

VOYAGER

SPACECRAFT Phase B, Task D

FINAL REPORT

OCTOBER 1967

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Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

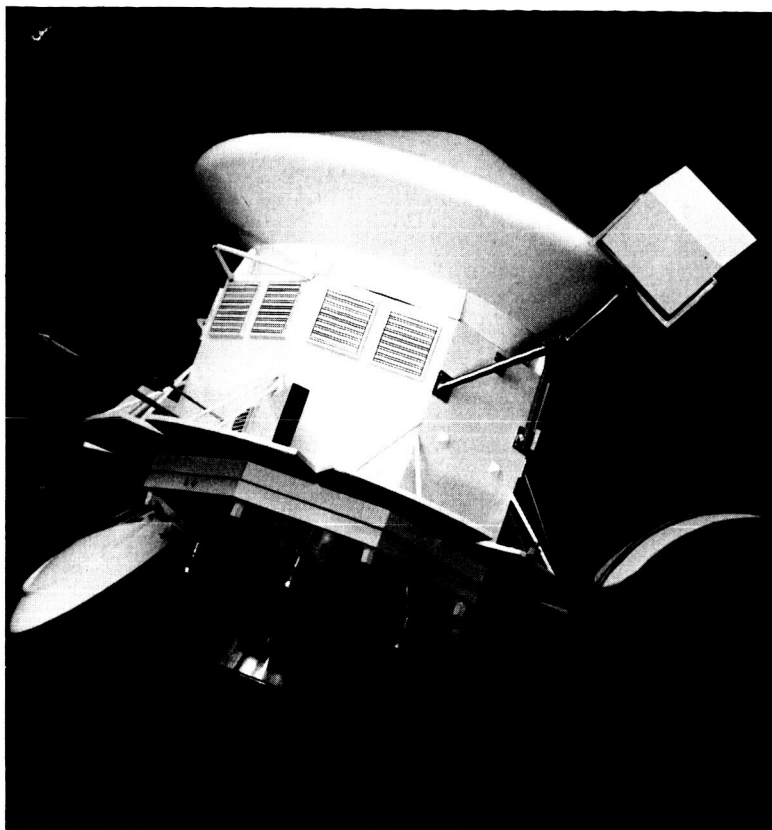
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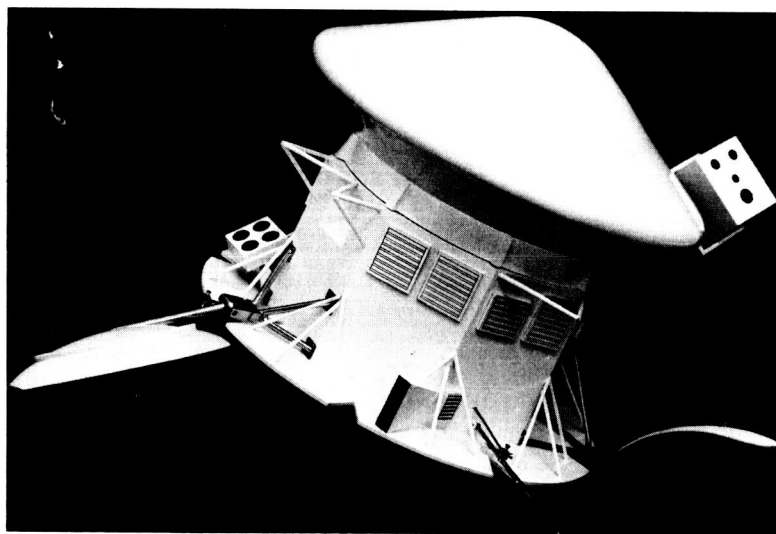
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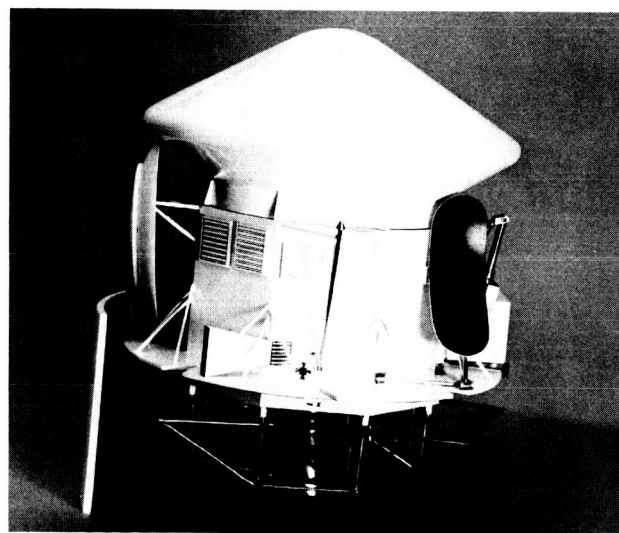


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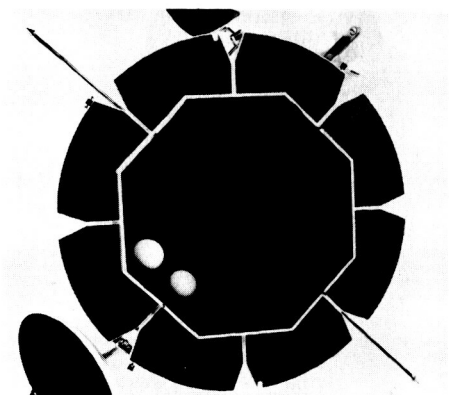
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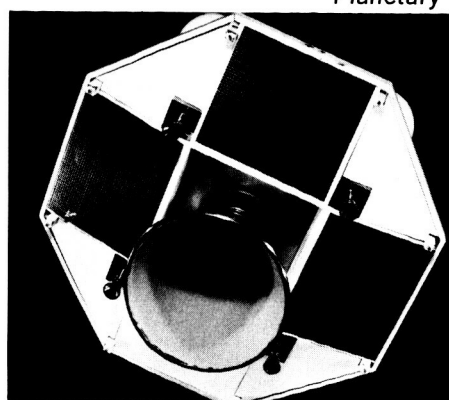
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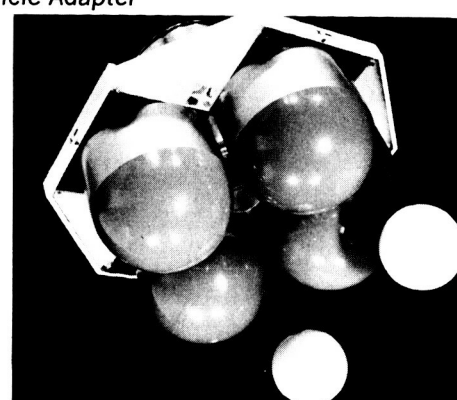
Stowed Configuration with Section of Shroud and Planetary Vehicle Adapter



Propulsion Module, Top View



Propulsion Module, Bottom View



Equipment Module, Bottom View

VOYAGER

SPACECRAFT Phase B, Task D

FINAL REPORT

Volume 2. Spacecraft Design and Performance

OCTOBER 1967

Prepared for

GEORGE C. MARSHALL SPACE FLIGHT CENTER

Huntsville, Alabama

TRW
SYSTEMS GROUP

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CONTENTS

	Page
1. INTRODUCTION	1-1
2. SPACECRAFT DEFINITION	2-1
3. THE 1973 MISSION	3-1
4. THE 1975, 1977, AND 1979 MISSIONS	4-1
5. REFERENCES	5-1

ILLUSTRATIONS

Figure		Page
2-1	Preliminary Specification	2-3
2-2	Planetary Vehicle External View	2-5
2-3	Spacecraft Inboard Profile.	2-17
2-4	Spacecraft Exploded View	2-19
2-5	Electronic Equipment Locations	2-20
2-6	Equipment Installation Layout	2-21
2-7	Sensor and Appendage Geometry.	2-25
2-8	Antenna and PSP Gimbal Geometry	2-27
2-9	Spacecraft Geometry and Reference Axes.	2-28
2-10	An Inboard Profile of the Propulsion Module.	2-31
2-11	Basic Structure of the Propulsion Module.	2-33
2-12	Capsule - Spacecraft Mechanical Interface	2-35
2-13	Spacecraft - Launch Vehicle Electrical Interface . . .	2-37
2-14	Spacecraft Functional Block Diagram	2-39
3-1	In the Launch Phase	3-3
3-2	S-IVB Second Burn	3-3
3-3	Acquisition of Sun-Canopus	3-4
3-4	In the Cruise Configuration	3-4
3-5	Typical Maneuver Sequence	3-5
3-6	Typical Interplanetary Cruise.	3-6
3-7	Orbit Insertion Maneuver.	3-6
3-8	Capsule Separation and Support Sequence	3-7
3-9	Spacecraft in Orbit	3-8
3-10	Post-Separation Capsule Support.	3-8



ILLUSTRATIONS (Continued)

Figure		Page
3-11	Interplanetary Trajectory Constraints	3-22
3-12	The Space Vehicle On-Stand Configuration	3-25
3-13	Saturn V Payload Capability for Voyager	3-27
3-14	Voyager Payload Versus Launch Azimuth	3-28
3-15	Weight of Each Planetary Vehicle	3-28
3-16	Voyager Shroud Geometry	3-29
3-17	Dynamic Envelopes	3-30
3-18	Division of the Planetary Vehicle Dynamic Envelope	3-37
3-19	The Characteristics of a Type I Trajectory	3-38
3-20	Maneuver Sequences of the Two Planetary Vehicles	3-39
3-21	Orbit Insertion Geometry	3-41
3-22	Velocity Increment	3-42
3-23	AIP Launch Period Identification Module	3-44
3-24	For a 1100 x 10, 000 km Orbit and a 20-Day Launch Period	3-45
3-25	Optimum 20-Day Launch Periods	3-47
3-26	For a 1000 x 20, 000 km Orbit and a 20-Day Launch Period	3-50
3-27	For a 1100 x 10, 000 km Orbit and a 30-Day Launch Period	3-50
3-28	For a 1000 x 20, 000 km Orbit and a 30-Day Launch Period	3-51
3-29	Tracking Uncertainty	3-60
3-30	Cumulative Distribution of the Velocity Increment Required	3-62
3-31	Cumulative Distribution of the B. T Component	3-63

ILLUSTRATIONS (Continued)

Figure		Page
3-32	Cumulative Distribution of the B. R Component	3-64
3-33	Cumulative Distribution of the Dispersion in Time of Flight	3-65
3-34	Uncertainties in the Orbital Parameters	3-68
3-35	Uncertainties After 14 Days of Tracking the Orbit . . .	3-69
3-36	Uncertainties in Orbital Tracking	3-70
3-37	Uncertainties in the Gravitational Constant of Mars . .	3-71
3-38	Effect of Survival and Growth Probability	3-73
3-39	Suballocation of Allowable Probabilities	3-75
3-40	Biasing to Avoid Launch Vehicle Impact at Mars	3-79
3-41	Orbits that Assure at Least 12 Years	3-84
3-42	Orbit Insertion Maneuver	3-85
3-43	Potential Sources of Contaminated Effluents	3-89
3-44	Post Encounter Cumulative Data	3-106
3-45	Structure Subsystem Reliability Model.	3-111
3-46	Guidance and Control Reliability Diagram	3-113
3-47	Command Unit Reliability Diagram	3-114
3-48	Computing and Sequencing Unit Reliability Diagram. .	3-115
3-49	S-Band Radio Reliability Diagram.	3-116
3-50	Telemetry and Data Storage Reliability Diagram	3-118
3-51	Electrical Power Subsystem Diagram	3-119
3-52	Electrical Distribution Subsystem Reliability Diagram.	3-120
3-53	Temperature Control Diagram	3-121
3-54	Reliability Block Diagram of PSP Equipment	3-123



ILLUSTRATIONS

Figure		Page
4-1	Direction of Earth	4-2
4-2	Characteristics of a Type I Trajectory in 1975 . . .	4-5
4-3	Characteristics of a Type II Trajectory in 1975 . . .	4-6
4-4	Characteristics of a Type I Trajectory in 1977 . . .	4-7
4-5	Characteristics of a Type II Trajectory in 1977 . . .	4-8
4-6	Characteristics of a Type I Trajectory in 1979 . . .	4-9
4-7	Characteristics of a Type II Trajectory in 1979 . . .	4-10
4-8	Performance Capability, 1975 Type I	4-12
4-9	Performance Capability, 1975 Type I	4-12
4-10	Performance Capability, 1975 Type II	4-13
4-11	Performance Capability, 1975 Type II	4-14
4-12	Performance Capability, 1977 Type I	4-15
4-13	Performance Capability, 1977 Type I	4-15
4-14	Performance Capability, 1977 Type II	4-16
4-15	Performance Capability, 1977 Type II	4-16
4-16	Performance Capability, 1979 Type I	4-18
4-17	Performance Capability, 1979 Type I	4-18
4-18	Performance Capability, 1979 Type I	4-18
4-19	Performance Capability, 1979 Type I	4-20
4-20	Dominant Factors	4-21
4-21	Dispersion Ellipses for Various Years	4-22
4-22	Post Encounter Cumulative Data, Type I	4-29
4-23	Post Encounter Cumulative Data, Type II	4-29
4-24	Post Encounter Cumulative Data, Type I	4-30
4-25	Post Encounter Cumulative Data, Type II	4-30
4-26	Post Encounter Cumulative Data, Type I	4-31



1. INTRODUCTION

This volume presents the design and performance characteristics for TRW's recommended spacecraft, and in particular its capability for performing the 1973 Voyager Mars mission. The applicability of the spacecraft system to the 1975, 1977, and 1979 missions is also discussed. The other volumes of this report cover related subjects such as spacecraft subsystems, science payload integration, spacecraft design alternatives and growth potential, operational support equipment, mission-dependent equipment, the implementation plan for the program, and selected engineering tasks.

A single recommended spacecraft preliminary design is presented as called for in the contract guidelines. Various alternatives are discussed and evaluated in Volume 6 to show the analysis leading to this recommended design. The required spacecraft sizing analysis for capsule Cases A and B is also covered in Volume 6, within the context of a general analysis of the broad question of performance margin and its utilization.

The question of spacecraft performance capability is felt to be a central one for the study. It arises from the great capability of the Saturn V and the resulting possibility of significant performance margin. Because of the inherent uncertainty in Mars exploration it is not possible to define just how this margin should be utilized. In the face of this indeterminacy, TRW has sought to implement a balanced approach. Rather than commit the realizable margin to any specific choice, an attempt has been made to achieve significant margin capability while retaining flexibility in its utilization. In particular, the recommended design provides options for substantial capsule weight increase, as well as spacecraft upgrading to support additional science with higher power and communication capability.

The main volume is organized into three parts. Section 2 immediately following presents the recommended design. The spacecraft's configuration, principal characteristics, and operation are covered, along with a discussion of the underlying design philosophy.

Design features are described, such as modularity, interfaces, and the physical arrangement of the structure and equipment. A system-level functional block diagram and a discussion of the spacecraft's operation is also included.

Section 3 covers the 1973 Mars mission and discusses the capabilities of the recommended spacecraft for performing this mission. A formulation of the overall mission problem is given, followed by a presentation to verify that the design described in Section 2 does indeed lead to satisfactory performance of the mission thus defined. The presentation treats adequacy of performance at the system and mission level, with the corresponding discussion at the subsystem level presented in Volumes 3, 4, and 5. Particular dimensions of performance covered are interplanetary trajectories and Mars orbit capability, trajectory accuracy capability, observance of the planetary quarantine constraint, accomplishment of scientific mission objectives, spacecraft communication performance, and probability of success.

Section 4 then deals with applicability of the recommended design to the 1975 through 1979 missions. This presentation utilizes the same approach as Section 3 but in view of the minimum modifications to the spacecraft, the main discussion relates to trajectory-oriented considerations.

Supporting appendices are contained in the annex to this volume. A detailed description of the mission is included along with definition of the related spacecraft operating modes. Selected Mars orbit data calculated for specific possible 1973 missions is also given. The supporting analyses for defining spacecraft performance in regard to accuracy and reliability are provided, along with detailed mass properties data and an equipment list giving component design parameters.

	Page
2. SPACECRAFT DEFINITION	2-1
2.1 THE SPACECRAFT	2-1
2.2 DESIGN PHILOSOPHY	2-5
2.2.1 Spacecraft Performance Capability . . .	2-6
2.2.2 Modularity	2-8
2.2.3 Configuration Design Criteria and Characteristics	2-8
2.3 DESIGN FEATURES	2-14
2.3.1 Equipment Module	2-15
2.3.2 Propulsion Module	2-29
2.3.3 Adapter	2-34
2.3.4 Spacecraft Interfaces	2-34
2.4 FUNCTIONS AND CHARACTERISTICS	2-38
2.4.1 Communications	2-38
2.4.2 Spacecraft Sequencing	2-46
2.4.3 Data Handling	2-49
2.4.4 Attitude Control	2-51
2.4.5 Propulsion	2-55
2.4.6 Electric Power	2-59
2.4.7 Temperature Control	2-61



2. SPACECRAFT DEFINITION

To meet the 1973 Voyager mission requirements as well as those for the 1975-1979 missions, TRW recommends a specific spacecraft design that is here described to the first few layers of detail. After a brief overall summary, this section presents the design philosophy underlying the recommended design. A description of the configuration is next provided in terms of spacecraft design features. Finally there is a functional description dealing with how the spacecraft operates to provide the necessary mission functions.

2.1 THE SPACECRAFT

The recommended spacecraft (frontispiece) is a simple modular design fashioned to carry out all Voyager spacecraft mission functions. It consists of an equipment module and a propulsion module which are mated during the later stages of assembly. The capsule rests on the equipment module, while the propulsion module fastens to a planetary vehicle adapter that mounts to the Saturn V shroud. The adapter remains with the shroud after each planetary vehicle is injected into the interplanetary trajectory. Figure 2-1 summarizes the overall physical and performance characteristics of the spacecraft.

The fully-stabilized spacecraft (Figure 2-2) points its body-fixed array at the sun for the entire mission except when oriented away for thrusting maneuvers to adjust the interplanetary trajectory or Mars orbit. The Mars-viewing scientific payload, enclosed in the double-gimballed planetary scan platform, automatically points downward to scan the planet from orbit regardless of spacecraft orientation, or can be commanded to point at places of interest, including the Martian satellites. Similarly, the double-gimballed high-gain antenna and the single-gimballed medium-gain antenna are continuously oriented toward earth. The PSP, the two antennas just mentioned, and two low-gain antennas deployed into fixed positions, constitute the appendages of the spacecraft.

To carry out the mission the spacecraft undergoes a complex multiphase mission profile. In doing so it must provide the following functions:

- Separate the planetary vehicle from the launch vehicle
- Execute a maneuver causing the two planetary vehicles to arrive at Mars on different dates
- Transport the capsule to Mars; before releasing the capsule, support it with power, telemetry data handling, and other necessary services
- Perform up to two interplanetary trajectory correction maneuvers as commanded from earth
- Insert the planetary vehicle into a specified Mars orbit
- Initiate the capsule-spacecraft separation sequence
- Execute orbital trim maneuvers, as commanded, before and after spacecraft and capsule have separated
- Update trajectory and time-dependent control sequences by ground command
- Measure and transmit engineering performance data to earth
- Execute the scientific mission as specified and transmit science information to earth either in real-time or by playback of stored data
- Relay capsule telemetry data to earth after the capsule has separated, during descent, and to the extent necessary after the capsule is operating on the Martian surface

The structures, electronics, instrumentation, and propulsion combine to form a spacecraft that, but for a limited number of maneuvers necessarily directed from earth, can complete its entire mission and return its planned complement of data automatically and without ground command. On the other hand, automatic sequences can be readily interrupted and modified from the ground and the entire scientific program will undoubtedly be directed closely from earth.

It was TRW's intent to contrive a spacecraft design that meets the Voyager requirements and performs the mission functions in a

Physical Characteristics

Standardization: Spacecraft hardware (less science) identical for all 1973-1979 missions

Science Payload Accommodation: Hypothetical 515 lb
Recommended 500 lb
JPL Mission Specification 400 lb

Capsule Accommodation: Both Case A (5,000 lb for all missions) and Case B (6,000 lb for 1973; 7,000 lb for 1975-1979)

System Reliability: 0.69 for a single spacecraft, 0.905 for at least one of two spacecraft (244 days interplanetary, 60 days Mars orbit)

Structure: 3 modules: equipment module, propulsion module, and planetary vehicle adapter; aluminum framing and honeycomb deck; semi-monocoque; meteoroid panels

Size of basic spacecraft: 14 ft, 1 in long; 19 ft, 10 in diameter

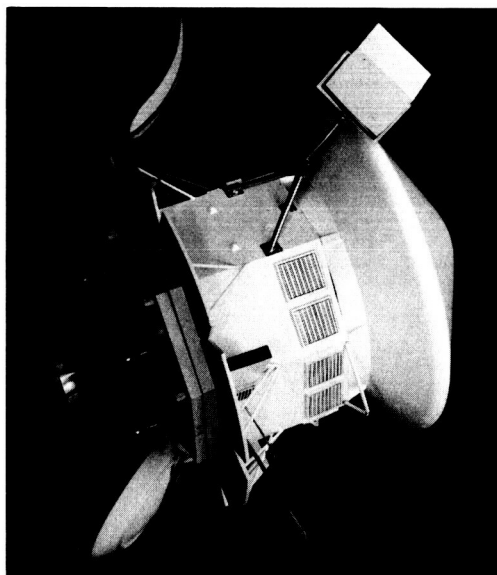
Appendage Mechanization: Planetary Scan Platform - double gimballed and limited range continuous deployment drive
High gain antenna - double gimballed
Medium gain antenna - single gimballed
Low gain antennas (2) - deployed to fixed position

Weight

	5,000	6,000	7,000
Capsule	5,444	5,513	5,516
Spacecraft (inert)	9,663	10,653	11,581
Usable propellant *			

	20,107	22,166	24,097
Total Planetary Vehicle			

* For minimum ΔV 1.95 Km/sec



Performance

Characteristics

TELEMETRY AND DATA STORAGE

4 tape recorders store 9×10^8 bits each
2 tape recorders store 7.5×10^6 bits each
Transmission in 32 6-bit biorthogonal code
Video transmitted adaptively between 51.2 and 3.2 kb/sec
Engineering data continuous at 512 bits/sec
Emergency Mode data at 8 bits/sec

Central telemetry data has plus two remote multiplex Planetary Scan Platform recorders. Flexible format selects programmed format or new formats.

S-BAND COMMUNICATIONS

High gain antenna: 34 db
3.2 degrees beamwidth
Medium gain antenna: 28 db
4 x 10 degrees beamwidth
Low gain antenna: 1 db

All digital communication 50-watt TWT's. Four rec antennas: high gain (9.5 gain; two low gain giving spherical coverage).

COMMAND

Rate: 8 bits/sec. Accepts up to 256 commands, processes serial data for 11 on-board destinations
Acquisition Time: 2 minutes maximum

Redundant bit synchronizer demodulate and synch from S-band radio.

GUIDANCE AND CONTROL

Limit Cycle: ± 0.5 degree per axis (coast phases)
 ± 0.25 degree per axis (photo-imaging and capsule separation)
Thrust Vector Control Accuracy: ± 0.43 degree per axis

Fully-stabilized, 3-axis nitrogen heated gas; sun references; redundant inert

COMPUTER AND SEQUENCER

Issues discrete command and serial messages for automatic on-board control. Computes pointing angles for high gain antenna and Planetary Scan Platform. Stores 512 20-bit words.

Primary sequencer plus counter meters and function generators, with decoder Planetary Scan Platform. Backup sequencer. Integrates accelerometer outputs during engine burn, gives cut-off

ELECTRIC POWER

Array Voltage: 37 to 50 VDC
Array Power: 836 watts at 1.62 AU
Bus Voltage: 50 VDC $\pm 1\%$
Individual DC/DC converters, 400 Hz to appendage drives, 819.2 KHz sync signal to subsystems

226 sq ft fixed solar array batteries (48 amp-hrs)

ELECTRICAL DISTRIBUTION AND PYROTECHNIC CONTROL

Distribution control unit feeds DC power to all subsystems. Pyrotechnic control unit operates all electro-explosives

Most harnesses removable

PROPULSION

Max. propellant load: 16,000 lbs
Propellants: N_2O_4 and Ap50 or MMH
Main Engine: Thrust 9850 lbs, I_{sp} 298 sec
1700 lbs 289 sec
Backup Engines: 400 lbs 292 sec
Total

Modified LMD main engine identical for all missions; automatic backup for Mars achieve reduced mission; gimballed and back-up engine duty-cycle-modulated for vector control

TEMPERATURE CONTROL

	Near Earth °F	Near Mars °F
Solar Array Temperatures (Average)		
Annular array	125	-7
Base array	246	86
Equipment panel	90	50
Planetary Scan Platform	84	35
Interior temperature		

Multilayer insulation; ins louvers, special surface f

FOLDOUT FRAME

BINARY SPECIFICATION

istics

Handling unit
or units on
Six tape
generator
or programs

15. Redundant
ivers, four
ft), medium
complete

rs and decoders
ze commands

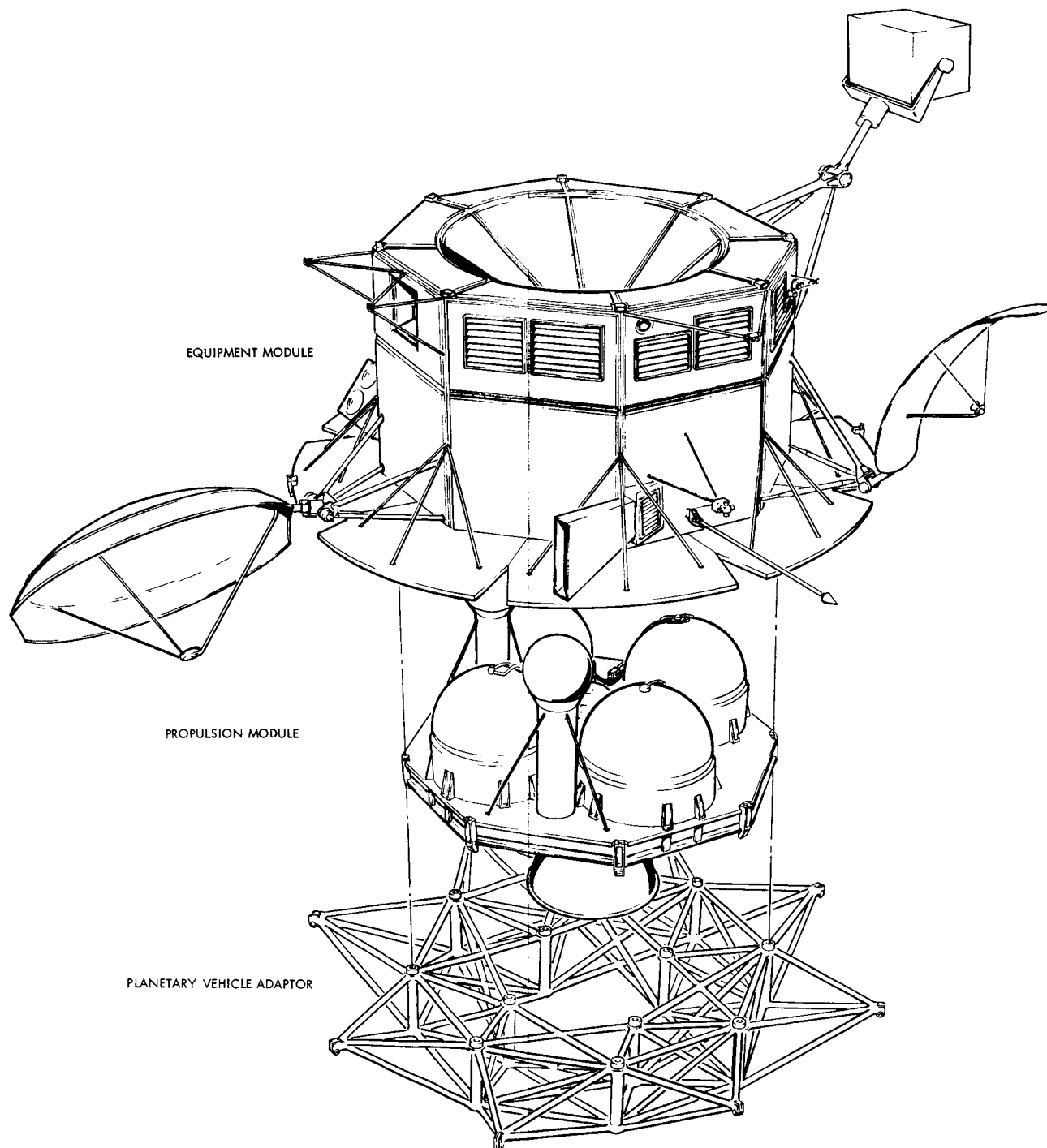
ontrol;
nd Canopus
tial reference

celero-
n
inates
ing
.

; three NiCd

ie with tankage
four C-1 engines as
orbit insertion to
main engine
engine
thrust

individually actuated
Inishes



FOLDOUT FRAME

Figure 2-1

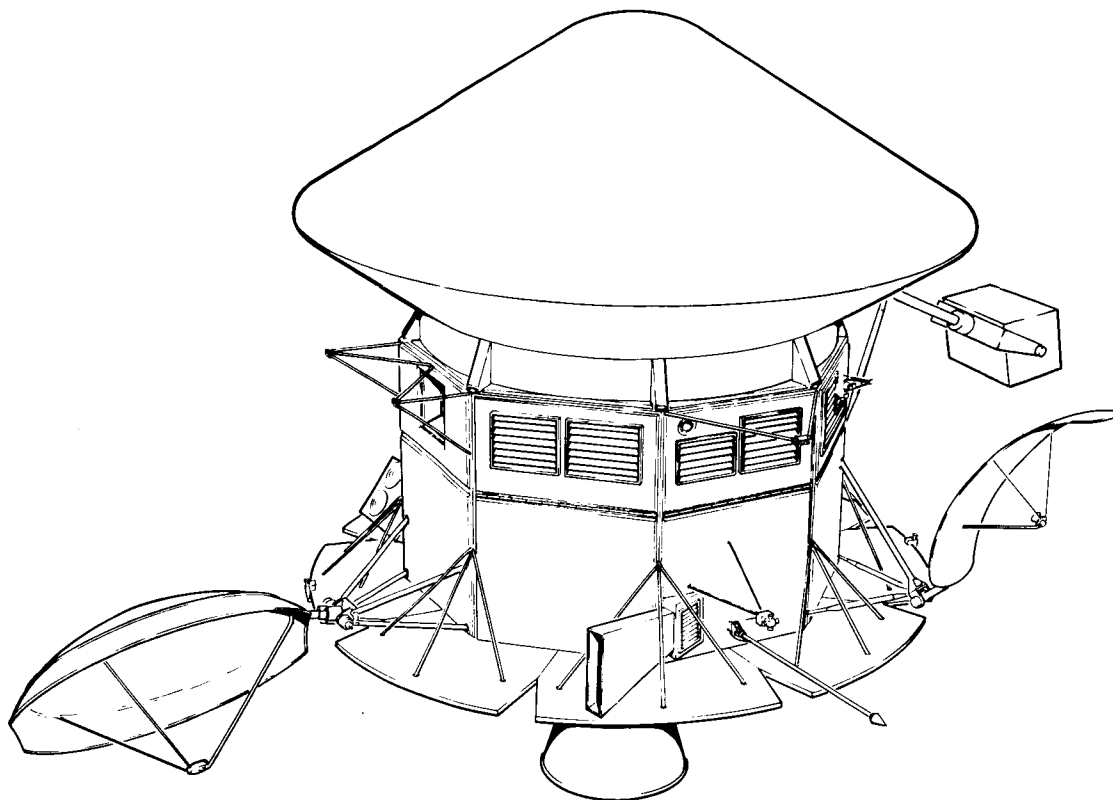


Figure 2-2

PLANETARY VEHICLE EXTERNAL VIEW with capsule as in cruise flight, shows the antennas, planetary scan platform, a sun-Canopus sensor with glint shield (one on each side of the spacecraft), attitude control gas jet, the louvered panels for temperature control, and the skirt of the modified LMD engine.

manner that satisfied all customer specifications, guidelines and goals. We believe that the intent is realized in the recommended design. The design is felt to offer significant performance margin with adjunct flexibility for its application so as to allow effective use of the Saturn V boost capability over the complete series of missions.

2.2 DESIGN PHILOSOPHY

This section presents some of the general aspects underlying the design approach followed to arrive at the recommended Voyager spacecraft. The central question of spacecraft performance capability and margin utilization is discussed as well as configuration modularity. Also covered are the general design criteria that structure the approach to configuration definition.

2.2.1 Spacecraft Performance Capability

A central question underlying the spacecraft design problem is the level of performance capability to be implemented. Basically, this question arises because of the unknown element represented by the exploration of Mars. Even such an unknown element does not lead to uncertainty in performance requirements, however, when capabilities are severely restricted; for only one mission at a time can be considered, and that within a limited special purpose framework. This was the case with Mars missions prior to the availability and adoption of the Saturn V as launch vehicle. For example, payload was severely restricted and the spacecraft design problem was dominated by minimum weight considerations. Similarly, design of the capsule and its payload was dominated by survival of hard landing, with ultra simplicity a prime necessity. Thus the vehicle design problem was well-defined, albeit a difficult one.

Use of the Saturn V has opened up a new level of capability for Mars exploration, and along with this the question of how best to allocate and apply such a capability. For example, for the 1973 mission, with a 30 day launch window, it is possible with the Saturn V to place a capsule of over 9000 pounds into a tight Mars orbit (1100 x 10,000 km) along with an apsidal rotation of 20 degrees, while complying with all trajectory and mission constraints. This is in contrast to the capsule weight requirements of 5000 pounds for Case A or 6000 pounds for Case B. The extra capability represents performance margin. It is sometimes convenient to measure such margin in terms of additional capsule weight, but this is not the only way such margin can be utilized. It can be translated into additional spacecraft weight for science or redundancy, increased rotation of the Mars orbit, additional orbit trimming, increased launch period, earlier arrival, etc.

The possibility of such margin raises the question of spacecraft performance capability to realize it and in particular the related aspect of sizing, or propellant capacity. This question is complex, and to determine spacecraft sizing for maximum performance is not straightforward for even a single mission, as the various ways to utilize



performance margin generally lead to individual optimum design points. In addition, the Mars exploration program is properly looked upon as an integrated, evolving series of missions. Coupled with this is the programmatic requirement to standardize hardware and minimize the total development effort required. These considerations further complicate the analysis of spacecraft sizing and margin optimization. However, the real element causing lack of tractability is the inability to define mission details at this time that affect the manner in which performance margin should be utilized. This stems directly from the unknown element for such an exploration program as mentioned at the outset.

In the face of this indeterminant element, a balanced approach to performance capability has been taken for the recommended spacecraft. Rather than commit the realizable margin to any specific approach, an attempt has been made to achieve significant margin capability while retaining flexibility in its utilization. This takes the form of extra strength capability for accommodating capsule weights up to 8000 pounds, a usable propellant capacity of up to 16,000 pounds, and configuration features that allow spacecraft upgrading to support additional science with higher power and communications capability.

This flexibility naturally results in a penalty compared with performance that could be realized for a specific commitment of margin utilization. However, at the same time it allows variation from mission to mission in the way that margin can be utilized. Actually, the penalty is felt to be modest in terms of the return. For example, the accommodation of the additional capsule weight over that for Case B entails only an additional 32 pounds. Also, the realized design for the recommended spacecraft retains essentially all the desired configuration attributes of modularity, equipment accessibility, simple assembly, etc., while also accommodating the upgradability feature.

Further discussion and analytical results dealing with this question of spacecraft performance margin and sizing are presented in Volume 6. Also presented in Volume 6 are details of the upgraded capability.

2.2.2 Modularity

Another important aspect of the TRW design approach is the feature of modularity. This arises from the importance of the assembly and test process in spacecraft implementation. A critical feature in this is the concentration of operations and the associated equipment and personnel in the relatively restricted area in close proximity to the spacecraft being processed. A considerable amelioration of this situation can be achieved, however, by having the article made up of modules that can be processed separately at successive levels of assembly. The design approach for the recommended configuration calls for such modularity.

Modularity applies to the black box level, but only begins to be important to the overall configuration in terms of major assemblies. Such modularity applies, for example, to the mounting of subsystem equipment on a single panel such that this panel assembly can be integrated and tested separately prior to integration into the spacecraft. This approach was followed for the recommended design; however, the most important modularity feature of the TRW design is the breakdown of the spacecraft into two major hardware modules. This breakdown into a propulsion module and an equipment module represents a primary feature of the configuration design approach.

The identification of the propulsion elements as suitable for making up a major spacecraft module follows from their size and because of the large amount of special testing and servicing required by this subsystem. The remaining elements then generally interact with each other sufficiently to call for their integration into a single (equipment) module; while at the same time these elements have only a modest amount of interaction with propulsion. This latter is helpful in implementing the modularity; nevertheless the achievement of an efficient, simple modularity configuration has represented a major aspect of the configuration design problem for the study.

2.2.3 Configuration Design Criteria and Characteristics

The design philosophy underlying spacecraft configuration definition is structured and made explicit by configuration design criteria. In particular this covers desired qualities and features concerned with



spacecraft geometry, arrangement of equipment, and other configuration-oriented aspects that should be embodied in the design. Some of these objectives are naturally competing, and in some instances project guidelines compromise such goals. Wherever possible, however, they have been incorporated into the recommended configuration. These considerations are discussed in this section.

A few of the more important of these items, in addition to performance capability and modularity previously discussed, are highlighted first before giving a more complete presentation.

- The control of center of mass location so as to maintain stability for propellant sloshing even under the condition of capsule removal prior to Mars orbit insertion. This condition is associated with the possible desire to obtain tighter orbits and additional maneuvering for photo-imaging effectiveness, if it is known before Mars encounter that the capsule has failed.
- Use of redundant reaction control so that each of the redundant systems has the complete capability for carrying out the entire mission, even in event of leakage of the other system with consequent depletion of its gas as a disturbing torque.
- Use of direct load paths to achieve a lightweight structure. This lead to a change from the outrigger approach of Task B to an underlying adapter.
- In addition to accessibility through use of a propulsion and equipment module, there is good accessibility to equipment by mounting on hinged panels. At the same time this allows physical entrance by personnel to the spacecraft interior for servicing, removal of equipment, etc.
- Easy installation of insulation by minimizing need for closeouts and providing clean external structure.
- Fixed solar array for enhanced reliability.
- Use of redundant guidance and control sensors, including oppositely located Canopus sensors. This provides for unobstructed planetary scan

platform viewing in either north or south inclined orbits by virtue of the two stable roll orientations. At the same time this allows a single location for the capsule relay link antenna for such orbits.

2.2.3.1 Structural and Mechanical Criteria

- a) Direct load paths are used to minimize the length of the load paths and to minimize the amount of structure subject to bending due to induced loads, reducing structural weight, and providing a clean design.
- b) The structural elements are designed for multiple functions wherever possible. For example, structural shear panels are used for equipment mounting, micrometeoroid protection, and some thermal environment control. The mechanical interface between the spacecraft and the adapter is both a field and separation joint.
- c) Masses are located as close as possible to the thrust axis and to the center of mass to minimize induced bending and the principal mass moments of inertia.
- d) The geometry of structural members and the materials from which they will be fabricated have been selected to maximize strength-to-weight ratios.
- e) The spacecraft surface area, volume, and the number and size of mechanical joints have been minimized to reduce weight consistent with the growth potential attributes included in the spacecraft system design.
- f) The number of deployed and articulated appendages are minimized while those incorporated are symmetrically located about the spacecraft for proper mass balance and to minimize the unbalanced solar torques.
- g) Spacecraft components are located close to the main structural elements to reduce secondary structure weight.
- h) Deployable and articulated appendages are stowed during launch to reduce structural requirements.
- i) Where multiple panels or members are used such as equipment-mounting panel doors, solar array panels, micrometeoroid protection



panels, and basic structural elements, component design has been identical in order to simplify fabrication and logistics and thereby minimize cost.

2.2.3.2 Control of Mass Distribution

- a) The mass moments of inertia are minimized to reduce reaction control gas requirements.
- b) Consummables are located symmetrically about the thrust axis, to the maximum extent possible, to prevent center of mass shift and rotation of principal axes during depletion of consummables.
- c) Provisions have been made for the longitudinal and lateral center of mass shift that occurs when the capsule is separated, to insure adequate spacecraft attitude control.
- d) The principal mass moment of inertia axis and the high-gain antenna and PSP axis are located as close together as possible to minimize center of mass shift and rotation of the principal axes as the high-gain antenna and the PSP are deployed initially and gimballed during the mission.
- e) In order to maintain a stable propellant sloshing mode without the use of tank baffles, the planetary vehicle and spacecraft center of mass locations are maintained forward of the centerline of the propellant tanks even under "worst case" conditions, i.e., with the tanks half full and the capsule removed.

2.2.3.3 Equipment Mounting and Accommodations

- a) The equipment compartment is configured as a complete module for ease of integration and test.
- b) The equipment-mounting door panels and the equipment compartment are sized to provide for growth in equipment mounting requirements, including provisions for an accompanying increase in thermal radiation requirements.
- c) Adequate space and mounting provisions are made for cable routing. Highly interconnected components are located together and the power amplifiers are located adjacent to the associated antennas to reduce cable lengths.

- d) All internally housed electronic equipment items are mounted on equipment door panels in order to provide easy access to the equipment for installation, test, and maintenance.
- e) All connections to articulated or deployed items are minimized, and slip rings and gimbals with continuous rotation in one direction are avoided.
- f) The number of equipment compartments is minimal in order to minimize the number of junction boxes and connections.

2.2.3.4 Propulsion Installation

- a) The propulsion system, including the propellant feed subsystem and the associated support structure, is modularized independently of the rest of the spacecraft for easy assembly and test in parallel with integration of the remainder of the spacecraft.
- b) The spacecraft center of mass is to remain on the thrust axis during propellant consumption to allow stable attitude control.
- c) The engine is located so as to provide adequate separation (at least 32 inches) between the spacecraft center of mass and the engine gimbal point for adequate thrust vector control authority.
- d) The thrust levels are constrained to minimize acceleration forces consistent with other propulsion design characteristics.
- e) Spherical tankage is utilized and pressure levels are selected for minimum weight consistent with other propellant feed requirements.
- f) Special attention is given to tankage support to minimize stress concentrations.
- g) Propellant mixture ratios are used that permit identical tank sizes for both oxidizer and fuel.

