defined herein will be restricted to those which will exist between the Capsule System and other systems, including NASA systems which are involved in the Project. These interfaces are defined from manufacture of the Flight Capsule through to the operations on the planet surface. The many interfaces within the Flight Capsule are not defined herein.

5.4.2 Hardware Interfaces

These are the interfaces which exist between the Capsule System and other systems where actual contact between the two exists. This contact may be mechanical, electrical, electromagnetic, etc.

5.4.2.1 Mechanical

1. Flight Spacecraft

a. Envelope -- The Flight Spacecraft shall be capable of accepting a Flight Capsule with an envelope as defined in Figure 3.

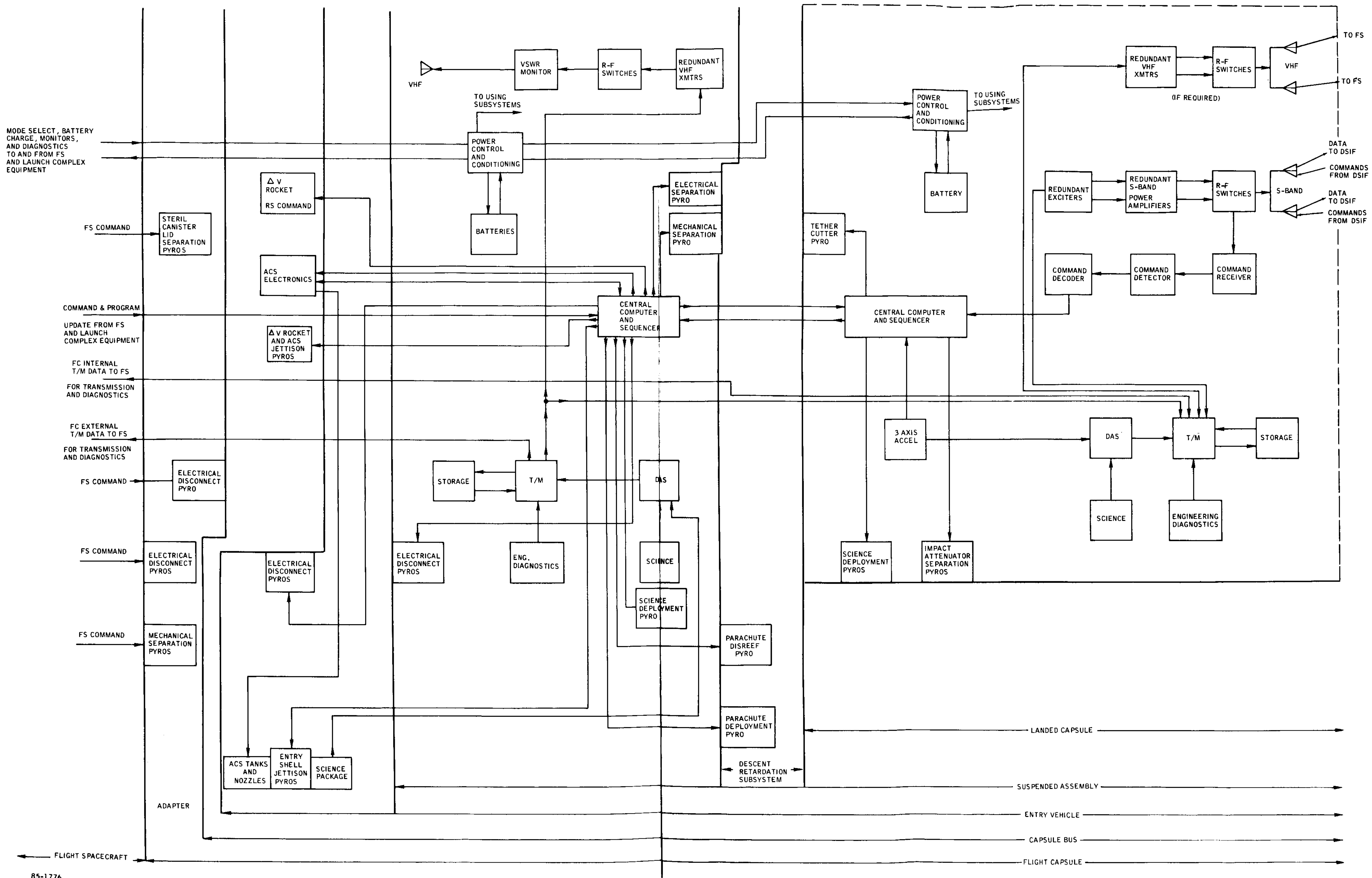

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Figure 6 CAPSULE SYSTEM FUNCTIONAL FLOW DIAGRAM

b. Mounting Pattern -- The Flight Capsule to Flight Spacecraft mounting adapter shall have an 80-inch diameter bolt circle field joint.

c. Alignment -- Alignment between the Flight Spacecraft reference axes and the Flight Capsule reference axis shall be maintained as specified in paragraph 4.4.

2. Launch Vehicle -- The Launch Vehicle shall be capable of inserting the Planetary Vehicle into the interplanetary trajectory. The weight of the Flight Capsule portion of the Planetary Vehicle will not exceed 2500 pounds. Adequate dynamic clearance between the Flight Capsule and the Launch Vehicle ascent fairing and instrument compartment shall be maintained throughout all phases of Launch Vehicle operation.

3. Operational Support Equipment

a. Handling -- The OSE must be capable of providing the Flight Capsule with suitable stands, jigs, fixtures, cradles, cranes, etc. for handling and positioning the Flight Capsule for various tests. These tests will include center of gravity (CG), moment of inertia (MI), product of inertia (PI), fit checks, alignment checks, mating checks and final mating with the Planetary Vehicle mounted on the Launch Vehicle.

b. Transportation -- The OSE shall provide shipping containers capable of maintaining the environmental conditions specified during handling and transporting the Flight Capsule from one facility to another.

c. Test and Checkout -- The OSE shall provide fixtures which can be used in performing fit and alignment acceptance tests.

5.4.2.2 Structural

1. Flight Spacecraft

a. Weight -- The Flight Spacecraft shall be capable of supporting the Flight Capsule weight of 2500 pounds maximum during all preflight, launch through injection, and post-injection phases of the mission.

2. CG, MI, PI

a. OSE -- The OSE must be capable of performing its functions with a Flight-Capsule weight that is not in excess of 2500 pounds.

5.4.2.3 Electrical

1. Flight Spacecraft

a. Umbilicals -- The Flight Spacecraft shall be equipped with umbilical connectors to both the Flight Capsule and to the Launch Vehicle. These connectors shall be capable of handling the signals, power and communications as specified below. The information supplied through these umbilicals will be used during ground checkout to feed signals, power and control functions from the OSE through the Flight Spacecraft to the Flight Capsule. In addition to ground operations, the Flight Spacecraft to Flight Capsule umbilical will be utilized during interplanetary cruise in support of the Flight Capsule for diagnostic purposes.

b. Power -- The Flight Spacecraft shall be capable of supplying 115 watts of 29 volt dc power to the Flight Capsule during the following modes of operation during the mission.

Launch pad checkout - 5 minutes

Launch to injection - 90 minutes

Planetary Vehicle cruise - 2 minutes daily

Preseparation - 4 minutes

The Flight Spacecraft shall be capable of continuously supplying the Flight Capsule with battery trickle charge power of approximately 0.3 amperes at 50 volts during the Planetary Vehicle cruise mode. The Flight Spacecraft shall be capable of continuously supplying the Flight Capsule with battery power during the Planetery Vehicle cruise mode to provide electrical power for thermal control.

c. Signal -- The Flight Spacecraft shall be capable of supplying the Flight Capsule with command information from launch pad operations through interplanetary cruise up to separation. These commands may be generated by the LCE, DSN, or in the Flight Spacecraft central computer and sequencer. These commands will include the initiation of diagnostic data readout from the Flight Capsule to the Flight Spacecraft during various phases of the mission, arming of the

Flight Capsule pyrotechnics, removal of the sterilization canister, separation of umbilicals, and separation of the Flight Capsule from the Flight Spacecraft.

d. Communications -- The Flight Spacecraft shall be capable of accepting telemetry data from the Flight Capsule, via either hard line through the umbilicals or via radio link, for storage and retransmission to the DSN. The Flight Spacecraft shall also receive command information and convert it to a form suitable for use by the Flight Capsule. The Flight Spacecraft shall also provide a means of determining a go-no-go status of the Flight Capsule just prior to separation and generating an appropriate command for post-separation operation.

e. Radio Interference -- The Flight Spacecraft and the Flight Capsule shall be in accordance with MIL-I-26600, Interference Control Requirements, Aeronautical Equipment plus applicable NASA specifications, with respect to susceptability, radiated interference, and conducted interference.

f. Grounding and Bonding -- The Flight Spacecraft shall utilize a single point ground system for all low frequency functions from the Flight Capsule. The mechanical joint between the Flight Capsule and the Flight Spacecraft shall be such as to provide unified behavior when the Planetary Vehicle is excited by a radio frequency field.

2. Launch Vehicle

a. Umbilicals -- The Launch Vehicle shall have umbilical connectors which mate with both the LCE and the PV which are properly interconnected to permit operation of the Flight Capsule from the launch complex including indicators, controls, monitors, etc.

b. Radio Interference -- The Launch Vehicle must conform to MIL-I-26600, Interference Requirements, Aeronautical Equipment plus applicable NASA specifications.

c. Grounding and Bonding -- All grounding and bonding of cable shields through the Launch Vehicle shall be in accordance with a single point grounding philosophy, In general, shields will be carried through umbilical connectors and not connected to the airframe. In instances where shields cannot be carried through connectors, these shields will be connected to airframe ground as close to the connector as possible, but connected to the airframe ground at one point only.

3. Operational Support Equipment

a. Systems Test Complex -- The OSE designed for use in the STC must be capable of functioning the Flight Capsule for complete electrical checkout and also for final acceptance tests. This equipment must be usable on a subsystem level as well as on a system level for these tests. The OSE must apply such functions as mode control, command control, timer updating, external power, battery conditioning, monitor loops, R F and hard line readouts, etc. This equipment shall, however, be designed such that no pyrotechnic or propulsion devices can be inadvertently operated.

b. Launch Complex Equipment -- The LCE portion of the OSE must supply the Flight Capsule with the same functions as the STC with the exception that it operates on a system level only.

c. Flight Capsule Simulators -- The OSE shall obtain devices which are electrically identical to the Flight Capsule at its interfaces. These devices will be used in compatibility testing between the Flight Spacecraft and the Flight Capsule, the Planetary Vehicle and Launch Vehicle, and between the Space Vehicle hardware and the LCE.

d. Mission Dependent Equipment -- The remote MDE used in Flight Capsule and/or Planetary Vehicle information format shall be capable of providing automatic readout of the Flight Capsule and/or Planetary Vehicle data during all phases of the mission.

4. Deep Space Network

a. Deep Space Instrumentation Facility -- The DSIF shall be capable of transmitting commands to and receiving telemetry data from the Flight Capsule, in combination with the MDE. The DSIF shall be capable of providing the MOS with range and tracking information for up-dating timers, selecting back up modes, etc.

b. Deep Space Operations Facility -- The DSOF shall be capable of accepting data from the DSIF, interpreting this data, and supplying the DSIF with appropriate information for command of the capsule.

5.4.2.4 Safety

1. OSE -- The OSE must be designed to provide the maximum safety to personnel and equipment during all handling, checkout, mating and launch phases of the mission. Specific areas which must be considered are pyrotechnic devices, pressure vessels, high voltage devices, grounding and bonding, etc.

2. Facilities -- Explosive safe facilities must be provided at ETR for propellant and pyrotechnic installation, and storage.

5.4.2.5 Thermal

1. Transportation and Storage -- During transportation and storage, the OSE shipping and storage containers shall keep the temperature experienced by the Flight Capsule within the range of 35 to 125°F.

2. Launch Pad -- While the Flight Capsule is mounted on the Flight Spacecraft the LCE shall provide (if necessary) air conditioning of the Flight Capsule such that the temperature environment limits do not exceed 35 to 125°F.

3. Flight Spacecraft -- The Flight Spacecraft and Flight Capsule shall provide a thermally balanced interface such that the two systems function properly when combined as Planetary Vehicle alone, while mounted on the Launch Vehicle, and during the interplanetary cruise phase; as well as operating separately in space after separation.

5.4.3 Software Interfaces

Software interfaces are those which exist between the Capsule System and the other systems and/or subsystems with respect to scheduling, planning, procedures, training, decisions, facilities, administration, etc.

5.4.3.1 Eastern Test Range

The major interface between the Capsule System and ETR will be scheduling and mission planning. A plan must be developed to provide a suitable daily launch window over the specified launch period. During these daily launch windows, all communications, tracking, and data acquisition facilities must be available for the program. In addition, ETR must provide storage, laboratory, assembly, test, and administrative areas for the Capsule-System contractor.

5.4.3.2 Operational Support Equipment

Interfaces between the Capsule System and the OSE will be one of developing schedules, test plans, test procedures, handling procedures, safety procedures, and test requirements.

5.4.3.3 Deep Space Network

Interfaces between the Capsule System and the DSN will include scheduling test plans, compatibility and test requirements.

5.4.3.4 Mission Operations System

Interfaces between the Capsule System and the MOS include scheduling, vehicle handling, and mission operations planning.

5.4.3.5 Flight Spacecraft

Flight Capsule interfaces include but are not limited to the following mission operations: scheduling, trajectory, arrival geometry, orbit geometry, lifetime, operating sequence, communications requirements, data acquisition and handling requirements, post launch decisions, logic, etc.

5.4.3.6 Launch Vehicle

Launch Vehicle interfaces with the Capsule System will include scheduling and performance capability.

5.5 EVENT SEQUENCE

The event sequence shall be considered to begin at such time as all systems have been integrated into the Space Vehicle ready to launch. The following sequence is typical of a Mars entry from approach trajectory mission, with emphasis on the Flight Capsule operations. The abbreviations shown in Table III are used in the sequence in addition to those defined in paragraph 1.3. Table IV is the event sequence from prelaunch terminal checkout of all the flight systems until separation of the Flight Capsule from the Flight Spacecraft.

Table V lists Flight Capsule events and Table VI lists Flight Spacecraft events after separation to the end of the Landed Capsule surface operation.

5.6 CONFIGURATION

5.6.1 Design Description

The inboard profiles of a Flight Capsule design are presented in Figures 7 and 8, in the launch and entry configurations respectively. The Flight Capsule utilizes the blunt cone (60-degree half-angle) as the entry shell configuration. The Landed Capsule is an oblate spheroid (lenticular) shape. The Flight Capsule configuration has several major assemblies: sterilization canister, Flight Capsule to Flight Spacecraft adapter, attitude control and ΔV propulsion assembly, Landed Capsule support structure, Landed Capsule and entry shell.

TABLE III

ABBREVIATIONS USED IN THE EVENT SEQUENCE

T	-	Time of SV launch
I	-	Time of PV injection into interplanetary transfer orbit
M	-	Time of PV or FS trajectory correction maneuver
S	-	Time of separation of FC from FS
E	-	Time of Entry Vehicle entry into Mars atmosphere
L	-	Time of Landed Capsule landing on Mars surface
P	-	Time of FS arrival at initial Mars orbit periapsis
RF	-	Radio frequency
AC	-	Attitude control
GC	-	Guidance and control
PS	-	Power supply
CCS	-	Central computer and sequencer
EDM	-	Engineering diagnostic module
comm	-	Communications subsystems
prop	-	Propulsion subsystem
accel	-	Accelerometer
pyro	-	Pyrotechnic

TABLE IV

EVENT SEQUENCE - PRELAUNCH TO SEPARATION

Event	Time Nominal	Commands		Comments
		Source	Destination	
1 FC terminal checkout	T - 180 mins	LCE - FC	FC-subsystems	Operate FC on ground power
2 LV and FS terminal checkout	T - 90 mins	LCE - LV LCE - FS	LV-subsystems FS-subsystems	Operate LV and FS on ground power
3 Switch LV and FS to internal power	T - 5 mins	LCE - LV LCE - FS	LV-subsystems FS-subsystems	Start of automatic count down procedures
4 Switch FC to FS power	T - 4 mins	LCE - FS	FS-CCS	
5 Up date all time dependent commands and enable all CCS's and LV GC	T - 3 mins	LCE - LV LCE - FS LCE - FC	LV-CCS+GC FS-CCS FC-CCS	
6 Release all lockouts	T - 2 mins	LCE - LV LCE - FS LCE - FC	LV-subsystems FS-subsystems FC-subsystems	
7 Turn on FC and FS diagnostic instruments and FS communications transmitter at low power appropriate to ground operations	T - 1 min	CCS - FS CCS - FC	FS-comm FC-EDM FS-EDM	FS and FC diagnostic data played out thru parasitic antenna on ascent fairing and thru hard line to LCE
8 Ignite first stage engines	T - 20 secs	CCS - LV	S-IB prop	
9 Release all launch pad to LV umbilicals	T - 5 secs	LCE - LV	Launch pad umbilicals	Discontinue all hard line monitoring and control of LV, FS and FC
10 Lift-off	T - 0	Event		
11 First stage cutoff	T + 140 secs	GC - LV	S-IB prop	
12 Separate first stage from second stage	T + 142 secs	CCS - LV	S-IB separation pyro	
13 Second stage ignition	T + 145 secs	CCS - LV	S-IV B prop	
14 Jettison ascent fairing	T + 160 secs	CCS - LV	ascent fairing pyro	Parasitic antenna jettisoned with ascent fairing at 350,000 feet altitude
15 Continue RF playout of FV diagnostic monitoring through low gain antenna on FS	T + 162 secs	CCS - FS	FS - comm	
16 Second stage cutoff	T + 590 secs	G C - LV	S-IV B prop	
17 Separate second stage from third stage	T + 592 secs	CCS - LV	S-IV B separation pyro	600,000 feet altitude
18 Start third stage propellant settling	T + 591 secs	CCS - LV	Centaur ullage prop	

TABLE IV (Cont'd)