2.2.3.5 Thermal Control Configuration Characteristics

- a) An insulated equipment compartment concept is utilized with louver controlled heat flow.
- b) Adequate louvered thermal radiation area is provided with a maximum view of space for efficient radiation



- c) The total spacecraft external surface area is minimized to reduce insulation requirements, thermal leakage, and weight.
- d) The thermal radiation louvered areas are shaded from the sun and other heat sources as much as possible for efficient performance.
- e) Maximum thermal interaction between spacecraft internal components, including the propellant and reaction control tankage, is provided to stabilize temperature levels.
- f) Heat-producing components are uniformly distributed about the spacecraft equipment panels to the maximum extent consistent with cabling, packaging, and mass properties considerations to provide uniform temperature conditions.
- g) Low-conductance structural attachments are utilized to limit the heat input by conduction from the propulsion system as a result of engine firings. High temperature insulation is utilized on the spacecraft surfaces in proximity to the engine head end as required to prevent overheating and damage to the insulation.
- h) The thermal radiation characteristics of the appendages are controlled to minimize solar torque and maintain balance. The concave surface of the high-gain antenna is coated black in order to minimize spacecraft windmilling caused by the solar pressure.

2.2.3.6 Guidance and Control Installation Criteria

- a) For high reliability, completely redundant high-pressure gaseous nitrogen systems are utilized for generating reaction control torques during cruise. Twelve reaction control nozzles in redundant pairs are provided about each control axis in order to produce pure torques and to achieve high reliability.
- b) The lengths of all lines between valves and nozzles are equalized to balance line losses, and the nozzles are located at a maximum distance from the spacecraft center of mass to maximize control moment arms.
- c) The gas impingement of the reaction control jets on the planetary vehicle is minimized for greater control efficiency.

- d) To minimize cross coupling in the event of a thruster failure during orbital operation, the roll thrust nozzles are located in the plane of the spacecraft center of mass at burnout with the capsule removed.
- e) Guidance and control sensors are redundant and located for unobstructed viewing angles and for maximum protection from exhaust effects.
- f) For uniform control characteristics, the pitch and yaw mass moments of inertia are equalized to the maximum extent, as are the pitch and yaw control torques.
- g) The positions of the control axes are located on or as close as possible to the principal moment of inertia axes in order to minimize cross coupling.
- h) For uniform control characteristics, the center of mass travel is minimized due to appendage articulations and depletion of consummables.

2.2.3.7 Electrical Power Configuration Criteria

- a) Solar power is employed utilizing nondeployable fixed solar array panels for minimum weight and high reliability.
- b) For maximum utility and efficiency during normal cruise attitudes, there is no solar cell shadowing and the back side of the solar array has a maximum thermal radiation view of space wherever possible.
- c) Where possible, the aft side of the solar array is left uninsulated for high efficiency.

2.3 DESIGN FEATURES

As shown in Section 2.1, the recommended spacecraft configuration is composed of three elements, the equipment module, the propulsion module, and the adapter. The adapter acts as the interstage between the launch vehicle and a planetary vehicle; that is, between the aft end of the propulsion module and the shroud, where it attaches by means of a field joint. The propulsion module, constituting the aft part of the spacecraft, contains the engine and its associated feed system and propellant and pressurization tanks. The equipment module is located above and



around the propulsion module and is attached to it at a transverse field joint. The capsule is placed at the forward end of the equipment module, attached at an eight-point field joint.

The inboard profile of the spacecraft assembly, i.e., the mated equipment and propulsion modules, is shown in Figure 2-3. The spacecraft weight analysis is given in Table 2-1 with a detailed discussion of mass properties data presented in Appendix F. As shown in the summary table, the weights of the spacecraft remain constant for all three capsules defined by Case A and Case B, except for propellants and attitude control gas. The arrangement of all equipment in the spacecraft is shown in Figure 2-4.

2.3.1 Equipment Module

The electronics for all subsystems of the spacecraft except propulsion are grouped together on the inside of the eight panels forming the octagonal circumference of the module (see Figure 2-5). In addition all appendages are mounted on the surfaces of this module. The compartment is completely enclosed by thermal insulation except for controlled radiation areas where thermal louvers are mounted.

Axial loads from the capsule are carried in the corner column members of the module directly to mating members of the propulsion module. The column members are stabilized by the eight panels, which carry shear loads from the capsule. Shear and torsion capability are provided at the forward end of the spacecraft by the dimpled upper micrometeoroid cover which closes out the top of the spacecraft. It is fabricated from the same type of material as the side panels.

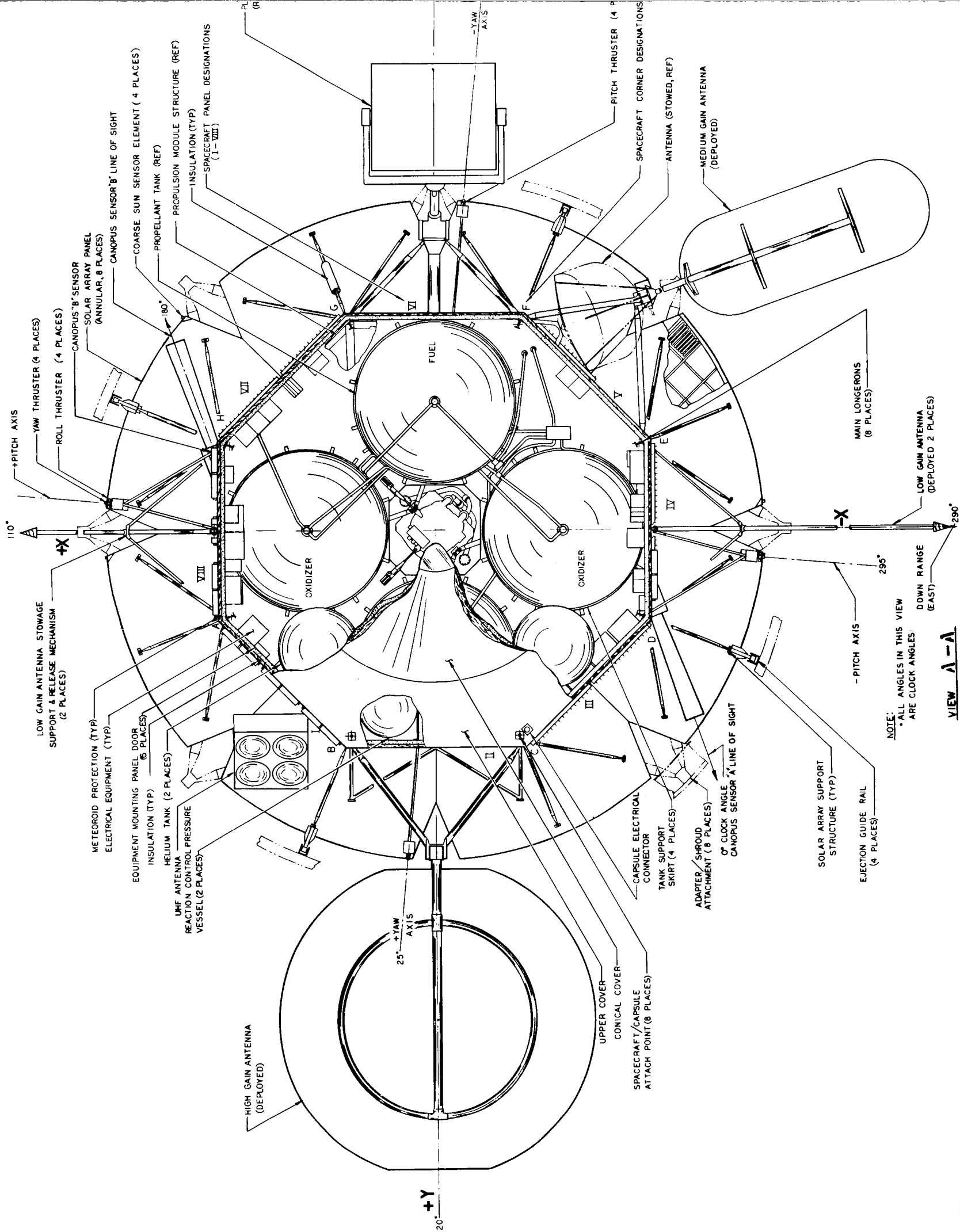
2.3.1.1 Equipment Mounting Panels

The electronic equipment is mounted to the panel doors as illustrated in Figure 2-6. The panels serve as access doors at the forward end of the spacecraft. A hinge on the aft side of each door allows it to rotate outboard and down for access to the interior and to the equipment installed on the panel. Each panel provides approximately 15 square feet of mounting surface.

Table 2-1. Summary Weight Analysis: Recommended Configuration

	Weight (lb)		
	Case A	Case B	
<u>Structure</u>	<u>913.0</u>	<u>913.0</u>	<u>913.0</u>
Basic Shell	211.4	211.4	211.4
Meteoroid Protection	562.6	562.6	562.6
Supports	110.4	110.4	110.4
Separation Mechanism	28.6	28.6	28.6
<u>Propulsion</u>	<u>1,568.7</u>	<u>1,568.7</u>	<u>1,568.7</u>
Engine (Main)	408.5	408.5	408.5
Fuel System	232.1	232.1	232.1
Oxidizer System	237.1	237.1	237.1
Pressurization	348.8	348.8	348.8
Supports	308.2	308.2	308.2
Secondary Engine Installation	34.0	34.0	34.0
<u>Equipment and Instrumentation</u>	<u>2,276.0</u>	<u>2,340.6</u>	<u>2,340.6</u>
Structure	180.0	180.0	180.0
Guidance, Control and Navigation	168.2	168.2	168.2
Instrumentation	397.7	397.7	397.7
Electric Power	443.5	443.5	443.5
Electric Networks	298.0	298.0	298.0
Temperature Control System	238.5	238.5	238.5
Attitude Control System	150.1	214.7	214.7
Science Equipment	400.0*	400.0*	400.0*
<u>Balance Weights</u>	<u>15.0</u>	<u>15.0</u>	<u>15.0</u>
<u>Contingency</u>	<u>216.1</u>	<u>217.7</u>	<u>217.7</u>
<u>Total Dry Spacecraft</u>	<u>4,988.8</u>	<u>5,055.0</u>	<u>5,055.0</u>
<u>Residuals</u>	<u>455.1</u>	<u>458.2</u>	<u>461.2</u>
Propellant	412.9	416.0	419.0
Pressurant	42.2	42.2	42.2
<u>Total Inert Spacecraft</u>	<u>5,443.9</u>	<u>5,513.2</u>	<u>5,516.2</u>
<u>Usable Propellant</u>	<u>9,663.1</u>	<u>10,652.5</u>	<u>11,580.5</u>
<u>Total Spacecraft at Liftoff</u>	<u>15,107.0</u>	<u>16,165.7</u>	<u>17,096.7</u>
<u>Capsule</u>	<u>5,000.0</u>	<u>6,000.0</u>	<u>7,000.0</u>
<u>Total Planetary Vehicle at Liftoff</u>	<u>20,107.0</u>	<u>22,165.7</u>	<u>24,096.7</u>

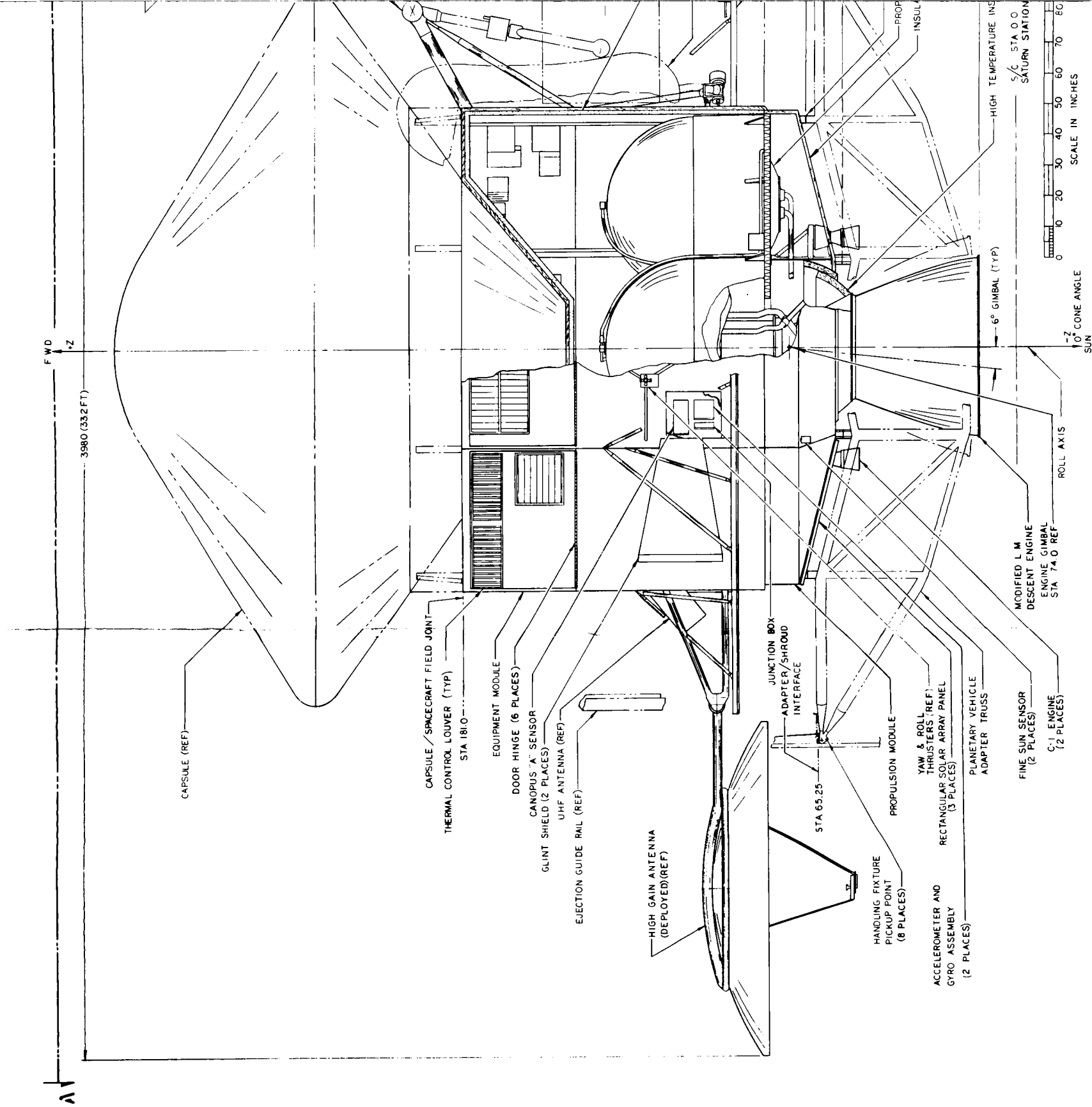
* JPL specification payload weight allocations. See Volume 5, Appendix G, for weight estimates of science payloads described in that volume.



FOLDOUT FRAME

FOLDOUT FRAME

2-17A

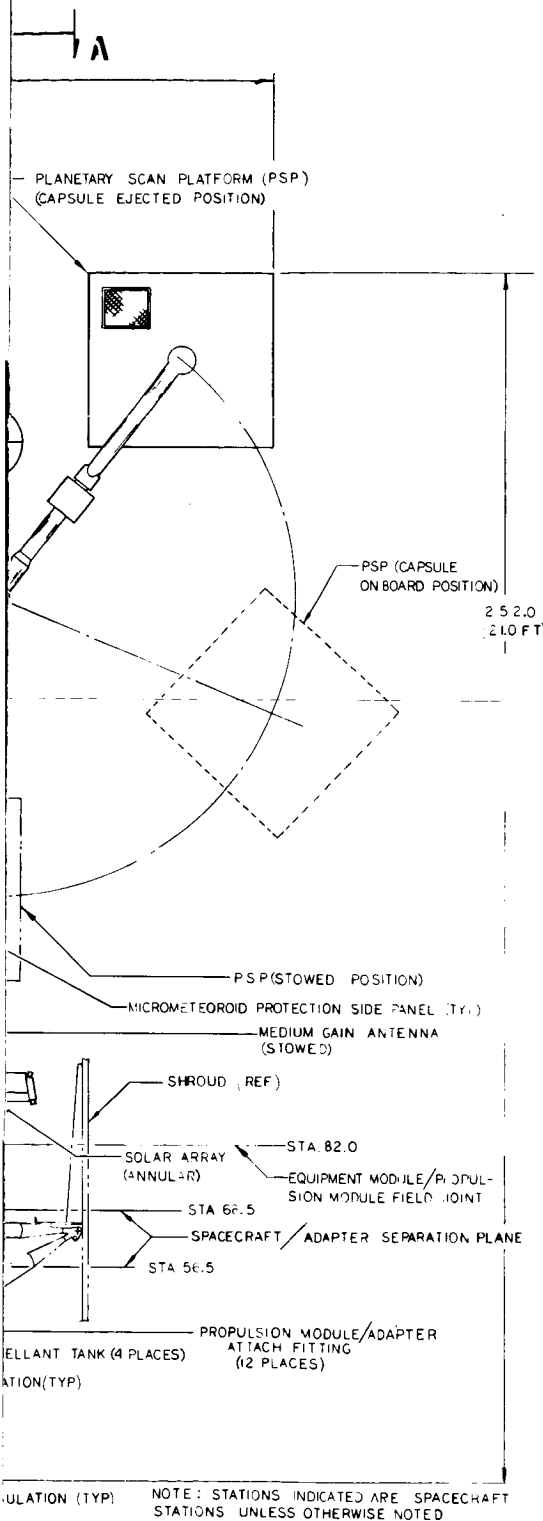


FOLDOUT FRAME.

FOLDOUT FRAME

2-170

2-17c



32595

90 100

Figure 2-3

SPACECRAFT INBOARD PROFILE is shown with appendages deployed and the capsule in place

FOLDOUT FRAME

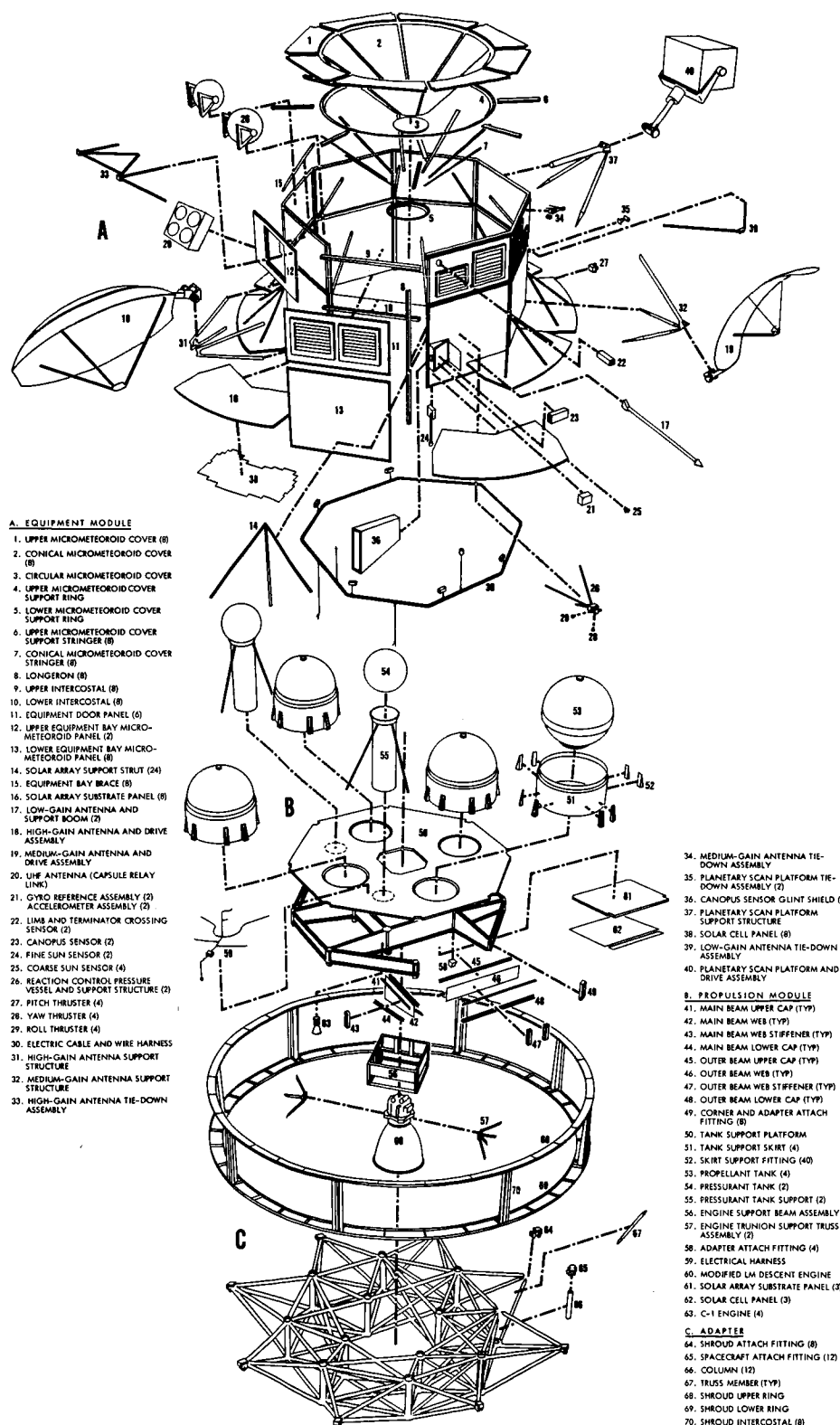


Figure 2-4
SPACECRAFT EXPLODED VIEW shows location of equipment items and structural breakdown.

AD 7-168

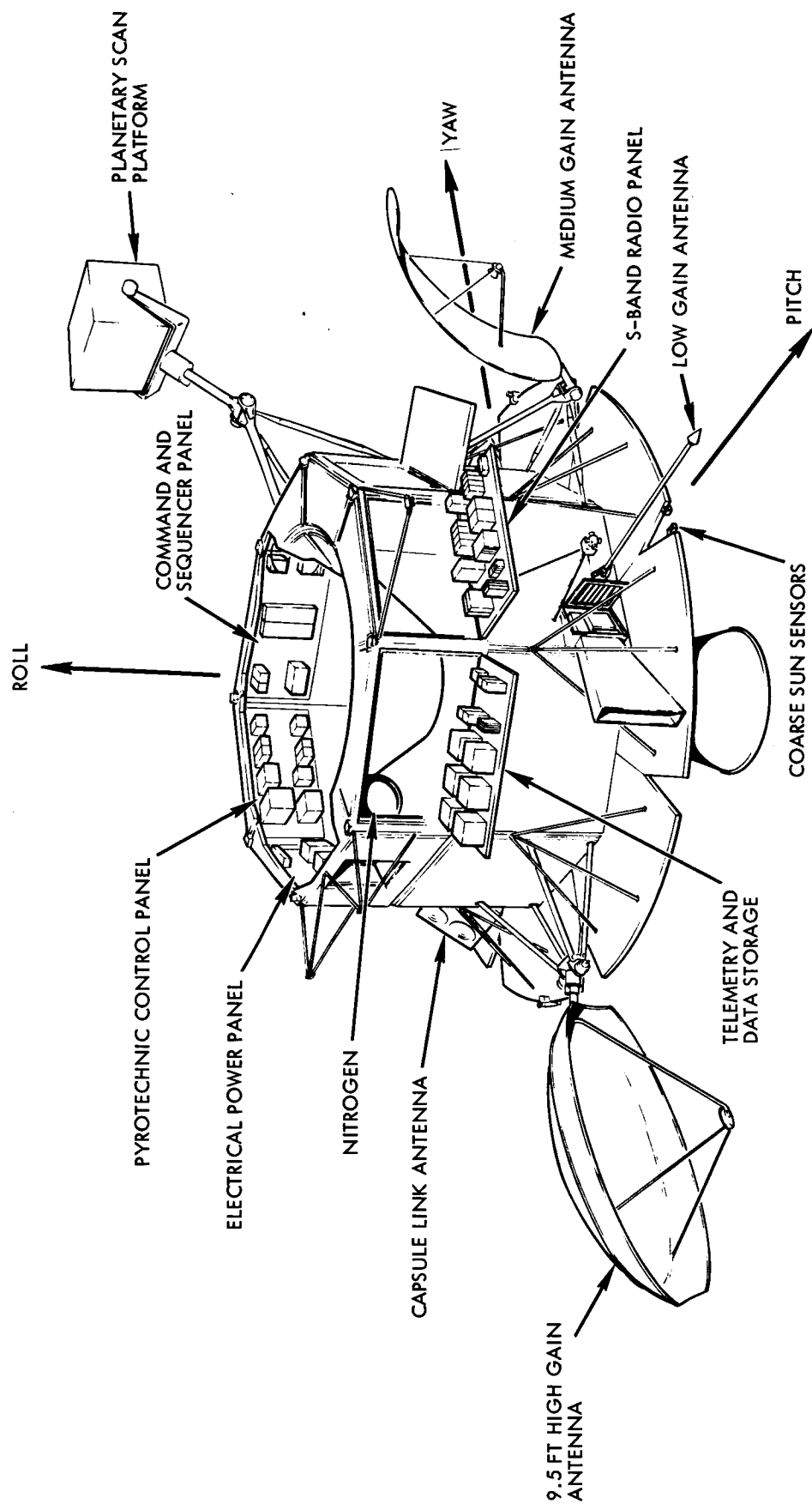


Figure 2-5
ELECTRONIC EQUIPMENT LOCATIONS for both internal and external components of the equipment module are shown. Equipment mounting panels are shown open as for servicing and are normally closed.

EQUIPMENT LIST

S-BAND RADIO SUBSYSTEM

1. S-BAND RECEIVER (4)
2. PRE-AMPLIFIER (4)
3. DIPLEXER (4)
4. CIRCULATOR SWITCH ASSY
5. BASEBAND ASSY
6. TRANSMITTER SELECTOR
7. RECEIVER SELECTOR
8. LOW GAIN ANTENNA SELECTOR
9. MODULATOR EXCITER (2)
10. ONE WATT TRANSMITTER
11. POWER AMPLIFIER & POWER SUPPLY (2)
12. ANTENNA DRIVE ELECTRONICS

COMMAND & SEQUENCER SUBSYSTEM

13. PRIMARY COMPUTER & SEQUENCER
14. BACKUP COMPUTER & SEQUENCER
15. COMMAND UNIT

TELEMETRY & DATA STORAGE SUBSYSTEM

16. TELEMETRY DATA HANDLING UNIT
17. SPACECRAFT & CAPSULE TV RECORDER (4)
18. ENGINEERING & SCIENCE RECORDER (2)

CAPSULE RADIO LINK SUBSYSTEM

19. RELAY LINK RECEIVER (2)
20. RELAY LINK DEMODULATOR (2)

ELECTRICAL POWER SUBSYSTEM

21. BATTERY (3)
22. POWER CONTROL UNIT
23. SHUNT ASSY
24. 400 H₂ INVERTER
25. DC/DC CONVERTER (8)

ELECTRICAL DISTRIBUTION & PYROTECHNIC CONTROL SUBSYSTEM

26. DISTRIBUTION CONTROL UNIT
27. PYROTECHNIC CONTROL UNIT
28. JUNCTION BOX (4)
29. JUNCTION BOX, SCIENCE

GUIDANCE & CONTROL SUBSYSTEM

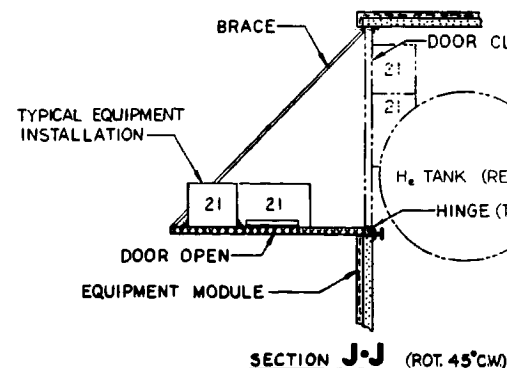
30. GUIDANCE & CONTROL ELECTRONICS ASSY
31. LIMB & TERMINATOR CROSSING SENSOR (2)

EQUIPMENT PANEL NO. VII
(COMMAND & SEQUENCER,
ELECTRICAL POWER,
GUIDANCE & CONTROL,
CAPSULE RADIO LINK)

EQUIPMENT PANEL NO. VIII
(ELECTRICAL POWER,
ELECTRICAL DISTRIBUTION &
PYROTECHNIC CONTROL)

DOOR LAND
(TYP)

EQUIPMENT PANEL NO. I
(ELECTRICAL POWER,
GUIDANCE & CONTROL)



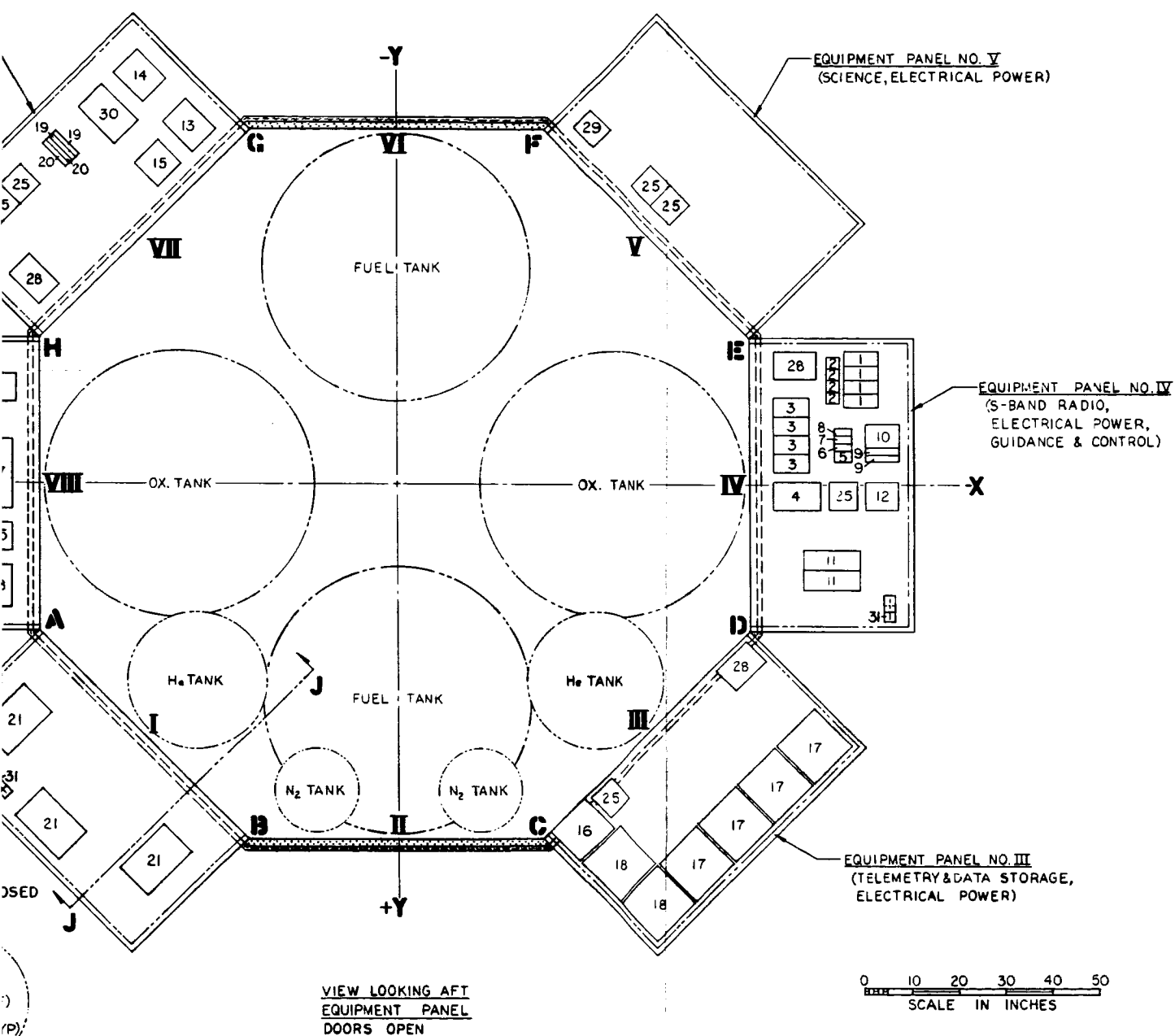


Figure 2-6
EQUIPMENT INSTALLATION LAYOUT
on six hinged mounting panels is
shown. Equipment items for each sub-
system are generally grouped together.



The components of individual subsystems are in general located together on a single panel insofar as possible in keeping with thermal distribution and mass balance requirements. Particular items such as the batteries, the power control units, and the power amplifiers are isolated on their respective panels to allow for adequate dissipation of heat. All components are mounted for efficient contact of their base areas with the panels.

2.3.1.2 Solar Cell Arrays

Two sets of solar cell arrays are mounted on the aft side of the spacecraft. The first is an annular array 238 inches in outside diameter. The second is a set of three rectangular panels attached to the propulsion module structure. Because the latter are directly aft of the propulsion subsystem, a layer of thermal insulation is placed between the panels and the structure of the module.

The eight panels of the annular array are each mounted to a substrate of beaded aluminum alloy sheet attached to a gridwork of aluminum channels. The inner edge of the array is mounted by high thermal impedance fittings and the outer edge is stabilized by a set of thermally insulated struts.

The panels are identical, to reduce fabrication costs. Therefore a cutout required in one panel is placed in each. However, the only large cutout required is at the periphery of the array to facilitate movement of the high-gain antenna. The same cutout on the appropriate panel is also utilized for the medium-gain antenna. Except for the four small coarse sun sensors, which are mounted directly to the solar array structure, no equipment is mounted to the array.

2.3.1.3 Antennas

S-band communications can use any of four antennas: a high-gain, a medium-gain, and two low-gain units, all of which are shown on Figure 2-7. The high-gain antenna is a double-gimballed parabolic reflector with a 9.5-foot diameter circular aperture. It is supported by a four-member tubular truss network. During launch, the antenna is stowed forward of the annular solar array, braced by stabilized struts whose ends bear on an antenna ring frame at two points. A single-point

stud built into the antenna feed housing engages a release fitting on the spacecraft. The antenna is structurally capable of holding any of its operating orientations during engine firing, including the high thrust level with the capsule removed.

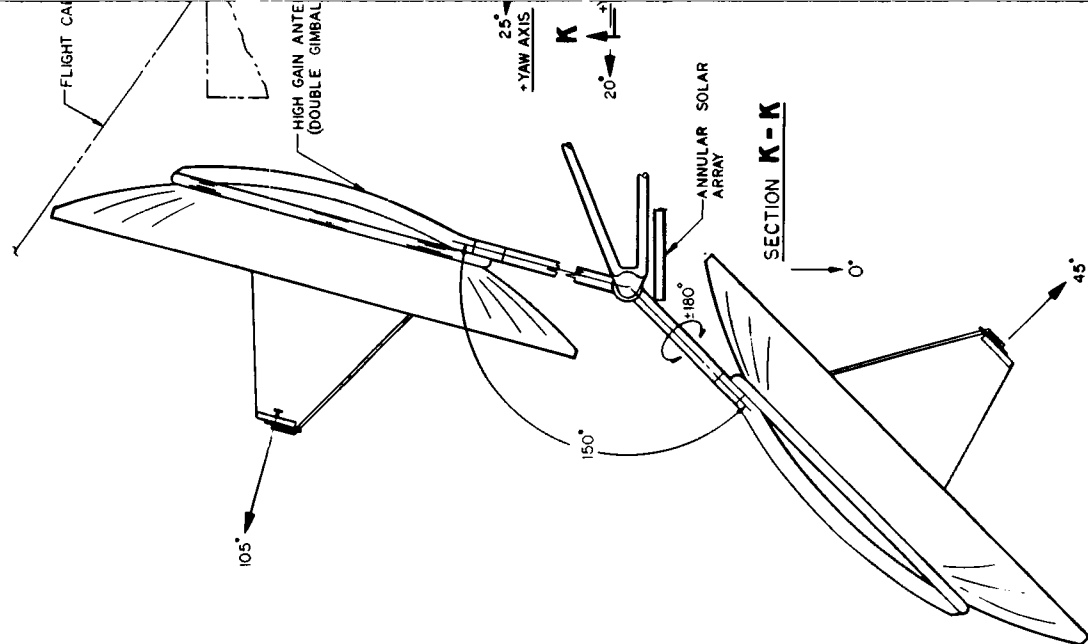
The medium-gain antenna is a single-gimballed parabolic reflector with a 36 x 94-inch rectangular aperture (with rounded ends). The antenna is supported by a tubular truss assembly; in its stowed position, it is stabilized by a bracket which supports the release mechanism between two bearing pads. For continuous earth coverage during orbit around Mars, the medium-gain antenna gimbal axis is aligned parallel to the Canopus sensors viewing angle (0 deg clock and 90 deg cone angles), as shown in Figure 2-8. The antenna is also structurally capable of holding any of its operating positions during engine firing.

The low-gain antennas are boom-mounted conical helices mounted on opposite sides of the spacecraft to provide omnidirectional coverage. In their stowed positions, the antennas are next to RF-transparent windows in the shroud for ground communication when the spacecraft is on-stand.

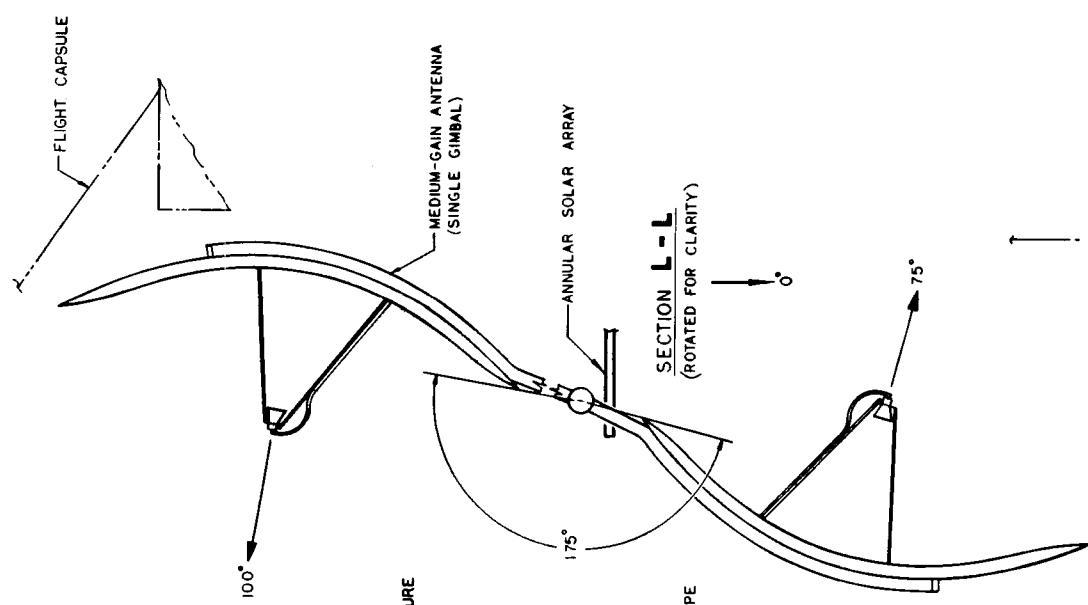
The UHF capsule relay link antenna is a quad-spiral array forward of the annular solar array, bracket-mounted to the equipment module. Its look angle is 20 degrees in clock angle and 140 degrees in cone angle.

2.3.1.4 Guidance and Control

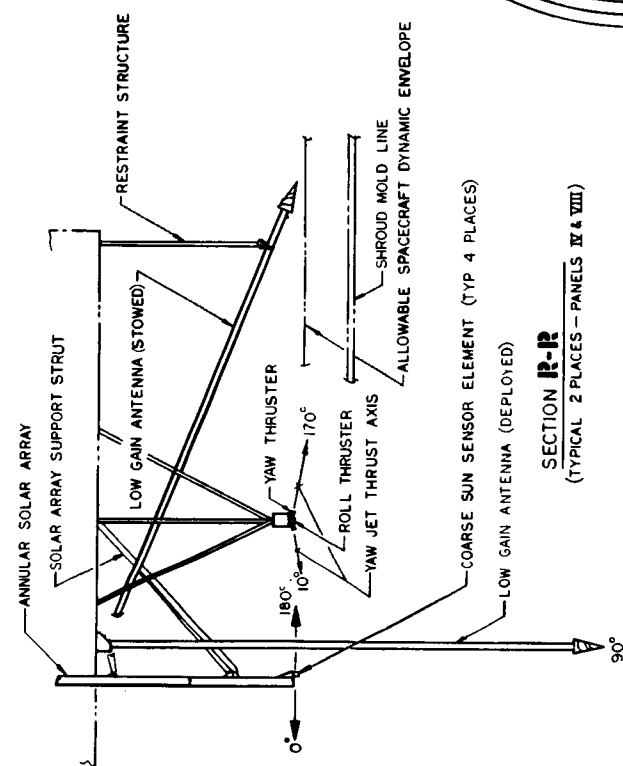
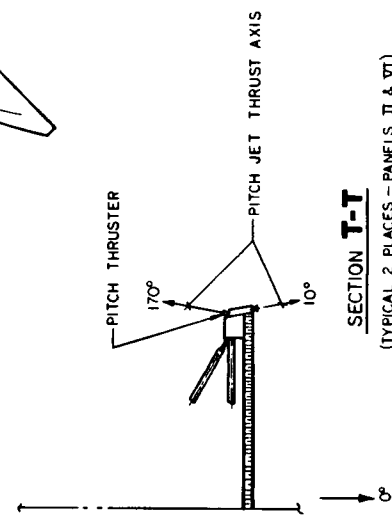
Two redundant sets of spatial reference equipment are located in identical installations on opposite sides of the spacecraft. Each location supports a Canopus sensor, a gyro assembly, an accelerometer assembly, and a fine sun sensor. As shown in Figure 2-9 the Canopus sensor at corner C is designated the Canopus "A" sensor, while that at corner G is called the "B" sensor. The "B" sensor is provided for redundancy and versatility in orbit. By definition, the direction of the Canopus sensor line of sight from the spacecraft always establishes the 0 degree clock angle position. To eliminate the possibility of stray or reflected light from impinging on the Canopus sensor apertures, internally baffled glint shields are installed as shown on the drawings.