EVENT SEQUENCE - PRELAUNCH TO SEPARATION

Event	Time Nominal	Commands		Comments
		Source	Destination	
19 Start third stage first burn, shut down ullage propulsion	T + 598 secs	CCS-LV	Centaur ullage and main prop	
20 Third stage first cut-off, start ullage propulsion	T + 998 secs	GS - LV	Centaur ullage and main prop	Begin parking orbit at 10^6 feet altitude
21 Reduce ullage propulsion thrust level	T + 1010 secs	CCS - LV	Ullage prop	
22 Begin third stage attitude maneuver	T + 1011 secs	GC - LV	AC - LV	
23 Turn PV cruise science on	T + 1020 secs	CCS - FS	PV science	
24 Third stage 2nd ignition, shut down ullage propulsion	T + 2510 secs	GC - LV	Centaur ullage and main prop	
25 Third stage 2nd cutoff	T + 2540	GC - LV	Centaur prop	
26 Third stage altitude maneuver	T + 2600	GC - LV	AC - LV	
27 PV injection into transfer orbit	I = 0 (T + 2600 sec)	Event		
28 PV separation from third stage	I + 3 mins	CCS - LV	Centaur separation pyro	
29 Start PV post separation sequence	I + 180 secs	separation switch	CCS - FS	
30 Turn on FS communications to full power	I + 181 secs	CCS - FS	FS comm	
31 Arm FS pyrotechnics	I + 181 secs	CCS - FS	FS pyro	
32 Deploy FS solar panels	I + 182 secs	CCS - FS	Solar panel pyro	
33 Activate FS AC	I + 183 secs	CCS - FS	FS - AC	
34 Activate sun sensor and acquire sun	I + 4 mins to I + 11 mins	CCS - FS	Sun sensor	FS AC is commanded by sun sensor for pitch and yaw reference
35 Begin trickle charge of FC batteries and active thermal control of FC subsystems with FS power	I + 11 mins	CCS - FS	FS - PS	
36 Activate Canopus tracker and acquire Canopus	I + 11 mins to I + 31 mins	CCS - FS	Canopus tracker	FS AC is commanded by Canopus tracker for roll reference

TABLE IV (Cont'd)

EVENT SEQUENCE - PRELAUNCH TO SEPARATION

Event	Time Nominal	Commands		Comments
		Source	Destination	
37 PV trajectory verification	I + 1 day	PV	DSN	DSIF tracking
38 Transmit trajectory correction commands as required: a - Pitch angle b - Yaw angle c - Velocity magnitude (burn time)	I + 2 days to I + 10 days	DSIF	FS-CCS	MOS compares actual trajectory to nominal requirements and selects commands for trajectory correction maneuver as required
39 Transmit maneuver start command	I + 2 days to I + 10 days M = 0	DSIF Event	FS-CCS	Two way communication is maintained between the PV and DSIF to allow the MOS to interrogate and command the PV. Status of all FC subsystems is continually monitored through use of FS power and played out with FS status as requested
40 Activate FS - AC gyros for warmup	M + 1 min	FS - CCS	FS-AC	
41 Switch FS to inertial control - (autopilot) and a. Deactivate canopus tracker b. Deactivate sun tracker	M + 60 mins	FS - CCS	FS-AC	
42 Start pitch turn	M + 60 mins	FS - CCS	FS-AC	
43 Stop pitch turn	M + 67 mins (maximum)	FS - CCS	FS-AC	
44 Start yaw turn	M + 67 mins	FS - CCS	FS-AC	
45 Stop yaw turn	M + 74 mins (maximum)	FS - CCS	FS-AC	
46 Ignite midcourse correction propulsion	M + 74 mins	FS - CCS	FS-prop	
47 Cutoff midcourse correction propulsion	M + 76 mins (nominal)	FS - CCS	FS-prop	
48 Switch FS to cruise trajectory acquisition cycle, deactivate autopilot	M + 76 mins	FS - CCS	FS-AC	
49 Reaquire sun	0 to 6 min	FS - CCS	FS-AC	
50 Reacquire Canopus	0 to 20 min	FS - CCS	FS-AC	
51 Confirm reacquisition of PV cruise condition	M + 102 mins	FS - CCS	FS-AC	DSIF track used as backup

TABLE IV (Cont'd)

EVENT SEQUENCE - PRELAUNCH TO SEPARATION

Event	Time Nominal	Commands		Comments
		Source	Destination	
52 Repeat Events 37 through 51 for each midcourse correction required to adjust PV trajectory	whenever required			
53 Switch PV RF playout to high gain antenna	as determined by range	DSIF	FS - CCS	Change in bit rate may effect amount of FS and FC diagnostic monitoring
54 Canopus sensor resets cone angle 7 to 12 times	preselected	FC - CCS	FS - AC	At descrete intervals throughout cruise
55 PV trajectory verification complete	S - 10 days	PV	DSIF	All commands for FS-FC separation maneuvers, FS orbit maneuvers and FC trajectory change/speed up are stored in the respective CCS units to be updated as required, based on latest trajectory verification by the DSN
56 Transmit updated separation, FC trajectory change/speed up and FS orbit commands based on latest trajectory verification, selected separation time and predicted entry time	S - 10 days to S - 1 day	DSIF	FS - CCS FS - CCS	A series of interrogations and verifications of the PV by the MOS will establish operability of PV for the separation phase
57 Turn on FC gyros for warm up	S - 240 mins	FC - CCS	FC gyro	FC still on FS power
58 Switch FC to internal power and turn on all FC subsystems for checkout	S - 210 mins	FC - CCS FS - CCS	FC subsystems	Preseparation checkout for MOS evaluation prior to final separation commands.
59 Receive, record, and transmit FC status	S - 210 mins	FS-comm	DSIF	
60 Shutdown FC systems except gyros and return FC to FS power	S - 208 mins	FC - CCS FS - CCS	FC-subsystems	FS continues transmission of PV status to MOS
61 MOS confirms PV system checkout and transmits separation commands or alternate commands for failure mode indicated in checkout - "Begin separation sequence"	S - 148 mins	DSIF	OFS	Normal operational sequence does not require 148 mins from "begin separation sequence" command to separation unless an PV maneuver is needed to support a back up mode of operation

TABLE IV (Concl'd)

EVENT SEQUENCE - PRELAUNCH TO SEPARATION

Event	Time Nominal	Commands		Comments
		Source	Destination	
62 Arm FS separation pyro and separation event switch-depressurize sterilization canister	S - 10 mins	FS - CCS	FS pyro Canister lid pyro	
63 Separate sterilization canister lid	S - 5 mins	FS - CCS	Canister lid pyro	
64 Complete trickle charge of FC batteries	S - 5 mins	FS - CCS	FS - PS	
65 Switch FC to internal power	S - 4 mins	FS - CCS FC - CCS	FC subsystems	FC gyro warm up continues of FC power
66 Turn on all FC subsystems for final preseparation checkout	S - 4 mins	FC - CCS	FC subsystems	
67 Checkout FC direct link transmissions	S - 4 mins	FC - comm	DSIF	Release mechanism armed if RF power is adequate
68 Shutdown FC direct link transmissions	S - 2 mins	FC - CCS	FC-comm	
69 Checkout FC relay link transmission	S - 2 mins	FC-comm	FS relay receiver	Release mechanism armed if RF power is adequate
70 Checkout of all FC subsystems complete	S - 1 min	FC - CCS	FC subsystems	Keep FC relay transmitter on for engineering diagnotic check throughout separation operations
71 Arm all FC subsystem pyro	S - 10 secs	FS - CCS	FC - pyro	
72 Separate FS to FC umbilical connectors	S - 1 sec	FS - CCS	Umbilical pyro	
73 Separate FC from FS	S = 0	FS - CCS	Separation pyro	

TABLE V

EVENT SEQUENCE - FLIGHT CAPSULE, SEPARATION TO END OF SURFACE OPERATIONS

Event	Time Nominal	Commands		Comments
		Source	Destination	
1 FC separate from FS	S = 0	Event		All FC subsystems in checkout mode relayed to FS via FC communications from EDM and stored for replay to DSIF
2 Actuate FC AC and start FC - CCS post separation sequence	S - 0	Separation switch	FC - CCS FC - AC	Maintain FC attitude to reference established by FS at separation, nullify tip off rates
3 Verify FC separation	S + 1 sec	FC accel	FC - AC	
4 Begin FC AC maneuver to acquire pitch and yaw changes from separation attitude to ΔV thrust application attitude	S + 100 secs	FC - CCS	FC - AC	
5 Initiate FC ΔV prop	S + 200 secs	FC - CCS	prop pyro	FC AC maintains thrust vector control during propulsion burn period
6 Verify ΔV	S + 200 secs	FC accel	FC - CCS	
7 Shutdown ΔV prop	S + 210 secs nominal	FC - CCS	prop pyro	Shutdown when preprogrammed ΔV is measured and attained
8 Begin FC AC maneuver to acquire pitch and yaw changes from thrust application attitude to zero angle of attack attitude	S + 210 secs	FC - CCS	FC - AC	
9 Spin up FC - Discontinue attitude control	S + 450 secs	FC - CCS	spin up prop pyro and FC - AC	Possible despin maneuver may be implemented to accommodate entry dynamics
10 Jettison prop and AC electronics support structure	S + 452 secs	FC - CCS	support pyro	
11 Complete transmission of separation and post separation operations	S + 500 secs	FC transmitter	FC receiver and DSIF	Data from FC - EDM played out realtime and replayed from storage to FS
12 Shutdown all FC subsystems except CCS timers, maintain monitor of critical subsystems to EDM to be recorded for later playout	S + 500 secs	FC - CCS	FC subsystems	

TABLE V (Cont'd)

EVENT SEQUENCE - FLIGHT CAPSULE, SEPARATION TO END OF SURFACE OPERATIONS

Event	Time Nominal	Commands		Comments
		Source	Destination	
13 Turn on all FC subsystems for FS relay to DSIF of data stored since last playout, plus prior data	S + 1 day	FC-CCS and FS command transmitter	FC subsystems	
14 Shutdown all FC subsystems except CCS timer. Maintain and monitor critical subsystems to EDM to be recorded for later playout	S + 1 day + 2 mins	FC-CCS and FS command transmitter	FC subsystems	
15 Repeat steps 13 and 14 daily until preentry phase	12 days			
16 Start trapped radiation detection - (if experiment is included)	E - 200 mins	FC-CCS and FS command transmitter	FC subsystems	CCS timer signals command receiver, DAS, telemetry, and trapped radiation detector
17 Begin playout accumulated trapped radiation data plus preentry FC subsystems status	E - 17 mins	FC - CCS	FC subsystems	
18 Maintain relay communication in playout status to start transmission of real time entry data	E - 15 mins	FC - CCS	FC subsystems	Tracking and instrument errors require 15 minute allowance between final entry maneuver and expected entry
19 Entry Vehicle enters Mars atmosphere at 800,000 feet altitude	S + 12 days E = 0	Event		
20 Accelerometer senses atmosphere at 0.1 G_e, start real time playout of entry, plus start recording entry data for post blackout playout	E + 20 secs	Accel	FC subsystems	Assume model 3 atmosphere for trajectory dynamics to impact. Transmission throughout blackout allows maximum transmission of data before and after blackout
21 10,000 fps velocity, end of communication blackout, continue transmission of stored data plus new data as available	E + 29 secs	FC comm	FS receiver	
22 10 G_e descending -- end of entry, beginning of descent	E + 35 secs	Event		

TABLE V (Cont'd)

EVENT SEQUENCE - FLIGHT CAPSULE, SEPARATION TO END OF SURFACE OPERATIONS

Event	Time Nominal	Commands		Comments
		Source	Destination	
23 Radar Altimeter senses 21,000 ft. altitude to deploy reefed chute and jettison entry shell	E + 42 secs	Radar altimeter	Chute pyro	
24 Accelerometer senses 1.38 G_e descending to deploy reefed chute and jettison entry shell	E + 42 secs	Accel	Chute pyro	Backup mode - if chute fails, data format to play-out priority information first- If chute operates as programmed, format to playout all data in series
25 Radar altimeter senses 19,000 ft. altitude to open chute fully	E + 44 secs	Radar altimeter	Chute pyro	
26 Open chute fully	E + 44 secs	FC - CCS	Chute pyro	Backup mode timed from deploy reefed chute event
27 Deploy science probes	E + to be determined	FC - CCS	Science pyro	Specific scientific, instrument probes to be deployed at selected descent conditions
28 Deploy tether, end of descent transmission, start storage of impact diagnostic data	L - 6 secs E + 244 secs (500 ft. altitude)	Radar altimeter	Tether pyro and FC comm	Assume 200 seconds of transmission time from deployment of reefed chute to deployment of tether adequate for two complete cycles of entry and descent data
29 Landed Capsule, beginning of landed operations	E + 250 secs L = 0	Event		
30 Jettison chute, external payload and tether, switch transmitter to landed operations mode	L + 1 sec	Impact switch	FC - CCS	
31 Shut off impact accelerometers and jettison impact attenuator	L + 10 secs	FC - CCS	FC systems	
32 Deploy articulated scientific and/or engineering instruments	L + 3 min	FC - CCS	Articulated subsystems	Allow three minutes for the payload to come to rest before activation of articulated systems
33 Start recording of landed data	L + 3 min	FC - CCS	Science subsystems	
34 Start relay and direct playout of recorded entry and descent data followed by landed data	L + 5 min	FC - CCS	FC comm to FS receiver and DSIF	

TABLE V (Concl'd)

EVENT SEQUENCE - FLIGHT CAPSULE, SEPARATION TO END OF SURFACE OPERATIONS

Event	Time Nominal	Commands		Comments
		Source	Destination	
35 Stop relay transmission - continue recording of science and engineering data to be played out at appropriate orbit of FS as commanded by FS.	L + A mins	FC - CCS	FC relay comm	If surface to FS relay link is implemented and FS is in favorable position in flyby trajectory – Time A of playout dependent on data quantity and rate
36 Stop direct transmission	L + B mins	FC - CCS	FC direct comm	Time B of playout dependant on data quantity and rate.
37 Start direct transmission of stored data since last interrogation, plus, original entry, descent and landed data	L + 4.5 hrs L + 28.5 hrs	FC comm and/or DSIF	FC direct comm	Favorable correlation of DSIF antennas will select playout time after L
38 Stop direct transmission continue recording of science and engineering data to be played out at next appropriate earth position	L + 5.0 hrs L + 29.0 hrs	FC - CCS	FC direct comm	Stop command based on FC-CCS selection of time needed to play out several sets of stored data
39 Start relay transmission to stored data since last interrogation, plus, original entry, descent and landed data.	L + X hrs	FS command transmitter	FC command receiver	Start command from FS command transmitter based on orbit FS attained– Time X based on FS orbital period
40 Stop relay transmission continue recording of science and engineering data to be played out at next appropriate FS orbit position	L + X hrs + N mins	FC - CCS	FC systems	Stop command based on FC-CCS selection of N time needed to play out several sets of stored data
41 Repeat steps 39 and 40 for each successive favorable orbit until exhaustion of landed power supply	L + Y hrs	FC comm	FS receiver	Time Y based on FS orbital period
42 End of landed mission	L + 24 hrs minimum			

TABLE VI

EVENT SEQUENCE - FLIGHT SPACECRAFT, SEPARATION TO END OF SURFACE OPERATIONS

	Event	Time Nominal	Commands Source	Commands Destination	Comments
1	FS Separates from FC	S = 0	Event		FS receiving FC diagnostic data during separation for storage, handling and retransmission to DSIF
2	Verify separation	S = 0	Separation Switch	FS comm.	Include separation event monitor with FC status information to DSIF
3	Discontinue receipt of FC separation operations	S + 500 secs	FS-CCS	FS receiver	
4	Command FC to transmit and turn on FS relay receiver to receive first FC cruise checkout data: to be relayed to DSIF	S + 1 day	FS-CCS and command transmitter	FS comm receiver and FC command transmitter	Alternate command by FC CCS
5	Turn off FS relay receiver	S + 1 day + 2 mins	FS-CCS	FS communications receiver	Time determined by bit rate and quantity
6	Repeat steps 4 and 5 daily until FC preentry	12 days			
7	Command FC to transmit and turn on FS relay receiver to receive FC preentry checkout, entry descent and initial surface operations data; to be relayed to DSIF	E-200 mins S + 12 days	FS-CCS	FS-Comm receiver	
8	Turn off relay receiver	L+A mins	FS-CCS	FS receiver	Time A of playout dependent of data rate and quantity

TABLE VI (Concl'd)

EVENT SEQUENCE - FLIGHT SPACECRAFT, SEPARATION TO END OF SURFACE OPERATIONS

	Event	Time Nominal	Commands Source	Commands Destination	Comments
9	Repeat prelaunch to separation event sequence events 37 through 51 for FS trajectory corrections prior to orbit insertion	P-3 hrs. to P-10 mins.			
10	FS orbit insertion maneuvers	P-10 mins to P=0	FS-AC and CCS	AC and retro prop	
11	FS orbit achieved	P = 0 L + 3 hrs	Event		
12	Confirm orbit achieved and position relative to Landed Capsule	P + 1 hrs to P+4 hours	FS	MOS	If Landed Capsule to FS relay link is implemented, series of interrogations and commands will be necessary between FS and MOS to establish orbit and update FS command transmitter program to be in sequence with landed FC for favorable relay transmission from Landed Capsule to FS - If no landed relay link to implemented, the FS has no further function for the Landed Capsule.
13	Command Landed Capsule to start transmission of data, start relay of data from FC to DSIF	P+X+3 hrs	FS command transmitter	FC command receiver	
14	FC stops transmitting data-shutdown FS receiver	P+X+3 hrs +N mins	FS-CCS	FS comm receiver	
15	Repeat steps 13 and 14 for each successive orbit until exhaustion of landed power supply				
16	End of landed mission FS remains in orbit to accomplish its orbiting science mission, etc.	L+24 hrs P+21 hrs (minimum)			