SECTION N-N
TYPICAL FOR BOTH
LOCATIONS - AT PAY



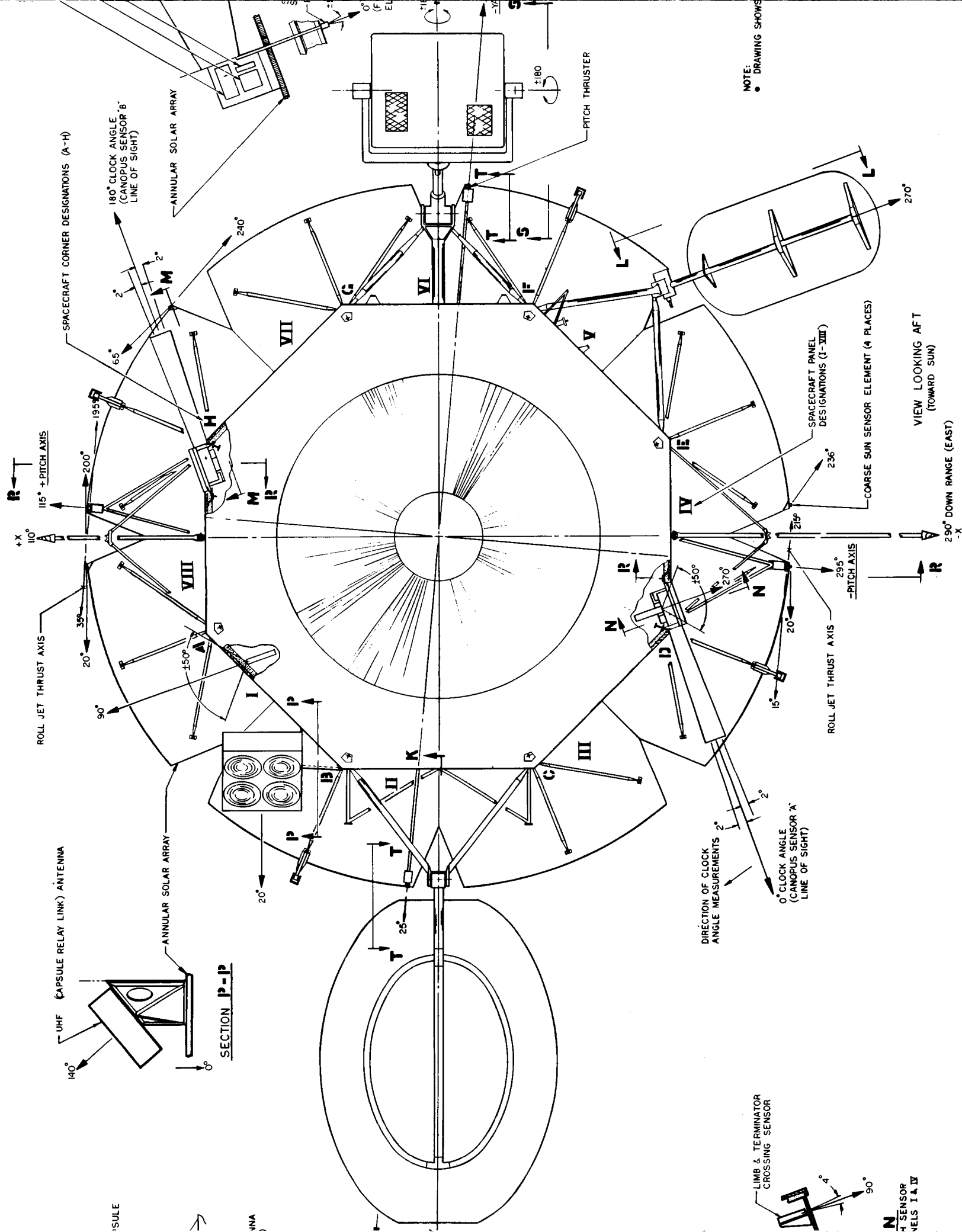
SECTION T-T
(TYPICAL 2 PLACES - PANELS II & VI)



SECTION B-B
(TYPICAL 2 PLACES - PANELS IV & VIII)

NOTES:

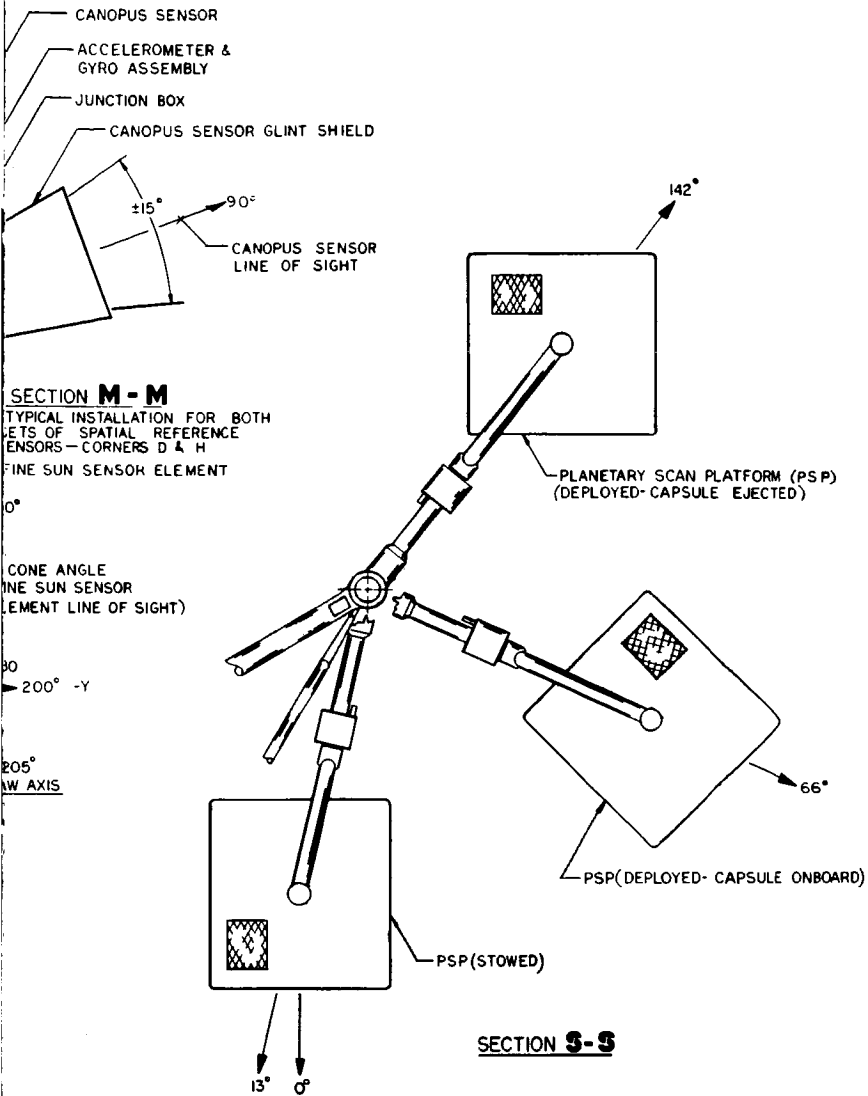
1. ALL ANGLES SHOWN IN ALL SECTIONS DEFINE THE CONE ANGLE AXIS FOR EACH SENSOR.
2. 0° CONE ANGLE IS DEFINED AS A LINE THRU THE SPACECRAFT POINTING AFT AT THE SUN.
3. ALL ANGLES IN MAIN VIEW DEFINE THE CLOCK ANGLE FOR EACH SENSOR AS MEASURED CLOCKWISE FROM THE LINE OF SIGHT OF THE CANOPUS 'A' SENSOR, LOOKING AFT.
4. 0° CLOCK ANGLE IS DEFINED AS THE LINE OF SIGHT OF THE CANOPUS 'A' SENSOR.



FOLDOUT FRAME

2-25B

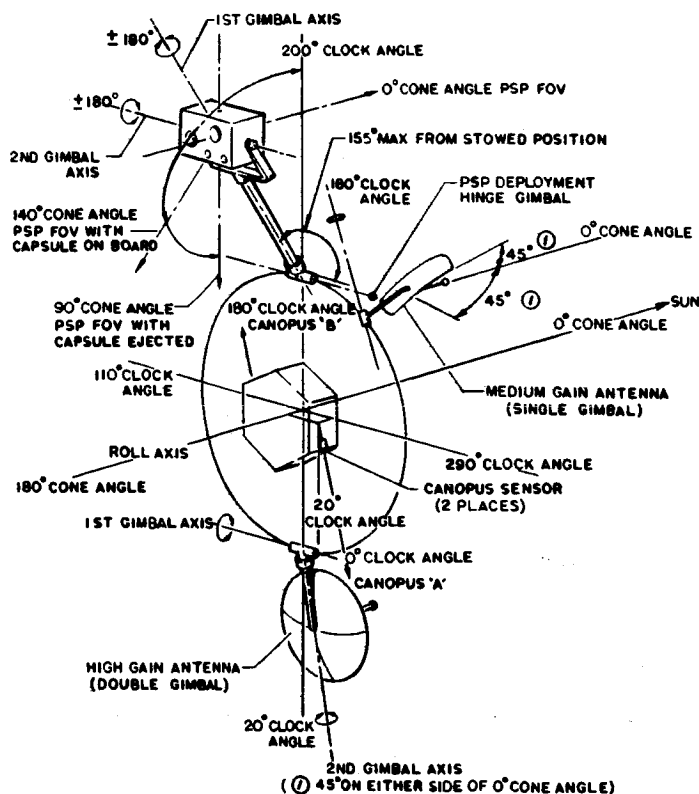
FOLDOUT FRAME 225C



SECTIONS K-K THRU S-S.

0 10 20 30 40 50
SCALE IN INCHES

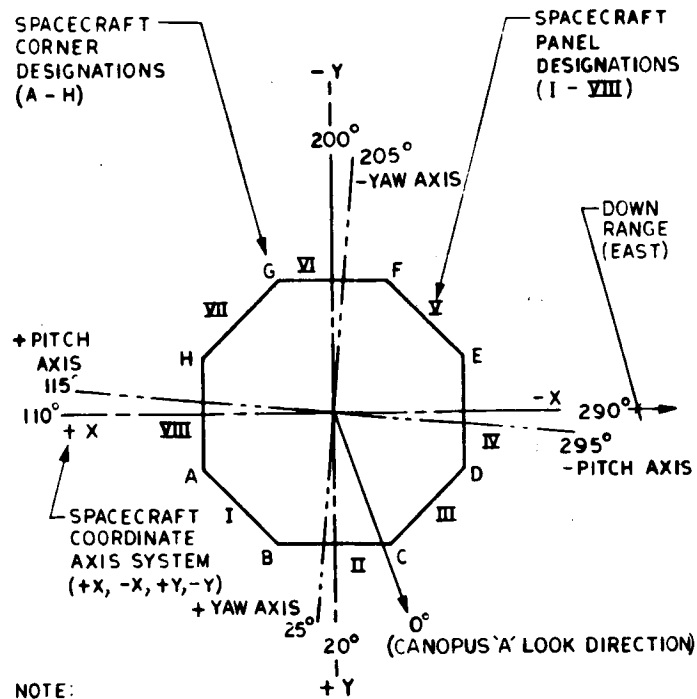
Figure 2-7
SENSOR AND APPENDAGE GEOMETRY
is shown, including antennas, planetary scan platform, guidance and control sensors, and reaction control jets.



AD7-149B

Two completely redundant high-pressure gaseous-nitrogen reaction control systems are provided, each one capable of performing the entire mission, even with leakage in the other system that causes gas usage in the non-failed one. Six pairs of thrusters (redundant pairs about each control axis) are angled 10 degrees outboard to prevent impingement of gas on the capsule, the annular solar array, or other portions of the spacecraft.

All of the scientific experiments for the recommended spacecraft are installed in the double-gimballed planetary scan platform. During



NOTE:

- SPACECRAFT PLAN VIEW LOOKING AFT (AT SUN DURING CRUISE).
- THE PLUS ROLL AXIS (+Z) IS OUT OF THE PAPER TOWARD THE VIEWER (FORWARD, AWAY FROM SUN DURING CRUISE).
- ANGLES SHOWN ARE CLOCK ANGLES.

AD7-177

Figure 2-9

SPACECRAFT GEOMETRY AND REFERENCE AXES includes corner and panel designations, control axes, and clock angle zero.

launch, the PSP is retained against three stabilizing supports by a single tie-down point. The PSP is deployed by an electric drive capable of continual articulation. While the capsule is still mounted on the spacecraft possible capsule contamination is avoided by keeping the deployment of the PSP restricted to a position from which there is no line of sight to any exposed capsule surface with the canister lid removed. After the capsule is separated, the PSP is able to deploy to its outermost operating position, where it can view across the entire forward surface of the capsule canister.



2.3.2 Propulsion Module

The propulsion module (Figure 2-10) is also octagonal in cross-section to match the equipment module. It measures 156 inches along the roll axis. To fit within the height limitations of the shroud, the paired tanks are symmetrically located about the geometric axes. The propellant mixture ratio permits the use of identical tanks; the oxidizer tanks are located on the x-axis and the fuel tanks on the y-axis.

The helium pressurization gas is stored in two identical tanks symmetrically located about the +y axis. The tanks are located as far forward as is practicable to permit the most favorable slosh stability environment and the maximum TVC moment arm by their influence on the forward movement of the spacecraft center of mass. The engine is mounted on the centerline as far aft as possible also to increase the TVC moment arm. However, the mouth of the engine bell clears the aft shroud cover sufficiently so that the TVC system of the encapsulated spacecraft can be tested by gimbaling the engine.

As shown in Figure 2-11, four additional adapter attach fittings are located at the four intersection points of the double cruciform beams to transmit the engine loads and a portion of the tankage boost loads to the adapter. The tops of the beam caps form a plane which supports a two-inch thick sandwich panel platform bearing the propellant and pressurant tankage and providing shear capability to the beam assembly. Skirt attachment flanges are integral with the tank walls at the equators of the propellant tanks. Honeycomb sandwich panel skirts connect the propellant tanks to the tank support platform via bathtub fittings (10 per tank) which attach to the platform assembly directly above the beam caps. The engine gimbal ring is supported by two tubular truss assemblies anchored at the stabilized midpoints of the cruciform beams. The two gimbal actuators are located forward of the platform, between tanks.

The equipment module furnishes complete protection from micro-meteoroids from the tank support platform forward. Below the platform, in the area of the peripheral beams at panels II, IV, VI, and VIII, protection is provided by 0.007-inch aluminum sheets spaced two inches from the surface of the beam webs. The aft rectangular areas of the



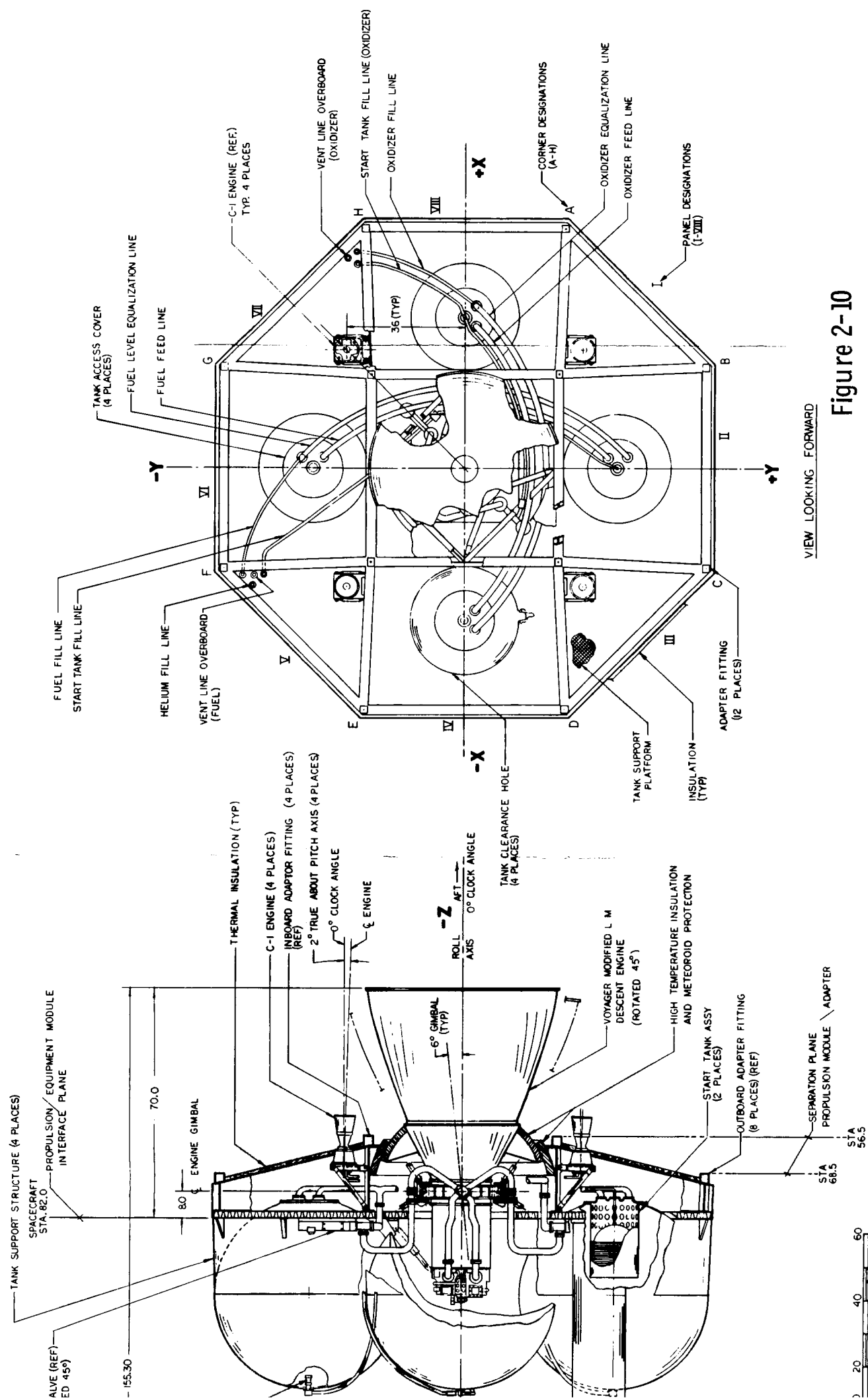


Figure 2-10

AN INBOARD PROFILE OF THE PROPULSION MODULE is shown. All propulsion elements are combined into this module and can be completely assembled and tested prior to integration with the rest of the spacecraft.

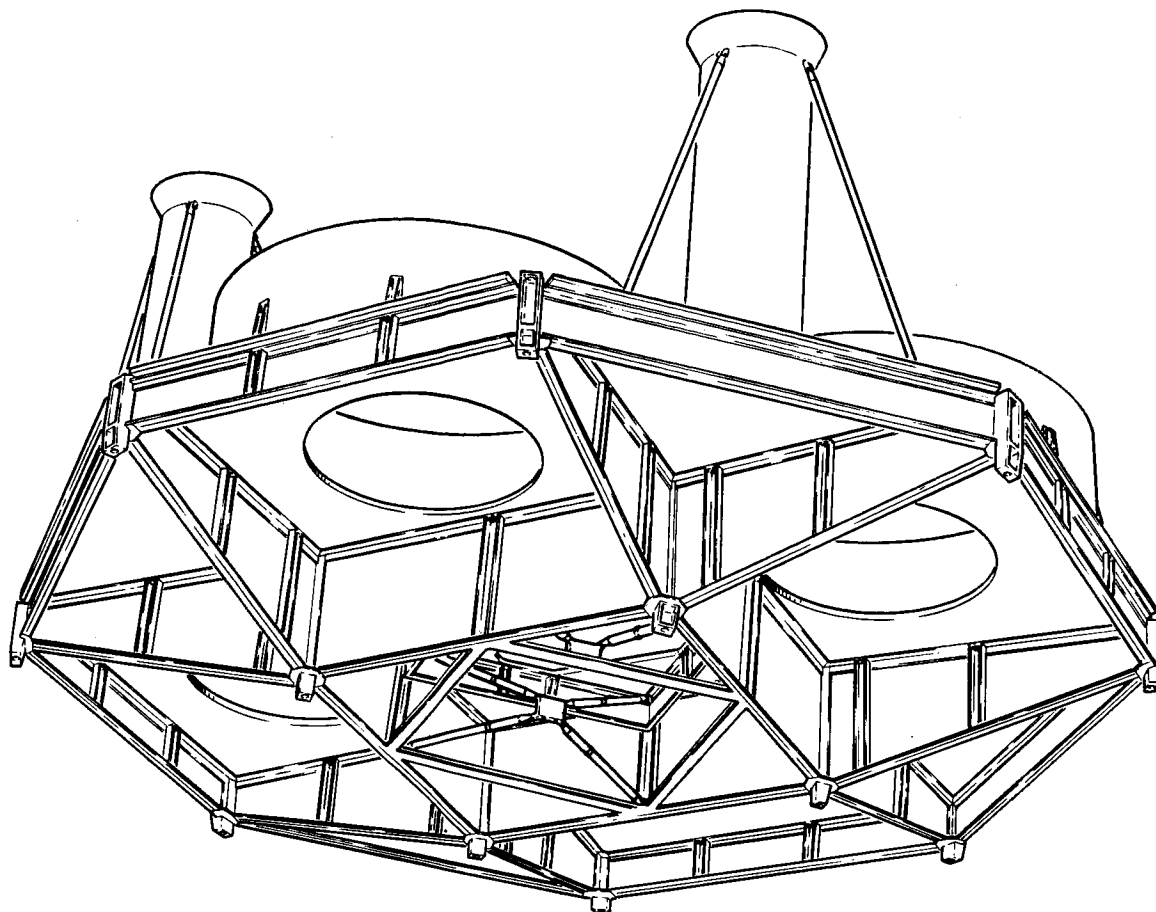


Figure 2-11

Shown here is the beam assembly which constitutes the BASIC STRUCTURE OF THE PROPULSION MODULE and provides the mechanical interfaces to the equipment module and the planetary vehicle adapter.

spacecraft are covered with solar cells and the associated substrates. An additional 0.025-inch aluminum shield over each of the exposed tanks and lines provides additional protection in these areas. Each of the triangular areas on the aft side require only a single 0.007-inch aluminum cover since no vulnerable equipment is located in this area and the tank support platform protects all the equipment above it. Around the upper end of the engine bell, the internal surface of the high temperature insulation is covered with a 0.005-inch sheet of titanium to provide micrometeoroid protection. To permit fueling after shroud emplacement, all propellant fill and vent lines are extended to the shroud, with special recessed boxes sealing them from the spacecraft interior. The lines then extend from the box through the adapter truss to a disconnect fitting at the separation plane. A circular pattern for routing the main

propellant feed lines is used since it minimizes the cutouts through the neutral axis of the beam webs and permits the shortest line lengths.

The pressurization system lines and tank vent lines are 3/4 inch in diameter. Lines extend from each helium tank to the pressure regulator and valve assembly panel where the valve and pressure regulator assemblies are clustered. From this panel the pressurization lines extend to each tank. Each tank has an internal helium diffuser assembly.

2.3.3 Adapter

The planetary vehicle adapter supports the planetary vehicle at 12 points on the aft surface of the propulsion module. The load from these 12 points is transferred to the shroud at a field joint. Holes drilled into the fittings at the ends of the adapter framework are matched and bolted to holes in the shroud. The 12-point attachment is at a biplanar separation joint. Each of the 12 support points is fitted with a compressed helical separation spring and a redundantly actuated bolt separation mechanism. The bolts are mounted in over-size holes and carry tension loads only. Shearing (transverse) loads across the interface are transferred by bosses machined on the aft ends of the propulsion module adapter attach fittings. The bosses engage matching holes in the adapter's spacecraft attach fittings.

To strengthen the shroud in the area of the adapter attachment, and in order to distribute the load from the equally spaced points of load concentration caused by the adapter attachment, TRW has designed a structural system of two circumferential rings, eight intercostals, and eight spacecraft attach fittings to be added to the shroud at each adapter field joint. This structure provides for the attachment of the adapter to the shroud at eight concentrated load points and the redistribution of those loads uniformly into the shroud shell.

2.3.4 Spacecraft Interfaces

2.3.4.1 Capsule

The mechanical interface between the capsule and the spacecraft is a structural field joint consisting of a 160-inch diameter bolt circle containing eight equally spaced attach points. (See Figure 2-12)

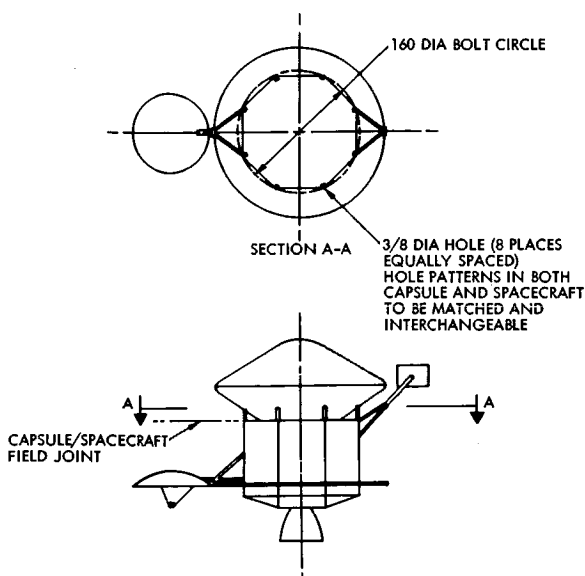


Figure 2-12
CAPSULE - SPACECRAFT MECHANICAL
INTERFACE gives structural support to
the complete capsule.

The mechanical interface maintains an installation alignment of ± 0.1 degree between the capsule and the spacecraft about the roll axis.

The following power and data connections are established between the capsule and the spacecraft.

- Power circuits from the spacecraft to the capsule power regulator or thermal control system.
- Power control circuits from the spacecraft to the capsule sequencing and timing unit.
- Capsule separation sequence initiation command circuits from the spacecraft to the capsule sequencing and timing unit.
- Capsule updating circuits from the spacecraft to the capsule sequencing and timing unit.
- Telemetry data rate and mode selection commands from the spacecraft command and sequencing unit to the capsule telemetry subsystem.
- Capsule telemetry data circuits to the spacecraft telemetry system.

2.3.4.2 Launch Vehicle

To allow launch pad checkout, monitoring, and command, RF windows in the shroud accommodate transmissions between ground stations and both planetary vehicles in the installed configuration. To facilitate transmission from the low-gain antenna during boost, the antenna is located approximately in the launch vehicle pitch plane and on the downrange side. During boost flight, the launch vehicle pitch plane is nominally in the local vertical, with the roll axis horizontal and in the direction of flight. This orientation is maintained approximately during powered injection flight and through separation of the planetary vehicle.

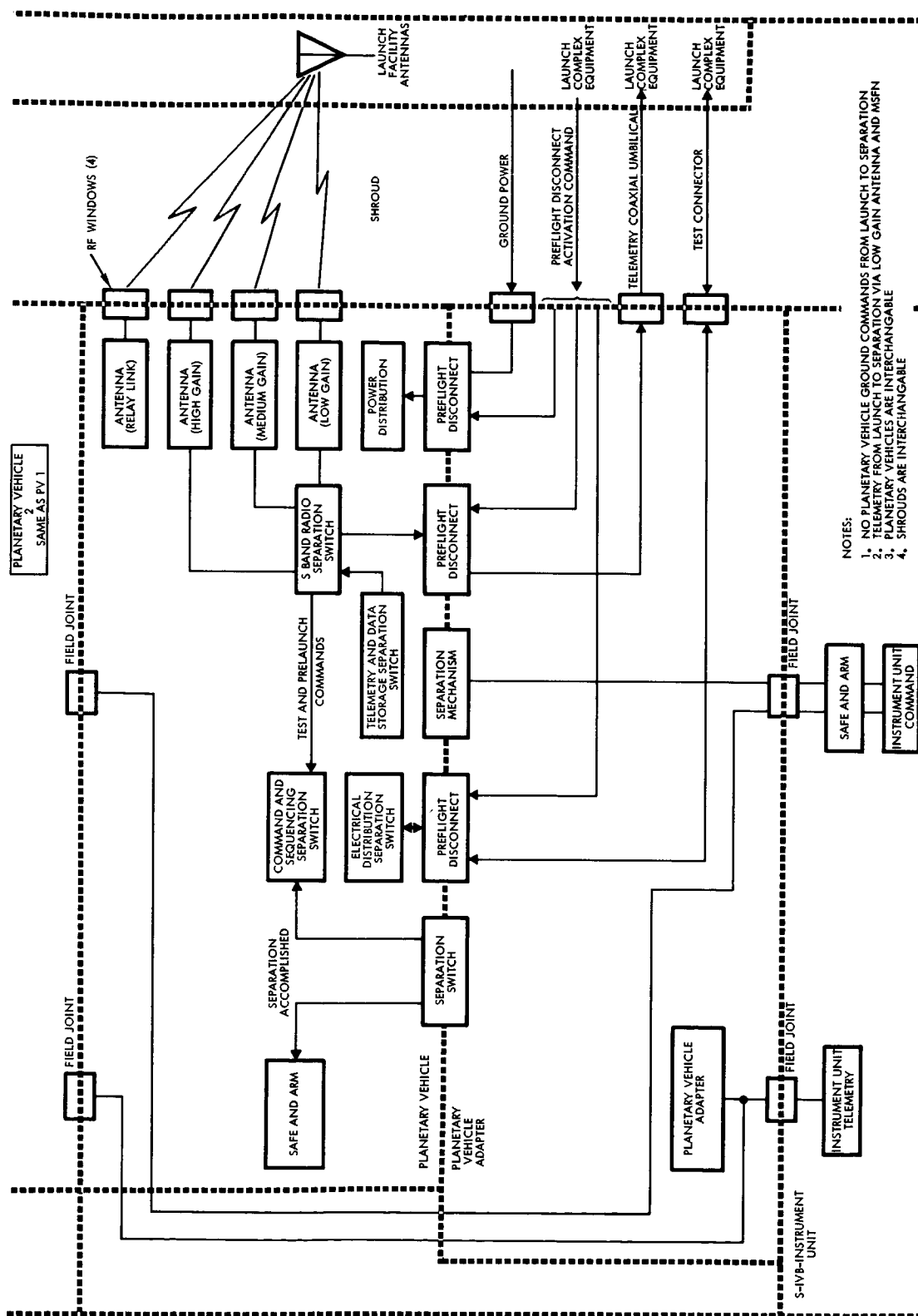
The propellant service lines are mounted to the top of the adapter truss. Near two of the eight outboard adapter fittings, a series of self-sealing fittings disconnect the oxidizer and fuel lines. Electrical contacts indicate the lines have been disconnected when fueling is completed for the oxidizer or fuel system.

Sensors, including accelerometers to monitor boost acceleration and vehicle vibrations, are mounted on the adapter to measure the flight environment during boost. This instrumentation, part of the launch vehicle system, is mounted on the adapter at the time of final field installation of the planetary vehicle with adapter in the shroud section.

Connectors are required to connect the boost-powered flight environmental instrumentation to the launch vehicle telemetry system. Since this instrumentation is installed on the planetary vehicle adapter, no in-flight disconnection is required.

The electrical interfaces between the spacecraft and the launch vehicle, through the adapter are illustrated in Figure 2-13 which also shows the interfaces with the launch complex. The following circuits are required:

- Umbilical circuits from the ground power source on the launch pad to the spacecraft, disconnected prior to launch after power transfer
- RF hardline umbilical circuits from the planetary vehicle and the launch control equipment, disconnected prior to liftoff



SPACECRAFT-LAUNCH VEHICLE ELECTRICAL INTERFACE includes the launch umbilicals, test cables, ground power umbilicals from the Launch Complex, and the separation circuitry from the S-IVB instrument unit.

- Launch pad test circuits between the planetary vehicle and the launch control equipment, disconnected prior to liftoff
- Separation circuitry and mechanisms for each planetary vehicle separation upon command from the Saturn instrument unit digital command system
- Switch and interconnection circuits to sense separation and signal the safe and arm unit and the command and sequencing unit
- Connectors for carrying signals from adapter-mounted instrumentation to the launch vehicle telemetry system

2.4 FUNCTIONS AND CHARACTERISTICS

The functions and relationships of various elements of the spacecraft which provide the capability for executing the Voyager mission are discussed here. In some cases, spacecraft functions are discussed in groupings which are different than the subsystem areas into which the hardware breakdown is organized. This has been done to bring out the essential interactions underlying the overall mission functions. The system functional block diagram, Figure 2-14 illustrates the flow of controlling signals to the various subsystems and the functional interdependency of these subsystems. Redundant circuitry and secondary power flow are not shown since the purpose of this diagram is to depict the functional quality of the spacecraft. Detailed diagrams and descriptions of the individual subsystems are in Volumes 3 and 4 of this report.

2.4.1 Communications

Communication functions include an uplink for commands and a downlink for telemetry data. Tracking is also provided by means of doppler range rate and pseudo random noise ranging utilizing both uplink and downlinks.

Four separate S-band antennas are provided. A two-axis steerable high gain antenna provides coverage for normal cruise and orbital operations. A second directional antenna is used for pre-maneuver operations near-Mars and as a backup to the high-gain antenna for

21

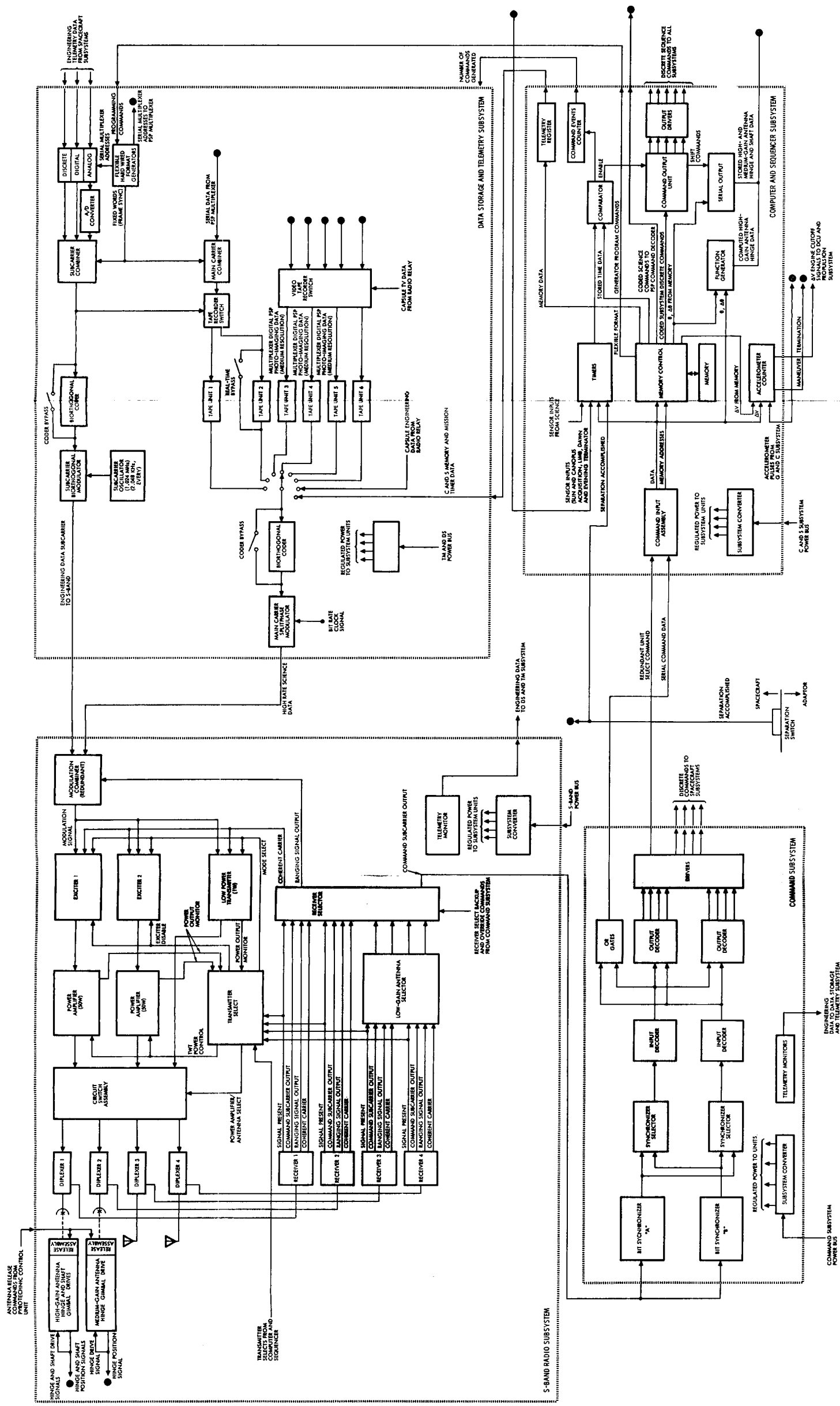


Figure 2-14 (sheet 1 of 2 sheets)

SPACECRAFT FUNCTIONAL BLOCK DIA-
GRAM indicates the primary relationships
between the subsystems. Detailed block
diagrams for the subsystems may be found
in their respective volumes.

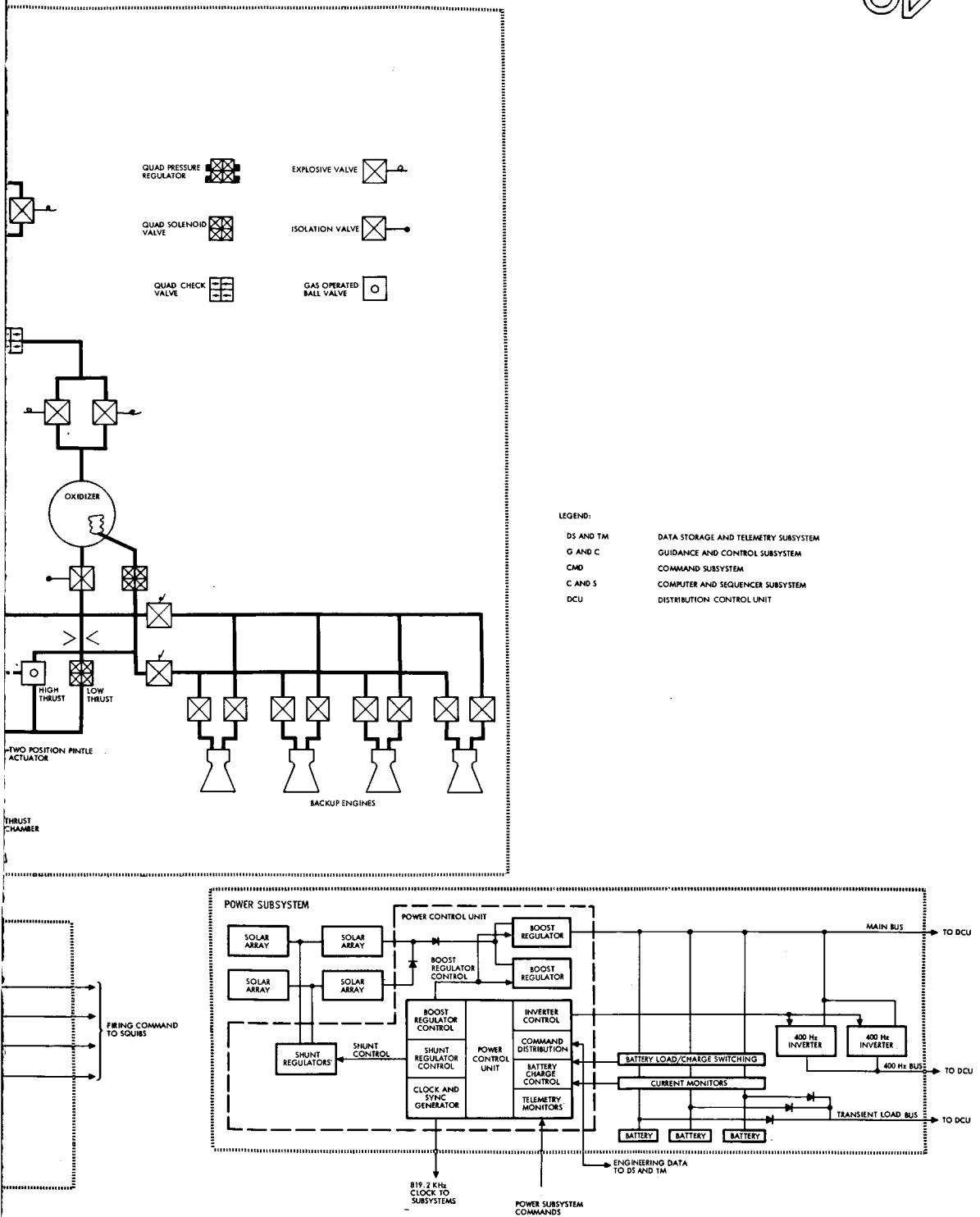


Figure 2-14 (sheet 2 of 2 sheets)
SPACECRAFT FUNCTIONAL BLOCK DIA-
GRAM indicates the primary relationships
between the subsystems. Detailed block
diagrams for the subsystems may be found
in their respective volumes (Continued)



orbital operations. Two low-gain antennas, one located on each side of the spacecraft, give spherical coverage for maneuver sequences and emergency (unoriented) communications. Each of the four antennas is connected through a diplexer which provides frequency-selective filtering to permit simultaneous transmission and reception.

2.4.1.1 Uplink

The S-band uplink receives and demodulates command and ranging signals transmitted from the Deep Space Network, and also provides a coherently related output carrier frequency to drive the downlink transmitter. The receiver channel output of each diplexer is connected to a separate phase-locked receiver which synchronously demodulates the uplink data and provides a narrowband output signal containing the command data subcarrier and a wideband output containing the pseudo random noise ranging signal.

Continuous command access is provided by the low-gain antennas independent of vehicle orientation. To provide a smooth transition between the two low-gain antennas in the presence of tumbling or antenna pattern anomalies, post-detection diversity combining is applied to the command subcarriers from the two individual receivers. Mutual interference is avoided by incorporating gating circuitry to inhibit a receiver channel output if it falls below a predetermined threshold signal-to-noise ratio. Reception over the high-gain or medium-gain antenna channels is not normally required for command operation. The primary function of these channels is to receive and demodulate the pseudo random noise ranging data. They do, however, provide backup as well as an additional command margin for normal mission operations.