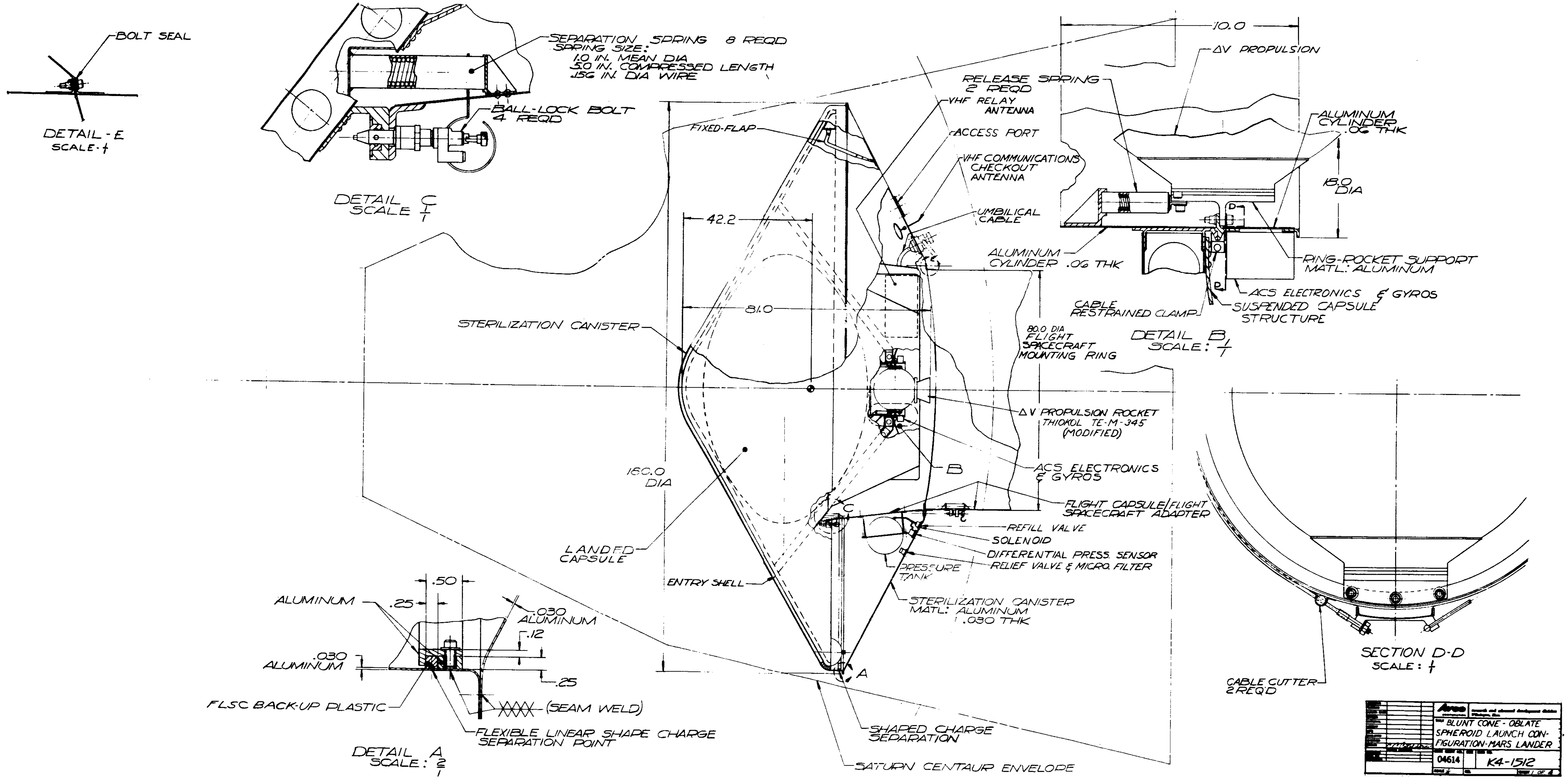


Figure 7 BLUNT CONE -- OBLATE SPHEROID LAUNCH CONFIGURATION

51-1

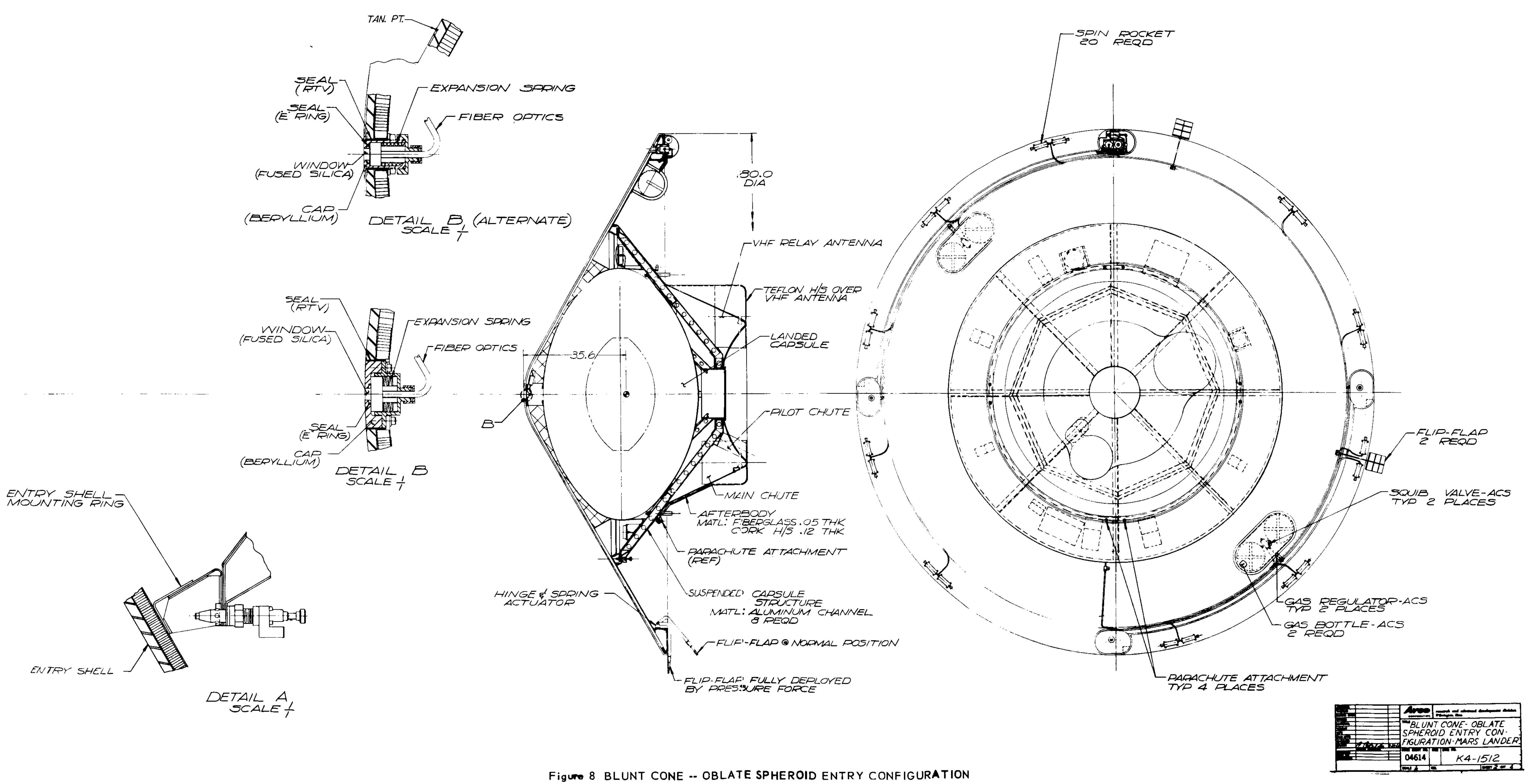


Figure 8 BLUNT CONE -- OBLATE SPHEROID ENTRY CONFIGURATION

52-1

5.6.1.1 Sterilization Canister

This assembly is constructed of a thin shell that completely encloses the Separated Vehicle plus a portion of the Flight Spacecraft to Flight Capsule adapter. Local access areas are provided for assembly and handling. A pressurization system is utilized to maintain a slight pressure differential across the canister from manufacture to separation of the canister lid.

5.6.1.2 Flight Spacecraft to Flight Capsule Adapter

This adapter creates the mounting and load path support system to the Flight Capsule during launch and interplanetary travel. It is constructed of a conical shell with mounting rings at either end. This adapter also houses the Separated Vehicle separation mechanism, located at the Landed Capsule support structure interface.

5.6.1.3 Attitude Control and ΔV Propulsion Assembly

These assemblies are mounted on the aft end (toward Launch Vehicle) of the Landed Capsule support structure and are jettisoned by a simple Marman clamp - spring release system after Flight Capsule separation, maneuver and thrusting. A portion of the ACS (tankage, regulators, nozzles, etc.) are mounted on the entry shell at the outer periphery, to be jettisoned with the entry shell.

5.6.1.4 Landed Capsule Support Structure

The support structure serves several purposes: 1) it provides a load-path system for the Separated Vehicle during launch loads, for ΔV thrusting loads and for parachute loads, 2) it forms the mounting surface for all of the external instrumentation and the parachute subsystem, 3) it structurally supports the Landed Capsule through its flight history, and 4) it provides the mounting and sepration functions for the entry shell. The basic structure consists of eight beams running radially from the ΔV propulsion unit to the entry shell interface.

5.6.1.5 Landed Capsule

The Landed Capsule consists of an impact attenuator, internal structures, internal payload (science, power, telecommunications, etc.) and science deployment mechanisms. Design details are presented in Figure 9. The impact attenuator is a foam-fille fiber-glass-honeycomb crushable-material, approximately 15 inches in depth. The crushable material is formed in a series of three layers, bonded together by thin fiberglass shells at assembly. A thin layer of balsa wood (approximately 1-inch thick) is bonded around the internal structure.

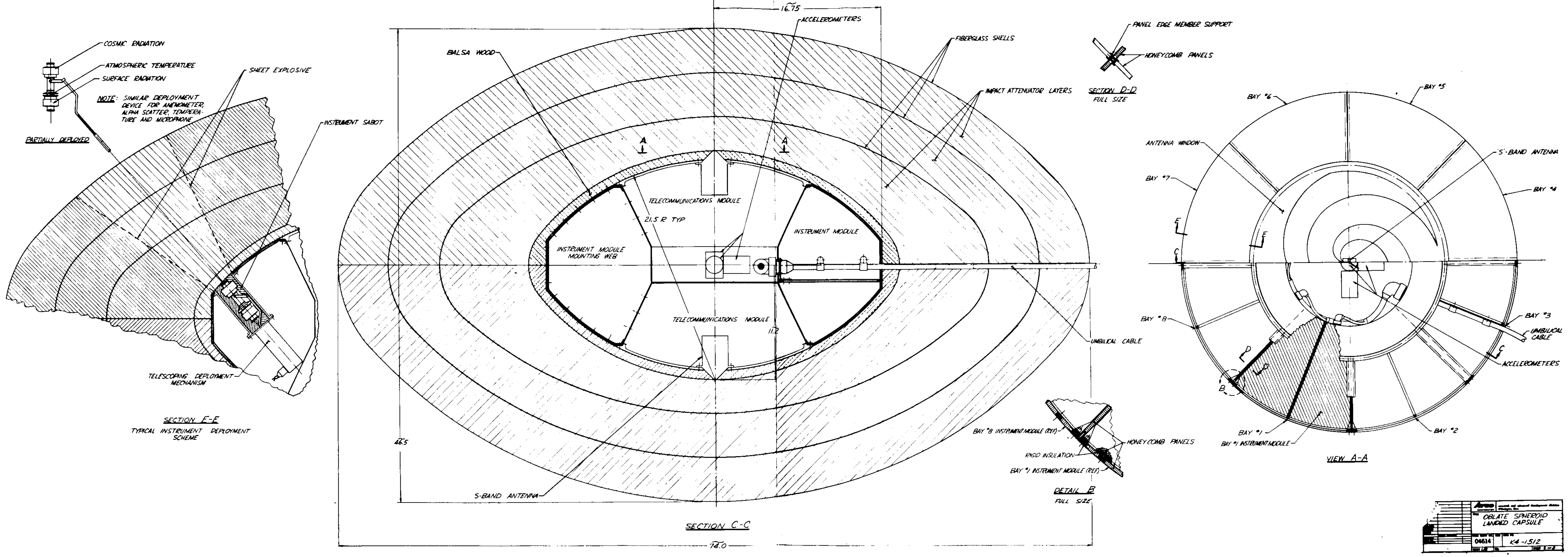


Figure 9 OBLATE SPHEROID LANDED CAPSULE

5.6.1.6 The internal structure

As indicated in Figure 9 the internal structure is composed of several bays which support modules of equipment. This modular construction facilitates the fabrication and checkout of the assembly. Each bay is a major structural element of the internal structure and hence provides the structural continuity. The inside wall of each bay supports the two VHF antenna cavities located in the cylindrical section in the center of the Landed Capsule. Inside the VHF antennas (surface spiral) are two S-band antennas (V-horn type).

5.6.1.7 Entry shell

The entry shell is constructed of a thermal protection, a primary structure and a secondary structure. The thermal protection utilizes a cork ablator as both the primary heat shield (on the forward side of primary structure) and secondary heat shield (afterbody of primary structure). The heat shield is bonded directly to the primary structure. The primary structure is constructed of gores of beryllium face sheets and stainless steel core. Stainless-steel weldments and beryllium splice plates are utilized to build up the complete structure. A beryllium-end closeout ring assembly forms the structural continuity at the outer edge of the primary structure. A secondary structure, composed of a beryllium ring assembly, is bolted to the primary structure at the location of the Landed Capsule support structure interface. This assembly is the mounting and separation point for the entire entry shell assembly.

5.6.2 Weight Summary

The summary weight breakdown for the above design is presented in Table VII. This breakdown is presented in terms of operational sequences starting with the Flight Capsule weight and concluding with internal weight of the Landed Capsule.

TABLE VII

WEIGHT SUMMARY

	Calculated (C) or Estimated (E)	Total Weight	
LAUNCH		(2500.0)	
Sterile Canister Lid	C		125.0
PRE-FC SEPARATION		(2375.0)	
Sterile Canister Base with Pressurization System	C		241.9
FC/FS Adapter	C		100.0
Electrical and Mechanical Connectors	E		50.0
SEPARATED VEHICLE		(1983.1)	
ΔF Propulsion	C		98.5
ACS Electronics	C		10.0
Spin Rocket Propellants	C		2.1
Propulsion Support Structure	C		10.0
Miscellaneous			12.5
ENTRY VEHICLE		(1850.0)	
Thermal Protection			290.0
Primary Structure			451.2
Thermal Control			25.0
Electrical and Mechanical Connectors			55.5
ACS Nozzles, Tanks, etc.			69.3
Spin Rockets and Supports			10.0
Contingency			25.0
SUSPENDED CAPSULE		(924.0)	
Science and Instrumentation			36.9
Telecommunications			20.6
Power			31.1
Miscellaneous			3.4
Contingency			22.0
Main Chute, Pilot, Mortar			74.0
Support Structure			65.0
Afterbody			76.0
LANDED CAPSULE		(595.0)	
Impact Attenuator			215.0
Electrical and Mechanical Connectors			15.5
INTERNAL WEIGHT		(364.5)	
Science and Instrumentation			48.0
Telecommunications			98.7
Power			70.1
Miscellaneous			2.0
Contingency			54.7
Thermal Control			15.0
Internal Structure			76.0

6.0 FLIGHT CAPSULE PRIMARY FUNCTIONAL AREAS

The following section presents a summary subsystem description of each of the primary functional areas within the Flight Capsule. The level of detail presented for each of these subsystems is compatible with the requirement for subsystem information needed to conduct a meaningful tradeoff analysis and the subsequent synthesis of the typical Capsule System defined herein.

6.1 STERILITY CONTROL

6.1.1 Functions

The sterilization canister shall maintain the required sterile environment and provide a continuous surface for passive thermal control to the enclosed Separated Vehicle, from the end of terminal sterilization until the canister lid is jettisoned.

6.1.2 Performance/Design Requirements

6.1.2.1 Performance

After sealing the sterilization canister with an auxiliary tank of gas connected to the fill line, the canister shall be brought up to sterilization temperature, 135°C (145 degrees for qualification). During temperature soak the expanding gas in the interior of the Flight Capsule will bleed off through a pressure relief valve. As the temperature drops at the end of the cycle, the pressure in the canister shall be brought up to greater than ambient by opening the valve to an auxiliary tank. Further investigation is needed to determine pressure requirements and allowables, and to determine if an alternate design approach is feasible which would include eliminating the auxiliary tank and plugging the relief valve outlet until the Flight Capsule has cooled to ambient temperatures.

The decrease in ambient pressure as the Launch Vehicle ascends from the launch pad shall be matched by a decrease in the internal pressure by bleed-off through the relief valve. The pressure regulator on the pressurizing tank shall sense both the internal Flight Capsule pressure and the ambient pressure and thus allow it to add gas to the internal package only when the differential pressure falls below a set value (less than 1 psi). As the ambient pressure stabilizes, the relief valve closes and further loss of pressure shall be by leakage only. The pressure regulator shall maintain the required differential pressure during the transfer through space.

During the Planetary Vehicle cruise phase, the sterilization canister shall act as a passive thermal control unit. Since the Flight Capsule will be on the shaded side of the Planetary Vehicle, most of its surface will be exposed to the near absolute zero temperature of outer space. The loss of heat from the Flight Capsule shall be minimized by low emissivity finishes on the sterilization canister.

The separation of the canister lid shall be accomplished with a flexible linear-shaped charge prior to planet encounter with a separation velocity greater than that of the subsequent velocity of the Separated Vehicle. Just prior to lid separation, the internal pressure in the sterilization canister shall be vented by opening the relief valve. (To reduce spacecraft attitude perturbations, the vent of the relief valve shall end in a tee.) An electric signal to the igniter circuit ignites the FLSC detonator which in turn ignites the flexible linear shaped charge to open the canister and move the lid away from canister base.

6.1.2.2 Design

1. Surface emissivity: Inside - to be determined
Outside - to be determined

2. Internal FC pressure (abs.): Max: Earth ambient + 1 psi
Min: Vented to space ambient prior to lid removal

3. Temperature: Max: 300°F
Min: -300°F

4. Pressurization tank pressure: 500 psi nominal

5. Max. Flight-Spacecraft leak rate: 7.5×10^{-6} ft^3/sec STD AIR

6. Probability of Mars contamination: 10^{-4} for mission

7. Minimum separation impulse: 3.4 lb-sec

8. Maximum separation impulse: 5.0 lb-sec

9. The sterilization canister-lid separation mechanism shall be designed to cleanly separate the lid from the canister base and to remove it from the flight paths of the Flight Spacecraft and Flight Capsule.

10. The shaped charge shall be sterilizable.

11. Redundancy is required in the lid ejection system.

6.1.2.3 Design Description

The sterilization canister assembly is composed of three main elements, the structure, the separation subsystem, and the pressurization equipment.

1. Structure -- The structure consists of two main assemblies, the lid and the base. The lid assembly is an aluminum dish closely contoured to the entry heat shield, extending onto a 183-inch diameter where it becomes cylindrical in shape.

The base is a conical aluminum sheet structure with a flat closed end, and an open flanged end to mate with the lid. The aft adapter for mating with the Flight Spacecraft is welded to this unit.

2. Separation -- The assembly, consisting of two rings and a shaped explosive charge, is spot or seam welded to the cylindrical section of the structure. The end of the cylindrical assembly is flanged to mate with the same type of flange on the base.

3. Pressurization -- The pressurization subsystem consists of a 12-inch diameter gas-storage tank, a pressure regulator, and a relief valve backed by a microfilter.

6.1.3 Functional Interfaces*

6.2 SEPARATION

6.2.1 Functions

A separation function is required to operate with each of the following subsystems during the Flight Capsule mission.

6.2.1.1 Sterilization Canister Lid

A flexible linear shaped charge mounted circumferentially around the lid cuts the lid free from the base and provides the necessary force to insure that it is propelled out of the expected trajectory of Separated Vehicle subsequent to separation from the Flight Spacecraft.

6.2.1.2 Separated Vehicle

The Separated Vehicle separates from the Flight Spacecraft with sufficient velocity to prevent impingement of the rocket engine exhaust plume on the Flight Spacecraft.

This separation function also includes separation of the electrical umbilical connection between the Flight Spacecraft and Landed-Capsule support structure. A secondary separation function is required to separate the sterilization canister base from the Flight Spacecraft to reduce the weight of the Flight Spacecraft after Separated Vehicle separation or to separate the entire Flight Capsule in the event of a major Flight-Capsule subsystem failure. These separation functions include the separation of the electrical umbilical connection between the Flight Spacecraft and the canister base.

6.2.1.3 ΔV Propulsion and ACS Electronics Assembly

After the ΔV rocket and ACS Electronics Assembly have completed their functions, they will be ejected from the Landed Capsule support structure to reduce entry weight.

6.2.1.4 Parachute

The pilot parachute shall be projected into the gas stream by means of a mortar. The aerodynamic drag on the pilot parachute will pull the reefed main parachute from its container into the gas stream.

6.2.1.5 Entry Shell

The entry shell shall be jettisoned after the parachute is deployed. The elimination of the entry shell from the suspended capsule reduces the suspended weight and descent rate and provides a free path for the subsequent separation of the Landed Capsule from the Landed Capsule support structure. This separation function also includes separation of the electrical umbilical connection between the entry shell and the Landed Capsule support structure.

6.2.1.6 Landed Capsule

The Landed Capsule shall be deployed on a tether from the Landed Capsule support structure to reduce the impact weight of the Landed Capsule. This function also includes the separation of the electrical connection between the Landed Capsule and the Landed Capsule support structure.

6.2.1.7 Tether

The tether shall be released to prevent dragging of the Landed Capsule by the parachute.

6.2.1.8 Impact Attenuator

The impact attenuator shall be ejected from the internal structure to provide a free path for the subsequent deployment of the scientific instruments.

6.2.1.9 Instrument Deployment

The sensing elements of the internal payload shall be deployed for atmospheric or surface measurements.

6.2.2 Performance/Design Requirements

6.2.2.1 Performance

The performance sequence of the separation subsystem related to the mission event sequence is as follows:

1. Sterilization canister lid ejection
2. Separated Vehicle separation
3. Separated Vehicle attitude orientation
4. ΔV propulsion rocket firing
5. Attitude orientation and spin-up of Separated Vehicle
6. Ejection of ACS electronics and ΔV propulsion assembly from Entry Vehicle
7. Despin Entry Vehicle
8. Parachute deployment
9. Release of entry shell from the Suspended Vehicle
10. Release of Landed Capsule from the Suspended Vehicle
11. Release tether from the Landed Capsule

12. Eject impact attenuator from the Landed Capsule

13. Deploy instruments from the internal payload.

6.2.2.2 Design

The separation subsystem shall meet the performance and design parameters listed in Figure 10 after exposure to the mission environments shown in Figure 2.

6.2.2.3 Design Description

To accomplish each of the separation requirements, the separation subsystem shall be implemented as follows:

1. Sterilization Canister Lid -- A five grain/ft flexible linear shaped charge shall be mounted circumferentially around the large diameter of the canister. The flexible linear-shaped charge shall sever the lid from the base. The pressure from the flexible linear-shaped charge shall provide the separating force.

2. Separated Vehicle -- Four pressure actuated ball locks, when actuated by a gas generator manifold system, release the Separated Vehicle. A separation velocity is provided by eight coil springs.

3. ΔV Propulsion and ACS Electronics Assembly -- A 20-inch diameter Marman clamp, consisting of 4, 20-degree clamp segments, restrained by an adjustable stranded cable, releases the assembly when the cable is cut by explosively actuated cable cutters. Four coil springs provide the separating force.

4. Parachute -- A pilot chute is propelled into the gas stream by a mortar which in turn pulls the reefed main chute from its storage canister into the gas stream.

5. Entry Shell -- Four pressure actuated ball locks, when actuated by a gas generator - manifold device, shall release the entry shell which drops away as the Suspended Vehicle is decelerated by the reefed main chute.

6. Landed Capsule -- The Landed Capsule shall separate from the support structure onto the tether when its supporting straps are released and the restraining cable is cut by explosively actuated cable cutters. The separating force is supplied by gravity.

Separation Functions \ Separation Parameters	Separation Velocity or Distance	Allowable Rotation Rate	Type	Separating Force	Operating Temperature and Pressure	Electrical Umbilical Separation	Wt. (lbs)	Design Requirements
1. Sterilization Canister Lid	1.5 ft/sec ± 5%	7 deg/sec maximum	Flexible Linear Shaped Charge	Reaction from explosively developed pressure	-160°F 80mmHg inside to 10^{-12}mmHg outside canister	No	13	Separation impulse = 15 lb secs
2. Separated Vehicle	3.0 ft/sec ± 5%	20 deg/sec maximum	Ball locks with explosive gas generator Mod. marman clamp, explosively actuated cable cutters	8 springs, K = 30 lb/in, Deflection: 5 in.	-60°F 10^{-12}mmHg	Yes	12 12	Separation impulse = 163 lb secs
3. Propulsion Rocket and ACS Electronics Assembly	1 ft/sec ± 5%	90 deg/sec nominal	Modified marman clamp with explosively activated cable cutters	4 springs, K = 1.2 lb/in, Deflection: 3 in.	-150°F to +200°F 10^{-12} mmHg	Yes	4	Separation impulse=5 lb secs
4. Parachute	63 ft/sec ± 20%	N A	Motar	Explosively developed pressure	Temp. to be determined 80mmHg inside to 10^{-12}mmHg outside parachute canister	No	3	Separation impulse=10 lb secs
5 Entry Shell	Dropped N A	N A	Ball locks with explosive gas generator	Inertia and gravity	Temp. to be determined 8 to 30 mb	Yes	12	No impulse required
6. Landed Capsule	Dropped N A	N A	Cable restrained straps with explosively actuated cable cutters	Gravity	Temp. to be determined 8 to 30 mb	Yes	3	No impulse required
7. Tether	100 ft/sec ± 25%	N A	Explosively actuated bolt	Explosively developed pressure and wind	Temp. to be determined 8 to 30 mb	No	1	To be determined
8. Impact Attenuator	5 ft ± 10%	N A	Sheet explosive and flexible linear shaped charge	Explosively developed pressure	Temp. to be determined 8 to 30 mb	No	1 1	To be determined
9. Instrument Deployment	4 ft ± 25%	N A	Gas pressure expanded telescoping booms	Explosively developed pressure	Temp. to be determined 8 to 30 mb	No	4	Separation impulse of 1 lb secs

Figure 10 SEPARATION SUBSYSTEM PERFORMANCE AND DESIGN PARAMETERS

7. Tether -- The tether shall be cut by an explosively actuated cable cutter.

8. Impact Attenuator -- Sheet explosive bonded between impact attenuation sections shall cut openings through the crush-up material and face structure.

9. Instrument Deployment -- Instruments shall be deployed by gas pressure expanded telescoping booms. Explosive gas generators and manifold system supplies the gas pressure.

6.2.3 Functional Interfaces

6.2.3.1 Electrical

1. Inputs

   a. FS CCS -- Separation commands while mounted to the Flight Spacecraft

   b. FC CCS -- Separation commands after separation from the Flight Spacecraft

   c. FC Power Supply -- Electrical power for activation of each pyro network.

2. Outputs

   a. Continuity monitoring circuits for system checkout.

   b. Discrete signals of subsystem operation for engineering diagnosis.

6.2.3.2 Mechanical

1. Inputs -- Attachment to adjacent structures by mechanical fastener, adhesive bonding and weldments.

2. Outputs -- Component operating forces as listed in Figure 10.

6.3 PROGRAMMING AND SEQUENCING

6.3.1 Functions

All Flight Capsule programming and sequencing functions shall be performed by the Central Computer and Sequencer (Flight Capsule CCS) subsystem, specifically:

1. Provide clock time from Flight Capsule/Flight Spacecraft separation to the end of the mission.

2. Provide all necessary Flight Capsule clock pulses and pulse rates.

3. Decode demodulated commands.

4. Safe, arm, and initiate Flight Capsule events by elapsed time, command, altitude, velocity and/or acceleration.

5. Sense and diagnose subsystem or component failures and issue commands for remedial action.

6. Control data handling modes.

6.3.2 Performance/Design Requirements

The Flight Capsule CCS shall be capable of controlling and/or handling the discrete functions listed in Table VIII. The primary mode of control for all of the functions shall be by an internal program which is formatted before launch but which can be updated throughout the mission. Whenever practical, a direct communication link command shall be used as backup.

6.3.3 Functional Interfaces

Detailed subsystem interface definition shall be implemented between each of the subsystems listed in Table VIII.

6.4 TRAJECTORY CHANGE PROPULSION

6.4.1 Functions

The primary function of the trajectory change propulsion subsystem is to alter the course of the Flight Capsule such that it enters the Martian atmosphere at a desired location, velocity and angle. A secondary function of the propulsion subsystem is to provide the necessary communications geometry, i.e., communications range and look angles, during the entry and descent mission phases.

6.4.2 Performance/Design Requirements

6.4.2.1 Performance

1. Description -- The propulsion subsystem shall consist of a spherical solid propellant rocket engine, a modified Titan Vernier engine TE-M-345 including burn-time control, diagnostic monitoring

TABLE VIII

CENTRAL COMPUTER AND SEQUENCER FUNCTIONS

1. Apply test voltage to each of 3 gyros
2. Uncage integrators
3. Apply 4 thrust vector reference voltages to ACS electronics
4. Open reaction system main gas valve
5. Ignite ΔV propulsion
6. Energize positive roll solenoids
7. Energize positive yaw solenoids
8. Energize positive pitch solenoids
9. Energize negative roll solenoids
10. Energize negative yaw solenoids
11. Energize negative pitch solenoids
12. Cut off ΔV propulsion
13. Shut off reaction system main gas valve
14. Apply spin up command to ACS
15. Jettison ΔV propulsion and ACS electronics assembly
16. Cruise phase power off
17. Cruise phase power on
18. Energize command receiver and Flight Capsule CCS
19. Switch command receivers
20. Actuate telemetry modes

TABLE VIII (Concl'd)

21. Energize direct transmitter
22. Switch RF amplifiers
23. Energize relay transmitter
24. Switch relay transmitter
25. Switch command receiver antenna
26. Switch direct transmitter antenna
27. Switch relay transmitter antenna
28. External payload battery of Flight Spacecraft power
29. Internal battery #1 on Flight Spacecraft power
30. Internal battery #2 on Flight Spacecraft power
31. Internal battery #1 relay in charge position
32. Internal battery #2 relay in charge position
33. External payload battery relay in charge position
34. External power on Flight Spacecraft
35. Switch data rate
36. Deploy reefed parachute
37. Dis-reef parachute
38. Tether landed capsule
39. Jettison parachute and landed capsule support structure
40. Arm entry pyrotechnics
41. Arm landed pryotechnics
42. Energize post impact engineering and diagnostic TM
43. Thermal control for ΔV propulsion and ACS on/off
44. Perform failure mode diagnoses and switching
45. Deploy landed science instruments
46. Control time (s) and durations of direct transmitter
47. Control time (s) and durations of relay transmitter

and mounting provisions. The primary modification is the replacement of the presently used propellant with a sterilizable propellant.

Total applied impulse shall be controlled by blowing off the engine nozzle by the firing of explosive latches. The variable impulse capability allows the use of the same engine to accommodate variations in payload weight, ΔV requirements and different mission thrust application angles.

The firing of the rocket engine shall be controlled by the FC CCS which stores the start time and duration commands, as updated by the DSIF-to-Flight Spacecraft-to-Flight Capsule communication link. After the ACS orients the Separated Vehicle to the correct thrust attitude, the rocket shall be ignited at the prescribed time, by an electrical signal originated in the Flight Capsule CCS. Thrust shall be terminated by a signal from the integrated output of the Flight Capsule accelerometers. This signal represents the ΔV attained. A back-up thrust termination signal shall be derived from the Flight Capsule CCS based on desired rocket burning time to achieve the required ΔV. Rocket-engine temperature measurement shall be used to modify the backup thrust-termination command.

2. Nominal Engine Vacuum Performance Summary

Impulse, total, lb-sec	4,850
Impulse, specific, seconds	255
Thrust, average, pounds	768
Thrust, maximum, pounds	808
Temperature limits	
Operation	20° to 130°F
Storage	10° to 130°F

6.4.2.2 Design

1. General Requirements

a. The motor case shall be spherical

b. The specific impulse (standard) shall be greater than 200 seconds.

c. The total impulse shall be 5000 lb-sec nominal.

d. A propellant mass fraction of 0.8 minimum is desirable but not critical.

e. The average nominal thrust shall be 800 pounds.

f. The engine shall have thrust termination capability over its entire performance range.

g. The engine exhaust products shall be gaseous only.

h. The rocket engine shall meet the performance specifications with a reliability goal of 99.9 percent after exposure to the following:

1) Flight Capsule static acceleration of 15g axial and 7.6g lateral and dynamic loads which can be simulated by a 3g rms axial and 2g rms lateral vibration input to a flight configuration on a hard mount using a sine sweep 1 minute/octave from 2 to 100 cps

2) Nominal ground transportation and handling loads

3) Earth storage at 80° ± 30°F for two years

4) Temperature in space of 80° ± 50°F for a period of up to one year at 10^{-6} mm Hg

5) Rocket firing during 50 rpm axial spin

6) Three sterilization cycles over a period of three months at 145°C

7) No inspection, by disassembly is permitted after sterilization.

i. Existing qualified rocket or components shall be used where applicable.

2. Selected Nominal Engine Design Summary

Envelope	13.5 inch diameter x 18.6 inches long
Weight	
Loaded, pounds	81.0
Propellant, pounds	63.9

Mass ratio, M_p/M_t	0.788
Burn time, seconds	Variable, (from 1 to 21)
Thrust termination	blow-off nozzle
Area ratio, exhaust nozzle	18.7
Propellant type	TP-H-3105
Engine assembly	sterilizable

6.4.3 Functional Interfaces

6.4.3.1 Electrical

1. Inputs

a. Rocket engine ignitors (2)

b. Explosive latches on engine exhaust nozzle

c. Engineering diagnostic transducers; temperature and continuity

2. Outputs -- None

6.4.3.2 Mechanical

1. The engine shall be bolted to the propulsion and ACS electronics support structure.

6.5 ATTITUDE CONTROL

6.5.1 Functions

6.5.1.1 Primary (Normal Operation) Functions

The attitude control subsystem (ACS) shall be capable of performing the following functions for the Separated Vehicle.

1. Null the tip-off rates due to separation from the Flight Spacecraft and realign the Separated Vehicle to the Flight Spacecraft reference attitude.

2. Maneuver the Separated Vehicle thrust axis to a preselected attitude, with respect to the Flight Spacecraft reference attitude, for trajectory change thrust application.

3. Provide three axis thrust vector control (TVC) during the trajectory change propulsion operation.

4. Maneuver the Separated Vehicle thrust axis from the thrust application attitude to the attitude required for applying the spin-up propulsion.

5. Provide spin stabilization for Entry Vehicle zero angle of attack at entry.

6.5.1.2 Secondary (Back-up Operation) Function

The attitude control subsystem shall be capable of performing the following back-up function for the Separated Vehicle if its primary operational mode is inoperative:

1. Spin stabilize immediately after separation to maintain the attitude attained by Flight Spacecraft maneuvering and to provide TVC during the Trajectory Change propulsion operation.

6.5.2 Performance/Design Requirements

6.5.2.1 Operation

The sequence of operations of the attitude control subsystem shall be as follows:

1. Subsystem turn-on and warm-up
2. Subsystem drift trimming
3. Subsystem checkout
4. FC electrical and mechanical separation from FS
5. Activate reaction subsystem and realign the Separated Vehicle to the Flight Spacecraft reference altitude
6. Reorient Separated Vehicle thrust axis to desired ΔV vector
7. Maintain three axis control while applying ΔV thrust

8. Reorient Separated Vehicle thrust axis to align with its flight path

9. Spin stabilize the Separated Vehicle

10. Jettison the propulsion and ACS electronics assembly

6.5.2.2 Design Conditions

The ACS shall be capable of accomplishing its intended functions in the appropriate operational sequence assuming the following design conditions:

1. Alignment Parameters

a. The Separated Vehicle cg location is known within 0.0833 inch (1 sigma) of its centerline and shall be 0.0833 inch fore or aft of the designated cg location.

b. The ΔV thrust rocket shall be located such that the rocket centerline is within 0.042 inch (1 sigma) of the Separated Vehicle centerline and the cg of the rocket engine shall be 0.042 inch fore or aft of its designated cg location.

c. The ΔV thrust vector misalignment shall be no greater than 0.167 degree (1 sigma) from the Separated Vehicle centerline.

d. The gyro package alignment error shall not exceed 0.09 degree (1 sigma) from the Separated Vehicle references axes.

e. The velocity change sequence shall be accomplished within 15 minutes after separation.

f. Flight Spacecraft sensor error is 0.053 degree 1 sigma (JPL data) from the reference axes of the Flight Spacecraft.

g. Flight Capsule mounting accuracy is 0.167 degree 1 sigma with respect to the reference axes of the Flight Spacecraft.

2. Cold Gas Reaction System Parameters

a. The specific impulse (standard) shall be greater than 60 seconds.

b. Storage capacity shall be available for a nominal useable total impulse of 764 lb-sec.

3. <u>Spin Rocket Parameters</u>

a. The specific impulse (standard) shall be greater than 200 seconds.

b. The total vacuum impulse of each rocket shall be 45 lb-sec, nominal.

c. The rocket burn time shall be 0.5 second, nominal.

d. The rocket assembly mass fraction is not critical.

e. The rockets shall produce gaseous exhaust products only.

4. <u>Operational Parameters</u> -- The ACS shall meet performance specifications with a reliability of 99.9 percent at 90 percent confidence after exposure to the following environments:

a. Flight capsule static acceleration of 15 g axial and 7.5 g lateral. Dynamic loads which can be simulated by a 3 g rms axial and 2 g rms lateral vibration input to a flight configuration on a hard mount using a sine sweep 1 minute/octave from 2 to 100 cps.

b. Nominal Earth transportation and handling loads.