The receiver outputs are processed through logic circuitry which selects the best receiver channel. Selection is based on a combination of command-controlled priority logic and relative signal-to-noise ratio. Normally the highest received signal-to-noise ratio is selected. To effect this selection, an output signal proportional to the received carrier amplitude is generated by each of the four receivers. Provided this signal exceeds a value proportional to the allowable command

error rate threshold, the uplink channel corresponding to the selected downlink antenna is selected. In the event this priority channel is below threshold, the channel having the highest signal-to-noise ratio is used. For example, during the normal cruise phase, and during orbital operations, the high-gain antenna is selected. If the receiver connected to that antenna drops below threshold due to antenna failure or a loss of orientation, the selection logic will then select the medium-gain antenna, provided it is being illuminated by the earth station. If not, it will select the diversity combined output from the low-gain channels. Provisions will be included to provide ground-controlled and failure-sensing override of the primary selection logic.

The command subcarrier from the selected receiving channel is sent to the command subsystem for demodulation and synchronization and decoding of the resulting digital commands. The command subsystem contains two bit synchronizers and two decoders configured for cross-strapped redundancy. Both bit synchronizers operate in parallel and command word bits are utilized to select decoders. The decoders receive data bits, bit synchronization pulses, and lock signals from the demodulator-bit synchronizers. The decoder performs command word format and parity checks, and, in response to routing addresses, transmits serial command word bits to the computer and sequencer or directly to other spacecraft subsystems and provides direct discrete commands on an immediate basis to other spacecraft subsystems.

2.4.1.2 Downlink

The downlink transmits engineering and scientific data to earth on essentially a continuous basis. It also transmits PRN ranging data along with a coherent carrier as required for range and range rate measurements. The coherent drive is supplied by the uplink when it is operative. In the absence of this drive, the downlink utilizes an on-board oscillation and operates noncoherently.

The associated hardware consists of modulation-combining equipment, two redundant 50-watt transmitter channels, and one low-power transmitter channel. Mode selection through the computer and



sequencer (backed up by direct command) normally determines the combination of exciter, power amplifier, and diplexer-antenna to meet the current mission transmission requirements. The choice between the two redundant 50-watt amplifiers and the transmitter exciter is also controlled by the transmitter selector using internal logic operating on signals from the power monitors so as to transfer operation from a unit when its RF output falls below a given threshold. The transmitter outputs are routed to a circulator switch assembly which is energized by the transmitter selector so that the selected transmitter output is connected to the designated antenna through the appropriate diplexer.

The modulator exciters (the low-power transmitter is basically also a modulator exciter) receive a composite modulating signal corresponding to the three channels for transmission of data from the spacecraft to the ground station. The high-rate channel directly phase modulates the S-band carrier and provides a baseband transmission bandwidth up to 800 kHz. This channel is time shared between the ranging code and the high rate science data. The data rate for the high rate channel is selected to be 3.2, 6.4, 12.8, 25.6, or 51.2 kbits/sec depending on range. This choice is completely adaptive in the sense that the data rate need not actually be switched down until the ground-received error rate exceeds the acceptable level. The second channel is operative continuously and provides 512 bits/sec engineering telemetry and low rate science data on a 1.024 MHz sub-carrier. The third channel provides 8 bits/sec telemetry data on a 2.048 kHz subcarrier. This channel is normally used only when the vehicle is performing a reorientation maneuver or when attitude stabilization has been lost.

2.4.1.3 Tracking

The S-band radio subsystem retransmits the received signal to permit measurement of spacecraft range and range rate by the Deep Space Network. Range rate is determined from the doppler shift obtained by retransmitting a carrier frequency coherently related to the spacecraft received frequency by a ratio of 240:221. This downlink frequency is then compared with the ground transmitted frequency to

derive the doppler shift. The ranging measurement is accomplished by measuring the phase delay of a pseudo random noise code sequence that is made up of five subcode words and has the property of requiring a correlation (acquisition) time proportional to the sum of the bits in the five code words and an unambiguous length proportional to the product of the bit count in each code word. The code has the advantage that it has a large unambiguous range measurement capability while at the same time it is capable of being acquired within a reasonable period of time. It has the disadvantage that it requires a transmission bandwidth of at least 2 MHz.

2.4.2 Spacecraft Sequencing

The basic sequencing of the spacecraft is accomplished automatically based on command data stored prior to launch in the computer and sequencer (C and S) or transmitted by the S-band uplink during the mission. Backup sequencing can also be accomplished directly by ground command, as well as the updating or modifying of any stored data.

The mission commands from the C and S are organized into distinct timed sequences controlled by on-board clock references. These sequences are initiated individually during the mission either upon receipt of a ground command, at a mission clock time determined by a stored command, or at the occurrence of certain mission events. The stored commands in each sequence are time-referenced to the start of the sequence and are issued at the indicated elapsed time from the point of initiation of the associated counter controlling the sequence. Decoded messages from the command subsystem are accepted by an input section of the C and S and routed to the appropriate internal location. This can be memory storage, in the case of a stored command, or a timer, accelerometer counter, or function generator in the case of numerical data.

2.4.2.1 Stored Commands

Each command output from the C and S is associated with a specific memory location. To load a command then entails storing

in the associated memory location the identification of a particular sequence for which the command forms a part, and also the time from sequence start at which the command is to be activated. An enable bit is associated with a stored command, and must be enabled or the command will be inhibited.

Resetting and initiating a timer activates a particular sequence as identified by a sequence code associated with the timer. There are four such computer and sequencer timers, each containing a sequence code and accumulated time when in operation. The four stored command sequences can be operative concurrently. The contents of the computer and sequencer memory are scanned serially once each second and compared for sequence code and time coincidence simultaneously with the four timers. If both time and sequence coincidence with any timer occurs, and the command word enable bit is set, either a command discrete or a serial message is issued, depending on the function of that memory location.

2.4.2.2 Timing

Several timing capabilities are required to meet the needs of the various stored command sequences. The mission timer for the overall mission sequence has a granularity of 1 second, a range of 25 months, and is used for sequencing long term events, beginning other sequences, and time tagging recorded data. Memory words are compared with the 14 most significant bits of this timer, thereby providing a 1.14-hour sequence granularity. An orbit timer has a 4-second granularity and 18.2-hour range, and is used for all orbit science sequences which require the terminator crossing signal. The remaining two timers are both maneuver and science sequences and have 1-second granularity with a 4.5-hour range. All four timers are capable of being loaded with a sequence code and initiated by stored program commands or ground commands via the command input unit.

2.4.2.3 Command Synchronization

The mission timer is used to sequence long term events both from the memory and from the function generator. A sequence can also be synchronized to a particular event. This is accomplished by

setting the associated enable bit at the time of the event. This is accomplished by a signal from the terminator, limb-crossing or fine sun sensors, by a separation signal, by a Canopus acquisition signal, or by ground command. Setting the enable bit permits the controlling timer for the sequence to begin timed pulse accumulation.

The computer and sequencer can repeat an orbit sequence in a periodic fashion, e. g. measured from the morning terminator. It can also perform a series of orbit or science sequences initiated by the relatively long-term orbital time reference such that the initiation of each sequence in a string is in a fixed relation to the terminator or limb crossings. This permits the sequence to prepare for events to occur in the next orbit sequence. A discussion of the use of the terminator and limb crossings for automatic sequencing of science operations was given in the TRW Phase 1A task B Final Technical Report, Volume 1, Appendix D, dated 17 January 1966.

Should the terminator sensors signal input controls fail to operate, sequencing initiate times may be controlled by properly timed ground commands. Orbit science sequencing can thus be synchronized to Mars lighting conditions even if the sensors themselves fail to operate.

2.4.2.4 Engine Sequencing

Engine cutoff commands are issued from accelerometer counters. The accelerometer counters accumulate accelerometer pulses during engine burn and issue a cutoff signal when the count reaches the required incremental velocity correction. The cutoff point is established by preloading the counter with the complement of the incremental velocity and letting the overflow pulse cut off the engine. The overflow pulse is sent to the correct cutoff valve by first enabling the appropriate one of five gates, one for each of five sets of engine valves.

If either of the accelerometer counters fail, the engine can still be shut down by using a stored program command controlled from a sequence timer. Stored start and stop commands are provided for each of the engine operating modes. The stop commands are issued over output lines to the same valves controlled from the accelerometer counter and can activate those valves if the counter fails.

Should the primary engine fail to start during the orbit insertion maneuver as indicated by no accelerometer output, stored commands are provided to start the backup C-1 engine and initiate the associated mode of vehicle attitude control.

2.4.2.5 Antenna Pointing

Earth pointing of the double-gimballed high-gain antenna and the single-gimballed medium-gain antenna is accomplished on a continuous basis. Pointing commands are generated against mission time by means of a function generator within the C and S. No sequence designation is required for the function generator, and the data output time is established by sampling the mission timer. Computations of pointing angles are performed from data obtained either from stored program commands or updated by ground command.

2.4.3 Data Handling

The telemetry and data storage subsystem takes data from the science subsystem, the capsule, and the diagnostic measuring circuits (engineering data) and converts it into a form suitable for transmission by the digital S-band communications system. The engineering and science data can be transmitted real-time or can be stored digitally on tape recorders for playback at a later time. Tape storage is used as a buffer whenever data is being generated faster than the communications link can accept it. Tape storage of engineering data is also used during maneuvers as a backup in case of interruption of communications. During capsule descent, the formatted TV and engineering data is received via the capsule-spacecraft radio link and either stored or directly transmitted to earth.

A versatile centralized telemetry data handling unit is supplemented by six remote multiplexers on the planetary scan platform (PSP), one for the PSP-mounted experiments and five others for the all-TV mapping system. This provides programmable data format capability and utilizes the seven data transmission rates that are available for the recommended spacecraft. Thus it is possible to meet the varying data requirements of different phases of the mission and of different missions without redesign of hardware.

2.4.3.1 Data Format Generation

All data except that from the television subsystem is ordered into a serial time sequence by data multiplexers. Analog data from the multiplexer is converted into digital words by an analog-to-digital converter. Under the direct control of a flexible format generator, the digital data is then combined with frame synchronization signals, identifications, etc., to form a composite serial bit stream.

Preprogrammed formats can be selected by ground command or new formats can be programmed into the format generator from the ground, which gives considerable flexibility. The flexible format generator provides serial addresses to the multiplexer. The address bits are decoded by the address decoder and driver within the multiplexers which in turn selects the desired data input. The flexible format generator is backed up with a hard-wired programmer capable of providing one fixed data format for each multiplexer. The hardwired programmers are brought into operation by ground command.

The high rate data emerging from the television mapping system is put into a fixed format by remote photo-imaging multiplexes on the PSP. These multiplexes are not under the control of the flexible format generator because the video data must be encoded and synchronized by the video data clock from each TV camera. The data contains all the required synchronizing and identification pulses to reconstruct one line of the picture on the ground. In addition, the remote photo-imaging multiplexer enters telemetry-frame sync code bits at the start of the scan line, and at the end of a picture, 22 TV camera engineering measurements are added. A separate video data encoder is provided for each camera, since some cameras are used simultaneously.

2.4.3.2 Telemetry Data Output

The serial bit stream of engineering data for continuous real-time transmission passes through a biorthogonal coder, biphase modulator, the S-band subcarrier, and is fed to the communication subsystem to be transmitted to earth. There are two data transmission formats as follows:



- Subcarrier format No. 1, engineering data only, includes high rate propulsion, guidance, and control data. This mode is used primarily during maneuvers and for failure analysis.
- Subcarrier format No. 2, is a combination of spacecraft and capsule engineering data, and would also include low rate science data such as field and particle measurements if this were on the spacecraft. This mode is used during cruise.

The basic S-band subcarrier (1.024 MHz) corresponds to a data rate of 512 bits/sec. Another S-band subcarrier (2.048 kHz) can be used in an emergency. This corresponds to a data rate of 8 bits/sec when utilizing the omni low-gain antenna.

All data other than the real-time engineering utilizes the high data rate channel on the S-band main carrier, which corresponds to the following formats:

- No. 1, playback of stored science data
- No. 2, playback of stored engineering data recorded during maneuvers
- No. 3, real-time transmission of science data
- No. 4, transmission of computer and sequencer memory contents
- No. 5, playback of stored video data
- No. 6, real-time relay of capsule low-rate data during capsule descent

The associated serial bit streams pass to the S-band transmitter input via a biorthogonal coder and a split-phase modulator and are transmitted as discussed in Section 2.4.1.2. The switchable data rates for the high rate channel are as follows: 51.2, 25.6, 12.8, 6.4, and 3.2 kbits/sec (emergency).

2.4.4 Attitude Control

The spacecraft generally operates in a fully attitude-stabilized mode utilizing the sun and Canopus as celestial references. At times, such as just after separation, during celestial reference acquisition,

and for reorientation and thrusting to carry out maneuvers, the spacecraft operates in a controlled maneuvering mode utilizing on-board inertial references. Control reverts automatically to an inertial holding mode during eclipse and occultation conditions, or at any time when the celestial references are lost. The celestial references are then established again when this is possible.

Control torque is generally obtained from a redundant reaction gas jet system that provides two levels of thrust. During main engine firings, control torque in pitch and yaw is obtained by gimbaling of the single thrust chamber about two axes, with roll control obtained by the reaction gas system. Thrust vector control for the backup C-1 engines is obtained by duty cycle modulation.

2.4.4.1 Planetary Vehicle Separation

For the period immediately following planetary vehicle separation, the spacecraft orientation is to be maintained approximately the same as it was prior to separation. This is to facilitate continued communication during the post-separation eclipse period until sun acquisition. At separation, a rate-nulling mode is activated which utilizes the reaction control high thrust level to reduce the initial tip-off angular motion. A reduction to less than 0.01 deg/sec each axis is achieved in about 15 seconds. The vehicle then continues to rotate at this low angular rate until sun acquisition is initiated.

2.4.4.2 Sun Acquisition

Upon emergence of the spacecraft into sunlight, the sun acquisition mode is automatically initiated by the illumination of the coarse sun sensors. Pitch and yaw angular rate signals are obtained from gyros in the rate mode (caged), with position errors obtained from the coarse sun sensor. Illumination of the fine sun sensor occurs when the aft roll axis is within 10 degrees of the sun line, and an acquisition gate then switches from the coarse to the fine sun sensor.

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



When the vehicle roll axis has acquired the sun, as signalled by the appearance of the sun in the field of view of the sun acquisition sensor and a low value of error signal from the fine sun sensor, the Canopus acquisition mode is commanded by the computer and sequencer. For this mode, a controlled roll rate of 0.1 deg/sec is established, and the Canopus sensor shutter is opened and its acquisition field of view is caused to sweep the star field. When the star of the chosen brightness range appears in the field of view, a star acquisition signal is generated which switches the control system from gyros to the output of the Canopus sensor.

The acquisition of Canopus is verified by prepositioning the high-gain antenna so that the antenna beam intersects the earth when the spacecraft is in the proper post-Canopus acquisition attitude. If a wrong star is acquired, as indicated by the absence of the RF signal, a command is sent to revert the control system to the Canopus acquisition mode and the roll search is resumed until Canopus is located.

2.4.4.3 Cruise

With the completion of sun and Canopus acquisition, the control system is operating in the cruise mode. This mode provides pitch and yaw control derived from fine sun sensor signals and roll control derived from the Canopus sensor signals. For this mode, reaction control is switched to the low thrust level. The gyros are turned off to conserve power and rate damping is provided by derived rate increment networks. The vehicle operates in stable limit cycles within the ± 0.5 degree deadband in each axis with limit cycle rates less than 10 deg/hr.

The cruise mode of operation is used for both transit and orbital phases of the mission. During the photo-imaging periods of the orbital phase, the limit cycle deadband is reduced to ± 0.25 degree to increase the pointing accuracy of the PSP.

2.4.4.4 Vehicle Reorientation

A maneuver mode is used before and after a midcourse trajectory correction, orbital insertion, or orbit trim firing. Essentially, it

consists of a reorientation of the vehicle to the desired direction using the on-board inertial reference assembly. Approximately one hour prior to maneuver, the computer and sequencer commands the inertial reference unit on to enable it to warm up and stabilize. At the start of the maneuver, reaction control is switched to the high thrust level and the control system is switched from the celestial sensors to the gyros. The gyros, which are operated in the position mode, have been previously initialized so that they are compensated for the attitude error existing at the time of transfer. The gyros are then torqued at the rate of 0.2 deg/sec by means of precision current generators and the vehicle turns through a corresponding angle under the action of the control system. Commanded turns are performed sequentially in the yaw, pitch, and roll axes. The polarity and turning time data transmitted by radio command are stored in the computer and sequencer.

Following completion of a reorientation maneuver, the new attitude is maintained by continuous inertial control.

Operation for inertial hold is similar to operation of the maneuver mode above except that the gyros are not torqued. Therefore, the vehicle maintains a constant attitude in inertial space within the limitations of gyro drift and the vehicle operating limit cycle. This mode is used subsequent to the maneuver mode and at other times when it is desired to maintain a constant attitude without continuous reference to the celestial sensors, as for instance during sun and Canopus occultations in the orbital phase of the mission.

The inertial hold mode can be used indefinitely subject only to gyro drift. However, it is primarily intended for relatively short periods of operation (several hours). Under these conditions, the celestial references can be reacquired simply by repeating any preceding maneuver in reverse, that is, torquing the gyros in the opposite direction at the same rate and for the same time under the command of the computer and sequencer. If it is not desirable to do this, or if the inertial hold mode has been used long enough that gyro drift renders this approach useless, the celestial references can be acquired automatically using the sun acquisition and Canopus acquisition modes.



2.4.4.6 Thrust Vector Control

Thrust vector control is required during operation of the main engines or C-1 engines. For this mode, the gyros are operated as position sensors and their outputs control the thrust vector direction via gimbaling of the main engine or duty cycle modulation of the C-1 engines. The high thrust reaction control jets are used for stabilization about the roll axis, using outputs from the roll gyro. Rate damping in pitch and yaw is provided by passive lead networks operating on the gyro signals. The variation between high thrust and low thrust operation of the main engine imposes the necessity to change gains for vehicle stability and control.

The velocity change during powered flight is sensed by a velocity meter and the velocity increment pulses are counted by the computer and sequencer during the engine operation. When the desired velocity increment is reached, the engine will be shut down automatically. Upon completion of the powered flight, the celestial references are reacquired either by commanded turns, as described for inertial hold above, or by switching to the automatic sun and Canopus acquisition mode of operation.

A test mode is used prior to operation of the main engine to assure proper operation of the TVC channels and the gimbal actuators. This test involves simulating an error signal at the inputs of the TVC channels, energizing the actuators, and observing their outputs for correct polarity and magnitude.

2.4.5 Propulsion

Propulsion is provided by a modified Lunar Module Descent Engine (LMDE) and a cluster of four C-1 engines as a backup to the LMDE. Both engines use storable hypergolic liquid propellants fed by a stored-gas pressurization system. The main engine is designed with two thrust levels; 9850 pounds for Mars orbit insertion, and 1700 pounds for planetary arrival date separation, interplanetary corrections, and orbit trim firings. Each C-1 engine has 100 pounds of thrust.

The fuel is either a mixture of hydrazine and UDMH in equal proportions or monomethyl hydrazine. The oxidizer is nitrogen tetroxide.

The four backup C-1 engines are located at the four corners of the propulsion module structure cruciform. The propellant lines for these engines are routed to a single set of explosive valves mounted on the tank platform. The engines are sealed from the propellant supply system until needed in case of main engine malfunction. The C-1 engines are capable of inserting the planetary vehicle into a 80,000 x 1000 km orbit. This orbit would be large compared to the orbits defined in the mission specification but would still constitute a successful mission because most experiments could be accomplished.

2.4.5.1 Pressurized Fuel Supply System

The helium pressurant is contained at an initial pressure of 4000 psia in two manifolded tanks. The tanks are sealed for all long-term coast periods by pyrotechnic valves. Pressurant is admitted to the regulators through pyrotechnic valves where it is stepped down to the nominal tank operating pressure of 235 psia. The oxidizer and fuel tanks are isolated from each other and from the helium regulator by pyrotechnic valves during all coast periods. When the tanks are being pressurized during the separation and the orbit insertion firings, mixing of the propellants is prevented by quad check valves located in the pressurization lines. For the trajectory correction and orbit trim maneuvers, the propellant quantity used is small compared to the ullage volume, and a blowdown mode of operation is used wherein the tanks are completely isolated from each other and from the pressurant supply system.

A burst-disc relief valve combination is provided to protect the propellant tanks in the event of inadvertent overpressurization. Explosive-actuated vent valves are incorporated in the pressurization system to allow venting of the propellant and the helium tanks after completion of all maneuvers.



A set of parallel redundant prevalves is included in the propellant feed lines for positive sealing of the propellants during coast periods. Corresponding pairs of fuel and oxidizer prevalves are mechanically linked to preclude mixture ratio shift in the event of failure of one of the parallel legs.

The propellants are contained in symmetrical pairs of identical fuel and oxidizer tanks. The tanks are manifolded in a parallel flow arrangement and gas-side and liquid-side equalization lines are provided between like propellant tanks.

2.4.5.2 Low-Thrust Maneuvers

Positive expulsion bellows tanks (start tanks) are located in one fuel and in one oxidizer tank. Bubble-free propellants from the bellows tanks are supplied to the engine during zero-g starts. The normal mode of operation is to open the start tank solenoid valves, open the engine low thrust solenoid valves, and then, at the expiration of 3.9 seconds settling time, open the feed system prevalves. Opening the prevalves allows settled propellant to flow to the engine. The start tank solenoid valves are then closed to prevent continued flow from the start tank and conserve propellant for subsequent starts. The engine continues to fire on settled propellant.

For a typical mission, the initial low-thrust maneuver is at 1700 pounds thrust. The initial nominal specific impulse will be 298 seconds. During this maneuver, the chamber walls will be charred; however, there will be essentially no degradation in performance since no throat area change takes place. Hence, the engine will deliver an average I_{sp} of 298 seconds for all maneuvers prior to orbit insertion. After the orbit insertion maneuver, all orbit trim maneuvers will be performed with an ablated chamber. Low thrust performance with the ablated chamber will be 289 seconds.

2.4.5.3 High-Thrust (Orbit Insertion) Maneuver

For orbit insertion, the start is identical to a low thrust start. After the low-thrust start, the engine is shifted to full thrust by opening the main ball valves and resulting pressure forces move the injector

pintle to the full thrust position. The ball valves are actuated with helium rather than fuel. The engine fires for about 300 seconds.

At the start of orbit insertion, at 9850 pounds thrust, the initial specific impulse will be a nominal value of 305 seconds. During the orbit insertion maneuver, erosion in the chamber throat will decrease the effective area ratio of the nozzle resulting in a decrease in specific impulse. Predicted change in performance during the orbit insertion maneuver is 2 to 3 seconds at full thrust. A secondary effect of throat erosion is the decrease in chamber pressure, which in turn increases the pressure drop controlling the main propellant flow. This results in a flow increase and an increase in engine thrust up to about 10,000 pounds. Since the amount of velocity increment will be controlled by an integrating accelerometer, the change in thrust level will not adversely effect the maneuver.

2.4.5.4 C-1 Engine Operation

Thrust imbalance between the C-1 engines because of spacecraft center of gravity shifts is balanced by modulating the engine thrust. Modulating the engine thrust to compensate for imbalance requires shutdown about once every 12 seconds for a period of about 0.15 second for pre-orbit insertion and orbit insertion maneuvers.

During orbit insertion the main engine fires for about 300 seconds. The backup engines, however, are required to operate continuously for about 9000 seconds.

The four C-1 engines are isolated from the main feed system by explosive valves. These engines are held in reserve, so that any time during the mission that a main engine failure occurred, the C-1 engines are available to take over as a fresh backup.

Spacecraft attitude control during the backup engine operation is obtained by duty cycle modulation of one of the diagonal pair engines for pitch and yaw control. Roll attitude is still stabilized by the four 3-pound cold gas attitude control thrusters.



2.4.6 Electric Power

Power is provided in suitable form for distribution to electrical equipment on board the spacecraft, and to the capsule until its separation. Primary power is derived from the sun by means of silicon photovoltaic cells, mounted on a fixed solar array which is oriented normal to the sun's rays under cruise conditions. Secondary nickel-cadmium batteries are used whenever the solar array is incapable of supporting the loads, as during launch, maneuvers, eclipses, or large transient demand. The batteries are recharged when surplus power is available from the array.

Power distribution from the main bus is between 37 and 50 VDC.* Transient loads for firing pyrotechnics are supplied directly off the batteries, and there is also a central 400 Hz AC bus for certain loads. Each spacecraft subsystem will be equipped with its own DC-DC converter. Load control is provided to automatically disconnect non-essential loads when current demand exceeds a preset limit or battery voltage falls below a preset minimum.

2.4.6.1 Solar Array Control

Because the array is sized to meet the critical power requirement during orbital operations, there is power to spare in the early parts of the mission. However, the transit from earth to Mars is not only accompanied by a decrease in solar power input, there is an accompanying drop in the solar array temperature which causes the output voltage to rise. This rise is such that when the array voltage is optimized for Mars orbit requirements, the array voltage near earth is far too low to charge the spacecraft batteries directly. A boost regulator raises the array output voltage to a level that can provide battery charging. As the spacecraft recedes far enough away from earth for the solar array voltage to be sufficient to charge the batteries directly, the boost regulator is taken out of the circuit and its losses are thereby removed. To limit the voltage supplied subsequently by the solar

* The main bus is generally regulated to 50 VDC ± 1 percent when on solar power, except when the batteries are charging.

array, a shunt regulator is brought into the array circuit. This regulator becomes operational at the point where it can be most efficient, for its losses are at a minimum when the load approaches the solar array power output.

In the shunt regulator, excess capacity is dissipated by means of shunt elements that cause array sections to operate at off-optimum points. The shunt regulator includes one linear element and several switching elements. Initially, the linear shunt element, which is essentially a linear power amplifier, shunts current from the lower section of the tapped array up to a maximum of about 5 amperes. As further array capacity must be dissipated, the switching shunt elements become active. These elements can either short out an array section completely or allow an array section to deliver its full power capability. The "on" points are set so that the first switching element shorts when the total shunt current requirement is 3 amperes; the second shunt element shorts when the total shunt current requirement is 5 amperes, continuing on until the N switching elements are saturated, at which time the linear element can then shunt an additional 5 amperes. During the sequential turn-on cycle of the switching shunt elements, the linear shunt element shunts currents of up to 3 amperes.

The combined solar array regulation scheme, using boost or shunt, gets progressively more efficient as its input voltage approaches the desired output voltage, and the shunt regulator becomes more efficient as the load approaches the solar array output power near the maximum solar distance.

2.4.6.2 Battery Control

The maximum battery energy requirements occur during orbital operations at Mars, specifically during solar eclipses. The eclipse conditions determine the battery design because of the associated magnitude of discharge and cyclic occurrence.

During the eclipse season, the battery is being cyclically charged during sunlight operation and discharged during eclipse. When the spacecraft enters an eclipse with the batteries fully charged, a voltage



monitor signal at about 47 VDC causes all batteries to be switched directly to the main bus. The batteries then supply power to the loads directly through quad-redundant relay contacts which minimize the power losses during discharge. This condition continues until the eclipse period ends, when solar array power increases. At this time, the bus voltage is clamped to the final discharge voltage of the batteries. When the array power exceeds load requirements, excess current is supplied directly through relay contacts to the batteries. The batteries charge toward an upper voltage limit (of 47 VDC) compensated by temperature. Since the charge rates of the batteries will not be identical, a circuit limits the current to any single battery. As each battery reaches its limit, it is switched to trickle charge rate. The batteries remain in trickle charge until another eclipse occurs. While in trickle charge, the batteries can support any transient overload which exceeds the illuminated array capacity.

A further feature included in the power subsystem is the capability to recondition the batteries prior to the major power drains that occur during Mars orbital operations. Nickel-cadmium batteries have a characteristics of losing stored energy when subjected to small repeated levels of discharge which is known as memory effect. This effect can be minimized by deeply discharging the battery and recharging it. The electric power subsystem contains the equipment necessary to accomplish this reconditioning upon receipt of the appropriate ground commands.

2.4.7 Temperature Control

The temperature control subsystem provides the appropriate thermal environments for all sections of the spacecraft during all phases of the Voyager mission by combining passive and active techniques. Use is made of multilayer insulation, high temperature molded insulation, louvers, special surface finishes, thermostatically controlled heaters, and varying degrees of structural coupling. An insulated compartment concept is utilized, whereby the external spacecraft surfaces are insulated except for designated radiation control areas. Unwanted heat from these areas is controlled by louvers and

dissipated to space as required to maintain the internal spacecraft temperature within desired limits. An additional feature is the use of appropriate surface coatings and finishes to achieve proper heat transfer between the spacecraft elements. Thermostatically controlled heaters are utilized for local control of critical components.

The system of louver-covered radiating surfaces affords active temperature control to provide a suitable temperature environment for internally-mounted equipment during mission life. In addition to accommodating relatively large, predictable, local and distributed changes in internal and external thermal environments during the mission, the louver system also has the capability of accommodating uncertainties in spacecraft thermal loads such as those occasioned by unpredictable degradation of surface properties, heat leaks, and failure-mode power dissipation. The electronic components are mounted on panels with the radiating area for each panel sized to dissipate the associated maximum heat load. The louvers open so that heat from the solar array is reflected to space rather than into the spacecraft. The radiating areas directly behind the louvers and other internal surfaces of the main compartment are coated with a high emissivity finish to enhance radiative heat transfer between large areas of the spacecraft and create a uniform environment. Faying surfaces of the component-mounting interfaces remain bare metal for good thermal and electrical contact.

The external multilayer insulation limits external heat gain to or loss from the spacecraft so that the heat to be dissipated is almost all internally generated and remains within the control range of the louvers. The surface finishes on external surfaces provide controlled radiation to space. In particular, the high emissivity finish on the back of the solar panels provides low solar cell temperatures for electrical efficiency.

Radiative heat transfer from the hot engine to the spacecraft is limited by the multilayer insulation covering the base of the propulsion module. The nozzle extension is wrapped with high temperature insulation to limit nozzle radiation to the base-mounted solar arrays. The



deployment mechanisms are covered with multilayer insulation to minimize heat exchange with the sun, space, and the solar array. They are also isolated from the adjacent structure by low conductivity attachment fittings.

The planetary scan platform provides a thermally controlled compartment for the installed science equipment. Thermal control of equipment in the PSP is accomplished by combinations of surface finishes, conductive coupling to the PSP box, and application of thermostatically controlled heaters. In the case of the IR spectrometers, which require cooling of the detectors, special radiators are installed in the surface of the PSP box to remove the heat from the detector elements. In addition to the radiators a cryogenic refrigerator is also provided to cool the detector of the broadband IR spectrometer to a temperature of 45°K.

	Page
3. THE 1973 MISSION	3-1
3.1 MISSION DEFINITION	3-1
3.1.1 The Voyager Mission	3-1
3.1.2 Spacecraft Mission Requirements	3-9
3.1.3 Comparison of 1971 Mission Requirements with Those for the 1973 Mission.	3-19
3.2 OTHER VOYAGER ELEMENTS.	3-23
3.2.1 Launch Vehicle System	3-23
3.2.2 Launch Operations System	3-30
3.2.3 Mission Operations System	3-31
3.2.4 Tracking and Data Acquisition System	3-31
3.2.5 Capsule	3-35
3.3 INTERPLANETARY TRAJECTORIES AND MARS ORBIT CAPABILITY.	3-37
3.3.1 Characteristics of Type I Earth-Mars Trajectories, 1973 Opportunity.	3-37
3.3.2 Mission Maneuver Sequence	3-39
3.3.3 Spacecraft Velocity Increment Requirements.	3-42
3.3.4 Summary Display of Spacecraft Capability.	3-44
3.3.5 Mars Orbit Capability	3-49
3.4 TRAJECTORY ACCURACY CAPABILITY.	3-53
3.4.1 Interplanetary Trajectory	3-53
3.4.2 Orbit About Mars	3-61
3.5 OBSERVANCE OF THE PLANETARY QUARANTINE CONSTRAINT	3-72
3.5.1 Effect of Survival and Growth Probability	3-72
3.5.2 Suballocation of Probabilities	3-74
3.5.3 Trajectory Control	3-76

CONTENTS (Continued)

	Page
3.5.4 Control of Contaminated Efflux and Ejecta	3-88
3.5.5 Controlling Cross Contamination of the Capsule	3-96
3.6 ACCOMPLISHMENT OF SCIENTIFIC MISSION OBJECTIVES	3-98
3.6.1 Orbital Photo-imaging	3-99
3.6.2 Other Orbital Scientific Experiments . . .	3-103
3.7 SPACECRAFT COMMUNICATION PERFORMANCE	3-104
3.8 PROBABILITY OF SUCCESS	3-106
3.8.1 Summary	3-106
3.8.2 Structures Subsystem Reliability Assessment.	3-109
3.8.3 Propulsion	3-111
3.8.4 Guidance and Control	3-112
3.8.5 Command and Sequencing Subsystem . . .	3-113
3.8.6 S-Band Radio.	3-115
3.8.7 Telemetry and Data Storage Subsystem.	3-117
3.8.8 Electrical Power Subsystem.	3-118
3.8.9 Electrical Distribution Subsystem.	3-120
3.8.10 Temperature Control Subsystem.	3-121
3.8.11 Planetary Scan Platform	3-122



3. THE 1973 MISSION

This section defines the 1973 Voyager mission and discusses the capabilities of the TRW recommended spacecraft for performing this mission. In effect this corresponds to a formulation of the overall mission problem, and a verification that the spacecraft design solution presented in the previous section does indeed lead to satisfactory performance of the mission thus defined.

The presentation treats adequacy of performance at the system and mission level, with the corresponding discussion at the subsystem level presented in Volumes 3, 4, and 5. The capability for accomplishing mission functions is covered by the detailed mission description in Appendix A, supplemented by a delineation of spacecraft operating modes given in Appendix B. Particular dimensions of performance covered below are interplanetary trajectories and Mars orbit capability, trajectory accuracy capability, observance of the planetary quarantine constraint, accomplishment of scientific mission objectives, spacecraft communication performance, and probability of success.

3.1 MISSION DEFINITION

The Voyager Mars program will ultimately lead to a comprehensive exploration of that planet. The initial Voyager Mars mission has been planned for 1973 as a first step in an integrated program directed towards that goal. In this program the individual missions form a logical sequence from both the scientific and engineering points of view. The 1973 mission calls for the spacecraft to accomplish extensive orbital reconnaissance using photo-imaging and complementary sensors. The spacecraft also provides support to the capsule through landing site selection, as well as carrying the capsule into a suitable Mars orbit and relaying data to earth from the capsule during its descent. The 1973 mission will also demonstrate and exercise the basic spacecraft and capsule capabilities for use in subsequent missions.

3.1.1 The Voyager Mission

Two planetary vehicles are to be launched on a single Saturn V boost vehicle during the mid-1973 Mars opportunity, with their arrival

at Mars occurring several days apart early in 1974. Each planetary vehicle will go into orbit about Mars, where the capsule will separate to descend and land on the Martian surface. The spacecraft will remain in orbit to carry out its scientific mission and support the capsule as required.

This Voyager mission represents a significant set of functional requirements for the spacecraft. The related mission functions are described in detail in Appendix A, along with a definition of how the recommended spacecraft operates and accomplishes the mission. This system operation is presented in terms of subsystem operating modes, which are then described in detail in Appendix B. Further information on the subsystem design underlying these operating modes is provided in Volumes 3, 4, and 5. A profile of the mission is presented below.

3.1.1.1 Launch and Injection

After final checkout and testing, the spacecraft is mated to the capsule, both are mounted and encapsulated in a shroud compartment, and two such planetary vehicle assemblies are mated to the Saturn V vehicle. The Saturn V is launched and the S-IC and S-II stages boost and are jettisoned; the S-IVB stage injects itself and the payload into a parking orbit. The forward shroud-nose fairing section is jettisoned, increases in altitude, and slows down (Figure 3-1). Upon on-board command, the S-IVB stage starts its second burn to inject the remaining space vehicle into interplanetary trajectory, after which the stage is shutdown (Figure 3-2). The two planetary vehicles will be separated in turn, with jettisoning of the mid-shroud element required, so as to uncover the aft planetary vehicle. Operation of the two vehicles will then proceed independently, but with coordination through mission operations.

3.1.1.2 Interplanetary Cruise

The separated planetary vehicle automatically acquires a sun-Canopus referenced orientation and deploys appendages to the cruise configuration (see Figures 3-3 and 3-4). This interplanetary cruise continues until Mars arrival except for interruptions to carry out mid-course maneuvers. An arrival date separation maneuver will be commanded between 2 and 20 days after injection to provide the time

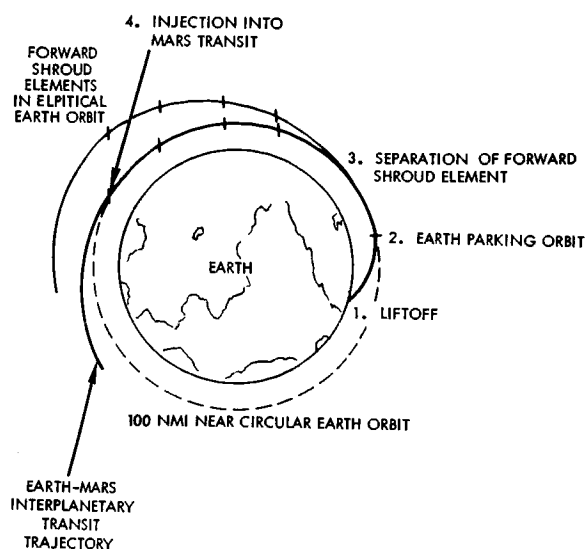


Figure 3-1

IN THE LAUNCH PHASE, the tandem-mounted planetary vehicles are inserted into a near-earth circular orbit. The forward shroud element is jettisoned, which causes it to increase in altitude and slow down so as not to interfere with the remaining vehicle.

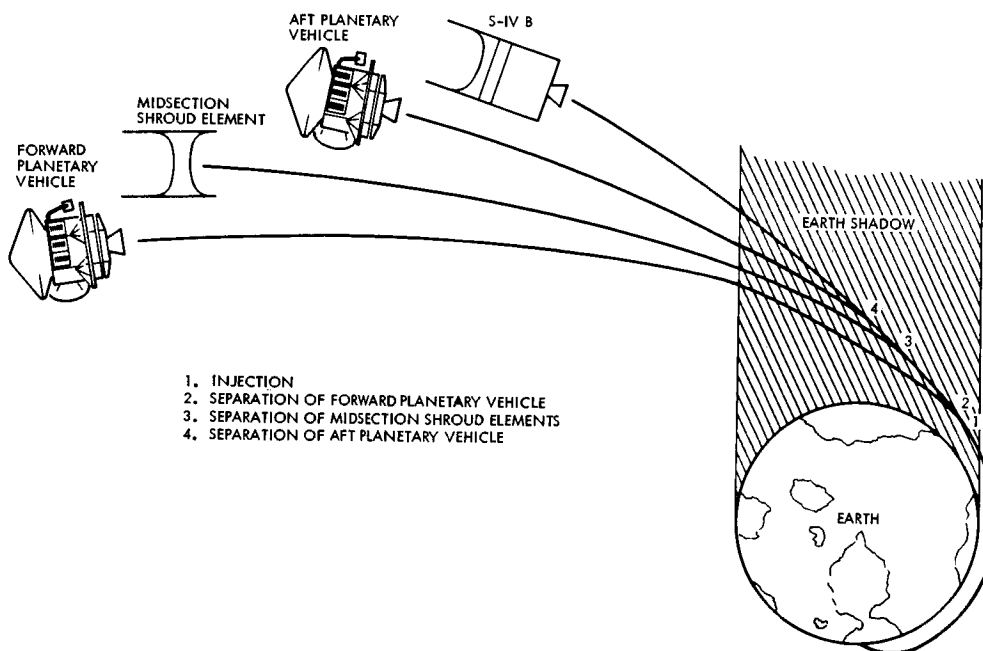


Figure 3-2

S-IVB SECOND BURN puts Voyager on interplanetary trajectory. The forward planetary vehicle separates, followed by jettisoning of the midshroud section and the aft unit, with the two planetary vehicles then proceeding independently to Mars.

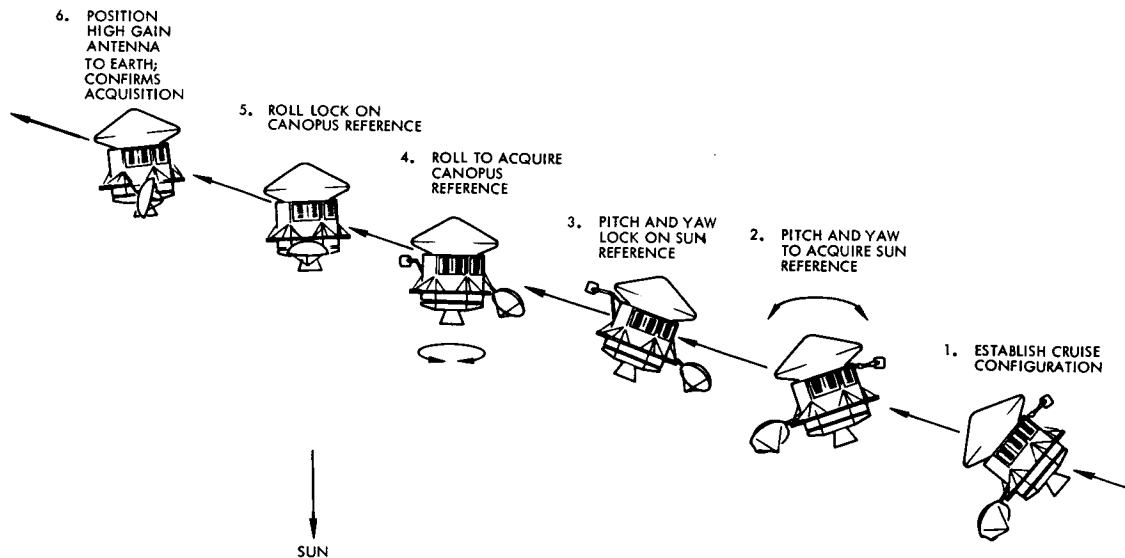


Figure 3-3

ACQUISITION OF SUN-CANOPUS references is the object of an early series of maneuvers during interplanetary cruise. These references establish three-axis stabilization to keep the solar array pointing at the sun and allow directional antennas to be pointed to earth during cruise operation.