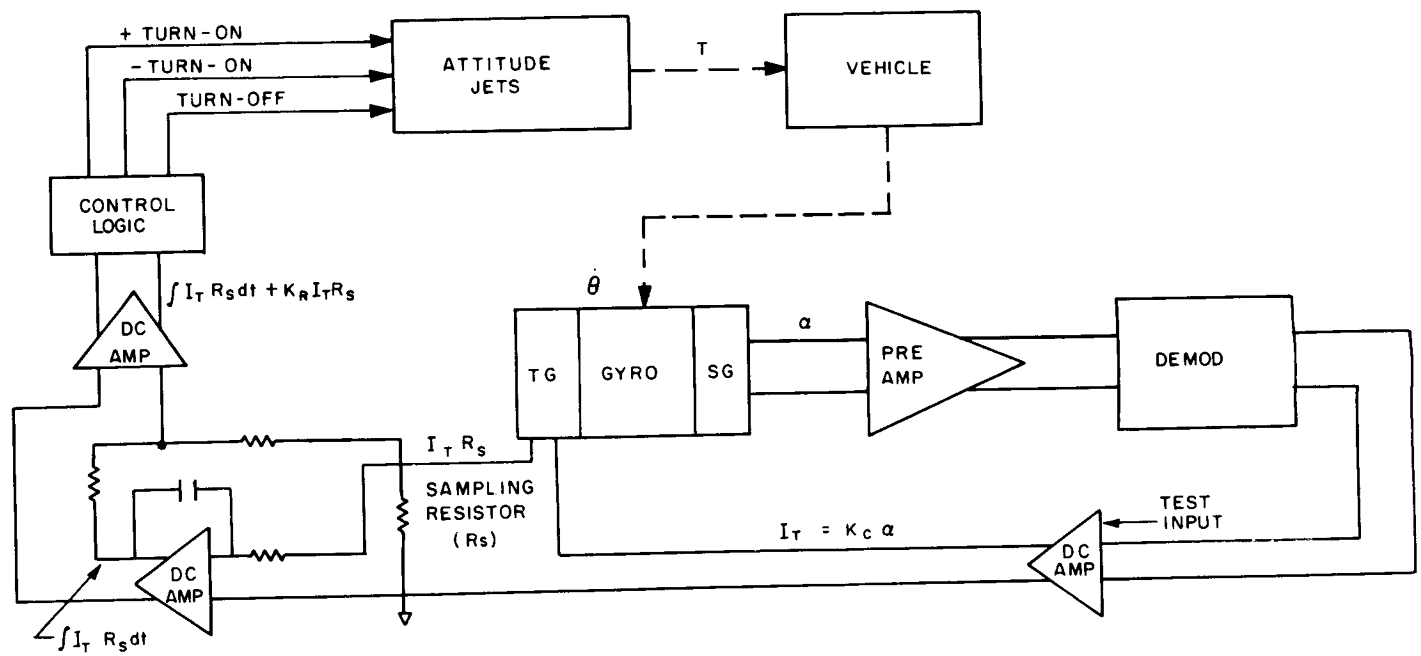
c. Earth storage at 80° ± 30°F for 2 years.

d. Three high temperature sterilization cycles over a period of three months; conditions of 145°C for 24-hour duration are to be attained inside the spin and ΔV rocket motors.

6.5.2.3 Subsystem Description

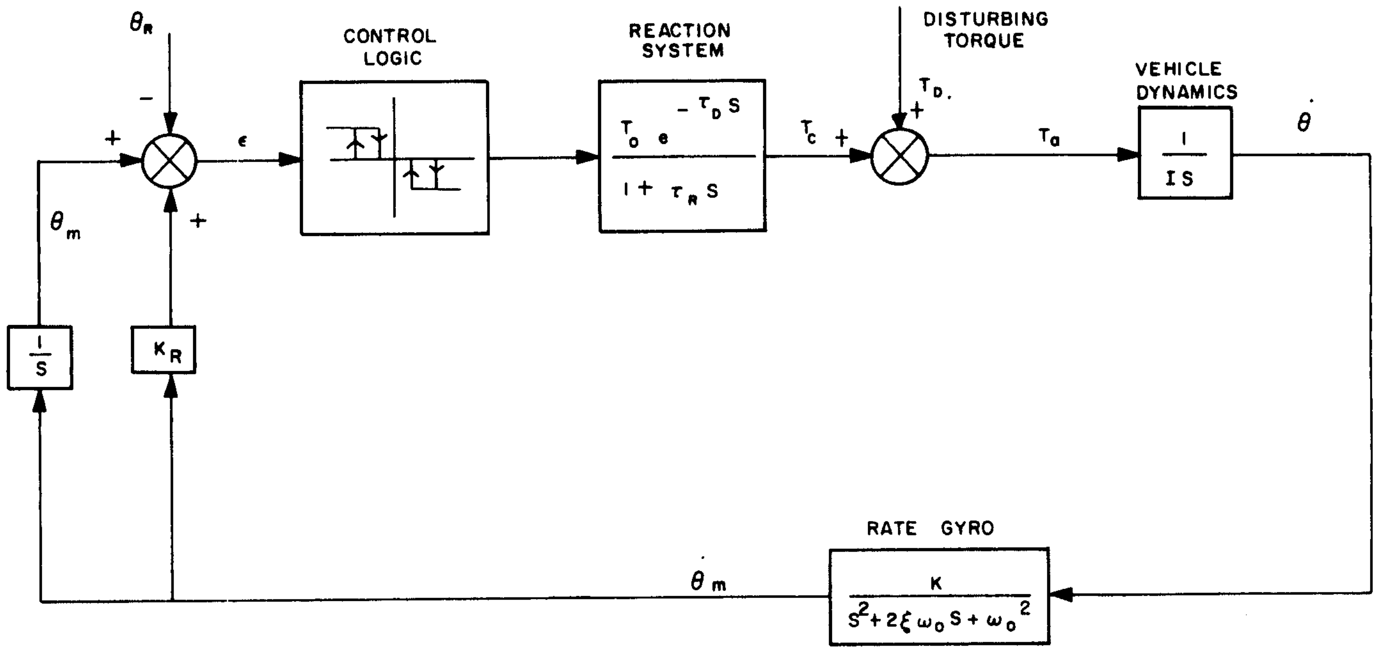
The subsystem schematic of a design solution to accomplish the ACS functions in the appropriate operational sequence within the design conditions noted in paragraph 6.5.2.2, is shown in Figure 11. Three axis control is provided by the cold gas reaction subsystem until the final orientation after ΔV thrusting. The Separated Vehicle is then spin stabilized with solid propellant rockets. A mathematical model showing the logic for this type of stabilization is presented in Figure 12.

Typical component solutions to implement the required ACS are as follows:



85-1782

Figure 11 ATTITUDE CONTROL SUBSYSTEM SCHEMATIC



85-1785

Figure 12 MATHEMATICAL MODEL OF ATTITUDE CONTROL SUBSYSTEM

1. Gyros -- The gyros are the Kearfott Alpha series. Modifications will be made to features such as the gyro gain, characteristic time, and gyro heater voltage to be more compatible with operational requirements.

2. Integrator -- An electronic analog integrator shall integrate the gyro output to provide an indication of angular position to be instrumented by using a dc operational amplifier with a capacitor in the feedback loop.

3. Control Logic -- A Schmitt trigger circuit appropriately biased for the correct firing level receives the error signal (proportional to attitude and attitude rate). The hysteresis of the Schmitt trigger is set to provide stability.

4. Reaction Subsystem -- A schematic of the cold gas nozzle and plumbing portion of the reaction subsystem is shown in Figure 13; spin rockets are not included. These 12 nozzles provide 3-axis control in couples. Spin stabilization is provided by solid propellant rockets. The spin rockets will be arranged in two groups. In the case of failure of one group, the other will be used for spin-up. Both sets of spin rockets will be used if the primary operational mode of the ACS fails and the backup mode is selected.

5. Spin Rockets -- The spin rocket is a solid propellant Scout spin motor MARC -4B2 (Atlantic Research), modified. The primary modification is the replacement of the presently used propellant with a sterilizable propellant.

6.5.2.4 Nominal Performance/Design Summary

1. Attitude Control Subsystem -- The overall ACS performance is characterized by:

Performance	0.23 degree (1 sigma)
Total operating time	15 minutes
System weight	90 pounds
Total stored impulse	764 lb-sec.
Limit-cycle amplitude	0.1 degree
Maximum turning rate in orientation phase	1.0 deg/sec

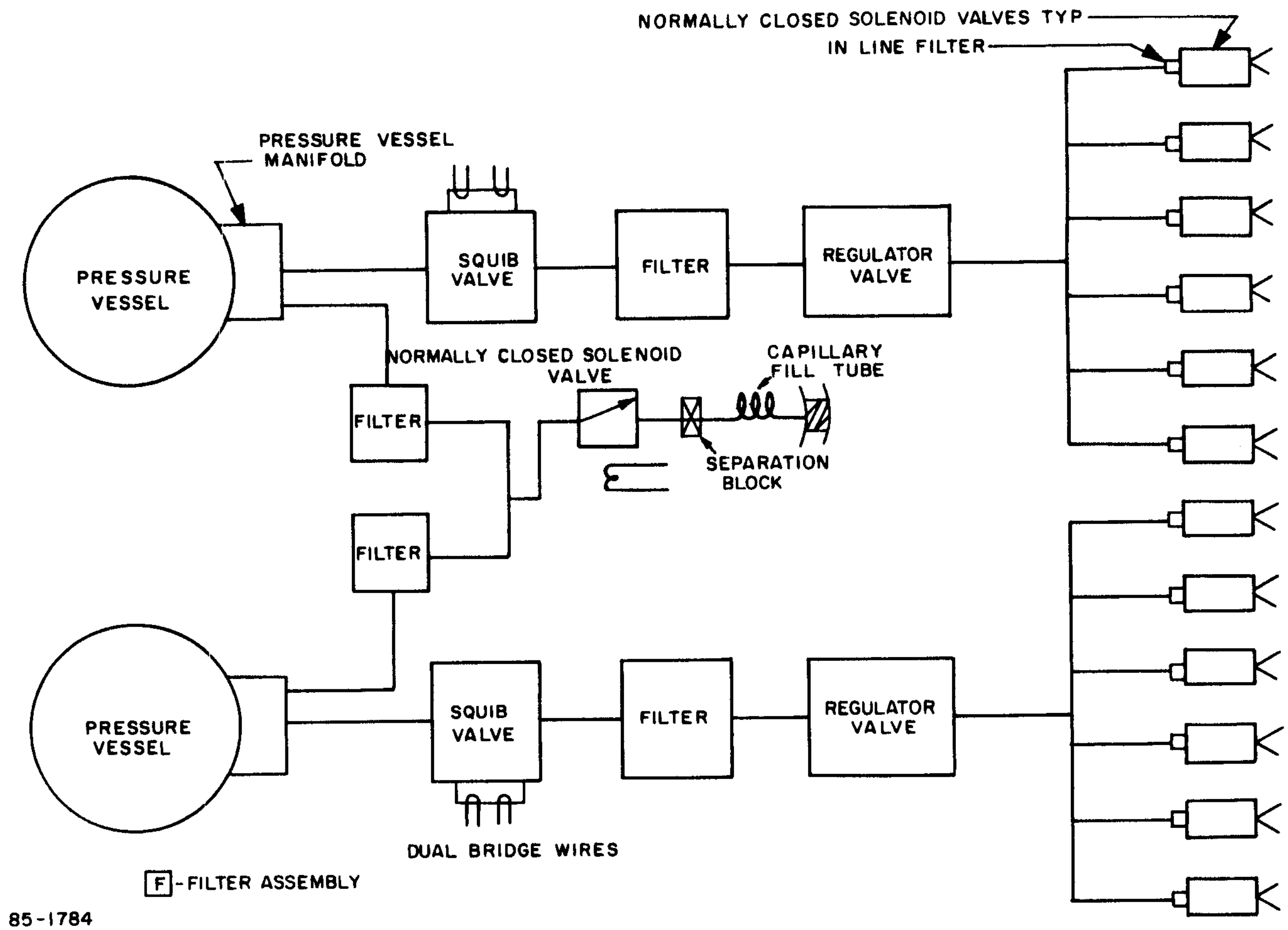


Figure 13 COLD GAS REACTION SUBSYSTEM

2. Sensors and Electronics --

a. Control Logic -- The control deadband shall be ±0.5 degree from the nominal zero position about each of the three control axes with the hysteresis factor of 10 percent.

b. Error Sources (1 sigma)

Pitch and Yaw Angles Computation Errors:	(Percent)
Torquer scale factor stability	0.50
Torquer nonlinearity	0.02
Wheel power frequency	0.01
D. C. voltage reference uncertainty	0.02
Integrator nonlinearity	0.05
Sampling resistor	0.02
Oscillatory Motion Rectification Errors:	
Torquer asymmetry	0.01
Difference between +dc and -dc (gyro torquing and integration)	0.01
Integrator rectivication	0.02

c. Accuracy:

Gyro drift	0.05°/hr
Scale factor	0.1
Integrator drift	0.10°/hr

d. Weight and volume:

Component	Weight (pounds)	Volume (in^3)
Inverter	1	40
Gyro package	4	50

Component	Weight (pounds)	Volume (in^3)
Control electronics	2	80
Integrator subsystem	1	40
Mounting cables	2	--
Total	10	210

The weight and volume figures above assume integrated circuitry will be used in the electronic modules. Battery power weight and volume is included with the electrical power and control subsystem.

3. Cold Gas Reaction Components

a. Thrust levels (pounds)

Yaw	6
Pitch	6
Roll	0.3

b. Total impulse (lb-sec)

Required	232
Stored	764

c. Response parameters (seconds)

Time Delay	0.012
Time Constant	0.002

d. Weight

Component	Weight Each (pounds)	Number	Total (pounds)
Nozzle valves	1.3	12	15.6
Pressure vessels	6.1	2	12.2
Gas (N_2)	12.6	--	12.6

Component	Weight Each (pounds)	Number	Total (pounds)
Squib valves	1.5	2	3.0
Vessel manifolds	0.3	2	0.6
Fill solenoid and capillary	0.8	1	0.8
Line coupler	6.0	2	12.0
Regulators	5.5	2	11.0
Filters	0.5	4	2.0
		Total Weight	69.8

4. Spin Rockets

a. Vacuum Performance

Impulse, total, lb-sec	45
Impulse, specific, seconds	205
Thrust, average, pounds	82
Burn time, seconds	0.55
Temperature limits	
Operation	-35 to +140°F
Storage	-65 to +140°F

b. Mechanical Properties

Envelope	1.53 inch diameter x 6.83 inch length
Weight, pounds	0.80
Propellant, pounds	0.22
Mass Ratio M_p/M_t	0.25

6.5.3 Functional Interfaces

6.5.3.1 Electrical

1. Inputs -- Discrete operational commands will be issued by the Flight Capsule CCS. Electrical power will be furnished by the electrical power and control subsystem.

2. Outputs -- Diagnostic measurements signals will be furnished to the telecommunications subsystem.

6.5.3.2 Mechanical

1. Inputs -- Structural support will be provided by the appropriate portions of the entry shell structure and the propulsion and ACS electronics structure as shown in Figures 7 and 8.

2. Outputs -- Control torques will be applied to the Separated Vehicle by cold gas jets located such that, for each direction about each control axis, two jets will produce a force couple in order to minimize cross-coupling and to provide protection against nozzle valve failure.

3. Thermal -- The ACS and all its components must be maintained within a temperature range of -65 to +140°F, except the spin rockets for which the minimum operating temperature is -35°F. During warm-up and operation of the gyros, they will be maintained at the proper operating temperature of +180°F by their own heaters.

6.6 DESCENT RETARDATION--ENTRY

6.6.1 Functions

The functions of the entry descent retardation subsystem hereafter termed the entry shell, are to provide suitable atmospheric deceleration and thermal protection for scientific instrumentation and associated subsystems. The entry shell consists of primary and afterbody structures and primary and secondary heat shields.

The primary structure shall provide a shape with high aerodynamic drag for atmospheric deceleration. It shall provide surfaces upon which the primary heat shield and thermal protection coatings are applied and it shall provide mounting surfaces for the Landed Capsule and other hardware.

The secondary structure shall provide aerodynamic shape designed to impact a pitching moment to reorient the vehicle to the blunt and forward position in the event of rearward entry. The secondary structure shall also provide support for the secondary heat shield and thermal coatings, mounting pads for antennas, propulsion engine, and the external payload.

The primary heat shield shall protect the structure and payload against the high temperatures associated with rapid entry of a high drag body into an atmosphere.

The secondary heat shield (that portion of heat shield over the afterbody and exposed interior portion of the primary structure) shall provide thermal protection to the aft portion of the Entry Vehicle during the normal blunted end forward entry mode. Additionally, it shall provide thermal protection in the event of rearward entry for that portion of the entry trajectory that the Entry Vehicle is oriented in a rearward entry mode.

6.6.2 Performance/Design Requirements

6.6.2.1 Performance

The entry shell shall accomplish its intended function under the following conditions:

1. Assembly to Entry

   a. Sterilization cycle temperatures and loads,

   b. Handling and transportation loads,

c. Ascent loads,

d. Spaceflight temperatures,

e. ΔV maneuvering and separation forces,

f. Entry temperatures and loads, and

g. Parachute opening shock loads.

2. Entry

a. Entry velocity between 18,400 and 23,800 ft/sec.

b. Entry angle of -90 to -20 degrees

c. NASA Model Atmospheres 2 and 3

d. Entry angle of attack -- 10 degrees nominally; -- 179 degrees failure mode (rearward entry)

e. Spin rates at entry of one to two rad/sec/max

f. The primary heat shield is required to supply sufficient heat protection to the primary structure to limit the maximum back-face temperature to 300° F considering a heat soak time up to Mach 1.0 of the the entry trajectory.

g. Ballistic coefficient of 0.20

6.6.2.2 Design Description

The following design description is an entry shell subsystem capable of accomplishing the intended functional and performance requirements.

The structural shape is a 60 degree blunted cone having a nose radius of 3.75 feet and a base radius of 15.0 feet. The primary structure is of honeycomb sandwich design using face sheets of 0.025-inch thick beryllium and a stainless steel honeycomb core is 0.60 inch thick. The afterbody is made of 0.050 inch fiberglass. The primary heat-shield material is cork silicone with a maximum thickness of 0.450 inch at the nose to 0.375 inch at the outer periphery. The secondary heat shield is made of the same material with an average of 0.125 inch over the exposed rearward surface of the structure and the afterbody. The hypersonic drag coefficient for the 60 degree blunted cone, including real gas effects, is 1.63.

6.6.3 Functional Interfaces

The entry shell shall provide support for the thermal control coating and also provide a part of a conduction path for heat flow.

The entry shell shall provide support and transmit control torques of the reaction control system.

The afterbody shall align and support the propulsion rocket and transmit thrust from the engine to Flight Capsule.

The afterbody shall form an electrical ground plane for the antennas.

6.7 THERMAL CONTROL

6.7.1 Function

A thermal control function shall provide a controlled temperature environment to all other functional areas. The thermal control function is defined as the implementation of all of the devices required for Flight Capsule temperature control from factory through surface operations.

6.7.2 Performance/Design Requirements

6.7.2.1 General

Thermal requirements are important at all levels of system design, i.e., components, subassembly, assembly, and system. Therefore, thermal requirements shall be considered at the component packaging level, suassembly and assembly level both. Attention shall be given to the location of heat dissipating components, attachments, exterior surface finish and the use of thermal insulation.