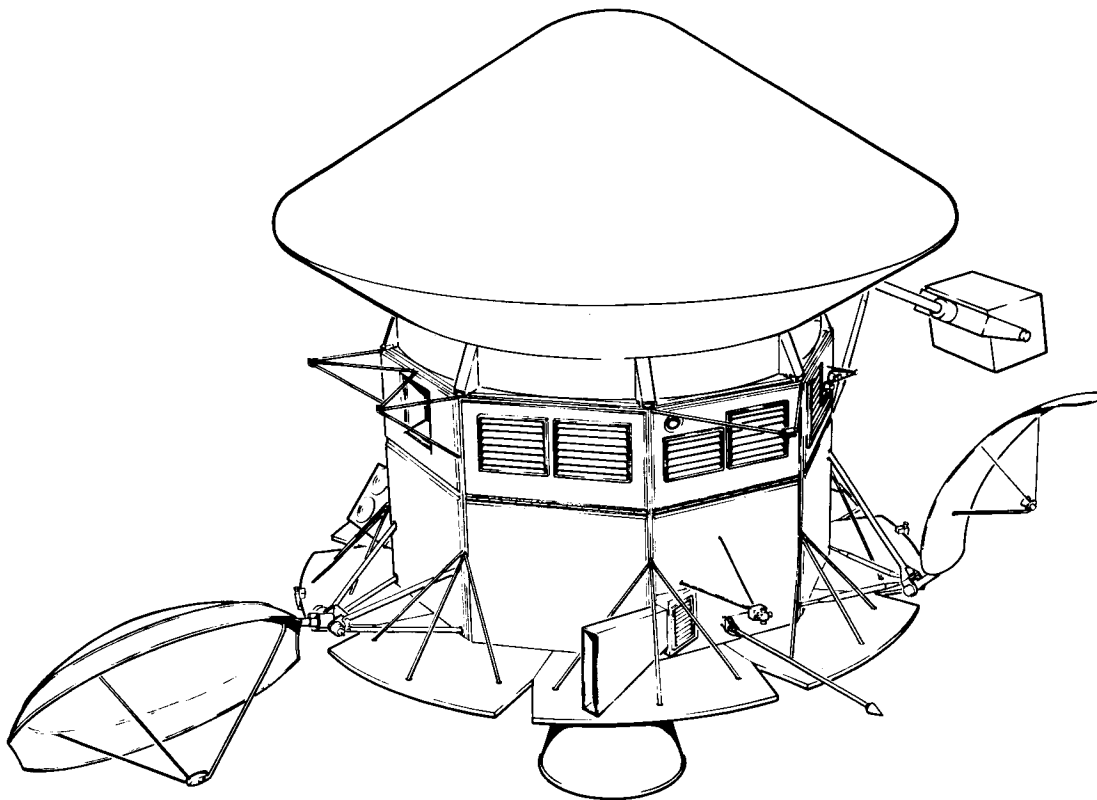


Figure 3-4

IN THE CRUISE CONFIGURATION, the antennas are deployed from their stowed positions to allow tracking by the DSIF and continuous transmission of engineering data to earth. The planetary scan platform moves to an intermediate position. It will move farther forward after the capsule has been dispatched to the Martian surface.



separation in planetary vehicle arrivals at Mars and to correct for the aiming point bias applied to the initial S-IVB injection trajectory. The maneuver direction and magnitude are based upon required arrival time and the premaneuver trajectory as determined by DSIF tracking. The proper vehicle orientation is achieved and verified, thrust is initiated and terminated when the desired velocity increment has been made, and the vehicle automatically resumes cruise (see Figure 3-5). Up to two

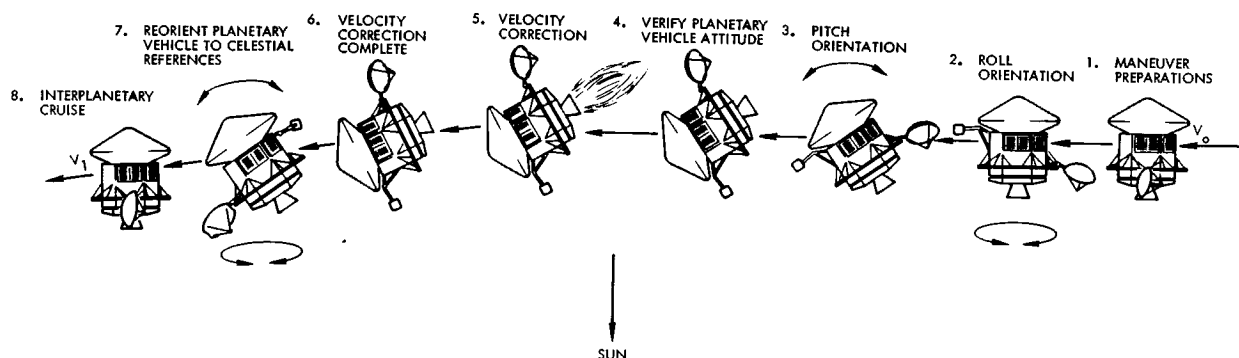


Figure 3-5

TYPICAL MANEUVER SEQUENCE for imparting velocity increment to the vehicle repeats several times during the mission. First maneuver ensures that the planetary vehicles will not arrive at Mars on the same date. Later maneuvers refine Mars arrival conditions.

additional midcourse maneuvers may be required during the course of the transit, depending on the accuracy of the initial arrival date maneuver. The final such correction places the planetary vehicle on a Mars flyby trajectory with a Mars-centered periapsis consistent with the location of the planetary orbit insertion maneuver (see Figure 3-6).

The Mars orbit insertion maneuver represents the major propulsive requirement for the spacecraft and the most critical aspect of the mission (see Figure 3-7). After its completion the planetary vehicle will be in a suitable orbit to survey the planet's surface for possible landing areas, and initial orbital operations are devoted to that purpose. The planetary scan platform surveys the planet's surface, either by automatic tracking or by target pointing, takes pictures as required, and acquires appropriate data. During this period, the remaining scientific instruments are also gathering data and transmitting it to earth as part of the overall mission.

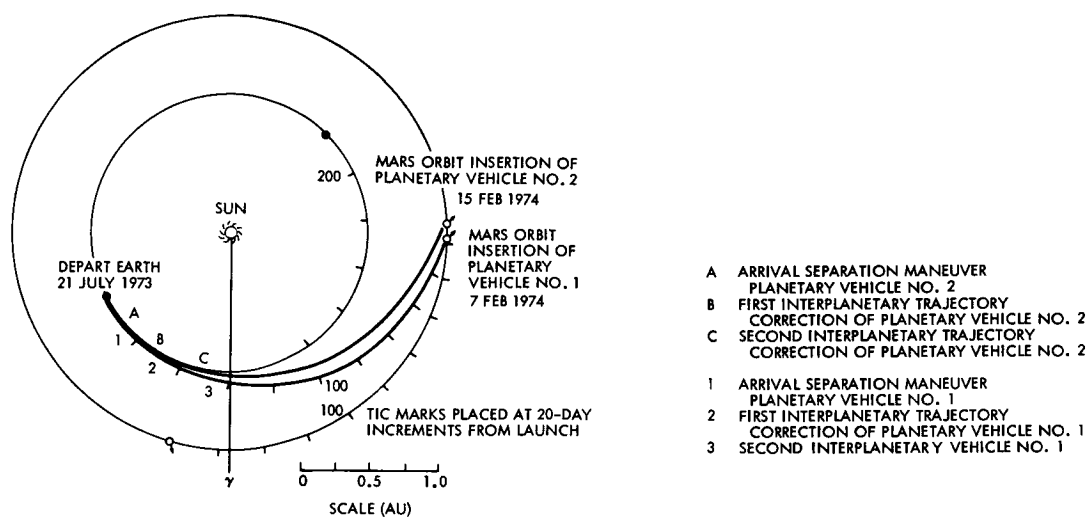


Figure 3-6

TYPICAL INTERPLANETARY CRUISE includes one or two maneuver sequences for midcourse correction to place the planetary vehicles on accurate Mars flyby trajectories.

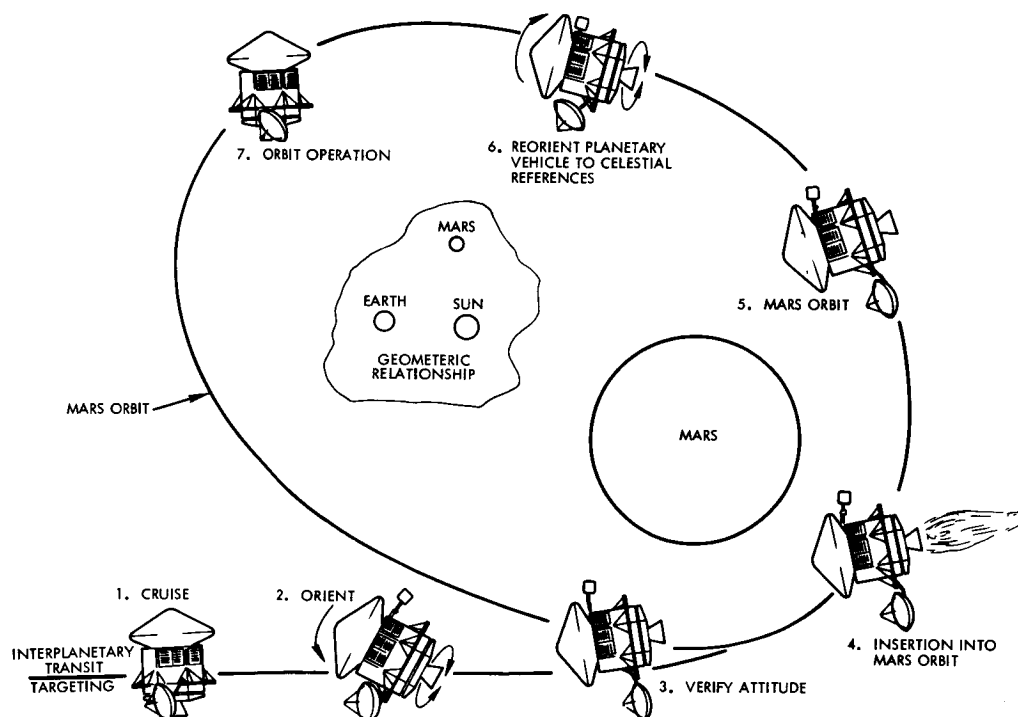


Figure 3-7

ORBIT INSERTION MANEUVER ends the interplanetary cruise by placing each planetary vehicle in a precise predetermined orbit about the planet. This is the most critical point in the mission because no second chance is possible.

3.1.1.3 Capsule Separation and Support

When the landing site has been selected, preparations for capsule separation will be initiated. An orbit trim maneuver may be required as part of this phase of the mission. The equipment in the capsule is turned on and checked out. At the proper time, the capsule jettisons its canister lid, separates itself from the spacecraft to a safe distance, and carries out a deorbit thrust maneuver to allow it to descend to the surface (Figures 3-8 and 3-9). During the descent and landing, the

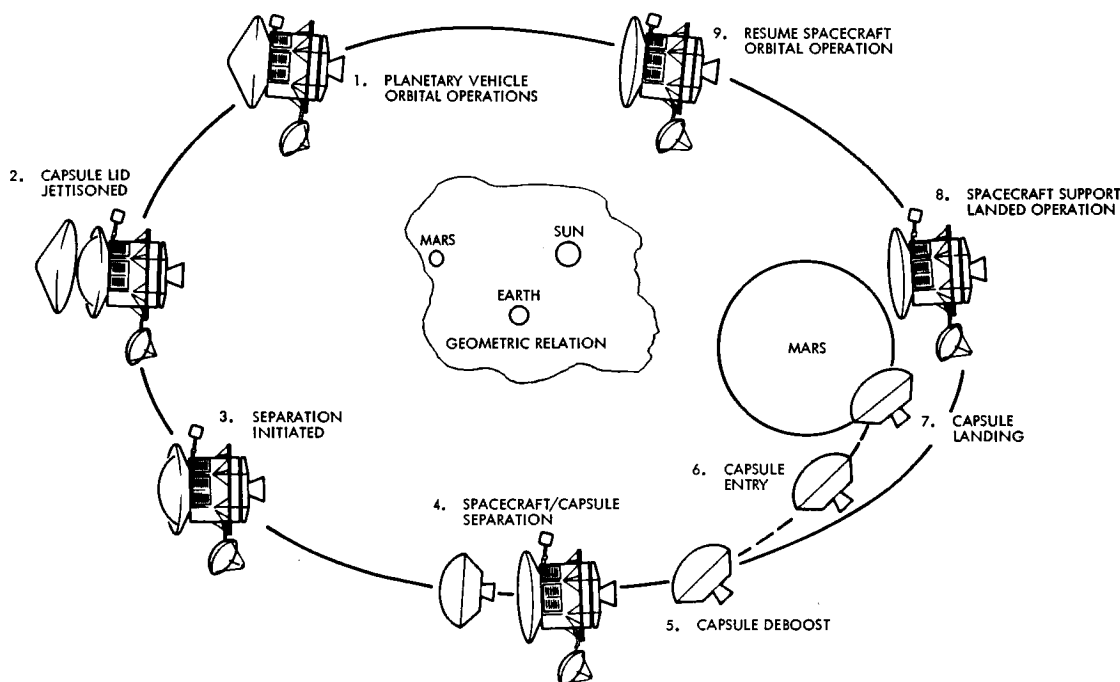


Figure 3-8

CAPSULE SEPARATION AND SUPPORT SEQUENCE occurs after mapping and selection of landing site. Instrumentation within capsule is energized, checked, Capsule then separates for de-orbit and descent to Martian surface.

capsule transmits data via a UHF radio link to the spacecraft (Figure 3-10). A low data rate channel from the capsule is transmitted to earth by the spacecraft in real-time, while a high data rate TV channel is recorded on board the spacecraft for later playback and transmission. After commencing surface operations the capsule will activate a direct radio link to earth, with the capsule-spacecraft relay link as a possible backup if required.

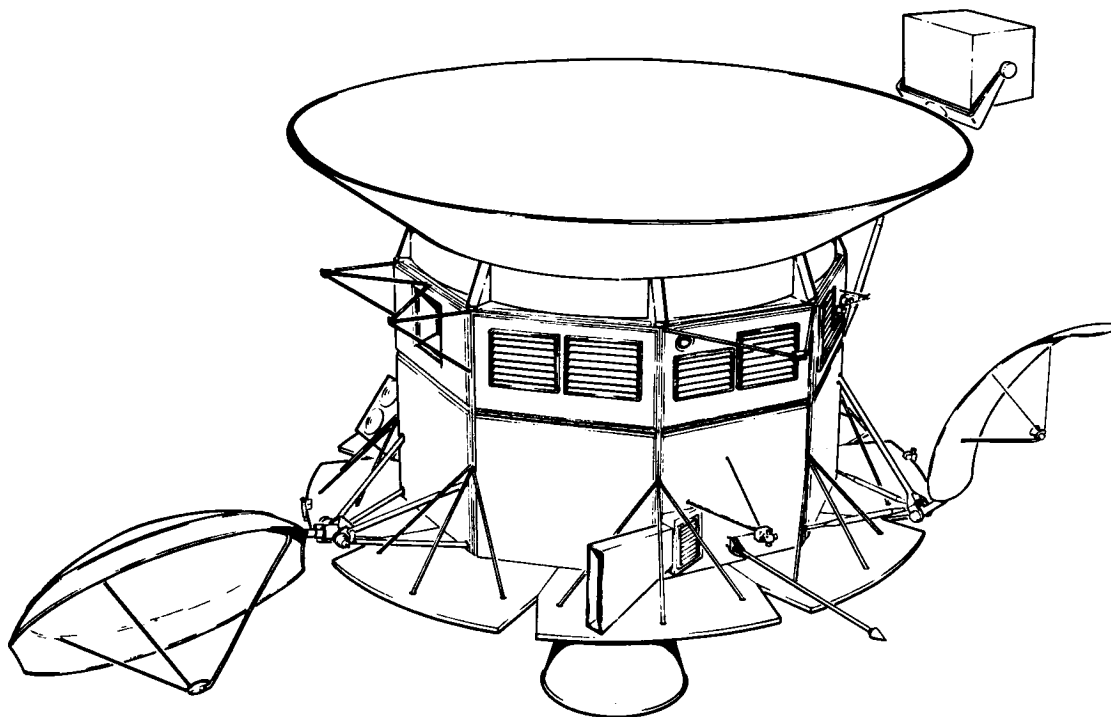


Figure 3-9

SPACECRAFT IN ORBIT deploys the planetary scan platform to its ultimate position for unobstructed view of Mars. An orbit trim maneuver may be executed to enhance the scientific mission return.

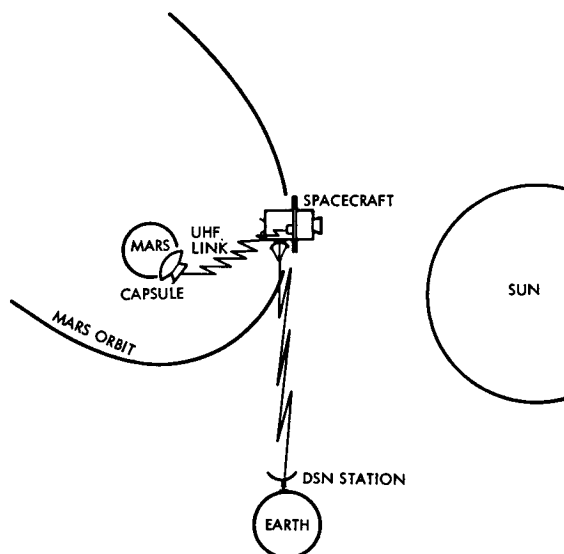


Figure 3-10

POST-SEPARATION CAPSULE SUPPORT consists of relaying data to earth until the capsule is able to establish its own communication link with earth. The spacecraft can continue to relay capsule data to earth if necessary.



3.1.1.4 Spacecraft Orbital Operations

For postlanding operations the spacecraft will continue with its orbital science mission. An additional orbit trim may be required to achieve a more suitable orbit. Extensive surveys from orbit will be carried out and correlated with results from the surface investigations by the capsule. The most important science experiment is photo-imaging, which will map the surface of Mars in low resolution (sometimes in color), with some areas to be photographed monochromatically in medium and high resolution. Spectrometric measurements will examine the atmosphere for water vapor and other constituents and determine altitude dependence of density and temperature. Atmospheric and ionospheric scale heights will be measured by the occultation of the radio link between the spacecraft and earth. The surface temperature will be mapped radiometrically and Rayleigh scattering in the lower atmosphere along with ultraviolet reflectivity of the surface will be determined.

3.1.2 Spacecraft Mission Requirements

The overall design problem specified in the Statement of Work was to define a single recommended spacecraft preliminary design that would be capable of performing the 1973 Voyager Mars mission as described in the JPL 1973 Voyager Mission Specification (Ref. 1). Additional guidelines were later provided by MSFC to take precedence over the information contained in the 1973 Voyager Mission Specification (Ref. 2, 3, 4). Particular mission requirements and study guidelines are summarized below to the extent they are applicable to spacecraft definition.

3.1.2.1 Project Guidelines

Overall guidelines that are applicable to the spacecraft on a project-wide basis are given below.

Launch Vehicle. The Saturn V with a suitable shroud will be the launch vehicle for Voyager Mars missions.

Launch Site. Complex 39 at Kennedy Space Center will be utilized for launching the Voyager space vehicle. Facilities of the AFETR will also be utilized as required.

Tracking, Telemetry, and Command. The Deep Space Net will be utilized to provide metric tracking, telemetry, data acquisition, and command coverage for the planetary vehicles from Mars transfer trajectory injection to the end of mission. The Manned Space Flight Net will also be utilized to support near-earth operations.

Operations. Launch operations will be controlled from the Launch Control Center at KSC. Space flight operations from Mars transfer trajectory injection to end of mission will be centrally controlled from the Space Flight Operations Facility at JPL.

Schedule. The 1973 Mars launch opportunity places a time constraint on the mission schedule; consequently, adequate schedule margins will be provided for all design, development, fabrication, testing, and deliveries are to conform to the established schedule milestones.

Adaptability. The spacecraft is to perform the 1975, 1977, and 1979 Mars missions with a minimum of hardware changes from the 1973 system.

Planetary Quarantine. The numerical allocation of allowable probability of contaminating Mars is 3×10^{-5} for each unsterilized spacecraft, including ejecta therefrom. This allocation is to include a pro-rated allocation for unsterilized items such as launch vehicle stages and adapters.

Provisioning. Three qualified spacecraft flight articles will be provided for the 1973 Mars launch opportunity along with appropriate qualified spares and all other required spacecraft system supporting elements. The three spacecraft will be essentially identical and interchangeable, and all three will be processed for flight.

Competing Characteristics. In the event of technical conflicts affecting mission design and operations, the relative priorities, in decreasing order of importance, are as follows:

- 1) Achievement of a planetary vehicle Mars orbit insertion
- 2) Achievement of a capsule landing
- 3) Performance of entry science experiments
- 4) Performance of orbital science experiments
- 5) Performance of landed science experiments



- 6) Contribution of technology and experience pertinent to subsequent missions
- 7) Design compatibility with future mission requirements
- 8) Achievement of interplanetary science experiments
- 9) Cost savings
- 10) Provision of additional 1973 mission capability beyond that specified here
- 11) Provision of alternate mission capability for exploration of planets other than Mars

3.1.2.2 Mission Operations

Requirements related to how the mission is implemented are given below.

Launch Mode. Two planetary vehicles are to be launched on a single Saturn V for each Mars launch opportunity.

Launch Period and Window. The launch period for Voyager missions is to be at least 20 days. The minimum daily firing window will not be less than 1 hour.

Arrival Date Separation. A propulsive maneuver to separate their arrival dates is to be carried out by each planetary vehicle at selected times between 2 and 20 days after injection. The maneuvers will separate the arrival times of the planetary vehicles by eight days for the 1973 mission and four days for later missions. The maneuver will also correct for the aiming point bias applied to the initial Saturn V injection trajectory and injection dispersion.

Interplanetary Trajectory Corrections. Up to two trajectory-correction propulsive maneuvers may be necessary in addition to the arrival date separation maneuver.

Capsule Delivery. The capsule will be taken into Mars orbit as part of the planetary vehicle prior to its separation and descent to the Martian surface.

Mars Orbit Insertion. The planetary vehicle will be placed into a suitable Mars orbit by a deboost maneuver utilizing spacecraft propulsion.

The insertion maneuver will be similar to the trajectory midcourse maneuvers. It is required that the capability be maintained to monitor and abort the maneuver if nonstandard events occur that would violate the planetary quarantine condition. In particular, the system will have the capability for verifying commands and that the desired planetary vehicle orientation has been achieved prior to the commitment of propulsion ignition. Updating of maneuver parameters (firing time, direction, and magnitude) is to be possible up to the closest point to firing that is practical, within monitoring constraints. The orbit insertion maneuver is to occur in view of the Goldstone DSIF station.

Mars Orbit Trim. The capability will be provided for accomplishing orbital trim maneuvers both before and after capsule separation.

Mars Descent. The capsule separation maneuver will nominally be performed between 3 and 12 days after orbit insertion. However, the capability is to be provided to support and separate the capsule for up to 30 days following orbit insertion.

The capsule bus de-orbit velocity increment capability will be consistent with landing on Mars within 300 km of the planetary vehicle orbit plane at any site selected within a central angle of at least 30 degrees subtended at the center of Mars.

Landing Site. Capsule landing sites for the 1973 Voyager missions will lie between 10 degrees north latitude and 40 degrees south latitude. In order to achieve the preferred lighting geometry for TV experiments during capsule descent, the capsule impact point is to lie between 15 and 30 degrees from the terminator. The capsule landing site will be selected so that it will be possible for the orbiter to obtain high resolution pictures of the region within 600 km of the specified capsule landing site and, if feasible, under lighting conditions similar to those experienced by the capsule. The deviation of the actual capsule landing site from the specified landing site is not to exceed 500 kilometers (3σ), with a design goal of 300 kilometers (3σ).

Termination of Mission Operations. Formally scheduled 1973 Voyager mission operations are to terminate by January 1975; mission



operations after this date will be limited to activities conducted on a noninterference basis with other Voyager missions and other programs.

Lifetime. Lifetime requirements for Mars operations are given below. Numbers in parentheses are design goals.

<u>Mission</u>	<u>Orbital Operations Lifetime (months)</u>	<u>Surface Operations Lifetime</u>
1973	2 (6)	1 day
1975	6	2 (6) months
1977	6 (12)	6 (12) months
1979	12	12 months

3.1.2.3 Spacecraft Sizing

General Philosophy. Spacecraft sizing is to take into account a balanced allocation of total performance capability between launch vehicle injection energy level and the desirability of increased spacecraft maneuver capability at the planet. That is, requirements for earth injection energy and deboost propulsive capability at Mars are not specified individually; rather, these are to be derived so as to enhance useful weight in orbit and orbit adjustment capability for the series of missions 1975 through 1979, within the trajectory constraints specified.

Injection Energy at Earth. The launch vehicle capability for payload versus injection energy at earth is given in Section 3.2.1.

Capsule Weight. Two mission cases are to be considered:

	<u>Capsule Weight (lb)</u>	
	<u>1973</u>	<u>1975-77-79</u>
Case A	5000	5000
Case B	6000	7000

Spacecraft ΔV Capability. The spacecraft will provide a ΔV capability in keeping with the general sizing philosophy above. A minimum of 1.95 km/sec total ΔV capability will be provided. This will cover the arrival date separation maneuver, interplanetary correction maneuvers, Mars orbit insertion, and orbit trim maneuvers.

Standard Propellant Capacity. The spacecraft propellant tank capacity is to be the same for the 1973, 1975, 1977, and 1979 Mars mission opportunities.

3.1.2.4 Trajectory Constraints

Trajectory Type.* A Type I transfer trajectory will be utilized for the 1973 mission. A Type II transfer trajectory is allowed, if desirable, for subsequent launch opportunities.

Declination of Launch Asymptote. The absolute value of DLA is not to be less than 5 degrees or greater than 36 degrees.

Ascent Mode and Parking Orbit. The parking orbit ascent mode will be utilized for the Mars 1973 mission. The nominal parking orbit is circular at 100-nautical-mile altitude. The capability to coast in parking orbit between 10 and 90 minutes is to be provided.

Launch Azimuth and Declination of Launch Asymptote. The launch azimuth and declination of launch asymptote limits are to be as follows:

<u>Launch Opportunity</u>	<u>Azimuth Limits (degrees east of north)</u>	<u>DLA Limits (degrees)</u>
1973	90 - 115	5 - 36
1975	45 - 115	5 - 50
1977	45 - 115	5 - 50
1979	90 - 115	5 - 36

Mars Orbit. The periapsis altitude of the Mars satellite orbits selected are to lie between 500 and 1500 km. The apoapsis altitude of the selected orbits are to lie between 10,000 and 20,000 km. These ranges are applicable for planetary vehicle and spacecraft satellite orbits both before and after orbit trim. The satellite orbit period may be controlled near a subsynchronous value in order to provide contiguous ground tracks of the orbiter.

* See Section 3.3 for a definition of Type I and Type II trajectories.



The capability will be provided to change the location of the periapsis of the ellipse in the plane of the approach trajectory by at least ± 20 degrees from the periapsis location of the hyperbola having the same periapsis radius. The satellite orbit periapsis location selected will be near either the evening or morning terminator in order to provide suitable lighting conditions for photography.

3.1.2.5 Trajectory Accuracy

General Philosophy. An adaptive guidance policy is to be employed during the flight to ensure that the final mission accuracy requirements are met. By adaptive is meant that the exact number and location of interplanetary trajectory corrections cannot be specified precisely before flight, but will depend upon prior dispersions, trajectory biasing needed to satisfy the quarantine requirements, maneuver size, operational considerations, and a number of other factors. The decision to perform a midcourse correction will be based primarily upon the difference between the predicted and nominal approach aiming intercepts in the $\bar{R}$, $\bar{T}$ plane as estimated at some time prior to Mars encounter.

As a design goal, the trajectory inaccuracy contributed by the spacecraft should have only a modest degrading effect over that due to DSIF tracking errors or other causes alone.

Launch Vehicle Injection Accuracy. Miss plus time-of-flight dispersions (1σ) of each planetary vehicle resulting from launch vehicle injection errors are to be correctable with an interplanetary trajectory correction velocity increment of 5 m/sec or less applied 2 days after injection.

Trajectory Perturbations. The unpredictable translational accelerations originating in the planetary vehicle are not to exceed a total average value of $0.6 \times 10^{-7} \text{ cm/sec}^2$ averaged over one hour. The design goal is not to exceed $0.2 \times 10^{-7} \text{ cm/sec}^2$. These values correspond to 3σ bounds and are to be interpreted that the actual errors must remain less than the bounds prescribed 99.7 percent of the time.

Spacecraft Accuracy Requirements. Spacecraft velocity increment accuracy requirements are given in Table 3-1. The requirements for command timing are given below.

Table 3-1. Velocity Increment Accuracy Requirements

Maneuver	Minimum Velocity Increments (m/sec)	Maximum Velocity Increments* (m/sec)	3 σ Error Component Parallel to Specified Velocity Increment (m/sec or percent of specified velocity increment)	3 σ Error Component Normal to Specified Velocity Increment along any two Orthog- onal Axes (m/sec or percent of specified velocity increment)
Interplanetary trajectory corrections	1.0 (0.3)	100 (200)	larger of 0.1 m/sec or 3.0 percent (larger of 0.03 m/sec or 2.0 percent)	larger of 0.1 m/sec or 3.0 percent (larger of 0.03 m/sec or 2.0 percent)
Mars orbit insertion	1.0 km/sec	2.0 km/sec	3.0 percent (1.5 per- cent)	5.0 percent (3.0 per- cent)
Mars orbit trim	5.0 (1.5)	150	larger of 0.5 m/sec or 5.0 percent (larger of 0.2 m/sec or 3.0 percent)	larger of 0.5 m/sec or 5.0 percent (larger of 0.2 m/sec or 3.0 percent)

*For purpose of error calculations only.

Note: Numbers not in parentheses are maximum values.
Numbers in parentheses are design goals.



<u>Command</u>	<u>3σ Error (sec)</u>	
	<u>Maximum</u>	<u>Desired</u>
Arrival date separation ignition	To be specified	
Interplanetary trajectory correction ignition	To be specified	
Mars orbit insertion ignition	To be specified	
Mars orbit trim ignition	30	To be specified
Capsule separation release signal	25	10

3.1.2.6 Spacecraft Science

The spacecraft is to provide support for a variety of science instruments for both planetary and interplanetary experiments. In particular, the spacecraft for the 1973 mission will accommodate the following hypothetical science payload.

- Photo-imaging equipment—all television system
 - a) One low resolution color television camera (1000 m resolution)
 - b) Three medium resolution television cameras (100 m resolution)
 - c) One high resolution television camera (10 m resolution)
- "Core" instruments for measurements of Mars surface and atmospheric constituents
 - a) High resolution infrared spectrometer
 - b) Broadband infrared spectrometer
 - c) Infrared radiometer
 - d) Ultraviolet spectrometer

Additionally, measurements of the Martian atmospheric properties (ionosphere) can be made using the shifts in signals from the spacecraft communication system during earth occultation periods.

3.1.2.7 Communication Requirements

The spacecraft system is required to operate in conjunction with the tracking and data acquisition system to provide tracking data, scientific and engineering telemetry data, and to receive commands throughout the various mission phases. These will be accomplished by means of a coherent two-way S-band radio link between the spacecraft and the DSIF.

Frequency. The earth-planetary vehicle uplink will employ a frequency in the 2115 ± 5 MHz range while the planetary vehicle-earth downlink will utilize a frequency in the 2295 ± 5 MHz range. The two spacecraft are to operate independently of each other and thus have different frequencies within the above mentioned bands.

Tracking. Tracking will be accomplished using the doppler shift on the carrier to provide range rate and the DSIF PRN ranging signal as modulation on both the uplink and downlink carriers for range. Thus, the spacecraft must provide for coherent carrier and ranging signal turn-around.

Commands. Commands will be sent to the vehicle by modulating the uplink carrier with a digital pulse train. No more than 30 minutes (30) will be required to lock the spaceborne decoder to the modulated uplink after carrier acquisition. Nominal lock-up times are to be substantially less. Commands are to be received with a bit error rate probability of less than 10^{-5} for all mission phases.

Telemetry. Telemetry will be sent to the DSIF by modulating the downlink carrier with a digital pulse train. During interplanetary cruise the spacecraft is to be capable of transmitting at least 2.5×10^6 bits of science data to earth each day. At the beginning of the pre-separation orbital operations phase, the spacecraft is to be capable of transmitting at least 6.5×10^8 bits of science data to earth per day. The spacecraft is to provide sufficient storage for data to ensure compatibility among engineering data handling requirements, science data handling requirements, and telemetry data transmission rates. No loss of data during the mission is permitted. The telemetry link will convey its data with a bit-error probability of less than 5×10^{-3} .



3.1.3 Comparison of 1971 Mission Requirements with Those for the 1973 Mission

One of the contractually expected results for the study is a comparison of the requirements and constraints imposed upon the spacecraft for a 1971 Mars opportunity with those applicable for the 1973 mission. For this purpose, the mission requirements for the 1971 mission are taken to correspond to guidelines and direction serving as the basis for the Voyager Spacecraft Task B Study (Refs. 6, 7, 8). Since the completion of the Task B study, there have been a number of changes which resulted in a need to update the Task B design. These changes are either related to changes in ground rules, to differences in solar system geometry between the 1971 and 1973 opportunities, or to the development of new information. This latter can often have the effect of changing ground rules, as interest is focused on a particular design approach.

3.1.3.1 Changes in Ground Rules and New Information

As would be expected, the Voyager ground rules are closely related to the current mission approach and project definition. As more understanding of the Mars exploration problem has developed in the intervening period since Task B, this has led to changes in the mission approach, with related changes to spacecraft requirements and constraints. Table 3-2 indicates in particular these changes for the 1973 mission, with all other 1971 requirements and constraints (Ref. 7) continuing to be applicable.

An important change in approach is the emphasis on the series of Voyager missions rather than just the initial mission. This calls for the spacecraft to be essentially standardized and designed for the series of missions rather than for a single flight. An associated feature is the need for propellant loading flexibility over the series of missions, and this consideration has had a significant effect on guidelines for propulsion. This is closely related to the overall spacecraft sizing problem, which is quite different for the 1973-1979 mission as compared to sizing for 1971 alone. (See Volume 6.)

Table 3-2. Ground Rule Changes

	1971 Mission		1973 Mission	
	Mission Description (Ref. 6)		Mission Spec (Ref. 1)	MSFC Additional Guidelines
1. <u>Adaptability to later missions</u>	Not required		Standardized for all missions	Minimum modifications and standardized propulsion hardware
2. <u>Weight Allocations (lb)</u>				
Spacecraft plus adapter	19,000		17,000	Spacecraft sizing based on launch vehicle capability
Capsule: Initial	3,000		5,000	Case A: 5000 Case B: 6000
Later	10,000		5,000	5000 7000
3. <u>Velocity Increments</u>				
Midcourse maneuvers ΔV (m/sec)	200		210	As required
Mars orbit insertion (km/sec)	2.0 deg required; 2.2 goal		1.76 required	As required
Orbit trim (m/sec)	100		150	As required
Total ΔV required (km/sec)	2.3		2.12	1.95 minimum
4. <u>Trajectory</u>				
Max C_3 (km^2/sec^2)	25		32.5	C_3 vs payload specified
Max V_3 at Mars (km/sec)	4.5		3.25	As required
Declination launch asymptote (deg)	5 to 38.4		5 to 36	
Launch azimuth limits (deg E of N)	60 - 115		90 - 115	
Arrival date separation (days)	10		8	
Minimum launch period (days)	45		30	20
Minimum daily launch window (hr)	2		1	
5. <u>Gross Launch Vehicle Payload</u>				
Capability (lb)	63,000		55,300	Net injected payload given vs C_3
($115^\circ A_z$, $C_3 = 32.5 \text{ km}^2/\text{sec}^2$)				
6. <u>Mars Orbit</u>				
Periapsis (km)	Not specified		500 to 1500	
Apoapsis (km)	Not specified		10,000 to 20,000	
Apsidal rotation (deg)	Not specified		20	
7. <u>Launch Vehicle Injection Accuracy (m/sec)</u>	10		5	
(ΔV correction at 2 days after injection)				
8. <u>Orbiter Lifetime (mo)</u>				
Initial	6		2 required, 6 goal	
Later	6		12	
9. <u>Planetary Contamination Probability</u>				
Total	10^{-4}		10^{-3}	
Spacecraft	5×10^{-5}		3×10^{-5}	
10. <u>Configuration</u>				
Dynamic envelope	208 in. long, no capsule identification		181 in. long, 34 in. capsule identification	Details of aft bulkhead specified
Capsule interface diameter (in.)	120		160	
Shroud concept	Nose cone separated at 350 ft, shroud to fold back to expose aft PV		Nose cone taken into parking orbit, over-the-nose concept	
Length of shroud cylindrical section (in.)	648		540	
Capsule emergency jettison	Required		Not specified	Not required
11. <u>Propulsion Options</u>	Use LMDS, Transtage, or solid with minimum modification		Not specified	Use LMD engine
12. <u>Capsule Support</u>				
Maximum capsule - spacecraft data rate (bits/sec)	50,000		Not specified	
Spacecraft-mounted capsule support (watts) (lb)	Not specified		50	
Pre-separation reconnaissance	Not required		50	Site selection data required
13. <u>Science</u>				
Total payload weight (lb)	400		400	Not limited to 400
Weight instruments on PSP (lb)	100		Not specified	
Weight instruments on bus (lb)	140		Not specified	
Spacecraft magnetic properties control to accommodate magnetometer	Required		Not specified	Not included; possible additional experiment
14. DSIF System Noise (210 ft antenna)* (deg K)	30 ± 5		45 ± 10	
15. Science Data Storage	5×10^7 required 5×10^8 goal, recorders specified		Sufficient to avoid loss of data	

* Achievable data rate in 1973 reduced by approximately 35 percent over 1971 for same ERP



An additional aspect that has come into focus during the intervening period since Task B is the indication that higher weights in orbit are achievable, providing that adequate propellant tankage is used. This information has coupled with other considerations to affect vehicle weight allocations.

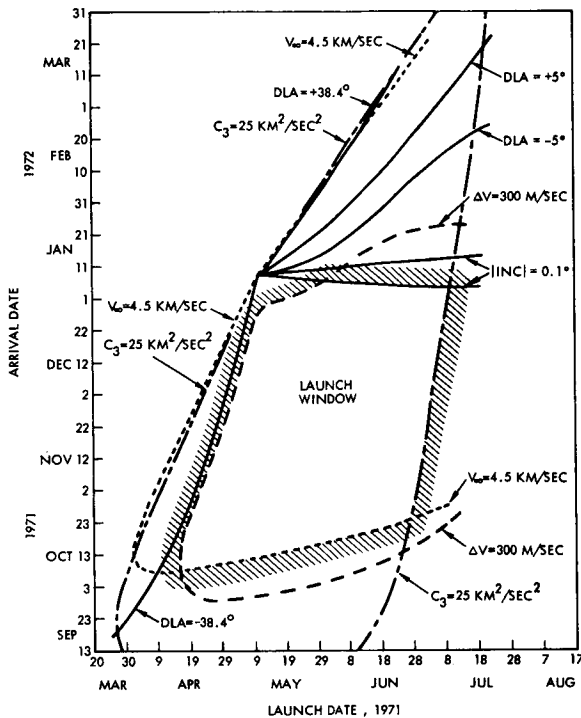
Another significant factor relative to spacecraft design is the removal of the Task B restriction (Ref. 6) to the Lunar Module Descent Stage, Transtage, or (Minuteman Stage II) solid rocket as possible design options. These were to be used with minimum modification, and the LMDS was selected by TRW as the most attractive of the three. The modified LMD engine was retained as the main engine for the current study along with some general features of the Task B design, but the structure and tankage have been optimized for the special considerations of the Voyager mission.

In addition to changes in the mission and project approach as noted above, another aspect of new information is advancement in state-of-the-art technology. An example of this is the development of improved solar cell mounting techniques (developed at TRW for the Intelsat program), which lower the allowable low temperature limit. This then may be reflected in a reduced solar array area requirement since back insulation of the array is no longer required to survive eclipse. The resulting lower operating temperatures improve the cell efficiency significantly.

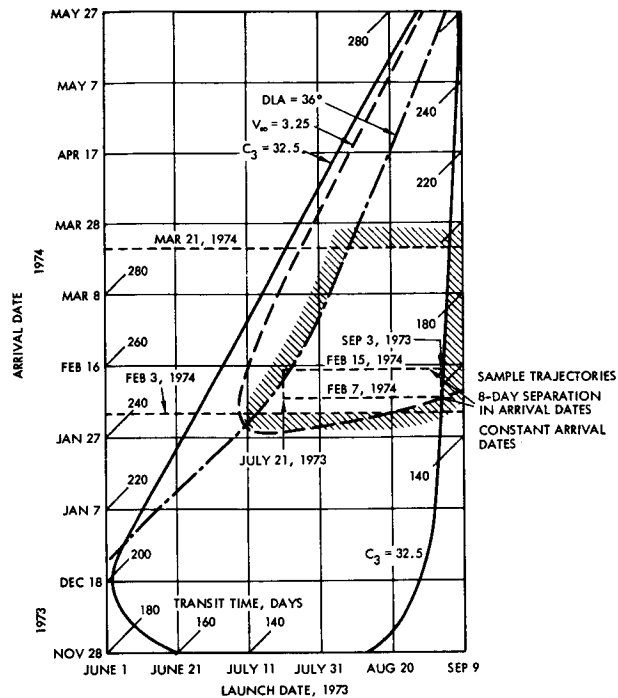
3.1.3.2 Trajectory Considerations

A comparison of trajectory requirements and constraints is shown in Table 3-2. Those for the 1971 mission description and the 1973 mission specification are shown graphically in Figure 3-11. The final 1973 constraints, in keeping with MSFC guidelines, subsume the trajectory capability question within the broader problem of spacecraft sizing. The corresponding trajectory discussion is given in Section 3.3 and a presentation on spacecraft sizing given in Volume 6.