During the design of functional equipment, consideration shall be given to thermal control requirements whereby actual heating elements shall be integrated into using equipment configuration, to minimize weight and power utilization. Where feasible, every effort shall be made to utilize passive thermal control consisting of coatings and finishes. Thermal control during entry shall be provided by heat-shield material bonded to the exterior of the entry shell with sufficient thermal properties to prevent excess heating of the payload or compromise to the scientific mission.

6.7.2.2 Design

1. The temperature control subsystem shall maintain the surface temperature of all sensitive components within the temperature ranges specified over the intended life of the component and all anticipated conditions of use.

2. The thermal control coatings and other thermal elements shall be be capable of withstanding sterilization temperature-time cycles.

3. Maximum use shall be made of passive techniques. Active controls shall only be used when it can be shown that passive control is difficult or impossible.

4. The temperature control subsystem shall be capable of maintaining component temperatures within specified limits during periods not to exceed two hours in which the Planetary Vehicle is not sun-oriented.

5. Temperature control of the Flight Capsule structure shall be provided as needed to keep thermal distortion within specified limits.

6.7.3 Functional Interfaces

Due to the nature of thermal control, all Flight Capsule functional areas shall have interfaces with the thermal control function.

6.8 DESCENT RETARDATION -- POSTENTRY

6.8.1 Functions

6.8.1.1 Primary

The primary function of descent retardation-post entry, hereafter termed the parachute sybsystem, is to decelerate the Landed Capsule to a tolerable velocity so that an impact attenuation subsystem can be utilized to absorb the balance of kinetic energy within acceptable deceleration limits.

6.8.1.2 Secondary

A secondary function of the parachute subsystem is to separate the Suspended Capsule from the entry shell.

6.8.2 Performance/Design Requirements

6.8.2.1 Subsystem Performance

1. The subsystem shall be designed to sense, analyze, and utilize entry parameters to initiate deployment of a single stage decelerator to decelerate the Suspended Capsule, separate the entry shell assembly, and provide the proper touchdown conditions within a system reliability figure of 0.999. The use of parachute reefing concept is acceptable within the definition of a single stage decelerator.

2. The subsystem shall decelerate the Landed Capsule vertical velocity to less than 80 fps in the terminal descent atmosphere (10 mb) and steepest entry angle (-52 degrees).

3. The subsystem shall be deployed full open at 15,000 feet altitude for the thinnest atmosphere (10 mb) and steepest entry angle (-52 degrees).

4. The subsystem shall reduce the Landed Capsule vertical velocity to less than 80 ft/sec in any of the atmospheres listed in Table I and any entry angle between -20 and -52 degrees if the parachute is disreefed at 15,000 feet altitude.

5. The subsystem shall be capable of successful operation following exposure to the environment specified below:

a. Prelaunch--Equivalent to 2 years shelf life plus transportation

b. Sterilization--

1) Terminal Sterilization: Three 36 hour cycles at 145°C in a dry nitrogen atmosphere

2) Surface Sterilization: Exposure to a chemical atmosphere of 12 percent ethylene oxide and 88 percent Freon 12 at a relative humidity of 50 percent for a 5-day period of 104°F.

c. Space--

Duration: 300 days

Vacuum: 1×10^{-8} Torr

Radiation: 6×10^{4} rads

Temperature: -85° to +120°F

d. Operation--

Duration: 60 minutes

Atmosphere: Carbon dioxide

Temperature: -100 to +150°F

Shock: 200 g

6.8.2.2 Component Performance

1. Deployment Initiation Sensing and Sequence Control

a. Sensing -- The sensing subsystem consists of three-single-axis accelerometers to sense peak acceleration, a timer to measure time from peak g to pilot parachute deployment, an analog computer relating peak g and time and a radar altimeter to initiate disreefing (by means of reefing cutters) of the main parachute. The subsystem shall sense and monitor the vehicle acceleration along the flight path with an accuracy of 1 percent full scale over a range of acceleration from 10 to 200 Earth g. The subsystem shall store a signal equal to the maximum acceleration level and transmit this signal to the sequence control subsytem.

b. Sequence Control -- The sequence control subsystem shall accept the maximum acceleration signal from the sensing subsystem and determine the proper time to initiate the deployment firing signal. The time measurement error shall be less than 3 percent for time delays from peak acceleration to deployment, ranging between 5 and 150 seconds.

c. Radar Altimeter -- A radar altimeter subsystem capable of measuring altitude above the surface with an accuracy of 2.0 percent over the range of altitudes from 500 to 30,000 feet, shall be used to perform the following operations:

--If, during entry, the Suspended Capsule reaches an altitude of 5000 feet with no deployment signal, an override firing signal will be transmitted to the pilot parachute system.

--A go-no-go circuit shall be provided for the disreef signal such that disreef cannot occur above an altitude of 16,000 feet.

d. Sequence Timer -- A timer shall be included in the central computer and sequencer subsystem to initiate the backup disreef signal. The signal will be generated 3 ± 0.3 seconds after the pilot parachute deployment signal. The circuit shall remain energized until the 16,000-foot altitude switch closes at which time the signal will be transmitted to disreef the parachute.

e. Pilot Parachute Deployment -- On receipt of the firing signal, the pilot mortar shall fire, breaking the pressure seal and deploying the pilot parachute. The mortar shall provide 50 ft/sec of separation velocity at a 10 g axial acceleration level.

2. Pilot Parachute -- The pilot parachute shall accomplish the following operations:

a. On inflating, the pilot parachute shall remove the main parachute canister cover and extract the main parachute in its deployment bag.

b. At main chute line stretch, the pilot parachute shall strip the deployment bag from the main parachute allowing it to inflate to its reefed condition.

3. Main Parachute -- The main parachute shall accomplish the following operations:

a. A storage canister shall maintain the main parachute in the required environment.

b. A deployment bag shall contain the main parachute until the main parachute is extracted during the deployment process.

c. The main parachute shall decelerate the vehicle to the required touchdown conditions.

d. Reefing lines, cutters, and pyrotechnics shall be utilized to control the main parachute opening characteristics and limit the descent time dispersion.

e. A mortar assembly shall eject the pilot parachute on a signal from the sensing system. The mortar assembly consists of a sabot, pilot parachute protective cover and a pressure gas generator.

f. The parachute canopy fabric, riser line and suspension lines shall withstand the opening shock loads for the worst design condition.

6.8.2.3 Design

The subsystem weight and main parachute configuration shown below are typical of a selected design to accomplish the subsystem function.

1. Subsystem Weight

Actuation mechanisms	3
Mortar	3
Pilot Chute	4
Main Chute	64
Total	74 pounds

2. Main Chute Configuration

Type	Ringsail
Diameter (disreefed)	85 feet
Reefed area of full open	18 percent

3. Design Parameters

   a. Blunt cone entry vehicle

   b. Lenticular Landed Capsule

   c. Suspended weight of 924 pounds

   d. Descending in a Terminal Descent atmosphere.

   e. Deployment Conditions (nominal)

-- Deploy (reefed) at Mach = 1.3

-- Disreef at 16,000 feet altitude

6.8.3 Functional Interfaces*

6.9 IMPACT ATTENUATION

6.9.1 Functions

6.9.1.1 Primary

The impact attenuation subsystem shall be capable of dissipating the Landed Capsule kinetic energy at impact while limiting the internal payload deceleration level to 500 Earth g.

6.9.1.2 Secondary

The impact attenuator shall allow the transmission of RF signals with minimum attenuation, reflection or distortion.

6.9.2 Performance/Design Requirements

1. Purpose -- The purpose of the impact attenuation system is to absorb the kinetic energy of the Landed Capsule at an impact velocity up to 130 fps as a primary mode of operation.

2. Operating Temperature Limits -- The operating temperature limits are from -135 to + 80°F.

3. Nonoperating Temperature Limits -- The non-operating temperature limits are from -300 to + 300°F.

6.9.2.1 Design

A typical impact attenuation subsystem to accomplish the intended function is herein described.

The impact attenuator shall be basically a fiberglass phenolic honeycomb, 1/8 inch or 3/16-inch cell size, with polyurethane foam filling the cells. Segments of this composite shall be bonded together in layers to build up the required thickness. The multi-layered segments shall in turn be bonded together, forming a complete ellipsoidal shell surrounding the internal payload. The segments of the attenuator shall be bonded directly to the outer structural shell of the internal payload. Shaped charges shall be provided at various places around the internal payload for the purpose of cutting through the attenuator after impact to allow the deployment of instruments. The attenuator shell shall be covered on the outside by a skin made of fiberglass fabric.

6.9.3 Functional Interfaces

6.9.3.1 Outer

The outer mechanical boundary shall be part of the mounting system which attaches the Landed Capsule to the external support structure which in turn is attached to the entry shell. A separate sling or netting attaches the Landed Capsule suspended under the parachute. These attachments shall be designed to prevent pre-crushing or cutting into the impact attenuator prior to impact.

6.9.3.2 Inner

The inner mechanical boundary is part of the outer structural shell of the internal payload.

6.10 DATA ACQUISITION -- SCIENCE INSTRUMENTATION

6.10.1 Function

The function of the science/instrumentation shall be to acquire the data required to fulfill the Capsule-System mission objectives.

6.10.2 Performance/Design Requirement

6.10.2.1 Scientific Data Acquisition Modes

Scientific data shall be acquired during the following modes of Flight Capsule operation:

Mode I - Separation to entry (if required)

Mode II - Entry to main parachute disreefed

Mode III - Parachute descent to impact

Mode IV - Surface operation

6.10.2.2 Design

The scientific instrumentation is classified in two categories; first, instrumentation to be used during the descent to the Martian surface, and second, instrumentation to collect data after impact on the Martian surface. The design parameters for each category are summarized in Tables IX and X. Also presented in these tables are requirements for the instrument weight, volume, power, bits per measurement,

TABLE IX

1971 SCIENCE INSTRUMENTATION (DESCENT)

Experiment	Acquisition	Weight lbs	Volume cu in	Power Watts	Bits per Measurement	Deployment Code	Sampling Rate
Accelerometer (3)	II	1. 8	15	2. 0	21	0	1 per sec
Atmos Pressure (2)	II	0. 6	6	0. 3	14	1	1 per 2 sec
Atmos Temp (2)	II	0. 6	6	0. 2	14	1	1 per 2 sec
Acoustic	II	3. 0	49	4. 0	14	1	2 (total)
Mass spectrometer	II	8. 0	400	10. 0	49	1	2 (total)
Gas chromatograph	II	4. 8	200	4. 0	56	1	2 (total)
Radar altimeter	II	8. 0	100	10. 0	7	1	1 per sec
Radiometer	II	2. 3	27	1. 5	42	1	1 per sec
Atmos pressure (2)	III	(0. 6)	(6)	0. 3	14	1	1 per 2 sec
Atmos temp (2)	III	(0. 6)	(6)	0. 2	14	1	1 per 2 sec
Beta scatter	III	0. 8	15	0. 3	7	1	1 per 5 sec
Mass spectrometer	III	(8. 0)	(400)	10. 0	49	1	3 (total)
Gas chromatograph	III	(4. 8)	(200)	4. 0	56	1	3 (total)
Radar Altimeter	III	(8. 0)	(100)	10. 0	7	1	1 per sec
Acoustic	III	(3. 0)	(49)	4. 0	14	1	3 (total)
TOTALS		29. 9	818				

TABLE X

1971 SCIENCE INSTRUMENTATION (LANDED)

EXPERIMENT	Acquisition	Weight lb.	Volume cu in	Power Watts	Bits per Meas.	Deployment Code	Sampling Rate
Gas chromatograph	IV	4.8	200	4.0	56	1	5 (total)
H_20 detector	IV	0.5	16	1.0	7	1	1/hr
Atmos, pressure (2)	IV	0.6	2	0.3	14	1	1/hr
Atmos. temp. (2)	IV	0.6	4	0.2	14	2	1/hr
Acoustic	IV	3.0	49	4.0	21	1	1/hr
Cosmic radiation	IV	2.2	100	0.4	14	2	1/hr
Surface radiation	IV	2.0	100	0.4	28	2	1/hr
Force anemometer	IV	2.0	14	1.0	112	2	5 (total)
Microphone	IV	0.5	33	0.4	70	2	5 (total
Surface temp. (2)	IV	0.6	4	0.2	14	3	1/hr
Alpha scatter	IV	0.7	12	1.4	1000	3	5 (total)
TOTALS		17.5	534				

sampling rate, and the acquisition mode. When the same instrument is to be used in more than one mode, its weight and volume in the second mode are given in parenthesis to indicate that they are not to be added to the total.

The deployment code, employed in the tables, is the following:

O -- Instrument requires no deployment and no access to the outside of the Entry Vehicle or Landed Capsule

1 -- Instrument requires access but no deployment

2 -- Simple deployment required, e. g., extension of arm from Landed Capsule

3 -- Complex deployment required, e. g., instrument must be deployed so that contact with surface is achieved.

6.10.3 Scientific Experiment Description

The following scientific experiment descriptions present instrument concepts and requirements to accomplish each of the considered experiments, plus a possible source of supply for the hardware.

6.10.3.1 Accelerometer

A standard type inertia accelerometer is recommended for measurement of the deceleration descent history in Mode II. A requirement of 0.1 percent accuracy of measurement is desired and is within the state-of-the-art. The accelerometer shall be located at the center of gravity of the Entry Vehicle.

6.10.3.2 Atmospheric Pressure

The atmospheric pressure shall be measured from the beginning of entry. Measurements will continue at the rate of one per two seconds until impact. A requirement of 1 percent accuracy has been placed on the pressure measurements. The descent pressure transducers shall be deployed to the rear of the Landed Capsule. For planetary surface operation only access to the atmosphere is required by the transducer.

6.10.3.3 Temperature Measurements

The atmospheric temperature measurement will begin at the end of blackout and will continue until impact and continue on the surface. Also, the Martian surface temperature will be measured. All measurements can be made with a platinum resistance wire thermometer.

One percent accuracy is required. On descent the temperature sensor shall be deployed on a boom to the rear of the Suspended Capsule and shall be thermally isolated from the structure. Boom deployment and isolation is also required for the atmospheric and surface temperature sensors to be used on the surface.

6.10.3.4 Electron Density Measurement

Electron density measurements will be made using an RF oscillator probe from entry to the beginning of blackout. The unit forms an integral part of the Landed Payload and as such requires no deployment.

6.10.3.5 Acoustical Densitometer

This instrument shall be used to determine the atmospheric density, mean molecular weight and the C_p/C_v ratio. Measurements shall begin at the end of blackout and continue to impact. A second unit shall make measurements on the surface. Measurements of 1 percent accuracy shall be made. Access to the atmosphere is necessary to make measurements.

6.10.3.6 Mass Spectrometer

The mass spectrometer will be used to determine the Martian atmospheric composition from end of blackout to impact. An accuracy of 10 percent is required for this instrument. Access to the atmosphere is necessary for operation.

6.10.3.7 Gas Chromatograph

Like the mass spectrometer, the gas chromatograph shall determine the atmospheric compositions at points from end of blackout to impact. A second unit shall be used to make measurements on the surface. An accuracy of 10 percent is required of the instrument. Access to the atmosphere is necessary.

6.10.3.8 Radar Altimeter

The radar altimeter shall determine height above the Martian surface from entry to impact. A requirement for 4 percent accuracy has been established. The antenna must be installed so that the entry shell does not obstruct its view.

6.10.3.9 Radiometer

The radiometer shall measure the intensity of selected molecular bands of the components of the Martian atmosphere excited by the shock wave of the Entry Vehicle to determine the atmospheric composition to 10 percent. A transparent window must be placed at the stagnation point of the Entry Vehicle to allow transmission of the shock layer radiation to the sensing devices.

6.10.3.10 Beta Scatter

The beta scatter device, which determines the atmospheric density by measuring the intensity of scattering of beta particles, shall be employed during the parachute phase of the descent (Mode III). A 5 percent accuracy is required. The entry vehicle must not obstruct the scattering path of the beta particles.

6.10.3.11 H_2O Detector

This device is more sensitive than the gas chromatograph also to be used on the surface for atmospheric analysis. If water is present in the atmosphere, it will follow the changes over a diurnal cycle. The instrument is accurate to ± 30 percent measuring a H_2O partial pressure of 10^{-4} millibars.

6.10.3.12 Cosmic and Surface Radiation

Solid state particle detectors arranged in a telescope configuration will be used to measure the cosmic and surface particle raidation at the Martian surface. Through the use of coincidence and anti-coincidence circuits, the direction and energy of the detected particles will be registered.

6.10.3.13 Anenometer Devices

A drag force sphere shall obtain data on average wind conditions and changes in such over a diurnal cycle. The anenometers must be deployed on a boom far enough from the Landed Capsule to obtain undisturbed wind force measurements.

6.10.3.14 Alpha Scatter Device

This instrument shall measure the intensity of alpha particles back scattered from the Martian surface. Information on the surface composition shall be obtained from this data. The instrument must be deployed to a position close to the planet's surface.

6.11 TELECOMMUNICATIONS

6.11.1 Functions

The Flight Capsule telecommunication subsystem shall perform the following functions:

1. Telemeter selected engineering parameters and scientific phenomena to the Flight Spacecraft.
2. Telemeter selected engineering parameters and scientific phenomena to Earth
3. Receive commands from the stations of the DSIF.

6.11.2 Performance/Design Requirements

6.11.2.1 General Definition

The telecommunication subsystem shall be comprised of the following subsystems, functionally integrated as shown in the block diagram in Figure 14:

1. Radio Subsystem -- consisting of transmission equipment for both Landed Capsule to Earth link and Suspended Capsule to Flight Spacecraft link

2. Telmetry Subsystem -- consisting of all data handling necessary to handle engineering instrumentation

3. Central Computer and Sequencer (CCS) -- consisting of all the timing and initiation logic for all Flight Capsule subsystems, including the computer logic for control of the ACS

4. Data Automation Subsystem -- consisting of all data handling equipment to be required for the scientific instrumentation

5. Power Subsystem -- which provides power to all Flight Capsule subsystems, separately discussed in paragraph 6.12, Electrical Power and Control

6. Command Subsystem -- consisting of transmission and reception equipment for the Earth to Landed Capsule link. This subsystem is part of the Mission Dependent Equipment required at the site of the DSN.