Although the constraints have changed from those previously applied to the 1971 opportunity, it is possible to draw some general conclusions relative to differences in solar system geometry between



Constraints on Launch Opportunity,
Earth-to-Mars, 1971



Constraints on Launch Opportunity,
Earth-to-Mars, 1973

Figure 3-11

INTERPLANETARY TRAJECTORY CONSTRAINTS have changed significantly from the 1971 mission description to the 1973 mission specification, and to the current approach. In addition, solar system geometry is changed so that the 1973 opportunity, although relatively favorable, is considerably less attractive than that for 1971.

1971 and 1973. Although 1973 is a relatively favorable launch opportunity, that for 1971 is even more favorable. The following general comparisons can be made for 1973 versus 1971:

- Flight times are substantially increased in 1973 compared with 1971.
- Communication distances between the spacecraft and earth are increased markedly at the time of arrival, at arrival plus one month, and at arrival plus six months. The minimum communication distance at arrival may increase by more than double for 1973.
- The spacecraft distance from the sun at arrival at Mars is increased noticeably. This is because Mars is farther from its perihelion at the time of 1973 arrivals than for 1971 arrivals. For all the 1973 trajectories, Mars passes through aphelion a maximum distance of 1.666 AU from the sun between one and six months after arrival.



- The declination of the launch asymptote at earth departure has the same range of magnitudes in 1973 as in 1971, but they are all northerly rather than southerly. This means that, in contrast to 1971, 1973 departures will be characterized by substantially longer coast times in parking orbit and injection loci over the Indian Ocean and Australia, rather than over the Atlantic.

3.2 OTHER VOYAGER ELEMENTS

The nature and quality of the 1973 and later Voyager missions are, of course, not attributable solely to the capabilities of the spacecraft system. The various aspects of mission performance which are assessed in the following portions of Section 3 and in Section 4 are due to the combined qualities of all the elements (or "systems") of the project. While the presentation emphasizes those aspects of performance which are most strongly influenced by the characteristics of the spacecraft system design, it is appropriate to list the qualities, capabilities, and requirements which are assumed embodied in the other project elements.

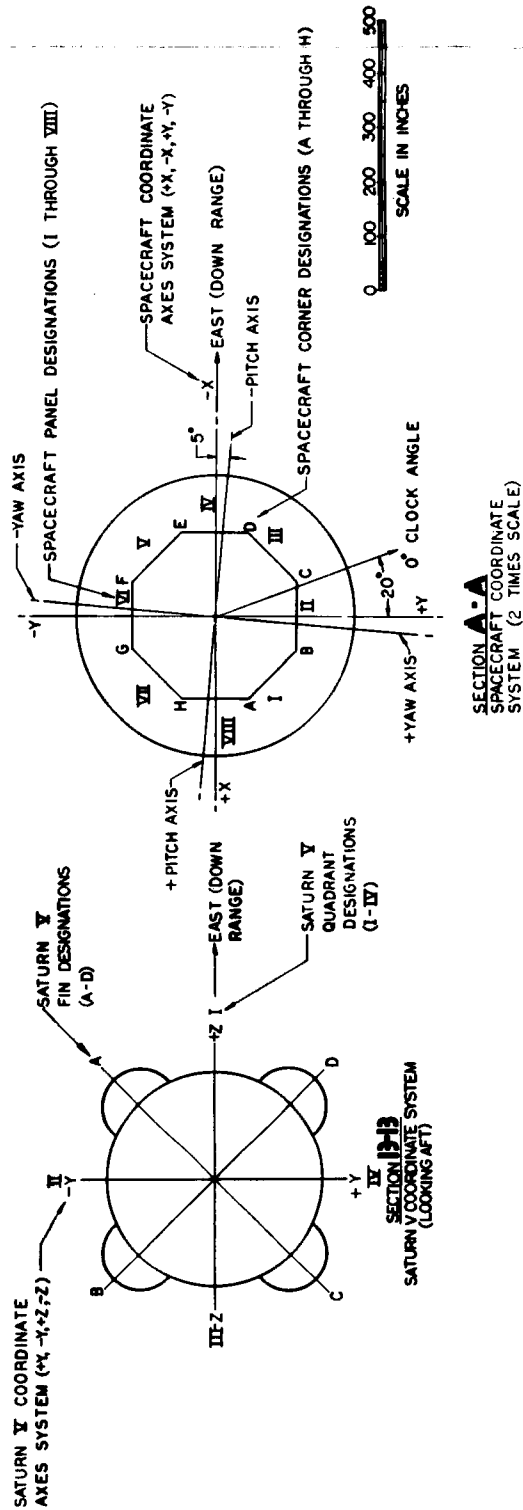
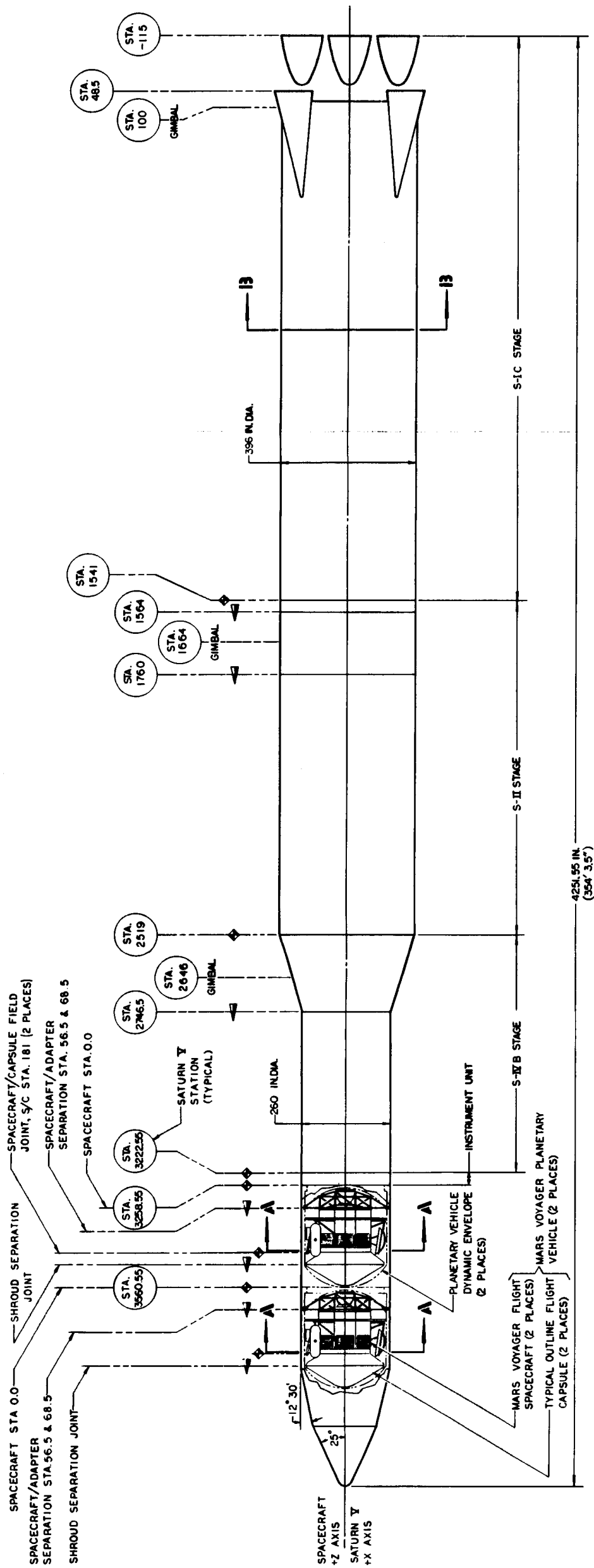
This listing is not intended to represent exhaustively or comprehensively all aspects of the specification of these other Voyager systems, or even of their interfaces with the spacecraft. It does call attention to the major characteristics assumed to be met by these other systems in order to assess the effect of the spacecraft design on the performance of the Voyager mission.

3.2.1 Launch Vehicle System

The launch vehicle system includes the Saturn V, Voyager shroud, launch vehicle OSE, the launch vehicle facilities, and the associated personnel. The Saturn V consists of the S-IC stage, S-IC/S-II interstage, S-II stage, S-II/S-IV interstage, S-IVB stage, and instrument unit. The space vehicle on-stand configuration showing the two planetary vehicles installed in the shroud is shown in Figure 3-12 along with the relation between spacecraft and launch vehicle reference axes.

3.2.1.1 Payload Capability

The primary attribute of the launch vehicle is the payload weight it can deliver onto a departure trajectory of given geocentric energy.



NOTES:
 ◆ FIELD SPICE
 ▲ SEPARATION
 ALL DIMENSIONS ARE IN INCHES

Figure 3-12
 THE SPACE VEHICLE ON-STAND CONFIGURATION is shown along with its reference axes and coordinates.



This capability is defined in Figure 3-13 (Ref. 2), which shows payload weight as a function of injection energy, C_3 (the square of the geocentric asymptotic departure velocity), for a launch azimuth of 115 degrees

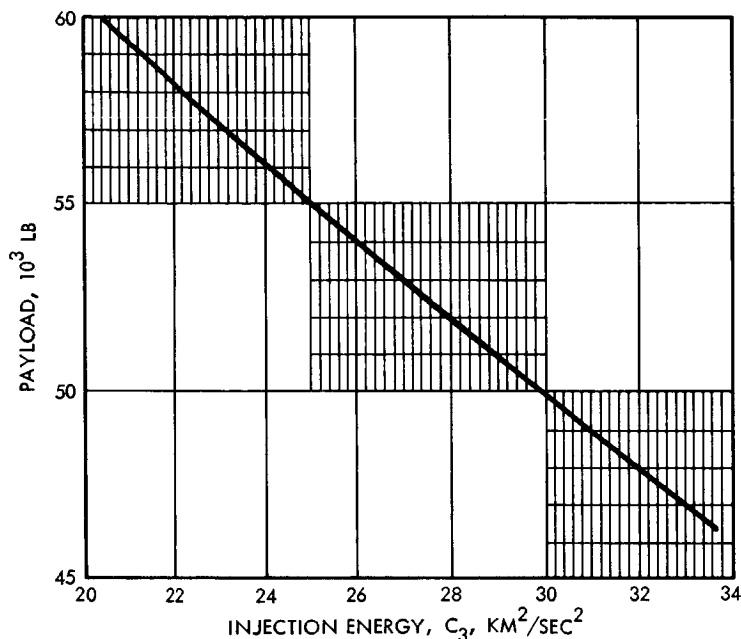


Figure 3-13

SATURN V PAYLOAD CAPABILITY. Launch azimuth is 115 degrees. Payload excludes the shroud, whose weight has been accounted for in this curve, but includes both planetary vehicles, both adapters, and a 5000-pound contingency.

from Kennedy Space Flight Center. Figure 3-14 shows how this capacity varies with launch azimuth, for the fixed value of $C_3 = 32.5 \text{ km}^2/\text{sec}^2$. Payload is defined as the weight of both planetary vehicles, both planetary vehicle-launch vehicle adapters, plus a contingency of 5000 pounds. (The weight penalty arising from carrying the shroud—the nose section to parking orbit, and the cylindrical section to injection—has already been accounted for by Figure 3-13.)

Subtracting 6300 pounds from the ordinates of Figure 3-13 and dividing by two accounts for the adapters at 650 pounds each as well as the 5000 pounds contingency, and leads to Figure 3-15 giving the weight of each separated planetary vehicle versus C_3 .

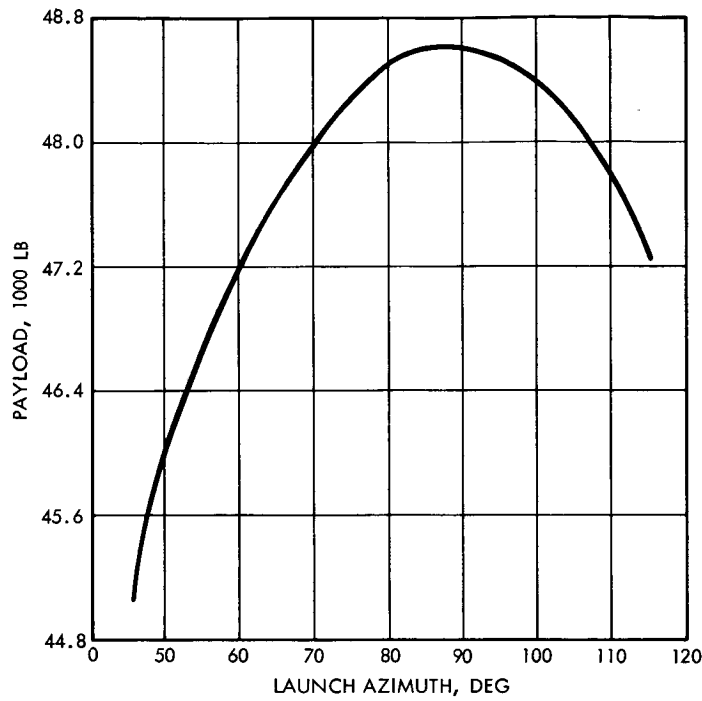


Figure 3-14

VOYAGER PAYLOAD VERSUS LAUNCH AZIMUTH. Voyager payload as defined in Figure 3-13 varies with the launch azimuth as shown, $C_3 = 3.25 \text{ km}^2/\text{sec}^2$.

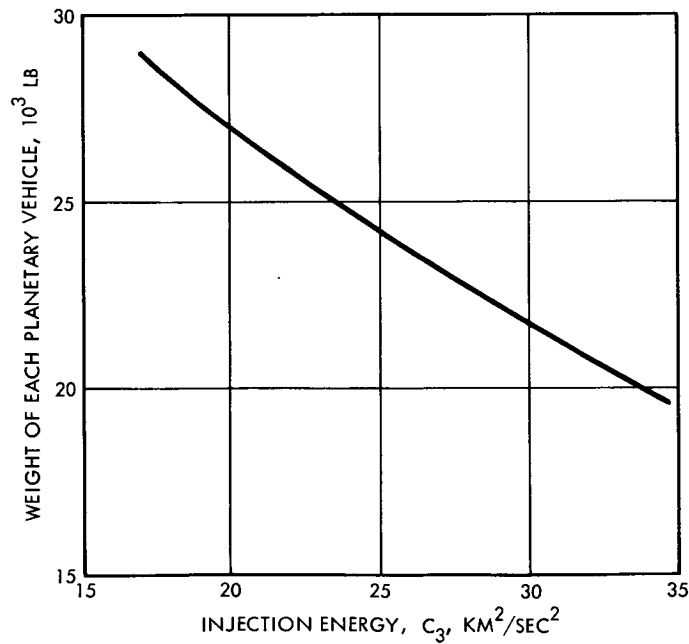


Figure 3-15

WEIGHT OF EACH PLANETARY VEHICLE launch azimuth is 115 degrees. Two adapters at 650 pounds and a contingency have been deducted from the capacity shown in Figure 3-13.

3.2.1.2 Injection Accuracy

One sigma dispersions (miss and time of flight) of each planetary vehicle resulting from launch vehicle injection errors are to be correctable with an interplanetary trajectory correction velocity increment of 5 m/sec or less, applied two days after injection (Ref. 1).

3.2.1.3 Parking Orbit Coast Time

To inject the Voyager planetary vehicles onto the desired interplanetary trajectories it is necessary to make use of an ascent mode incorporating a parking orbit. The launch vehicle is to have the capability to coast in parking orbit for times between 10 and 90 minutes (Ref. 1)

3.2.1.4 Shroud Geometry

The Voyager shroud consists of a nose fairing, two identical cylindrical sections and dust covers for encapsulating the planetary vehicles, and a cylindrical shroud spacer section. The geometry for the shroud and its elements is shown in Figure 3-16 (Ref. 3).

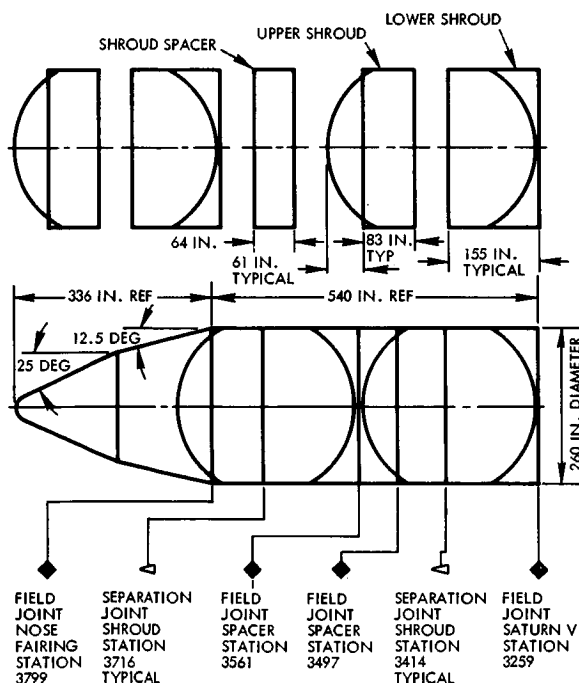


Figure 3-16
VOYAGER SHROUD GEOMETRY and the various elements are shown for the "over-the-nose" concept of shroud separation

3.2.1.5 Planetary Vehicle Dynamic Envelope

The envelope available within the shroud for the two planetary vehicles is shown in Figure 3-17 (Ref. 3).

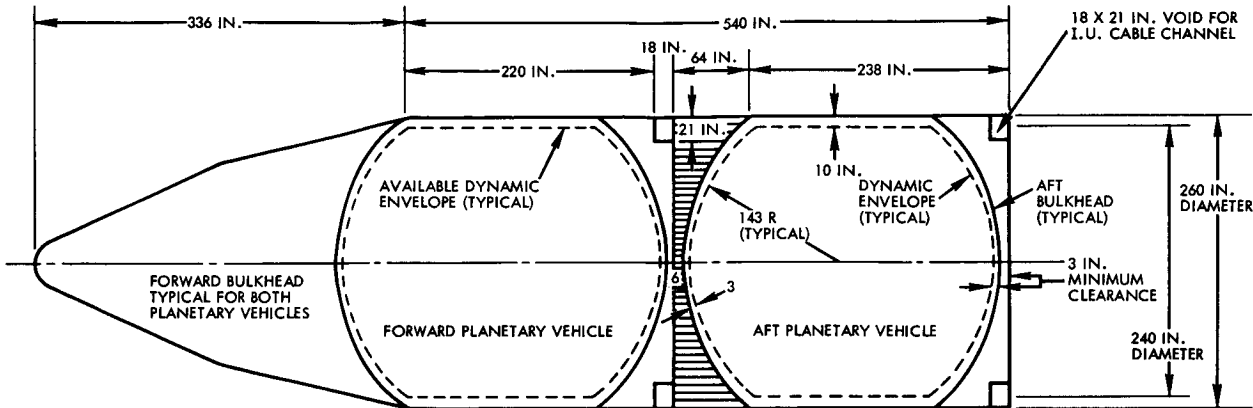


Figure 3-17

DYNAMIC ENVELOPES for the two Planetary Vehicles are identical. Clearance is allowed for all relative motion between the shroud and the installed vehicles

3.2.2 Launch Operations System

Many aspects of launch operations have a strong interaction with the spacecraft in the areas of spacecraft handling, assembly, and test; mating with the capsule and with the launch vehicle; and establishing flight readiness. Generally, these interactions do not have an easily definable influence on the operational (postlaunch) performance of the spacecraft.

However, operational constraints due to considerations of safety and earth-spacecraft communications during the launch phase do influence the mission performance. In this regard, the principal constraint which is imposed is the limitation of permissible launch azimuth as given in Section 3.1.2.4. The inclusion of northeast launches for the 1975 and 1977 opportunities is in recognition of the extreme launch azimuth declinations associated with Type I trajectories of those opportunities. If Type II trajectories (traversing more than 180 degrees heliocentric angle between earth and Mars) are employed, the more conservative 90- to 115-degree range can be adhered to.



3.2.3 Mission Operations System

The primary qualities of the mission operations system, which are assumed for the spacecraft performance assessments, are associated with a reaction capability to update spacecraft commands promptly, on the basis of data received on the ground. This capability permits:

- The planning of adaptive maneuvers, e.g., orbit insertion, in which the timing of the maneuver and the magnitude and direction of the velocity increment can be programmed based on the most recent estimate of the spacecraft trajectory
- The most effective detection of failures or non-nominal conditions and institution of backup modes, where ground analysis and/or command is necessary
- The greatest capability to detect and rectify a trajectory (attained as the result of a non-nominal maneuver) which threatens to violate the planetary quarantine constraint.

3.2.4 Tracking and Data Acquisition System

The tracking and data acquisition system is to provide telemetry data acquisition, tracking, command, and data handling support during mission operations. Existing and planned DSN facilities are to be utilized as defined in Reference 5, along with other NASA facilities and Department of Defense facilities, as appropriate. The interface between the spacecraft and the tracking and data acquisition system is the two-way S-band communication link, which is required to provide one-way and two-way doppler tracking, turnaround ranging, command, and telemetry data acquisition for the two planetary vehicles. The requirements imposed upon the TDAS are summarized in Table 3-3 and discussed below.

3.2.4.1 Support Coverage

Telemetry data acquisition, tracking, and command coverage is to be provided as follows (Ref. 1):

- 1) Launch Through Interplanetary Trajectory Injection. Provide metric tracking coverage and telemetry data acquisition coverage for the space vehicle. After injection, approximately 2 hours of S-IVB metric tracking coverage is required for S-IVB orbit determination.

Table 3-3. Tracking Data System Requirements

Mission Phase	Stations	Antenna and Mode	Nominal Gain		Transmitter Power	Receiver Noise Temperature	Remarks
			Transmitter (db)	Receiver (db)			
Prelaunch	Kennedy Space Center, No. 71	4-ft diplexed	23	25	5 watts	3,000°K	- Single-channel transmitter used for checkout* - Two-channel receiver required
Launch (L)	Kennedy Space Center, No. 71	4-ft diplexed	23	25	5 watts	3,000°K	Two-channel receiver required for both Stations 71 and 72
Parking Orbit through Injection (I)	Ascension Island, No. 72	30-ft diplexed	42	43		290°K	
	Ascension Island, No. 72	30-ft diplexed	42	43			Two-channel receiver required at each DSN site
	Johannesburg, No. 51	3-ft acquisition aid antenna only	19	21		300°K	Continuous coverage from DSN not possible in 100-nmi parking orbit
	Madrid, No. 61						Command through spacecraft command link not possible due to doppler and extended acquisition time
	Woomera, No. 41						
	Tidbinbilla, No. 42						Apollo/Saturn net needed for telemetry through S-IVB in parking orbit for emergency command if required
	Goldstone, No. 11/12						
Post-Injection to I + 30 days	Apollo Network						
	Ascension Island, No. 72	30-ft diplexed	42	43	10 kilowatts	290°K	Station No. 72 only until I + 8 hours, if in view
	Johannesburg, No. 51	acquisition aid and 85-ft dish					Simultaneous two-channel transmitter and receiver at each site
	Madrid, No. 61		19	21	10 kilowatts	300°K	Uplink is initially multiplexed via the acquisition aid (for Station 72) until I + 8 hours; after that, option exists for multiplexed operation or separate 85-ft sites for each spacecraft
	Woomera, No. 41		50.1	52.3		55°K	
	Tidbinbilla, No. 42						
	Goldstone, No. 11/12						Two-channel receiver on each antenna
I + 30 days to Encounter (E) - 5 days	Johannesburg, No. 51	85-ft dish					Time shared uplink required - 100 kw site only required after approximately I + 150 days for: a) turnaround ranging via spacecraft high-gain antenna, or b) for emergency command via spacecraft low-gain antenna
	Madrid, No. 61						Two-channel receiver on each antenna required
	Woomera, No. 41						
	Tidbinbilla, No. 42						
	Goldstone, No. 11/12						
E - 5 days to E + 8 days	Goldstone, No. 14	210-ft dish	59.8	60.9	100 kilowatts	45°K***	
	Johannesburg, No. 51	85-ft dish	50.1	52.3	10 kilowatts	55°K	Simultaneous two-channel uplink required only when both spacecraft are in their respective E - 5 to E + 8-day period (spacecraft have a minimum arrival separation of 10 days)
	Madrid, No. 61						85-ft sites continue to track second spacecraft from E - 5 days of first spacecraft to E - 6 hours of second spacecraft (approximately 15 days)
	Woomera, No. 41						
	Tidbinbilla, No. 42						
	Goldstone, No. 11/12						210-ft sites establish two-way communication with first planetary vehicle from E - 5 to E + 8 days (for first planetary vehicle) then switch to two-way communication with second planetary vehicle from E - 6 hours to E + 8 days (for second planetary vehicle) while continuing to receive from first planetary vehicle****
	Goldstone, No. 14	210-ft dish**	59.8	60.9	100 kilowatts	45°K***	
E + 8 days to End of Mission	Madrid and Australia						Two-channel receiver on each antenna
	Goldstone, No. 14	210-ft dish**	59.8	60.9	100 kilowatts ***	45°K***	- Time shared uplink utilized - Two channel receiver required at all sites

* Uplink not established until after injection.

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

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

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



- 2) Interplanetary Trajectory Injection to Injection Plus 30 Days. Provide continuous metric tracking telemetry data acquisition, and command coverage for each planetary vehicle. Approximately 1/2 hour of overlapping metric tracking coverage is required when a planetary vehicle is within view of two DSN sites.
- 3) Injection Plus 30 Days to Mars Encounter Minus 20 Days. Provide continuous telemetry data acquisition and command coverage for each planetary vehicle. Provide 12 hours of continuous metric tracking coverage every two days for each planetary vehicle during cruise. During each 12-hour period, approximately 1/2 hour of overlapping metric tracking coverage is required when a planetary vehicle is within view of two DSN sites. For planetary vehicle interplanetary trajectory corrections provide 5 days continuous metric tracking coverage prior to correction and 10 days continuous metric tracking coverage after correction.
- 4) Mars Encounter Minus 20 Days to Spacecraft-Capsule Separation. Provide continuous metric tracking coverage, telemetry data acquisition coverage, and command coverage for each planetary vehicle.
- 5) Spacecraft-Capsule Separation to Termination of Orbital Operations. Provide continuous telemetry data acquisition coverage and command coverage for each spacecraft. Provide continuous metric tracking coverage of every other orbit plus continuous metric tracking coverage during occultation experiments for each spacecraft. To get full spacecraft telemetry capability, the use of the 210-foot antenna is required.

3.2.4.2 Telemetry Data Acquisition

For compatibility with the mission requirements, each receiving station of the existing and planned DSN, including Stations 71 and 72, must be capable of receiving, processing, and recording data at two different RF frequencies, and at data rates from a few bits per second up to the peak total telemetry data rate from both planetary vehicles (slightly over 100 kbits/sec). Functional or simple block redundancy is to be provided whenever practical, including, as a minimum, redundant receivers, processing, and recording equipment. Each station must have the capability to receive signals which are either coherent or non-coherent with the uplink carrier.

In the coherent mode of operation, each station should also be capable of simultaneously receiving and demodulating telemetry data and

sync and the turnaround ranging code from both planetary vehicles, excepting only periods when the command link is operative.

3.2.4.3 Command

During prelaunch preparations at KSC, the integrity of the command link will require verification via the RF link, using DSIF Station 71. No command uplink is required after liftoff until after injection into the interplanetary trajectory.

From interplanetary trajectory injection to the end of the mission, each station of the DSN should be capable of transmitting commands, required for midcourse maneuvers, Mars orbit insertion, orbital sequence updating, spacecraft-capsule separation, and as backup for on board commands to both planetary vehicles sequentially. Addressing each planetary vehicle sequentially is accomplished by discrete RF uplink frequencies for each vehicle and a spacecraft address word at the beginning of each command.

3.2.4.4 Tracking Quality

The tracking data quality is shown in Table 3-4.

Table 3-4. Tracking Data Quality

Quantity	3 σ Error		Assumptions
	Required	Design Goal	
Two-way doppler (continuous count, range difference in meters)	1.0	0.1	$\Delta\rho$ accumulated over 10,000 sec at $\rho = 1.5$ AU (near zenith)
	0.01	0.001	Resolution of $\Delta\rho$ accumulated over 60 sec at $\rho = 1.5$ AU
Range (meters)	50	10	Absolute error at $\rho = 1$ AU (assuming c known)
	8	0.8	Difference between two range measurements taken 1 day apart
Angles (degrees)	0.03	0.01	Only used near launch. Total angle error for all error sources with correlation widths of ≤ 60 sec
	0.05	0.01	> 60 sec



3.2.4.5 Performance Requirements

In establishing spacecraft communications design, tracking and data system performance requirements in Table 3-5 have been assumed.

3.2.5 Capsule

The capsule system has a strong influence on mission definition and hence upon the spacecraft. Support of the capsule in carrying the capsule to Mars and into Mars orbit, reconnoitering possible landing sites, initiating separation, and relaying capsule data, are first order requirements affecting spacecraft design and operation. These aspects have generally been dealt with in Section 3.1 covering mission definition. Some additional considerations are given below.

3.2.5.1 Power Required from the Spacecraft

The spacecraft is to provide 200 watts of power to the capsule and 50 watts of power to spacecraft-mounted capsule bus support equipment. The spacecraft weight allocation includes 50 pounds for this capsule bus support equipment.

3.2.5.2 Attitude Control

At the time of separation of the capsule from the spacecraft, the spacecraft is in its cruise orientation. After separation, the capsule, using its own internal attitude references, executes the deorbit maneuver. The contribution of the spacecraft to the descent trajectory accuracy of the capsule is limited to the pointing accuracy of the spacecraft when capsule gyros are calibrated or uncaged, and when the capsule is separated, and to the accuracy of separation timing.

3.2.5.3 Weight

Two mission cases have been identified for capsule weight allocations as given in Section 3.1.2.3. A measure of successful spacecraft performance in a mission will be its ability to carry capsules of the weights given to Mars and into Mars orbit.

3.2.5.4 Capsule Envelope and Interface

The capsule is mounted forward of, and is structurally supported and transported by, the spacecraft. With the planetary vehicle in the

Table 3-5 . Tracking and Data System Performance Requirements

Item	Value
Transmitter power	
10-kw sites	70 dbm + 0.5 db - 0 db
100-kw sites	80 dbm minimum
Transmitter circuit loss	
10-kw sites	-0.4 db + 0.1 db
100-kw sites	-0.4 db $\pm$ 0.1 db
Transmitter antenna gain	
Acquisition aid	20 db + 2.0 db
85-foot dish	53 db $\pm$ 1.0 db - 0.5 db
210-foot dish	61.0 db $\pm$ 0.8 db
Transmitting antenna pointing loss	
Acquisition aid	0 db
85-foot dish	0 db
210-foot dish	-0.5 db
Receiving antenna gain	
Acquisition aid	21.0 db + 1.0 db
DSIF, 71	25.0 db minimum
85-foot dish	53.0 db + 1.0 db - 0.5 db
210-foot dish	61.0 db $\pm$ 1.0 db
Receiving antenna pointing loss	
Acquisition aid	0 db
DSIF, 71	0 db
85-foot dish	0 db
210-foot dish	-0.5 db
Ellipticity	
Acquisition aid	< 1.5 db
DSIF, 71	Linearly polarized
85-foot dish	0.7 db + 0.3 db
210-foot dish	0.3 db $\pm$ 0.2 db
Noise temperature	
Acquisition aid	270°K + 60°K
DSIF, 71	3000°K maximum
85-foot dish	55°K + 10°K
210-foot dish	45°K $\pm$ 10°K

normal sun-oriented cruise attitude, the capsule is shaded by the spacecraft. The capsule envelope and interface geometry is shown in Figure 3-18.

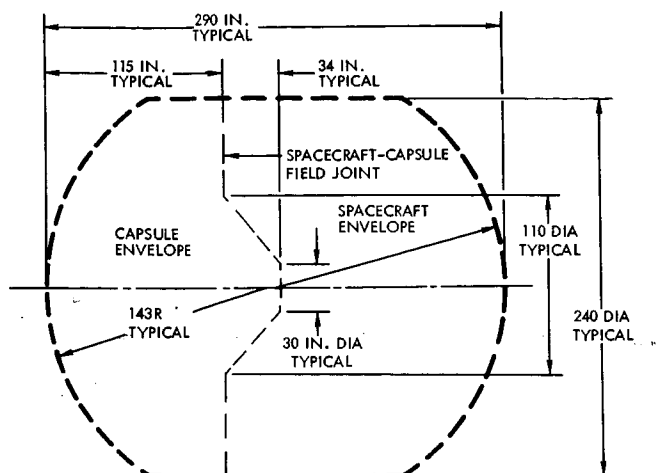


Figure 3-18
DIVISION OF THE PLANETARY VEHICLE DYNAMIC ENVELOPE into a portion for the capsule and a portion for the spacecraft.

3.3 INTERPLANETARY TRAJECTORIES AND MARS ORBIT CAPABILITY

3.3.1 Characteristics of Type I Earth-Mars Trajectories, 1973 Opportunity

Figure 3-19 displays the characteristics of the 1973 Type I earth-Mars trajectories. As will be shown later, Type I trajectories permit excellent spacecraft performance. The longer Type II trajectories were not evaluated for this opportunity.* Injection energy, C_3 , asymptotic approach speed at Mars, V_{∞} , and declination of the launch asymptote, DLA, are plotted as contours on a departure date-arrival date coordinate

* Type I trajectories are defined as those in which the planetary vehicle traces a central angle of less than 180 degrees about the sun between departure from the earth and arrival at Mars. In Type II trajectories, the angle is greater than 180 degrees. The two types are effectively noncontiguous, because when the central angle is very near 180 degrees, the position of Mars out of the ecliptic causes the interplanetary trajectory to be highly inclined to the ecliptic, leading to excessive launch energy requirements.

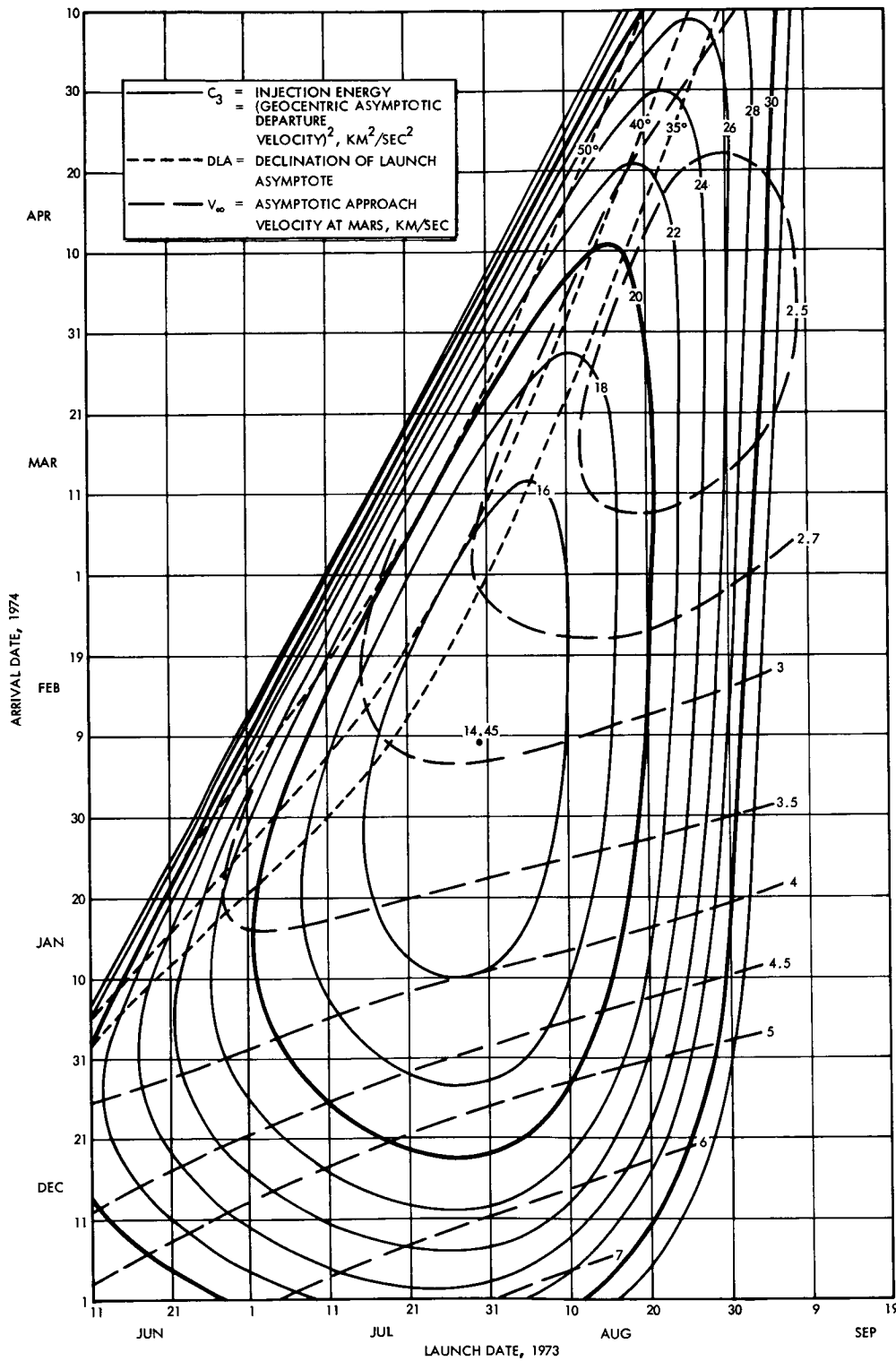


Figure 3-19
THE CHARACTERISTICS OF A TYPE I TRAJECTORY IN 1973 for earth-Mars transit include variations in injection energy, declination of launch asymptote, and asymptotic approach velocity at Mars with changes in launch and arrival dates.



system. A maximum allowable DLA of 36 degrees is permitted, corresponding to an extreme launch azimuth range of 90 to 115 degrees for a one-hour daily launch window.

3.3.2 Mission Maneuver Sequence

The performance capability of the recommended spacecraft is predicated on the maneuver sequences depicted in Figure 3-20. These maneuvers are described below in the order in which they are performed by the Voyager planetary vehicles.

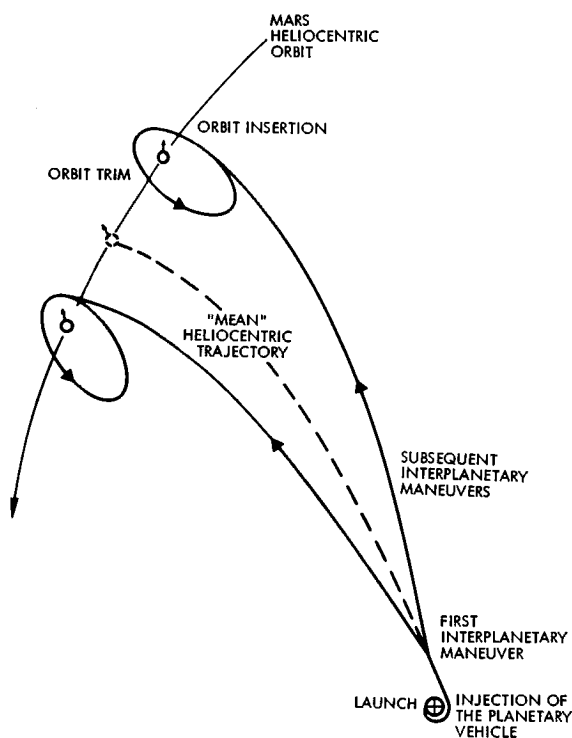


Figure 3-20
MANEUVER SEQUENCES OF THE TWO PLANETARY VEHICLES require each to depart onto separate transit trajectories from the initial (or "mean") trajectory on which they are placed at injection.

3.3.2.1 First Interplanetary Maneuver

Two planetary vehicles are injected by the Saturn V onto a "mean" heliocentric transfer trajectory directed close to Mars, but biased away so as to avoid (with a very high probability) an impact at Mars by the launch vehicle. This trajectory is denoted in Figure 3-20 with a dashed line. Two to five days after injection, the planetary vehicles are diverted

by propulsive maneuvers onto two separate trajectories so as to encounter Mars on separate days. The pertinent vector diagram is illustrated in Figure 3-40. The hyperbolic excess velocity difference at earth ($\Delta V_{\infty e}$) is divided equally between the two planetary vehicles. An injection bias and spherical injection error are vectorially combined with $(\Delta V_{\infty e})/2$ to give the total velocity increment required of each planetary vehicle for this maneuver

$$\Delta V = \sqrt{\left(\frac{1}{2} \Delta V_{\infty e}\right)^2 + (\text{Bias})^2} + 3\sigma,$$

where $\Delta V_{\infty e}$ = the vector difference between the V_{∞} vector velocities of geocentric departure of the two planetary vehicles. For the analyses of this section, the following relations were assumed:

$$\text{Bias} = 3.6\sigma$$

$$\sigma = 15 \text{ m/sec} - \text{figure of merit of the launch vehicle} \\ \text{(equivalent spherical injection error, determined two days after launch).}$$

(This value of σ is pessimistic, compared with the 5 m/sec of the JPL 1973 Voyager mission specification, leading to a conservatively high allocation to ΔV .)

3.3.2.2 Subsequent Interplanetary Maneuvers

An impulsive velocity allocation of fixed magnitude has been made for all subsequent interplanetary maneuvers, i.e., midcourse corrections.

3.3.2.3 Orbit Insertion

Both planetary vehicles are to be inserted into elliptic orbits at Mars arrival. The optimum orbit insertion maneuver—that requiring minimum velocity increment—is a cotangential periapsis transfer. Figure 3-21a illustrates the optimum orientation geometry. The common periapsis radius of the hyperbolic approach trajectory and the ellipse, together with the magnitude of the hyperbolic excess velocity at Mars arrival, determine the optimum orientation of the line of apsides of the ellipse with respect to the planetocentric asymptotic approach direction (the $-V_{\infty}$ vector, or $-\bar{S}$, of the Mars-centered $\bar{R}$, $\bar{S}$, $\bar{T}$ coordinates). The optimum orientation angle, ψ_{opt} , is simply

$$\psi_{\text{opt}} = \cos^{-1} \left[\frac{-1}{1 + r_p (V_{\infty})^2 / \mu} \right],$$

where $\frac{\pi}{2} \leq \psi_{\text{opt}} \leq \pi$

r_p = planetocentric radius at periapsis

μ = gravitation parameter of Mars.

For Voyager missions, ψ_{opt} is usually in the range 105 to 130 degrees.