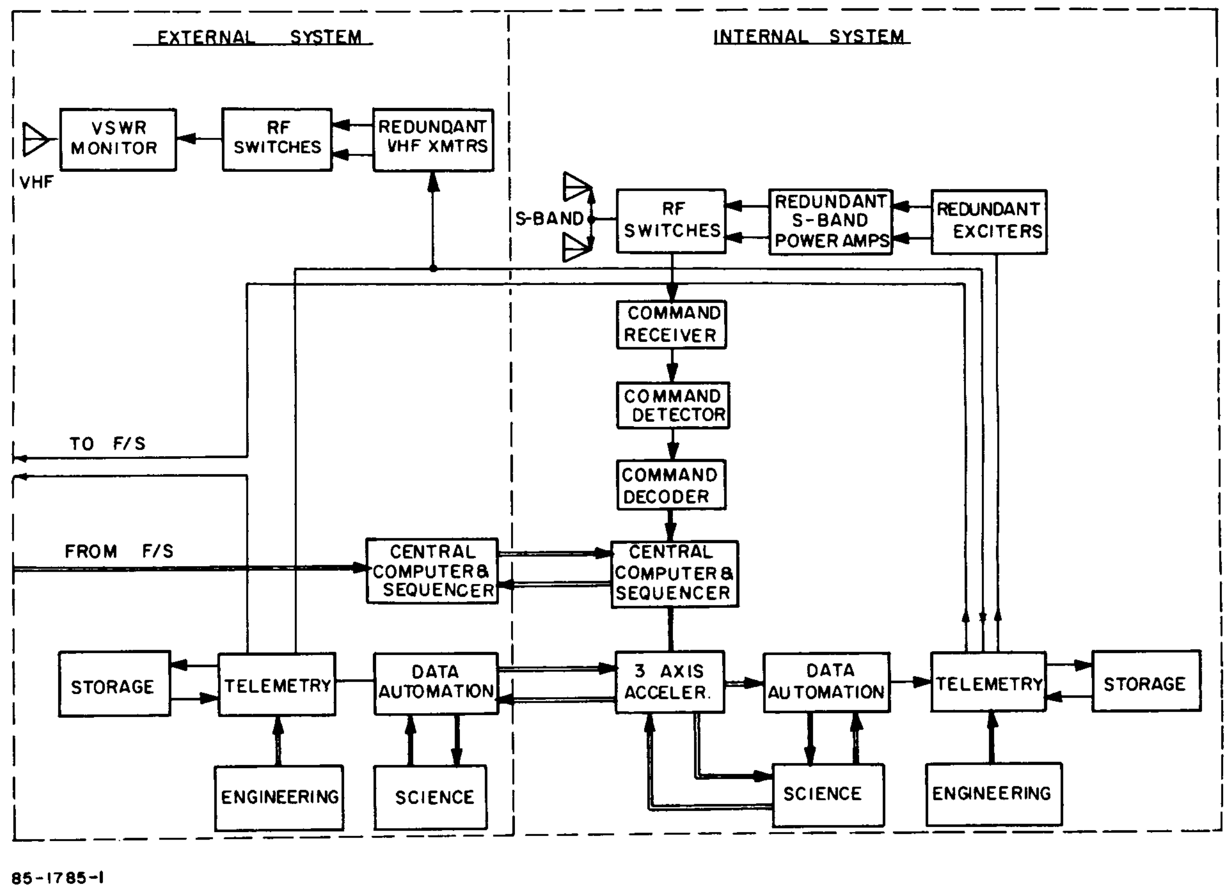


Figure 14 NOMINAL PERFORMANCE COMMUNICATION SUBSYSTEM

6.11.2.2 Telecommunication Subsystem

In the design of the Flight Capsule telecommunication subsystem to meet the functional requirements in paragraph 6.11.1, the following design constraints shall be assumed for the Flight Capsule/Flight Spacecraft relationship:

1. The Flight Capsule telecommunications equipment for the direct link shall be compatible with nonmission-oriented Deep Space Instrumentation Facility equipment

2. The bit error probability for Flight Capsule telemetry at threshold shall be less than $P_e = 10^{-3}$

3. The bit error probability for the Flight Capsule command link at threshold shall be less than $P_e = 10^{-5}$

4. The Flight Capsule command equipment shall be compatible with the command verification equipment (Spec GMG-50109-DSN-3)

5. The Flight Capsule command equipment shall employ pseudo-noise synchronization techniques

6. The maximum direct link communication range shall not exceed 3×10^8 km

7. The maximum relay link communication range between the Flight Capsule and Flight Spacecraft shall not exceed 75,000 km

8. The Flight Capsule operating frequency shall lie within the band 267 mc to 273 mc

9. The Flight Capsule net antenna gain shall be greater than -1db within ±70 degrees of the peak value

10. The peak radiation vector of the Flight Spacecraft relay antenna shall be oriented to accommodate the typical relationships shown in Figure 15 (The shaded portion of the Flight Spacecraft trajectories represents those periods of time in which Flight Capsule to Flight Spacecraft communications are possible)

11. The peak gain of the Flight Spacecraft relay antenna shall be not less than 9 db

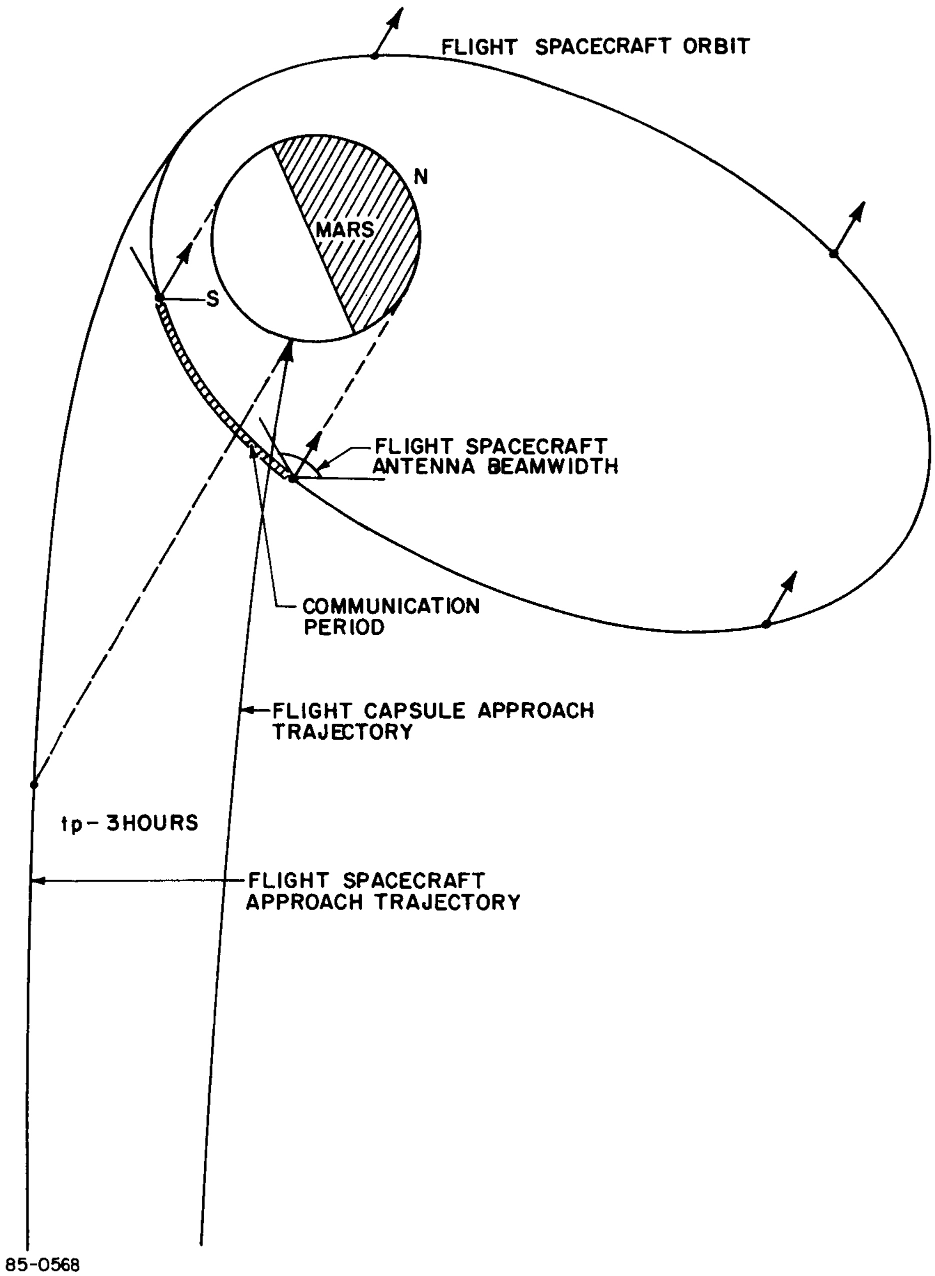


Figure 15 ORBIT/CONTACT ANALYSIS

12. The Flight Spacecraft receiver design is dependent on its data rate capability to DSIF, as follows:

a. For bit rate capability greater than 1000 bps - Direct feed through will be utilized whereby the received data subcarrier will directly modulate the transmitter

b. For bit rate capability less than 1000 bps - Bit detection will be performed on the Flight Spacecraft

13. The Flight Spacecraft recorder capacity shall be a minimum of 100,000 bits.

6.11.2.3 Radio Subsystems

1. General Requirements -- The Flight Capsule radio subsystem shall:

a. Modulate the transmitter signal with the telemetry signal with the telemetry signal and, depending on the particular communication link selected, transmit a modulated RF signal to the Flight Spacecraft or to DSIF. Direct link modulation shall be multiple frequency shift (MFS) in which data is transmitted word by word rather than bit-by-bit, that is, 2^n discrete frequencies representing the 2^n combinations in an n bit digital worked (n is 5). Relay link modulation to be determined. (Frequency shift keyed (FSK) and/or phase shift keyed PSK) will be selected.)

b. Receive the RF-signal transmittal from DSIF.

c. Demodulate the receiver RF signal and send a composite command signal to the central computer and sequencer.

2. Specific Requirements - Direct Link-S-Band --

a. The antenna shall be capable of handling an input power of 25 watts.

b. The input VSWR of the subsystem shall not exceed 1.50:1.00 at the center transmitting frequency of 2300 mc.

c. The antenna radiation pattern shall provide broad hemispherical coverage in the sector of the antenna - in a conical sector of 120 degrees about the antenna vertical, the gain shall average 0.0 db.

d. The input impedance of the antenna shall be 50 ohms nominal.

3. <u>Specific Requirements - Relay Link - VHF</u>

a. The input impedance of the antenna shall be 50 ohms nominal.

b. The antenna shall be capable of handling 25 watts cw power.

c. The input VSWR to the antenna shall not exceed 2.0:1.0.

d. The radiation pattern shall provide broad hemispherical coverage in a region ±90 degrees from the antenna boresight with a peak gain of at least 5.0 db.

4. <u>Specific Requirements - Antenna Transfer</u>

For the direct communication link, there shall be three antennas; one located external to the Landed Capsule and on the entry-vehicle after-body; used from separation of the Flight Capsule from the Flight Spacecraft to surface impact. The other two shall be located on either side of the Landed Capsule to be used after impact. The selection of which antenna will be energized, will depend on the Landed Capsule attitude. For the relay link, there shall be only one antenna.

6.11.2.4 Telemetry Subsystem

1. <u>Requirements</u> -- The telemetry subsystem shall:

a. Transduce selected engineering parameters into appropriate electrical signals.

b. Condition selected engineering telemetry signals to match their characteristics to the telemetry encoding circuits.

c. Time multiplex (commutate) engineering measurement signals.

d. Convert each commutated engineering data sample to a seven-bit binary work.

e. Appropriately modulate (PSK, MFS, FSK) the transmitter(s) with the binary wave train.

2. Data Modes

a. The external portion of the telemetry subsystem shall have the following five data modes:

Initiation	Mode	Function
Flight Spacecraft/command	1	In-transit and separation checkout
Flight Spacecraft/command	2	Postseparation checkout - data transmission in real time, alternating science and engineering frames
g	3	Entry data storage - science data and critical engineering stored in external memory
g and time	4	Entry descent data transmission -10g plus 11 seconds to impact - Data transmission via relay
g and time	5	Standby - used in cruise - only external CC&S and master timer operating

b. The internal portion of the telemetry subsystem shall have the following three data modes:

Tether event	1	Landed data storage - store impact data and post impact science and engineering data
Landed event	2	Post-impact relay transmission - playout of data stored in Mode 1
Time	3	Post-impact direct transmission - playout of stored data at 1 hour and 24 hours after impact

6.11.2.5 Central Computer and Sequencer

1. Requirements -- The subsystem shall:

a. Detect commands in the form of binary PSK modulation on a square-wave subcarrier which is the output of the radio subsystem demodulator.

b. Decode the digital commands, routing real time commands to recipient Flight Spacecraft subsystems.

c. Self-sense equipment failures and exercise remedial action.

d. Provide the necessary timing signals to all subsystems.

e. Generate appropriate signals for initiation of selected events.

2. Command Mode -- There shall be two modes of operation, real-time discrete commands and internal program.

6.11.2.6 Data Automation Subsystem*

6.11.2.7 Channel Assignments

Channel assignments for the commutators of both the telemetry subsystem (internal and external) and the data automation subsystem (internal and external) are required. Commutation of the channels shall be performed by a digital multiplexer. Each channel is equipped with an oscillator whose frequency corresponds to the analog output from the instrument. When enabled by the channel signal, the oscillator generates a wave train and the number of pulses counted in a given time represent the digital word. Most engineering measurements are binary indicators, and seven such measurements may be accommodated in a single channel. A summary of the number of channels required and their use is given in Table XI.

TABLE XI

CHANNEL SUMMARY

Use	Internal DAS	External DAS	Internal TM	External TM
Synchronization	3	3	5	5
Identification	1	4	1	1
Measurement and Calibration	32	258	106	106

The required science and engineering measurements to be implemented by the telecommunications subsystem are shown in Tables XII and XIII, respectively. The subscripts, i and e, in the Mode column in these tables refer to the internal and external instrumentation, respectively.

TABLE XII

TELEMETERING CHANNEL ASSIGNMENT (Science)

	Measurement	Sampling Rate	Analog/Binary	Mode
1.	Spare	12/hour	A	$(2, 3, 4)_e$
2.	Beta scatterer	12/min	A	$(2, 3, 4)_e$
3.	Gas chromatograph	2/min	A	$(2, 3, 4)_e$
4.	Mass spectrometer	12/min	A	$(2, 3, 4)_e$
5.	Radiometer	1/sec	A	$(2, 3, 4)_e$
6.	Pressure transducer	30/min	A	$(1, 2, 3, 4)_e$
7.	Temperature transducer	30/min	A	$(1, 2, 3, 4)_e$
8.	Radar altimeter	1/sec	A	4_e
9.	Spare	30/min	A	4_e
10.	Accelerometer	1/sec	A	$(2, 3, 4)_e$
11.	Acoustic transducer	30/min	A	4_e
12.	Spare	1 sample	A	1_i
13.	Spare	1 sample	A	1_i
14.	Alpha scatterer	6/day	A	1_i
15.	Cosmic radiation	1/hour	A	1_i
16.	Gas chromatograph	6/day	A	1_i
17.	Spare	6/day	A	1_i
18.	Force anemometer	6/day	A	1_i

TABLE XII (Concl'd)

	Measurement	Sampling Rate	Analog/Binary	Mode
19.	Dust particle detector	6/day	A	1_i
20.	Surface radiation	1/hour	A	1_i
21.	Pressure transducer	2/hour	A	1_i
22.	Temperature transducer	1/hour	A	1_i
23.	Acoustic transducer	1/hour	A	1_i
24.	Water detector	1/hour	A	1_i
25.	Atmos temperature	2/hour	A	1_i

TABLE XIII

TELEMETERING CHANNEL ASSIGNMENT (Engineering)

	Measurement	Sampling Rate	Analog/Binary	Mode
1.	Sterilization canister pressure	1/min	A	1_e
2.	Sterilization canister separation	1/min	B	1_e
3.	Continuity loop sterilization separation	1/min	B	1_e
4.	Separation switch, sterilization canister	1/min	B	1_e
5.	Separation switch, flight capsule	1/min	B	1_e
6.	Continuity loop, umbilical	1/min	B	1_e
7.	Gyro output	1/min	A	1_e

TABLE XIII (Cont'd)

	Measurement	Sampling Rate	Analog/Binary	Mode
8.	ACS error signal	1/min	A	1_e
9.	ACS system gas pressure	1/min	A	1_e
10.	Safe & initiate, ACS and Δ velocity	1/min	B	1_e
11.	ACS solenoid current	1/min	B	1_e
12. a	ΔV continuity loop	1/min	B	1_e
12. b	Spin rocket continuity loop	1/min	B	1_e
13.	ΔV Rocket ignition	1/min	B	1_e
14.	Spin rocket ignition	1/min	B	1_e
15.	Velocity	1/min	A	1_e
16.	External battery voltage	1/min	A	$(1, 2, 3, 4)_e$
17.	Internal battery No. 1 voltage	1/min	A	1_e, 1_i
18.	Internal battery No. 2 voltage	1/min	A	1_e, 1_i
19.	Propulsion and ACS electronics assembly separation switch	1/min	B	1_e
20.	Relay transmitter No. 1 current	1/min	B	All
21.	Relay transmitter No. 2 current	1/min	B	All
22.	Command receiver No. 1 current	1/min	B	1_i

TABLE XIII (Cont'd)

	Measurement	Sampling Rate	Analog/Binary	Mode
23.	Direct exciter No. 1 current	1/min	B	1_i
24.	Direct exciter No. 2 current	1/min	B	1_i
25.	Direct amplifier No. 1 current	1/min	B	1_i
26.	Direct amplifier No. 2 current	1/min	B	1_i
27.	Safe & initiate, chute heat shield, external payload	1/min	B	$(3, 4)_e$
28.	Continuity; chute, heat shield, external payload	1/min	B	$(3, 4)_e$
29.	De-spin rocket ignition	1/min	B	$(1, 2, 3, 4)_e$
30.	Chute deployment squib current	1/sec	B	$(3, 4)_e$
31.	Chute deployment separation switch	1/sec	B	$(3, 4)_e$
32.	External battery temperature	1/min	A	$(1, 2, 3, 4)_e$
33.	Internal battery No. 1 temperature	1/min	A	1_i
34.	Internal battery No. 2 temperature	1/min	A	1_i
35.	Heat sink temperature	1/min	A	All
36.	Capsule internal temperature	1/min	A	All

TABLE XIII (Cont'd)

	Measurement	Sampling Rate	Analog/Binary	Mode
37.	RF amplifier No. 1 temperature	1/min	A	1_i
38.	RF amplifier No. 2 temperature	1/min	A	1_i
39.	External science current	1/min	A	$(1, 2, 3, 4)_e$
40.	Internal science current	1/min	A	1_i
41.	ACS system current	1/min	A	1_e
42.	S-band antenna No. 1 forward power	1/min	A	3_i
43.	S-band antenna No. 2 forward power	1/min	A	3_i
44.	VHF antenna No. 1 forward power	1/min	A	2_i
45.	VHF antenna No. 2 forward power	1/min	A	2_i
46.	VHF antenna forward power	1/min	A	$(1, 2, 4)_e$
47.	Command receiver No. 1 received signal strength	1/min	A	1_i
48.	Command verification	1/min	B	1_i
49.	Command verification, quantitative, as required	1/min	B	1_i
50.	Chute disreef squib current	1/sec	B	4_e
51.	Chute disreef continuity loop	1/min	B	4_e

TABLE XIII (Cont'd)

	Measurement	Sampling Rate	Analog/Binary	Mode
52.	Heat shield deployment squib current	1/sec	B	4_e
53.	Tether cutter squib current	1/sec	B	1_i
54.	Safe & initiate; impact attenuator, science deployment	1/min	B	1_i
55.	Impact attenuator deployment squib current	1/min	B	1_i
56.	Science experiments deployment command	1/min	B	1_i
57.	Cosmic radiation detector deployed	1/min	B	1_i
58.	Surface radiation detector	1/min	B	1_i
59.	Penotrometer deployed	1/min	B	1_i
60.	Alpha scatter deployed	1/min	B	1_i
61.	Hot wire anemometer deployed	1/min	B	1_i
62.	Force anemometer deployed	1/min	B	1_i
63.	Surface temperature sensor deployed	1/min	B	1_i
64.	Dust particle detector	1/min	B	1_i
65.	S-band antenna No. 1 reverse power	1/min	A	3_e
66.	S-band antenna No. 2 reverse power	1/min	A	3_e

TABLE XIII (Concl'd)

	Measurement	Sampling Rate	Analog/Binary	Mode
67.	VHF antenna No. 1 reverse power	1/min	A	2_i
68.	VHF antenna No. 2 reverse power	1/min	A	2_i
69.	VHF antenna reverse power	1/min	A	$(1, 2, 4)_e$
70.	Relay transmitter No. 1 oven temperature	1/min	A	$(1, 2, 3, 4)_e$
71.	Relay transmitter No. 2 oven temperature	1/min	A	$(1, 2, 3, 4)_e$
72.	Central timer & sequencer clock rate	1/min	B	All
73.	Power conditioning voltages	1/min	A	All
74.	Reference voltages	1/min	A	All
75.	Storage	1/min	B	3_e, 4_e, 1_i
76.	Capsule time	1/min	B	All
77.	Fuze monitoring on all power users	1/min	B	All
78.	Redundancy indicators	1/min	B	All

6.11.2.8 Design Parameters and Performance Summary

1. Parameters -- The values of the principle parameters that determine the overall telecommunications subsystem performance are shown in Table XIV.

2. Performance Summary -- The performance margin of these selected subsystem parameters are plotted in Figures 16 and 17 for both the Earth command link and the direct telemetry link as a function of arrival date. The following definitions were used as the ground rules in the establishment of the performance summary shown on these figures:

a. Performance Margin -- The performance margin is defined as the ratio of the nominal received signal level to the nominal threshold signal level expressed in db. It shall be considered acceptable when the margin is equal to or greater than the linear sum of the adverse system tolerances.

b. Nominal Received Signal Level -- The nominal received signal is defined as the received signal level as calculated from the nominal system parameters (gains, losses, and power levels). The calculation excludes as far as possible, all arbitrary margins, pads or unknown factors.

c. Nominal Threshold Signal Level -- The nominal threshold signal level shall be the received signal level required to achieve a specified threshold signal-to-noise ratio in the effective noise bandwidth of the detector or demodulator given the system noise spectral density.

d. Threshold signal-to-noise ratio required at the detector that will result in the minimum acceptable performance.

6.11.3 Functional Interfaces*

6.12 ELECTRICAL POWER AND CONTROL

6.12.1 Functions

6.12.1.1 Primary

The primary function of the electrical power and control subsystem shall be to provide an electrical energy source whose output is appropriately conditioned and controlled to the requirements of the Flight Capsule subsystems.

TABLE XIV

PRINCIPLE PERFORMANCE PARAMETERS

A-	Radio Transmission	(Relay Link)	(Direct Link)
	Nominal transmitter power, watts:	30	20
	Frequency, mc:	267 to 273	2300
	Bit rate, bits per sec:	1056 and 64	2
	Antenna, type:	Logarithmic Spiral	V
B-	Command		
	Mode:	Direct Link	
	Frequency, mc:	2113	
	Quantity:	34	
	Type:	Discrete	
C-	Telemetry		
	Sampling rate, average samples per min:	1	
	Accuracy, bits:	5	
	Measurements, quantity:	78	
D-	Data Automation		
	Sampling rate, average samples per sec:	1	
	Accuracy, bits:	7	
	Measurements, quantity:	25	
E-	Command Computer, and Sequencer		
	Modes, quantity:	10	
	Control Function, quantity:	51	

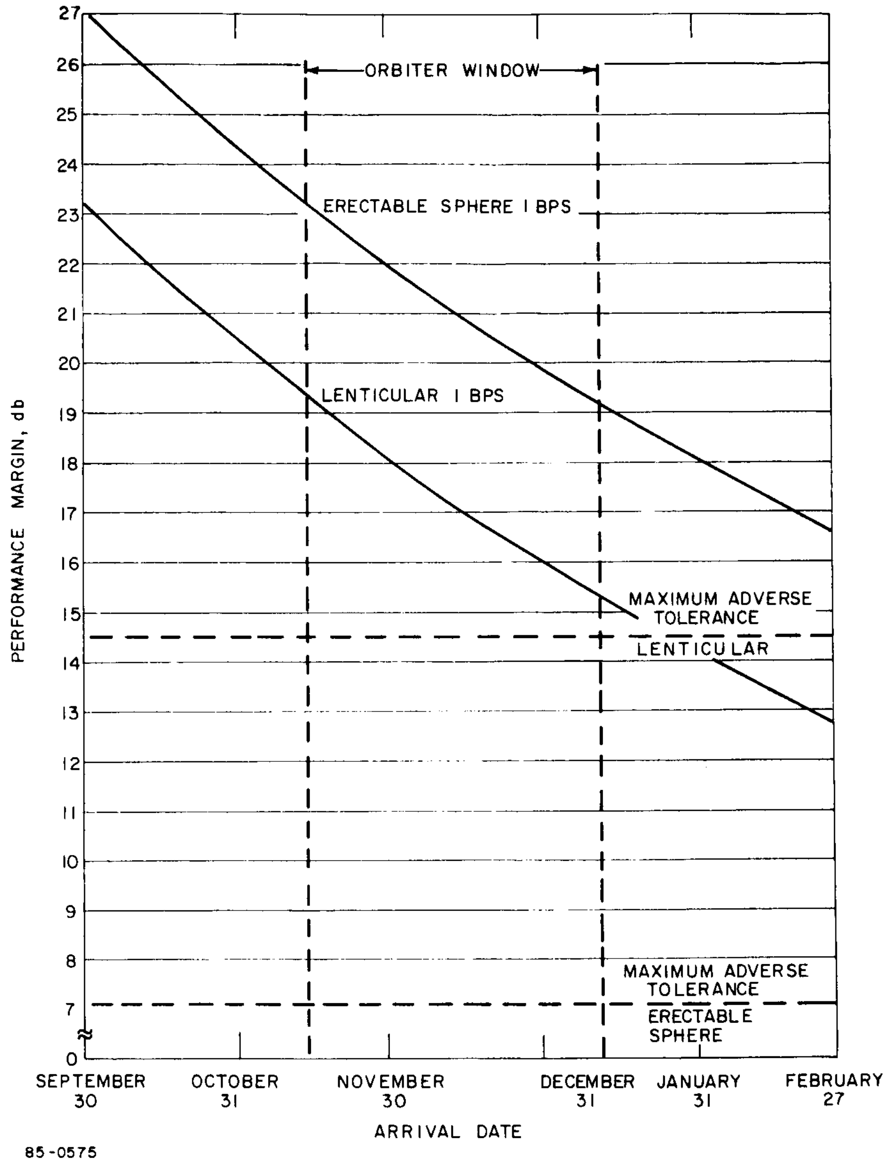


Figure 16 COMMAND-LINK PERFORMANCE MARGIN

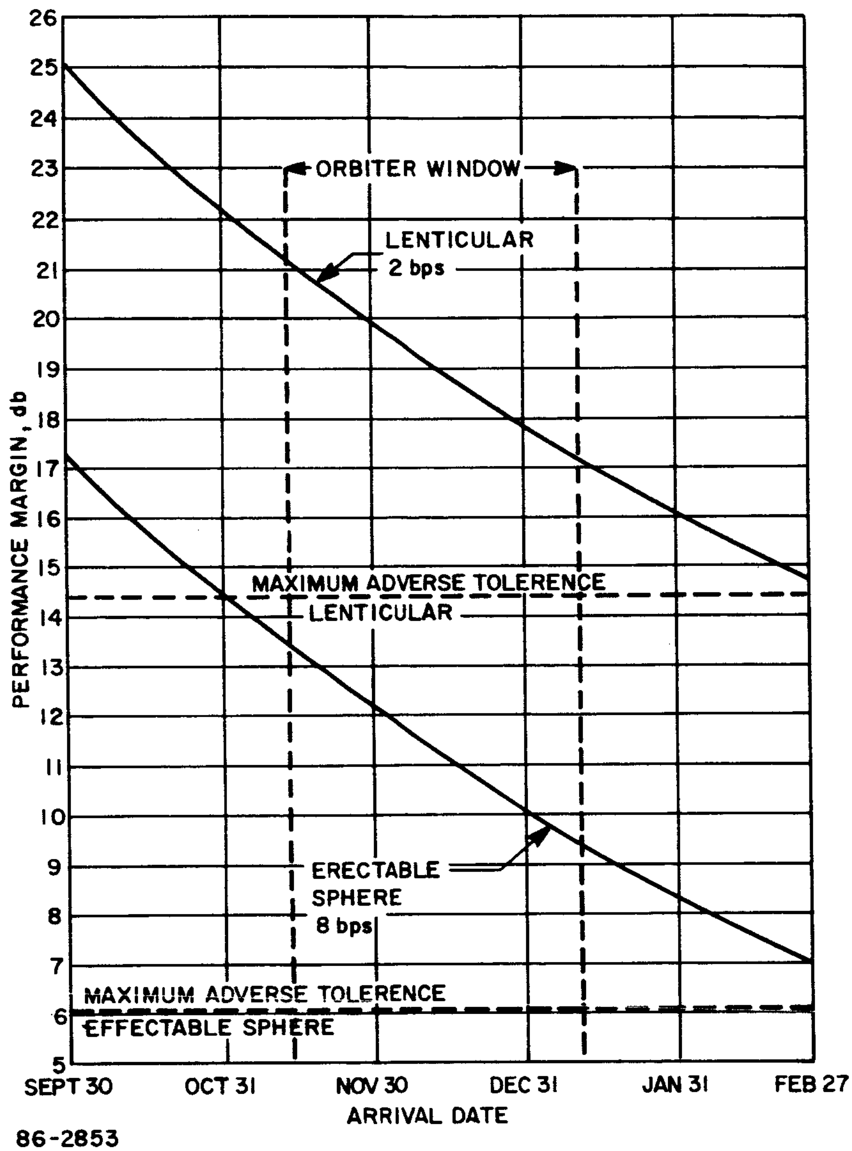


Figure 17 DIRECT – LINK PERFORMANCE MARGIN

6.12.1.2 Secondary

As a secondary function, it shall be capable of receiving and conditioning electrical energy from an external Flight Spacecraft or OSE source, to supplement its primary function and to maintain a specified level of stored electrical energy during all mission phases prior to separation from the Flight Spacecraft.

6.12.2 Performance/Design Requirements

6.12.2.1 Performance

1. Cruise to Encounter -- A portion of the Flight Spacecraft solar panel power output shall be continuously furnished to the Flight Capsule for active temperature control and battery trickle charge, in addition to periodic requirements for subsystem status checkouts.

2. Encounter to Separation -- Power shall be supplied by the Flight Capsule battery to the Flight capsule communications and instrumentation, in order to checkout Flight Capsule status. The battery shall be recharged by Flight Spacecraft power just prior to separation.

3. Separation to Entry -- After separation from the Flight Spacecraft, the Flight Capsule battery shall supply power to all subsystems during the attitude control maneuver and Flight Capsule status check. Thereafter, during the nominal 12-day cruise period, the battery shall supply power as required for environmental control, subsystem checkout and communications.

4. Entry Through Post-Impact -- The battery shall supply power as required to the science instruments, communications and environmental control.

6.12.2.2 Design

1. Location of the Charge Regulator -- The battery charge regulator shall be located aboard the Flight Spacecraft, to minimize Landed Capsule weight and eliminate shock at impact which could cause physical damage to the charge regulator and create a battery-to-ground short.

2. Maximizing Energy Available to Flight Capsule -- The battery shall be kept charged by the Flight Spacecraft power supply until separation to maximize the energy available to the Flight Capsule power subsystem after separation from the Flight Spacecraft.

3. Isolation of Battery Sections -- The battery shall be split into three 28-volt sections, each capable of supplying power for separate functions. Each battery section will be electrically isolated from the others immediately after separation. Before separation, each section will be connected in parallel to the Flight Spacecraft power supply charge regulator. These separate battery sections shall be supplied for each of the following phases of Flight Capsule operation:

a. Separation to impact, all subsystems

b. Post-impact, direct-link transmitter and all instrumentation

c. Post-impact, relay-link transmitter (if required)

4. Redundant Switching Regulators -- A switching regulator shall be provided for each battery section to ensure that the failure of any one battery section or regulator will not terminate the mission.

5. Ground Isolation -- All components of the electrical power and control subsystem shall be designed with greater than 1000-ohms dc resistance to the Flight Capsule structure.

6. All the battery sections shall be kept fully charged until separation.

7. After separation from the Flight Spacecraft, the stabilized temperature of battery sections shall not be allowed to exceed 100°F or drop below 40°F.

8. From launch to separation from the Flight Spacecraft, the stabilized temperature of the battery sections during charge or discharge must not exceed the limits of +40°F to 120°F.

9. The energy demand from the battery sections at any one time from launch to 16 hours before separation shall not exceed 50 percent of the total battery rated capacity of 1375 watt-hours at 75°F.

10. The weight required by the electrical power and control subsystem, excluding battery charge regulator, shall not exceed 120 pounds.

11. The weight of the battery charge regulator shall not exceed 5.5 pounds.

12. Power Availability -- The power available to the Flight Capsule from launch to separation is limited by the energy conversion ability of the solar panels on the Flight Spacecraft, the rating of the electrical conversion equipment on the Flight Spacecraft and necessary power demands of the Flight Spacecraft systems. The power available to the Flight Capsule after separation is limited by the capacity of the battery sections. At no time is the discharge power to exceed the 1-hour rate (i.e., power level that would discharge battery capacity in 1 hour).

13. Battery Ratings -- The minimum total rated capacity of all the battery sections shall be 945 watt-hours at 40°F discharge temperature, allocated to each battery section as follows:

a.	Separation to impact supply	184 watt-hours
b.	Post-impact, direct-link and all instruments	751 watt-hours
c.	Post-impact, relay-link supply (if required)	10 watt-hours

6.12.3 Functional Interfaces

6.12.3.1 Inputs to the Electrical Power and Control Subsystem

1. From the launch complex, the electrical power and control subsystem receives:

   a. External power to operate the Flight Capsule subsystems during prelaunch checkout at 28 Vdc nominal,

   b. External power to keep the batteries charged at 15 milliamps per ampere-hour of battery capacity for the trickle charge and the current equivalent of the 16-hour rate for full charge.

2. From the Flight Spacecraft photovoltaic solar panel power supply, the electrical power and control subsystem receives a dc input of 33 volts minimum and 52 volts maximum to charge the battery and supply environmental control.

6.12.3.2 Outputs from the Electrical Power and Control Subsystem

Outputs from the electrical power and control subsystem are as follows:

1. To the launch complex equipment; indications and measurements for prelaunch monitoring of subsystem performance. Specific measurements to be made are listed in Table XIII.

2. To telemetering subsystem, a number of analog signals representing measurements of power subsystem operating parameters. Specific measurements to be made are listed in Table XIII. Each output will be conditioned to the standard telemetry measurement range of 0 to 5 volts.

3. To both the relay-link and direct link transmitters of the communication subsystem; raw dc power with the following characteristics:

a. Voltage range: -20 to -32 Vdc

b. Rated load: 125 watts

c. Maximum load: 166 watts, including all transients

4. To all the other subsystems, regulated dc power with the following characteristics:

a. Amplitude: 28 Vdc ± 1 percent

b. Ripple: less than 0.10 volt peak-to-peak

c. Transients: less than 0.5 volt for 5 amp load charges

6.13 SIGNAL AND POWER INTERCONNECTION

6.13.1 Functions

The signal and power interconnection subsystem shall provide all means of signal and power transmission within the Flight Capsule. In addition, the same equipment shall be utilized in a like manner during PV integration, on-the-pad operations, conditioning and checkout through appropriate umbilical connections.

6.13.2 Performance/Design Requirements

6.13.2.1 Applicable Specifications

1. Mil Specs -- MIL-W-8160 Wiring, Guided Missile Installation of, General Specification for QQ-S-571 solder; lead alloy, lead tin alloy, and tin alloy; flux-covered ribbon wire and solid form.

2. NASA Specs -- MSFC-PROC-158 Soldering of Electrical Connections (high reliability)

MSC-ASPO-S-5
MSC-ASPO Soldering Specification

6.13.2.2 Harness Requirements

1. The electrical system harnessing design shall include control of magnetic and electrostatic interference, power losses, IR drop in conductors and grounds, and insulation electrical characteristics. In addition, consideration shall be given to mechanical aspects including insulation strength, protection from heat, ability to survive low temperatures, dispersion of heat, tolerance for vibration, adequate support, and accessability during construction, rework adjustment and test.

2. Detail cabling design information shall be developed as part of the final design.

3. Wires shall be installed in accordance with MIL-W-8160. Soldering shall be in accordance with MSC-ASPO-S-5 using solder SN 60 in accordance with QQ-S-571 Type AR. Work shall be done by certified personnel.

4. Shield ground wires shall be No. 20AWG unshielded.

5. Component reference designations shall be marked per Avco Specification RAD-P-100001-12B adjacent to connectors - cable part number shall be marked approximately 6.0 inches from mating face of connectors.

6. Connectors shall be segregated where possible according to the characteristics of current flowing, to reduce cross talk.

7. Cables passing through restricted areas containing wires carrying currents which may interfere with each other shall be segregated into logical groups as soon as practical.

8. Wires shall not interfere with the withdrawal or installation of electrical components - Connectors shall be located to permit mating and unmating with minimum disassembly.

9. Separate signal return and power lines shall be provided for components using power converters, inserter motors, or other decoupling devices.

10. Separate signal return lines shall be provided for each signal in components having multiple output signals.

11. Shielded twisted groups shall have connector pins in close proximity.

12. Wires shall be grouped in any given connector by signal and power level of current characteristics to avoid transfer of deleterious energies.

6.13.3 Functional Interfaces*