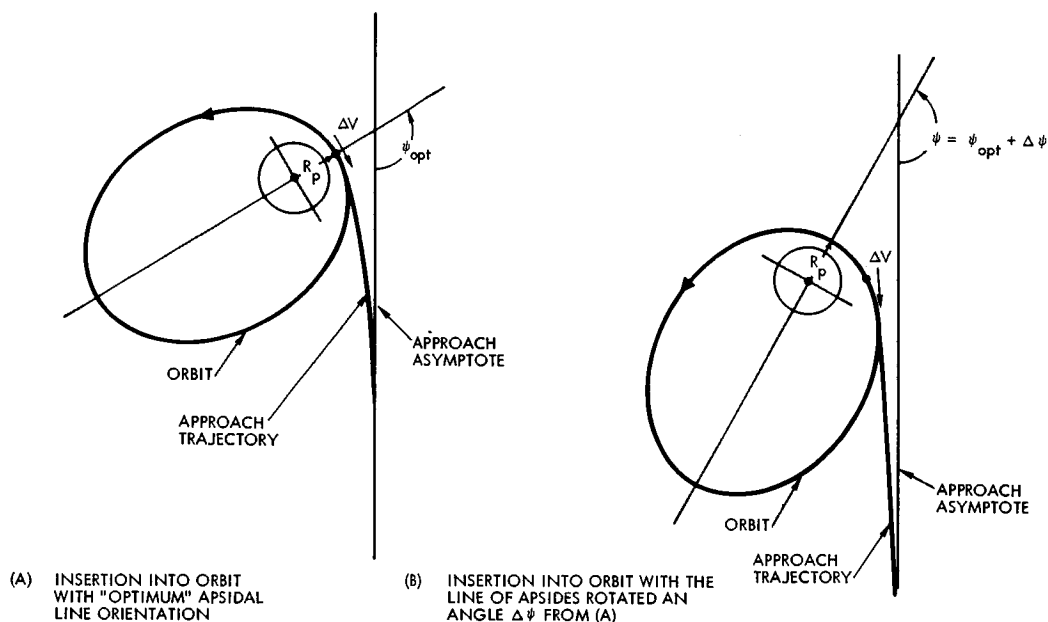


Figure 3-21

ORBIT INSERTION GEOMETRY. The "optimum" orientation of the line of apsides, (A), requires the least ΔV for transfer from an approach of given V_{∞} . As it is rotated from this orientation, the best point for transfer moves away from the periapsis points of both the elliptical orbit and the hyperbolic approach trajectory, and the required ΔV magnitude increases.

Figure 3-21b illustrates the situation wherein the elliptic orbit's line of apsides has been rotated in the direction of motion, away from the optimum orientation depicted in Figure 3-21a. Figure 3-22 illustrates the orbit insertion velocity increments required for nonrotated and rotated orbits as a function of asymptotic approach velocity at Mars. The effects of a 20-degree posigrade apsidal rotation are presented for two representative Mars orbit sizes.

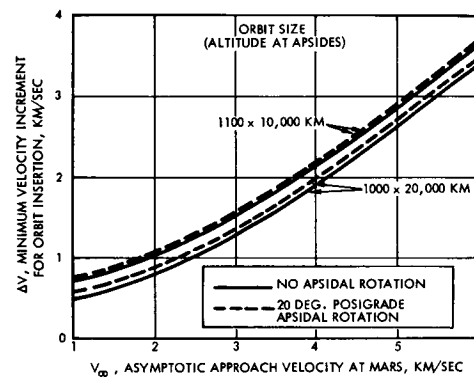


Figure 3-22
MINIMUM VELOCITY INCREMENT requirements for insertion into nonrotated and rotated elliptic Mars orbits.

Only coplanar orbit transfers are considered. With that limitation, the choice of the approach trajectory target point determines the orientation and inclination of the orbit about Mars. Of course, a plane change can be effected by the orbit insertion maneuver, but the velocity requirements are large for significant angles of change.

3.3.2.4 Orbit Trim

An impulsive velocity allocation of fixed magnitude has been made for an orbit trim maneuver to take place prior to the separation of the capsule and its descent to the Martian surface. While the spacecraft propulsion system has the capability to perform additional maneuvers after the capsule has landed, no propellant is specifically allocated for this purpose.

3.3.3 Spacecraft Velocity Increment Requirements

Several assumptions have been made for the magnitudes of the various spacecraft impulsive velocity requirements. These assumptions,



together with the specific impulses for each maneuver, are tabulated in Table 3-6 for the 1973 mission. $I_{sp} = 285$ seconds corresponds to use of the LM descent engine in its low-thrust mode; 305 seconds corresponds to high thrust. (Increasing the low thrust level from 1050 to 1700 pounds (discussed in Volume 3) has raised the I_{sp} from 285 to 298 seconds. This improvement was not incorporated into the analyses of this section.)

Table 3-6. Impulsive Velocity Requirements
1973 Mission

Maneuver	ΔV Allocation (km/sec)	I_{sp} (sec)
First interplanetary ("arrival date separation maneuver")		285
• Arrival date separation	Variable	}
• Injection bias	0.054	
• Spherical injection error	0.045	
Subsequent interplanetary ("inter- planetary trajectory correction maneuvers")	0.010	285
Orbit insertion	Variable	305
Orbit trim	0.150	285

* Combined according to the equation of Section 3.3.2.1.

The arrival date separation and orbit insertion maneuvers have been termed "variable" in Table 3-6 because they vary with both launch date and arrival date. A module has been added to TRW's Analytic Interplanetary Program (Ref. 9) to evaluate planetary vehicle gross weight, impulsive velocity requirements, weight placed in Martian orbit, and usable propellant requirements associated with any specified launch date interval, mean arrival date, and arrival date separation. Given a launch period duration, this module also identifies the precise launch

date interval which maximizes weight in Martian orbit subject to constraints on DLA. Figure 3-23 gives a functional flow chart of this launch period optimization process.

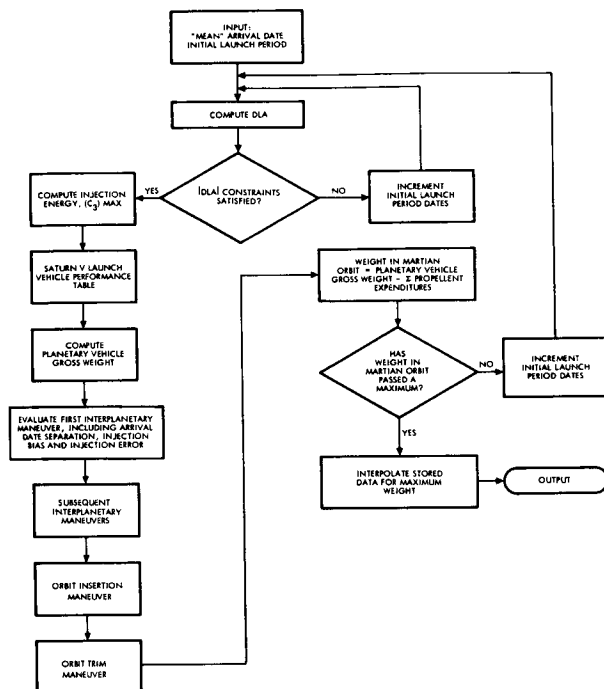


Figure 3-23
ATP LAUNCH PERIOD IDENTIFICATION MODULE-FUNCTIONAL FLOW CHART showing input requirements, computation sequence, and output.

3.3.4 Summary Display of Spacecraft Capability

Sections 3.3.1 through 3.3.3 have delineated the trajectory-related assumptions pertaining to the 1973 Type I opportunity, including launch azimuth limits, maneuver sequences, and incremental velocity requirements. In addition, an arrival date separation of ± 4 days from the mean arrival date has been stipulated. This section summarizes the performance capabilities of the recommended spacecraft for this mission opportunity. Several combinations of launch period duration and Mars orbit size have been evaluated. These are shown in Table 3-7.

Figure 3-24 shows the variations of required usable propellant with Mars arrival date and periapsis location relative to the optimum orientation for a 20-day launch period and the $1100 \times 10,000$ km orbit. Positive values of $\Delta\psi$ correspond to posigrade rotations of the line of

Table 3-7. 1973 Type I Trajectory Parameters

Launch Period Duration (days)	Mars Orbit Size Altitudes at Apses (km)
20	1100 x 10,000
20	1000 x 20,000
30	1100 x 10,000
30	1000 x 20,000

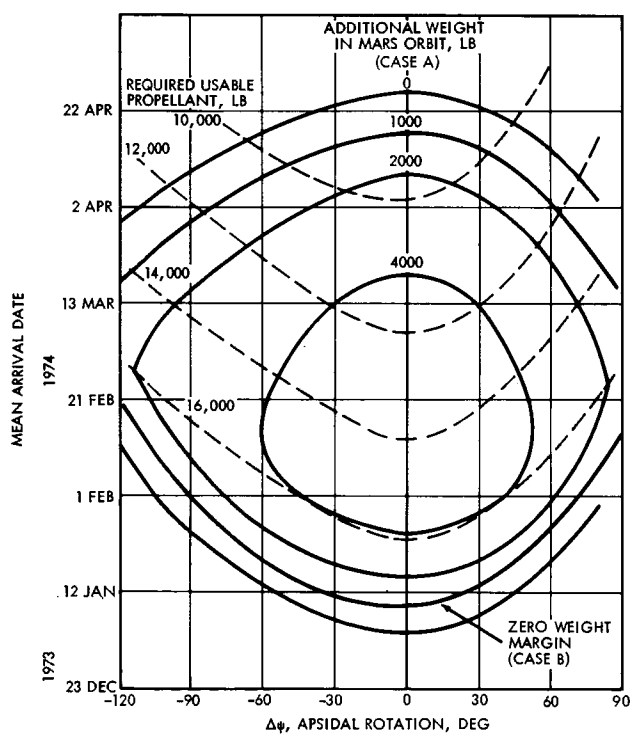


Figure 3-24
FOR A 1100 X 10,000-KM ORBIT AND A 20-DAY LAUNCH PERIOD, performance capability can be applied to reducing the transit time, carrying heavier payload, or altering the position of periapsis at Mars. Required propellant in the spacecraft for these options is shown up to a maximum of 16,000 pounds. Only Type I trajectories are included.

apsides (see Figure 3-21b); negative values refer to retrograde rotations. Also shown are contours of additional weight of spacecraft or capsule which may be realized for mission Case A (see Section 3.1). This configuration, which corresponds to the zero additional weight contour, weighs 10,370 pounds, including allocations of 400 pounds for spacecraft science,* 50 pounds for spacecraft capsule support, 5000 pounds for the capsule, and 4920 pounds for the spacecraft and all its subsystems (including propulsion), but excluding usable propellant. Additional weight contours for Case B are simply those for Case A, reduced by 1000 pounds. For example, the 1000-pound contour for Case A corresponds to zero additional weight for mission Case B.

Contours of additional weight illustrate the reduction in arrival date span and allowable apsidal rotation as more of the performance margin is devoted to increasing the payload weight. The maximum additional weight capability of 5480 pounds (Case A) is achieved with a mean arrival date of February 11, 1974, and no apsidal rotation. Required usable propellant contours show the propellant weight which must be utilized to achieve the indicated performance. Since the recommended spacecraft design accommodates a maximum of 16,000 pounds of usable propellant, weight contours for arrival dates earlier than the ones shown for the 16,000-pound propellant contour reflect the penalty incurred by restricting usable propellant to this design limit.

There is an optimum 20-day period of launch days associated with each mean arrival date of Figure 3-24. Figure 3-25 shows the position of the optimum launch period on a departure date-arrival date coordinate system for several selected mean arrival dates. Representative C_3 and DLA contours are noted. For any given mean arrival date, the gross weight of the planetary vehicle (including propellant) is limited by the maximum value of launch energy, C_3 , which is required during the launch period.* As may be seen from Figure 3-25, $(C_3)_{\max}$ may occur at either the beginning or end of the optimum launch period. Note that injection requirements and constraints (C_3 and DLA)

* JPL specification payload weight allocation. See Volume 5, Appendix G for weight estimates of science payloads described in that volume.

** C_3 , the vis viva energy, is the square of the geocentric asymptotic departure velocity.

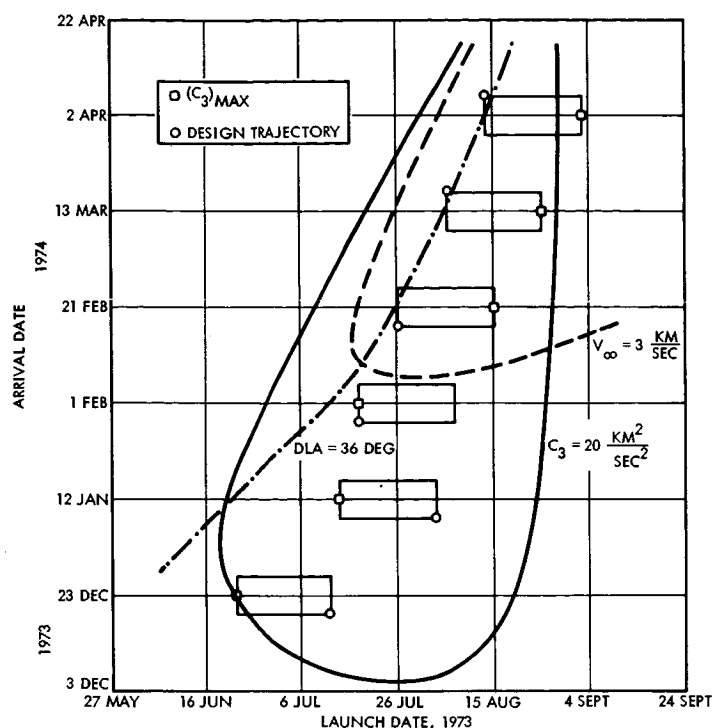


Figure 3-25
OPTIMUM 20-DAY LAUNCH PERIODS for Type I trajectories in 1973, are determined by the selection of mean arrival dates. The maximum launch energy, C_3 , required during that period can occur either at its beginning or end. Representative values of approach velocity, declination of launch asymptote, and injection energy are plotted in this departure-arrival coordinate system to show the range of these values involved with these launch periods.

are based on an earth-Mars trajectory with the indicated launch date, but with the "mean" arrival date—that halfway between the scheduled arrivals of the two planetary vehicles.

The gross weight of each planetary vehicle is determined by $(C_3)_{\max}$ and the Saturn V launch vehicle performance. Figure 3-15 displays gross weight of each planetary vehicle as a function of C_3 for a launch azimuth of 115 degrees. It is based on MSFC-supplied launch vehicle performance, and accounts for a 5000-pound contingency, as well as the two planetary vehicles and their adapters, within the launch vehicle capability. These data have been used to compute planetary vehicle gross weight.

The use of a 115-degree launch azimuth is conservative since launch vehicle performance also varies with launch azimuth (Figure 3-14) and in general $(C_3)_{\max}$ is associated with launch azimuth requirements

which are less than 115 degrees. Referring to Figure 3-25, extreme required launch azimuths of 115 degrees, which correspond to $|DLA|_{\max} = 36$ degrees, occur only on the first day of the optimum launch periods corresponding to the later mean arrival dates, i.e., 21 February 1974, etc. $|DLA|$ and launch azimuth then decrease through the launch period. As long as the required launch azimuth at the location of $(C_3)_{\max}$ is in the range 90 to 115 degrees, the actual launch vehicle capability is no less than that which corresponds to the 115-degree launch azimuth.

Figure 3-25 also illustrates the locations of the design trajectories which determine the subsequent weight which can be placed in Martian orbit. A design trajectory is defined here as that which requires the greatest propellant-to-gross-weight ratio of any trajectory during the launch period under consideration. Since the velocity increments for both the arrival date separation and orbit insertion maneuvers vary throughout any given launch period, this ratio must be determined at both extremities and at intermediate points in the launch period in order to determine this maximum, which must be provided. The analysis is complicated by the fact that the orbit insertion maneuver ΔV varies with both launch date and date of arrival at Mars. All orbit insertion maneuver ΔV 's have been evaluated for the larger V_{∞} requirement associated with the pair of arrival dates corresponding to any given launch date. Selection of the larger V_{∞} is conservative, since only one of the two spacecraft is required to enter Martian orbit from this V_{∞} .

The ranges of launch dates indicated for the launch periods shown in Figure 3-25 are optimum for their corresponding mean arrival dates. Each optimum launch period is determined by maximizing the limiting value of weight in Martian orbit subject to the aforementioned DLA constraint. Optimum launch periods for the very early mean arrival dates favor V_{∞} . As mean arrival date is increased, the optimum launch periods are located in the central portion of the C_3 and V_{∞} contours where a balance between these quantities is obtained. For later mean arrival dates, the optimum launch period favors C_3 , and then is limited by the DLA constraint.



Figures 3-26, 3-27, and 3-28 illustrate the performance capability of the recommended spacecraft for the remainder of the launch period duration—Mars orbit size pairs given in Table 3-7. Increasing the apoapsis altitude results in a greater spacecraft capability since the orbit insertion maneuver velocity increment is reduced. On the other hand, an increased launch period duration reduces spacecraft capability since both C_3 and V_∞ are necessarily increased (for a given mean arrival date).

3.3.5 Mars Orbit Capability

The discussions and performance charts above have shown the capability that the spacecraft has for achieving various Mars orbit orientations with respect to the hyperbolic approach velocity vector. These orientations have been defined in terms of the rotation of the line of apsides of the Mars orbit from the optimum orientation determined by V_∞ and r_p (see Section 3.3.2.3). The performance analyses are independent of the inclination of the Mars orbit, but do depend upon the size of the orbit, i.e., altitudes at the apsides. Thus, optimum heliocentric transfer trajectories may be identified from these performance data, but supplemental Mars orbit data must be evaluated in order to select an orbit size and orientation which satisfies specific engineering constraints and scientific objectives.

Appendix C contains such data for a number of orbits about Mars. These orbits are entered from seven selected 1973 Type I earth-Mars trajectories covering three arrival dates: February 7, March 21, and April 25, 1974. A number of orbit sizes and orbit inclinations relative to the Mars equator have been studied, together with several apsidal orientation angles, ψ . The quarantine requirement that the orbit not decay before 1986 essentially dictates the minimum orbit size, in particular the altitude of closest approach. The range of orbit sizes given in the JPL 1973 Voyager mission specification, 500 to 1500 km periapsis altitude and 10,000 to 20,000 km apoapsis altitude, is compatible with the orbit lifetime requirement, and orbits in this size range were studied. It was observed that the orbit constraints were rather insensitive to this range of orbit sizes. The following discussion

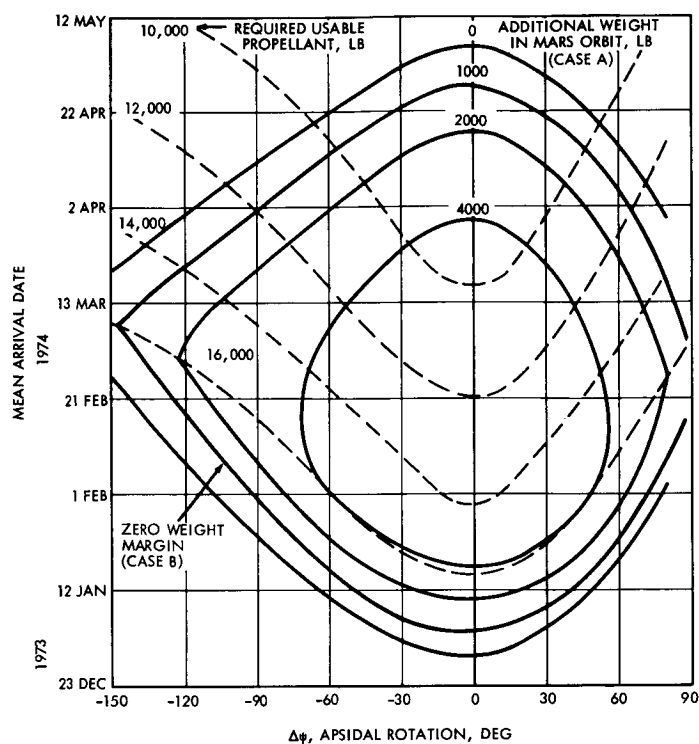


Figure 3-26
FOR A 1000 X 20,000-KM ORBIT AND A 20-DAY LAUNCH PERIOD, performance capability can be applied over a wider range than for the smaller orbit.

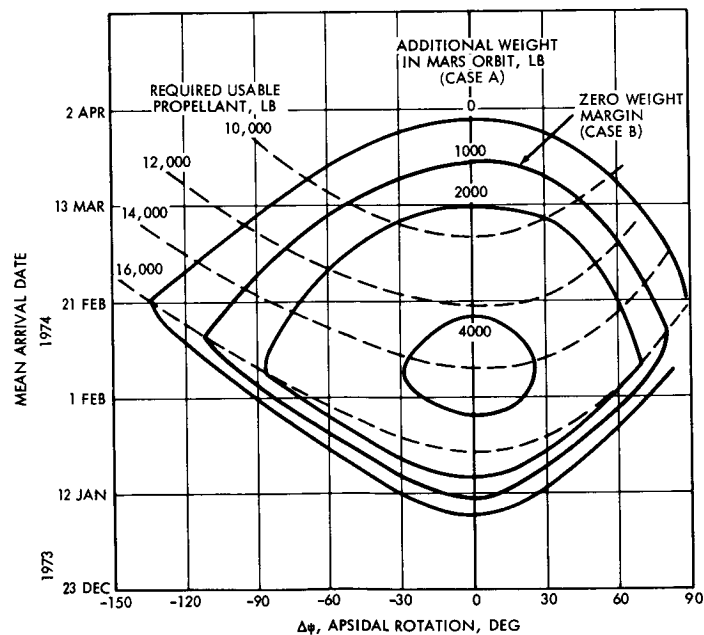


Figure 3-27
FOR A 1100 X 10,000-KM ORBIT AND A 30-DAY LAUNCH PERIOD the constraints on possible applications of performance capability are narrower than for a 20-day launch period.

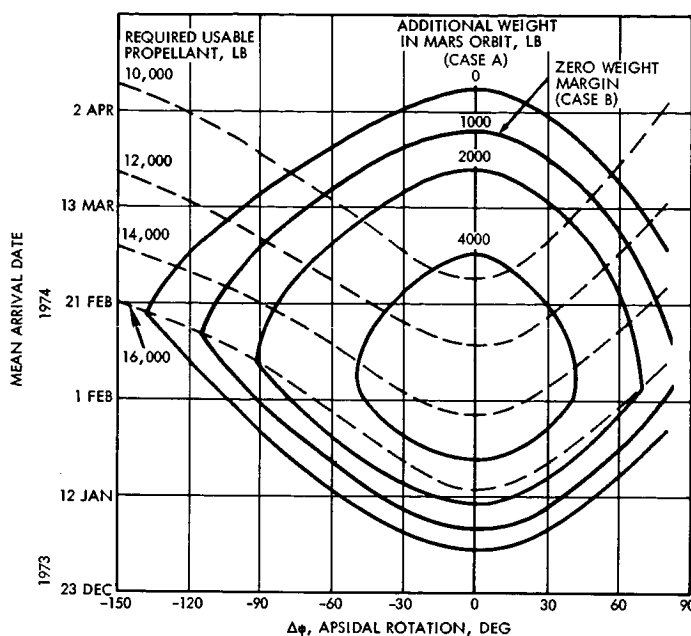


Figure 3-28
FOR A 1000 X 20,000-KM ORBIT AND A 30-DAY LAUNCH PERIOD the range of options for applying available performance capability are similar to that for the 1100 X 10,000, 20-day case, the less demanding requirements of the more elliptical orbit being traded for the shorter launch period.

pertains to the 1000 x 15,000 km orbit, and addresses constraints on orbit selection which are identified in the JPL specification.

The choice of orbit inclination obviously affects the occultation geometry with other bodies such as the sun, earth, and Canopus. Low inclinations increase the chances of occulting the former two bodies but leave Canopus well in view, while high inclinations tend to have the opposite effect. Due to the earth-occultation experiments, the orbital inclination must be carefully selected so as to achieve the proper amount of earth occultation but a minimum of sun occultation (since the orbiter is solar-powered).

Referring to the Mars orbit data for the first trajectory, inclinations of 20 degrees (both North and South)* result in sun occultations at

*Inclination is considered North if the initial passage and approach target point is in the northeast quadrant. South inclination implies approach trajectory in the southeast quadrant. Retrograde orbits (northwest or southwest quadrant targeting) were not studied in the appendix.

Mars arrival which last for 20 days past insertion. For northern inclinations up to 60 degrees, sun occultations occur during the first 180 days, violating the requirement for no sun occultation during this period. Thus, all northern inclinations are eliminated by this constraint.

For an apsidal orientation angle $\psi = 150$ degrees, sun occultation occurred for the first 15 days. This indicates that large posigrade apsidal rotations should be avoided.

Earth occultation occurred for all inclinations considered. For higher inclinations, earth occultation occurred shortly after closest approach during the first few days past insertion. This may be undesirable for communication purposes. Retrograde apsidal rotation will increase the time from periapsis to enter earth occultation, but only small rotations should be considered to avoid sun occultation. However, large retrograde rotations may be desired to allow the capsule to land near the dawn terminator so that communications with earth can be maintained for a reasonable length of time before the earth sets, as seen from the landed capsule.

Good lighting conditions exist for all southern inclinations with no apsidal rotation. The optimum periapsis orientation is approximately 5 degrees (60-degree inclination) to 20 degrees (20-degree inclination) from the nearest terminator (the evening terminator). The terminator moves away from periapsis after encounter.

Constraints which are placed upon the choice of orbits from the standpoint of earth-based determination of the spacecraft's orbit include: (a) the angular rate of the orbit plane with respect to the earth-Mars line of sight should be non-zero, (b) the orbit inclination with respect to the plane generated by the earth-Mars line of sight should be non-zero (this latter plane is the Mars "ecliptic" plane), and (c) the orbit plane should not be normal to the earth-Mars line. All data indicate that no problem exists with regard to these constraints.

The location of the subperiapsis point lies in a region of high scientific interest for all southern inclinations and for northern inclinations below 60 degrees.



As may be seen from this brief discussion of the Mars orbit data for earth-Mars Trajectory 1, there is no a priori selection criterion. Southern inclinations from approximately 30 to 60 degrees with small retrograde apsidal rotations appear to be most favorable for this trajectory. Similar Mars orbit data comparisons may be made for the remainder of the data presented in Appendix C.

3.4 TRAJECTORY ACCURACY CAPABILITY

3.4.1 Interplanetary Trajectory

This section discusses the results of a guidance and navigation analysis to determine the capability of the recommended spacecraft configuration to meet the interplanetary trajectory accuracy requirements of the Voyager missions subject to propellant limitations. These trajectory requirements reflect the need to approach Mars close enough to permit accurate orbit insertion, and to observe the planetary quarantine constraint. The fundamental conclusion drawn from these results is that the spacecraft described earlier in this volume will satisfy these requirements. The analysis shows that the accuracy of the approach trajectory is highly dependent upon the tracking estimate of the spacecraft before the final midcourse correction. This conclusion is based upon the analysis of the squared multiple correlation coefficient relating final accuracy statistics to the statistics prior to the final midcourse correction.

The accuracy capability of the spacecraft will be discussed in terms of three primary considerations, namely:

- Trajectory biasing and correction policy
- Maneuver execution errors
- Uncertainty due to tracking

The overriding consideration in establishing the biasing and correction policy is the quarantine constraint. Thus, aim points for injection and the several midcourse maneuvers were selected such that the probability of contaminating Mars by impact of a nonsterile vehicle would be less than 3×10^{-5} . (The most recent NASA quarantine plan, as suballocated in Section 3.5, indicates probabilities of contamination of 10×10^{-6}

for the launch vehicle and 5×10^{-6} for each spacecraft by impact resulting from injection and interplanetary maneuvers. The 3×10^{-5} figure used in this analysis, while less stringent, nevertheless verifies the adequacy of the biasing principles described.)

The maneuver execution error model of the TRW recommended spacecraft represents a system which is within the state of the art. The results of the analysis show that this system does adequately perform the mission.

The tracking uncertainty is a function of the equipment capabilities of the DSIF and the control center, as long as the spacecraft transponder (S-band receiver and transmitter) maintains frequency lock and exceeds the threshold signal-to-noise requirement at the ground station. The tracking uncertainty is also dependent upon the knowledge of the physical constants (i.e., the value of the astronomical unit and the ephemeris of Mars). The AU uncertainty will be refined as a result of the 1967 Mariner/Venus mission, and current interplanetary radar experiments. Mariner 1969 (Mars) should further increase knowledge of the AU and improve Mars' ephemeris accuracy.

The conclusions of the analysis associated with each of these considerations will be discussed in the following subsections, while the details of the analysis used to support these conclusions are presented in Appendix D.

The particular trajectory selected for studying the accuracy capabilities was chosen because that trajectory places the most demands upon the proposed spacecraft, namely (1) the launch day is the first day of the available launch window, thus the trajectory is at the boundary of the mission constraints; (2) the separation velocity is the greatest of any trajectory within the launch window; and (3) this trajectory is very likely the most sensitive to injection and execution errors.

A spacecraft following the selected trajectory would be injected into the geocentric escape trajectory on 21 July 1973 and would pass near Mars 205 days later. The injection parameters are shown in Table 3-8. Five days after injection, a separation maneuver is performed which



Table 3-8. Injection Parameters

Parameters	
R, km	7015.9895
Geocentric latitude, deg	-23.791894
Longitude, deg	97.321896
Inertial velocity, km/sec	11.357226
Inertial flight path angle, deg	77.999976
Inertial azimuth angle, deg	61.395050
Epoch, 21 July 1973	15 ^h 54 ^m 35 ^s GMT

changes the aim point and the flight time of each of the two vehicles. The total flight time of the first vehicle is changed to 201 days and the total flight time of the second vehicle is changed to 209 days. Midcourse maneuvers are simulated at 30 days after injection and 10 days before arrival time (time of distance of closest approach).

It was decided to analyze the trajectory having a 201-day flight time in this study. It was noticed that the nominal arrival-date separation maneuver required to decrease the flight time from 205 to 201 days was 103 m/sec while the nominal separation maneuver required to increase the flight time from 205 to 209 days was 176.7 m/sec. However, when the velocity required to perform the deboost maneuver is considered, the velocity requirements of both spacecraft are almost identical. Results similar to these presented in this study would be obtained had the 209 day flight time trajectory been analyzed. However, in this case the increased arrival-date separation maneuver ΔV would propagate into correspondingly larger uncertainties in the control of the trajectory effected by this and interim maneuvers. Final approach accuracy is about the same for the two spacecraft, the limiting factor being tracking accuracy before the final maneuver in each case.

3.4.1.1 Injection Biasing and Trajectory Correction Policy

All aim points were selected such that the probability that a trajectory impacting Mars would be less than 3×10^{-5} . The technique used

to compute the aim points is extremely conservative and this technique was selected to provide a conservative upper bound on the magnitude of the various maneuvers. The results show that the spacecraft always satisfies the contamination constraint. Other less conservative methods of selecting the aim points are discussed in Section 3.5.

It is convenient to think of the Mars-centered state vector in terms of the impact parameter plane. This plane contains the center of Mars and is perpendicular to the incoming V_{∞} vector. This plane contains the R and T vectors by definition. The impact parameter, B, lies in the R-T plane and has components of $B \cdot T$ and $B \cdot R$.

A more complete definition of the R-T plane and the $B \cdot T$ and $B \cdot R$ coordinate system is given in Appendix D. State vector uncertainties are defined to be density functions whose contour of constant probability density is an ellipse in the R-T plane. The probability, P, of being inside a $k\sigma$ ellipse is given by

$$P = 1 - e^{-k^2/2}$$

Solving the problem in reverse (i.e., given P, compute k), we find that $k = 4.561$ when $P = 0.99997$. Placing the impact area outside the 4.561σ ellipse guarantees that the probability that an impact trajectory will result is less than $1 - P = 0.00003$ since the impact area occupies only a small fraction of the region outside of the ellipse.

The initial aim point was selected to be near the line of minimum sensitivity so that execution errors would have a small effect. The selected initial aim point also required the same expenditure of fuel for the sum of the arrival date separation maneuver and the deboost maneuvers.

The aim point for the arrival date separation maneuver and the subsequent midcourse maneuvers was selected to lie on a line containing the center of Mars and the final aim point. This strategy has been shown to maximize the probability of being within the final success zone after the arrival date separation maneuver, subject to planetary quarantine constraints. This strategy also minimizes the probability of needing additional interplanetary corrections. The aim points of the various maneuvers are shown in Table 3-9.



Table 3-9. Aim Points

Maneuver	B · T (km)	B · R (km)	T _f (days)
Injection	273136	140965	205
Arrival date separation	29400	-14900	201
First midcourse	6328	-4690	201
Second midcourse	6328	-4690	201

The final aim point (first and second midcourse maneuvers) corresponds to an approach trajectory with a minimum altitude of 1070 km.

The initial aim point was selected to be along the semiminor axis of the uncorrected miss ellipse such that the center of the 4.56σ ellipse was at $B \cdot T = 273136$ km and $B \cdot R = 140965$ km. The method of obtaining the uncorrected miss ellipse is discussed in Appendix D.

The aim point for the separation maneuver was selected to lie on the line containing the final aim point and the center of Mars. The aim point for the separation maneuver was obtained by combining the tracking covariance matrix (tracking from injection to five days) with the expected separation maneuver execution error covariance matrix.

The aim point for the first midcourse correction was obtained by combining the tracking covariance matrix (tracking from 5 to 30 days) with the expected midcourse execution error covariance matrix. It was found that the aim point for the first midcourse correction could be the final target without violating any of the constraints.

3.4.1.2 Maneuver Execution Errors

The maneuver execution error characteristics of the proposed spacecraft represent state-of-the-art design. The results of the analysis show that the accuracy of the approach trajectory is not strongly dependent upon the maneuver execution errors.

The model of the maneuver execution errors of the spacecraft are made up of two parts: The first part consists of errors proportional to the magnitude of the maneuver (ΔV), and the second part consists of nonproportional errors. The 3σ values of the error parameters are as follows:

<u>Direction</u>	<u>Proportional</u>	<u>Nonproportional</u>
Along thrust axis	(0.001) (ΔV)	0.024 m/sec
Perpendicular to thrust axis	(0.014) (ΔV)	0

The covariance matrix of execution errors is shown in Appendix D. The 0.014 ΔV component perpendicular to the thrust axis corresponds to a thrust vector pointing accuracy of 0.8 degree.

3.4.1.3 Tracking Model

This section will discuss the effect of tracking uncertainty upon the accuracy of the approach trajectory. The results of this study show that 70 percent of the dispersion in miss after the final midcourse correction is due to the uncertainty in the estimate of miss before the final midcourse maneuver. The uncertainty in the estimate of the state vector of the spacecraft is a function of the capabilities of the DSIF and the ground computing center. The spacecraft contribution to tracking is the satisfactory performance of the onboard transponders. The tracking uncertainties will be reduced when the knowledge of the physical constants (value of the astronomical unit and the ephemeris of Mars) is improved. The tracking data presented in this section are consistent with JPL tracking data, as shown in Reference 10.

The recommended spacecraft will be tracked by the DSIF stations at Goldstone, Madrid, and Woomera. The station locations are shown in Table 3-10 and the 1σ noise and biases are shown in Table 3-11. Figure 3-29 shows the semimajor axis of the 1σ uncertainty ellipse and the uncertainty in arrival time as a function of time for two cases. The first case solved for systematic errors (data biases and station location uncertainties), while the second case solved for the systematic errors and considered state noise. State noise may be considered as random forces which act upon the spacecraft and thus slightly alter the

Table 3-10. Station Locations

Station	Latitude (deg)	Longitude (deg)	Altitude (meters)
Goldstone	+ 35.067	243.205	1213.
Madrid	+ 40.334	355.751	800.
Woomera	- 31.212	136.886	145.

Table 3-11. Measurable Uncertainties (1σ)^{*}

Range (meters)		Azimuth (mr)		Elevation (mr)		Range Rate (m/sec)	
Noise	Bias	Noise	Bias	Noise	Bias	Noise	Bias
5	5	0.1	0.1	0.1	0.1	1×10^{-3}	1×10^{-3}

* Same for all three stations

trajectory. Samples of state noise are outgassing, meteoroids impacting the spacecraft, uncertainties in solar pressure, and round-off errors in the computer programs. The value of state noise uncertainty was 0.037 m/sec and this value was for the entire flight time. This value is 250 times greater than present estimates and thus provides a very conservative estimate of the tracking capabilities. A more complete discussion of tracking and state noise is given in Appendix D.

The semimajor axis of the combined estimate (solve for systematic errors and consider state noise) uncertainty ellipse is seen to be 9 km after 191 days (10 days before encounter) and is seen to remain constant until the time of deboost (201 days). The covariance matrix was increased by a factor of $(2)^2$ to account for error sources not included in the study such as uncertainty in the astronomical unit and the uncertainty in the ephemeris of Mars. The semimajor axis assumes that spacecraft will be sent to Mars in 1969 and 1971 so that the uncertainty in the AU and the ephemeris of Mars in 1973 will be reduced from the present value. Under these assumptions, a 20 km uncertainty is reasonable and this value agrees with unpublished JPL data.

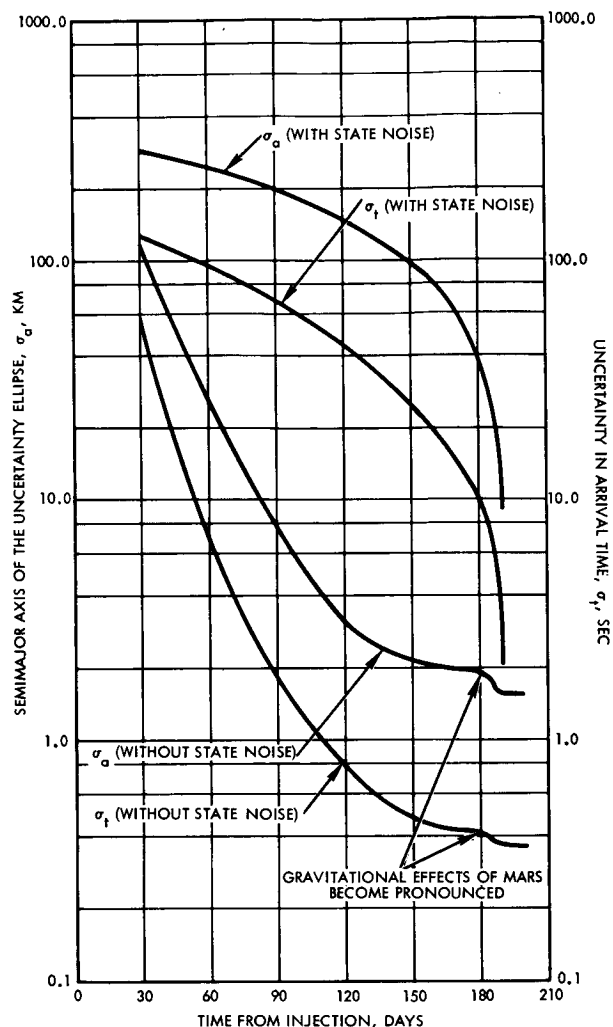


Figure 3-29
TRACKING UNCERTAINTY decreases with time as more tracking data is obtained so that at arrival the ellipse within which the spacecraft can be located has a semimajor axis of 2 km or 9 km (1σ) depending upon whether state noise is included in the calculation. State noise incorporates all random forces such as outgassing or meteoroid impacts. A conservative value of 0.037 m/sec is used for state noise in these calculations.

Notice that the semimajor axis of the uncertainty ellipse and the uncertainty in arrival time decrease rapidly as the spacecraft nears Mars and the gravitational effects of Mars become pronounced; however, the uncertainties remain constant after the initial decrease.

3.4.1.4 Key Numerical Results

The accuracy of control of the approach trajectory and the velocity requirements were computed using a Monte Carlo simulation of the Voyager mission from injection to the end of the second midcourse



maneuver. The total number of samples was 1001; thus the confidence level is greater than 0.995 that 99.5 percent of the population lies below the largest value of the random variable under consideration. For example, Figure 3-30 shows that the largest value of total velocity increment is 159.869 m/sec. It may be said with 99.5 percent confidence that the probability is 99.5 percent that sample number 1002 will be less than 159.869 m/sec. The details of the Monte Carlo simulation are shown in Appendix D.

Figure 3-30 shows the cumulative distribution (CD) of the velocity required to perform the three maneuvers. Referring to the figure, the + sign represents every 10th data point while the solid line represents a normal cumulative distribution having the sample mean and standard deviation. The nominal total velocity is 105 m/sec and the mean is seen to be 107.7 m/sec. If the total flight time had been increased to 209 days, the nominal separation maneuver would have been 176.7 m/sec and the total nominal velocity would have been 179.4 m/sec. However, when the deboost maneuvers are included, the total expended velocities of both vehicles are almost identical. It should be noticed that the distribution of total velocity is non-gaussian.

Figures 3-31, 3-32, and 3-33 show the CD of the final dispersions in B·T, B·R, and time of flight, respectively. The distributions are seen to be non-gaussian with a mean of almost zero. The non-normal distributions are due to the proportional velocity execution errors; thus, the actual distributions could only be obtained from a Monte Carlo analysis and not by conventional analytical methods.

The multiple squared correlation coefficients were examined and 70 percent of the final dispersions were caused by the tracking uncertainty before the second midcourse correction.

3.4.2 Orbit About Mars

Two aspects of trajectory accuracy which pertain to the orbit about Mars are discussed. These are 1) the accuracy with which the attained orbit can be controlled – dependent on the orbit insertion maneuver execution errors – and 2) the accuracy to which the elements of the attained orbit can be estimated.

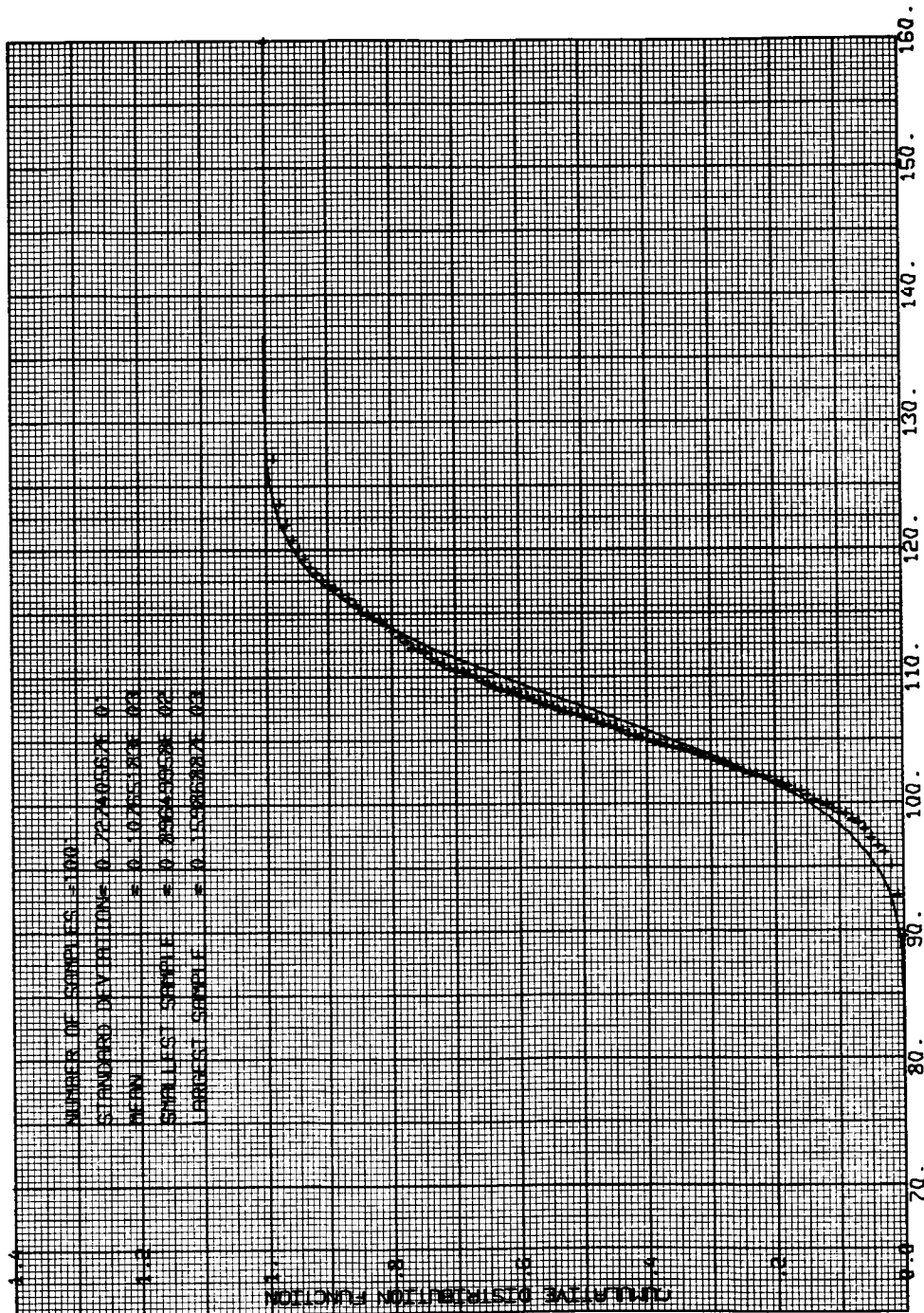


Figure 3-30
 CUMULATIVE DISTRIBUTION OF THE VELOCITY INCREMENT REQUIRED as plotted by a Monte Carlo simulation from injection to the end of a second midcourse maneuver. The points show the simulation; the superimposed curve is a normal distribution having the sample mean and standard deviation. The printout shows a mean total velocity of 107.7 m/sec (0.10765183×10^3).

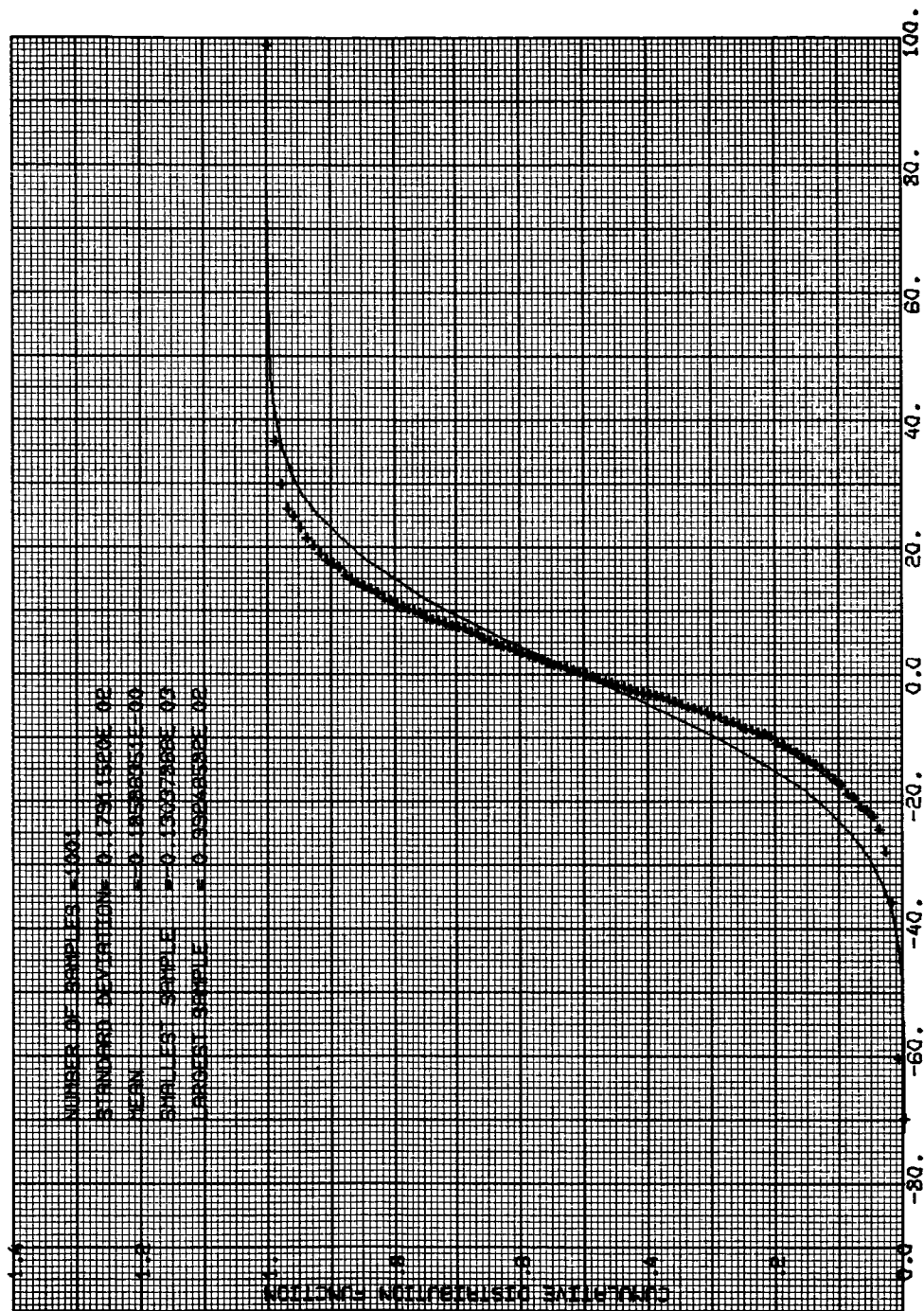


Figure 3-31
 CUMULATIVE DISTRIBUTION OF THE B.T COMPONENT of the impact parameter plane
 (R - T) as computed by the Monte Carlo simulation shows a mean of almost zero
 (-0.186 km).

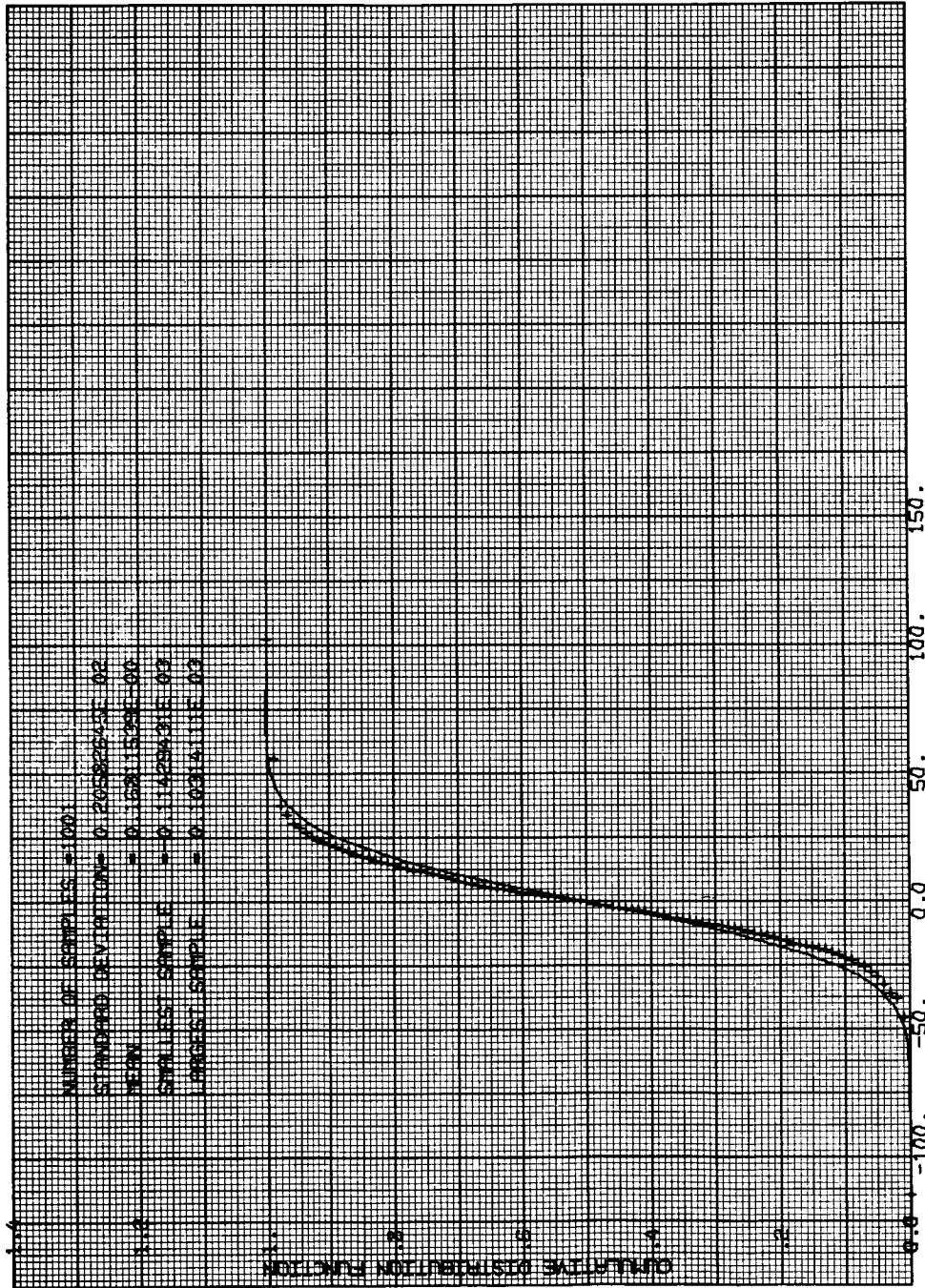


Figure 3-32
CUMULATIVE DISTRIBUTION OF THE B-R COMPONENT of the impact parameter plane
(R - T) as computed by the Monte Carlo simulation shows a mean of almost zero
(0.168 km).

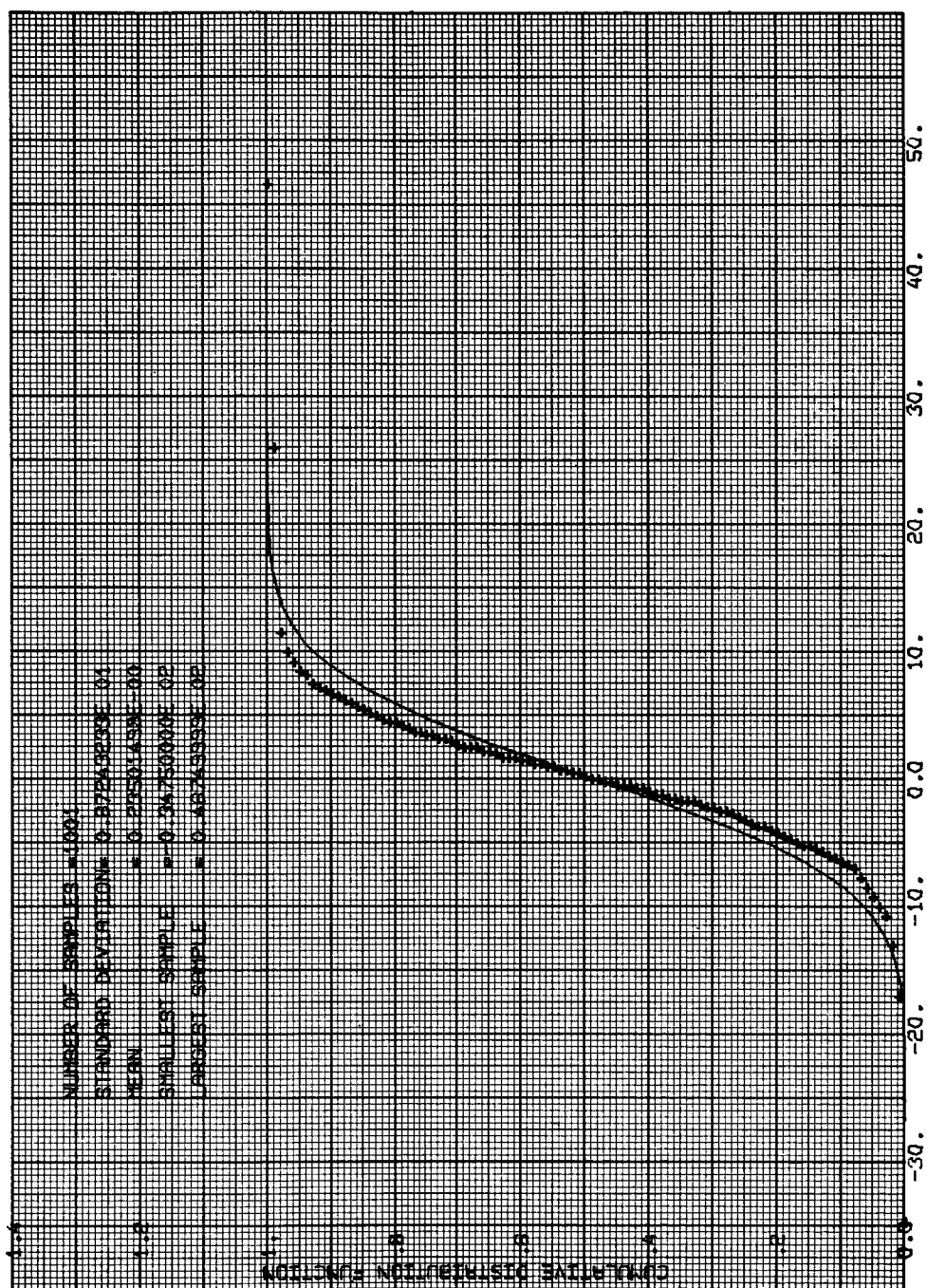


Figure 3-33
CUMULATIVE DISTRIBUTION OF THE DISPERSION IN TIME OF FLIGHT as computed
by the Monte Carlo simulation shows a mean of 0.235 second.

The 3σ accuracy of control of the orbit is about 18 and 55 km in periapsis and apoapsis altitudes, respectively, which appears to be satisfactory, although requirements for these quantities have not been given. The 3σ accuracy of the estimate of the orbital elements is based on two-way range-rate doppler data at ground stations, and improves markedly from about 2 and 7 km (periapsis and apoapsis altitudes) one day after arrival to 20 to 50 meters two weeks after arrival.

3.4.2.1 Orbit Insertion Maneuver Execution Errors

In this analysis, only the deboost maneuver execution errors were considered. The model of the maneuver execution errors has been discussed in Section 3.4.1.2. For this analysis, the deboost maneuver performed was taken at the periapsis of the Martian-centered approach hyperbola (an altitude of 1000 km). This deboost maneuver places the spacecraft into an orbit whose periapsis and apoapsis altitudes are 1,000 and 15,000 km, respectively. The 3σ values of the errors in the orbital elements resulting from deboost maneuver errors are shown in Table 3-12.

Table 3-12. 3σ Values of the Errors of the Orbital Elements of the Martian Orbit Due to Vehicle Execution Errors

Parameter	3σ Value
Semimajor axis	33.218 km
Eccentricity	1.12424×10^{-3}
Period	161.466 sec
Apoapsis altitude	55.136 km
Periapsis altitude	18.114 km

3.4.2.2 Accuracy of Estimate of the Orbit About Mars

This section discusses the results of a study to determine the accuracy of estimating the elements of an orbit about Mars by earth-based sensors. The conclusion drawn from these results is that the orbital parameters may be determined to a high degree of accuracy. The orbit studied is described in Table 3-13.



Table 3-13. Orbital Parameters (Martian Equatorial Coordinate System)

Semimajor axis	1.1584600×10^4 km
Eccentricity	0.620789963
Inclination	44.857675 deg
Ascending node	207.73972 deg
Argument of perigee	131.25327 deg
Mean anomaly	359.988290 deg
Epoch	7 February 1974, 15 ^h 42 ^m 33 ^s GMT

This orbit has altitudes of approximately 1,000 and 15,400 km at the ap-sides, is entered from an interplanetary trajectory with launch date July 21, 1973, arrival date February 7, 1974, and arrival V_{∞} of 3.018 km/sec. It is representative of candidate orbits for the 1973 Voyager mission.

The values of the uncertainties in the estimated orbital parameters (in a Martian equatorial coordinate system), after 14 days of tracking are shown in Table 3-14.

Table 3-14. Uncertainties in the Estimated Orbital Parameters After Fourteen Days of Tracking

Parameter	1 σ Value
Perigee altitude	8.011×10^{-3} km
Apogee altitude	1.494×10^{-2} km
Semimajor axis	1.132×10^{-2} km
Eccentricity	3.412×10^{-7}
Inclination	1.322×10^{-5} deg
Ascending node	5.455×10^{-5} deg
Argument of perigee	1.102×10^{-4} deg
Mean anomaly	1.514×10^{-6} deg
Second zonal harmonic of Mars	1.323×10^{-9}
Gravitational constant of Mars	$32.66 \text{ km}^3/\text{min}^2$

The spacecraft was assumed to be tracked by the DSIF stations at Goldstone, Madrid, and Woomera. The stations measured only range-rate data and the data sampling rate was 1 data point per minute. Figures 3-34 through 3-37 show the uncertainties in the estimated orbital parameters (tracking from insertion to 14 days). In this analysis, systematic errors (data biases, station location uncertainties, gravitational constant, and second zonal harmonic of Mars) were solved for. The details of this analysis are presented in Appendix D.

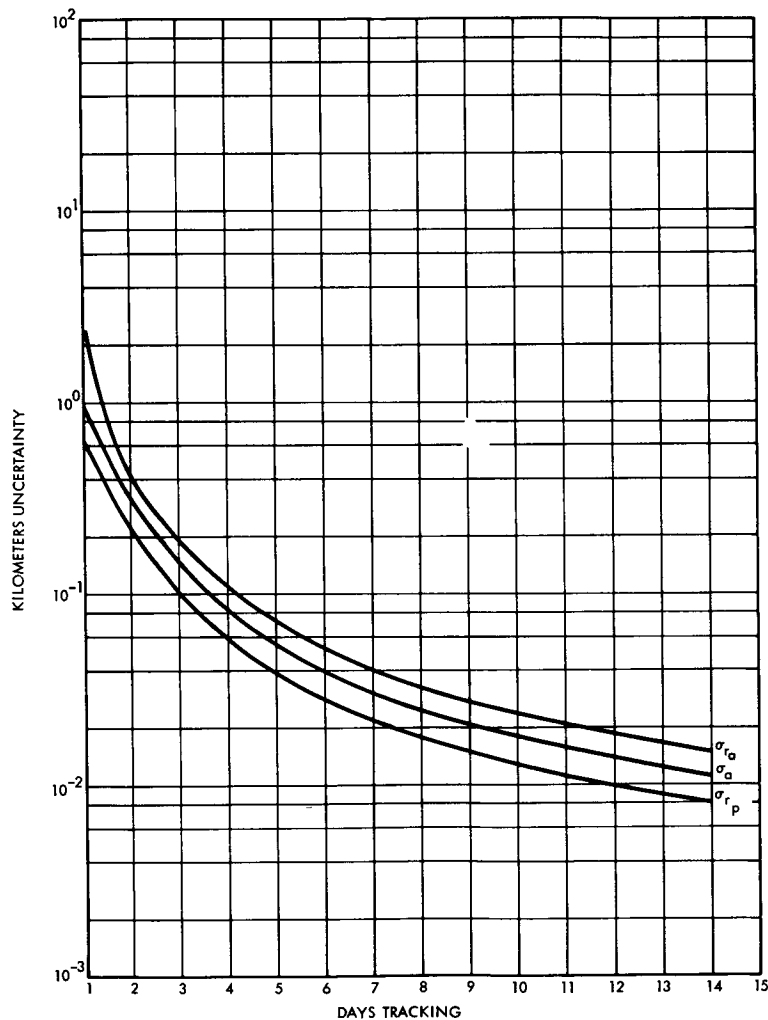


Figure 3-34
UNCERTAINTIES IN THE ORBITAL PARAMETERS at Mars are reduced after 14 days of tracking to 0.008 km for the radius of periapsis (σ_{rp}), 0.011 km for the semimajor axis (σ_a), and 0.015 km for the radius of apoapsis.

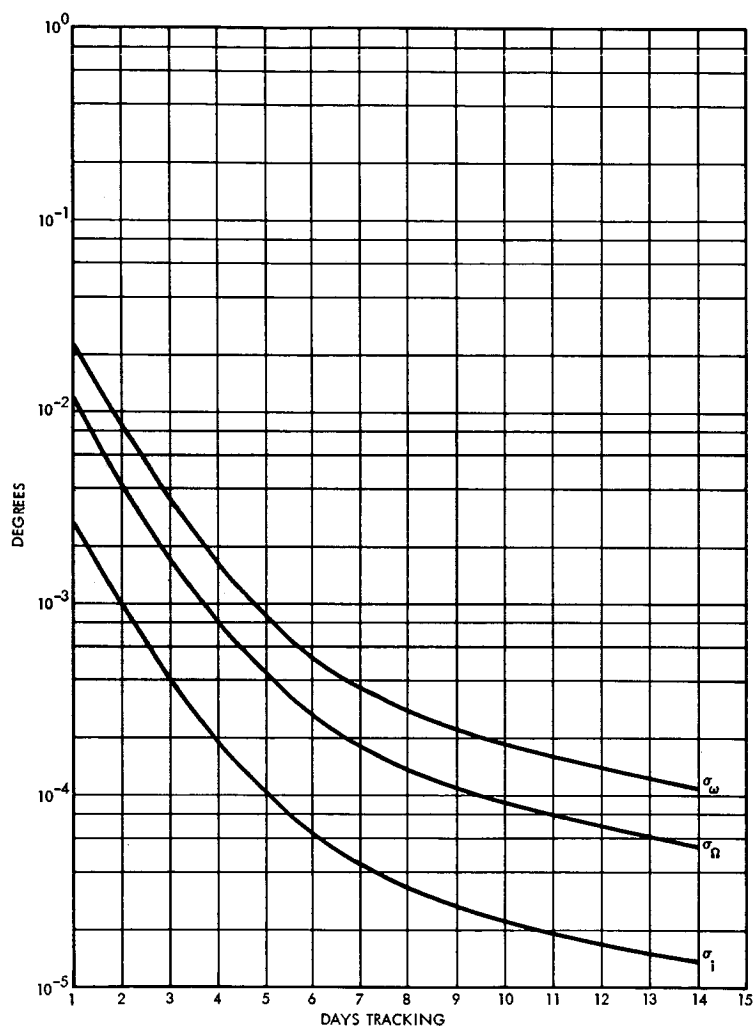


Figure 3-35
UNCERTAINTIES AFTER 14 DAYS OF TRACKING THE ORBIT at Mars are reduced to 0.00013 deg for the inclination (σ_i), 0.00051 deg for the ascending node (σ_{Ω}), and 0.0011 deg for the argument of perigee (σ_{ω}).

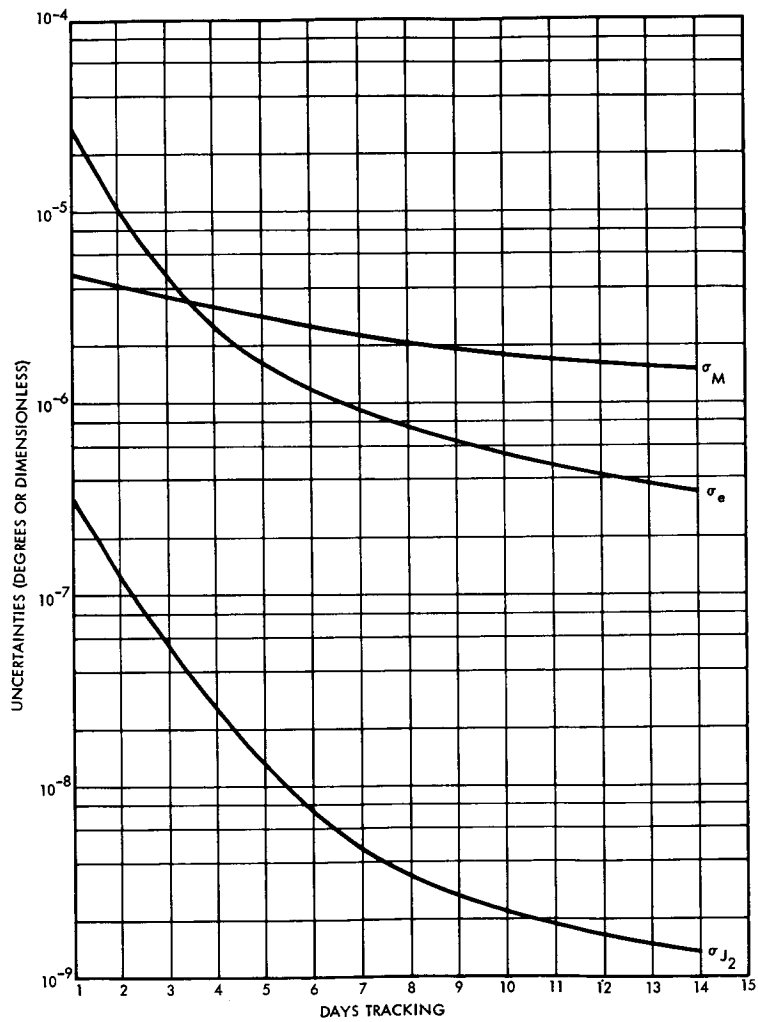


Figure 3-36
UNCERTAINTIES IN ORBITAL TRACKING 14 days following insertion are reduced to 1.3×10^{-8} for the second zonal harmonic (σ_{J_2}), 4.4×10^{-6} for the eccentricity (σ_e), and 1.5×10^{-5} deg for mean anomaly (σ_M).

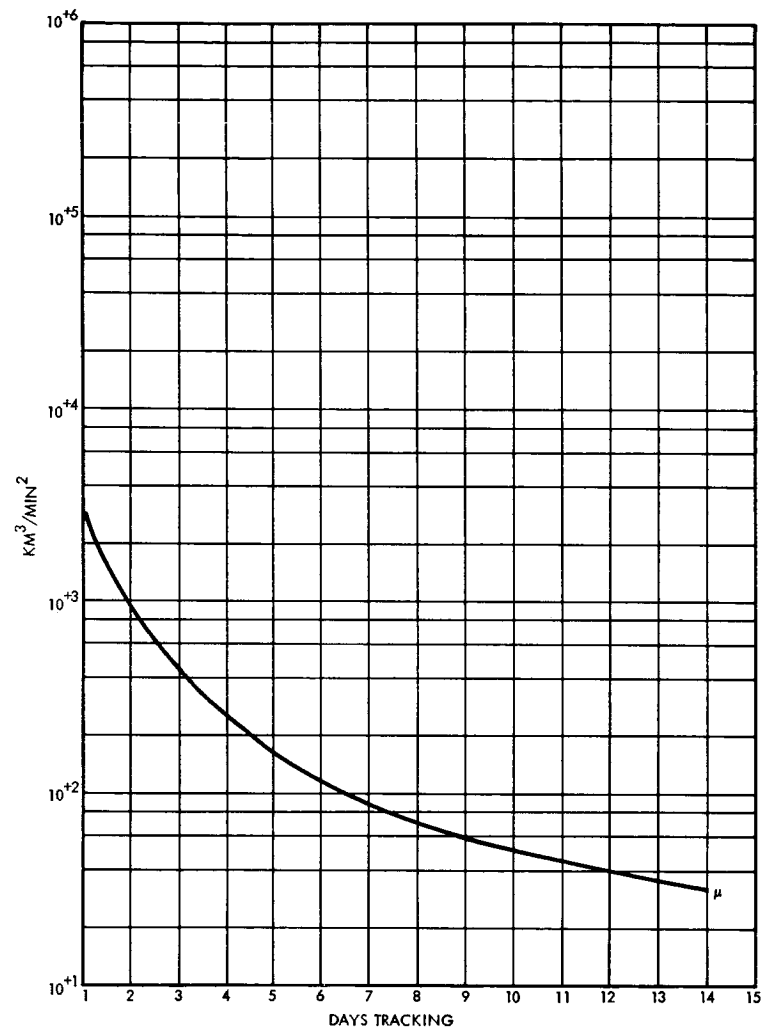


Figure 3-37
UNCERTAINTIES IN THE GRAVITATIONAL CONSTANT OF MARS will have been reduced to 31 km³/min² after 14 days of tracking the Voyager orbit.

3.5 OBSERVANCE OF THE PLANETARY QUARANTINE CONSTRAINT

The NASA Planetary Quarantine Plan for the Voyager project (Ref.11) gives the most recent statement as to NASA policy and requirements in assuring the observance of a biological quarantine of Mars by the Voyager project. In addition to outlining the management, procedural, testing, and documentation aspects of adherence to the provisions of the plan, it also covers contamination allocations of permitted probabilities of contaminating Mars. The following numerical allocations of allowable probabilities of contaminating Mars have been assigned to the individual sources of contamination for elements pertinent to the Voyager project, for use pending the establishment of detailed quarantine criteria.

- For each sterilized capsule, 10^{-6}
- For each unsterilized spacecraft, including ejecta therefrom, 3×10^{-5}
- For unsterilized items common to more than one spacecraft, such as launch vehicle stages and adapters, probability of contamination will be included in the sum of probability allocations noted above for the unsterilized spacecraft and will be suballocated between project systems by the Voyager manager
- For viable organisms released on Mars, the probability of survival and growth will be considered as 10^{-3}
- For any sources of contamination that continue to be present over long periods of time, such as those that would apply to an unsterilized orbiter, the above constraints will apply to all potential contaminations up to, and including, the year 1985
- The probabilities of transit to Mars, survival during transit, and release of viable organisms on Mars will be determined by the systems and project offices, as appropriate, and approved by the project manager

3.5.1 Effect of Survival and Growth Probability

The 10^{-3} probability of survival and growth for unsterilized items given above is to be applied as follows. Where the probable number of viable microorganisms on a component (e.g., the capsule) which may

impact Mars is small compared with 10^3 (this extends down to statistical numbers much smaller than 1), the probability of contaminating the planet is reduced by multiplying this number by 10^{-3} . However, where the mechanism of contamination is via a component or fragment which bears and delivers more than 10^3 viable organisms, no relief is afforded by the 10^{-3} survival and growth probability.

As an example of the first category (Figure 3-38a), suppose the probability of any microorganisms accompanying the capsule, and surviving descent, entry, and touchdown, is 10^{-3} . Then, in one of 1000 such missions we would expect one microorganism to be landed. But in only one of 1000 of these instances would the microorganism be expected to survive and grow on Mars. Thus, the probability of contaminating Mars—to the extent of altering its ecology—would be only one in 10^6 .

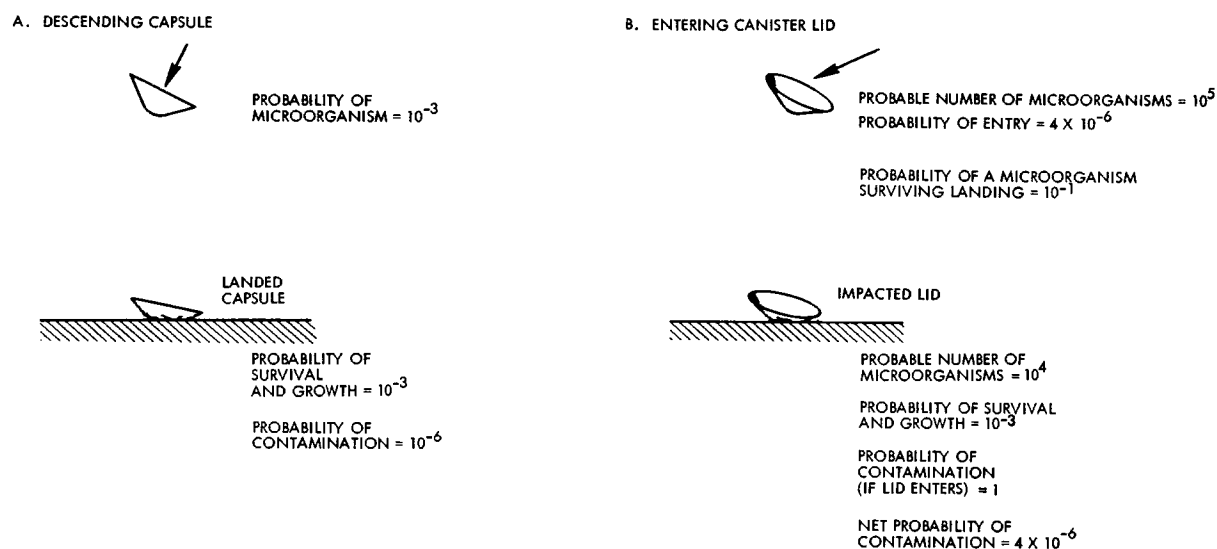


Figure 3-38

EFFECT OF SURVIVAL AND GROWTH PROBABILITY of 10^{-3} varies. As the capsule has a high probability of entering and (statistically) a very low number of microorganisms, the 10^{-3} growth probability reduces total probability of contamination significantly. The canister lid has a low probability of entering, but carries many microorganisms, so the 10^{-3} probability does not reduce the total probability of contamination.

As an example of the second category (Figure 3-38b), suppose the probable number of microorganisms on the lid of the capsule canister is 10^5 , and that the probability of the lid assuming an orbit which decays prematurely into the Martian atmosphere is 4×10^{-6} . Also, suppose that one microorganism in 10 will survive the entry and impact. Then, with some 10^4 microorganisms landed on Mars, the 10^{-3} probability for

survival and growth is not low enough to reduce the probability of contamination (should such an entry occur) appreciably below one. Thus, the probability of contamination of Mars via the canister lid is 4×10^{-6} , the same as the probability that the lid will enter the atmosphere.

One possible method of contaminating Mars is that in which the capsule is contaminated by efflux from the spacecraft, after the canister lid has been opened. For this particular avenue it is not clear which allocation of the Planetary Quarantine Plan applies. That is, both the capsule and the spacecraft are involved. The above discussion has a significant bearing in any case. The probability that any viable microorganisms be transferred to the capsule in this manner must necessarily be kept small. This mechanism is of the first category above, and the 10^{-3} probability of growth and survival does apply. Therefore, it appears appropriate that the probability of such cross contamination is to be reduced by a factor of 10^{-3} before adding it into the appropriate allocation.

3.5.2 Suballocation of Probabilities

Figure 3-39 summarizes the NASA allocation of allowable probabilities of contaminating Mars, and distributes or suballocates these probabilities to contributing items. To the extent that control of these means of contaminating Mars is within the purview of the spacecraft contractor, they are treated in the paragraphs below. The suballocations have been determined with qualitative recognition of these modes which are most difficult to control.

The allocations to the spacecraft for impact resulting from interplanetary trajectory and to the launch vehicle for impact resulting from injection are both relatively small. These probabilities can be restricted as much as necessary by the policy of biasing the injection and each interplanetary maneuver except the last so that it aims at a target point farther from Mars than the final target point. The category "efflux from spacecraft" has a minimal allocation, because the gaseous nature of such efflux makes any microbes which are therefore released particularly susceptible to ultraviolet radiation from the sun. (The same consideration does not necessarily apply where efflux from the spacecraft contributes to cross contamination of the lander, rather than direct contamination from Mars.) However, when the statistical microbial burden is

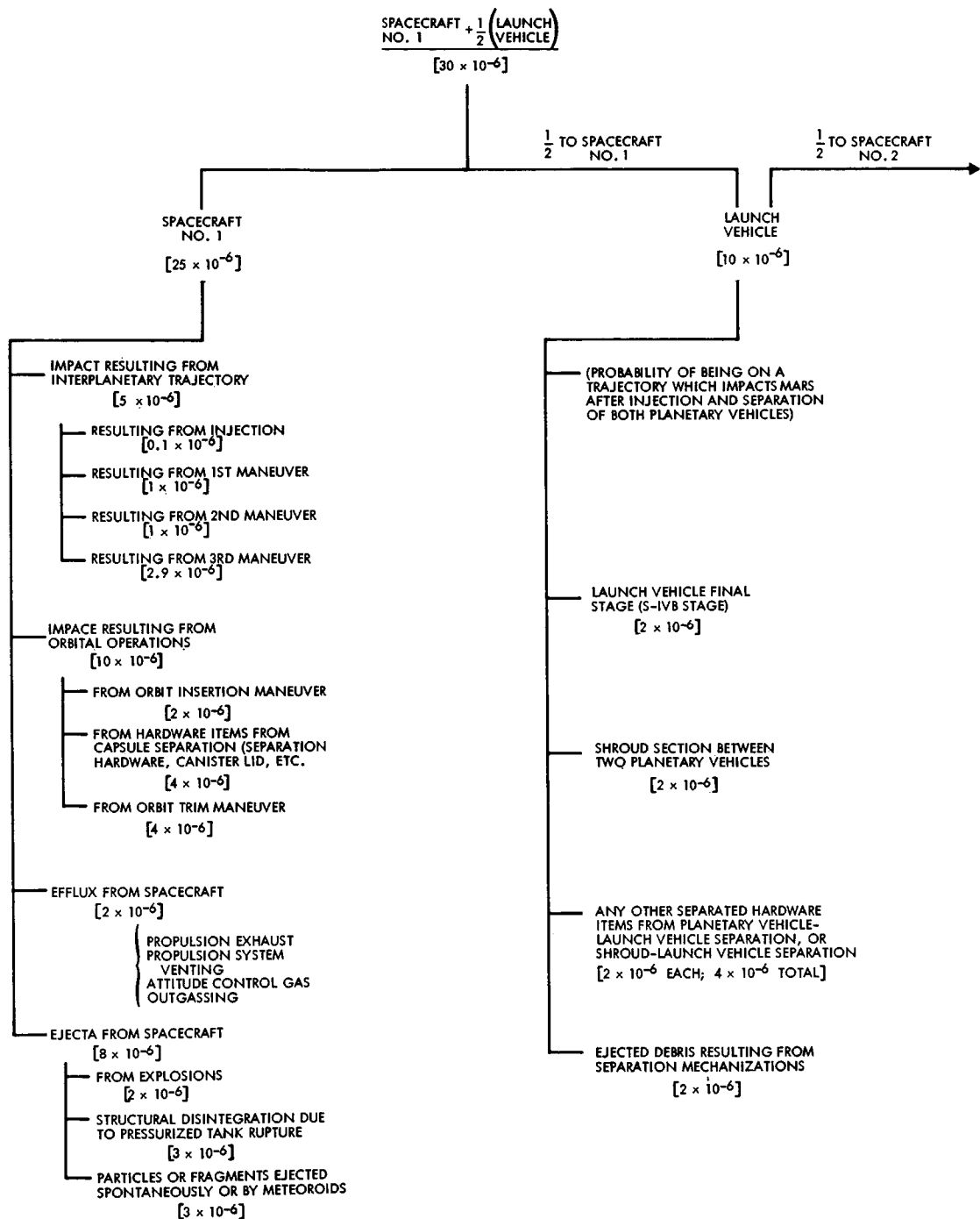


Figure 3-39
SUBALLOCATION OF ALLOWABLE PROBABILITIES of contaminating Mars to various modes is shown. The allocations for impact resulting from the interplanetary trajectory are relatively small, with the major portion associated with orbital operation. Modes involving low statistical microbial burden receive a 1000 to 1 reduction due to low probability of growth and survival on Mars.

very low, the reduction due to the 10^{-3} probability of survival and growth applies as discussed above.

The major portion of the total spacecraft allocation has gone to orbital operations (to account for the uncertainty of our ability to predict atmospheric drag in orbit about Mars) and to ejecta from spacecraft (recognizing a difficulty in guaranteeing to a very high probability that explosions, structural disintegration, and fragmentation of the spacecraft will not take place).

3.5.3 Trajectory Control

In the observance of the planetary quarantine constraint, trajectory control applies to the prevention of unintentional entry into the Martian atmosphere of all elements of the planetary vehicle whose sterility is not guaranteed. This applies to all portions except for the sterilized capsule which is intended to land on the planet. It includes the final stage of the launch vehicle, the shroud and planetary vehicle adapters, the spacecraft, and portions of the capsule canister. While the application of ethylene oxide gaseous decontaminant (ETO) to some of these elements is under consideration, such a procedure would be adopted only to limit the possibility of contamination of Mars by efflux and ejecta from the affected element, and would not be expected to actually sterilize interior portions of the spacecraft or other elements.

Trajectory control is exerted in the targeting and execution of all propulsive maneuvers, and in the tracking and orbit determination of the spacecraft. The application of this control would also extend to the programming and execution of propulsive maneuvers designed solely to avoid unintentional entry of a nonsterile body.

3.5.3.1 Injection

At injection, the launch vehicle final stage, the shroud, and the two planetary vehicles are placed on an interplanetary trajectory generally directed towards a subsequent encounter with the planet Mars. Soon after injection, separation processes lead to four major elements, all on almost the same interplanetary trajectory: the S-IVB stage of the launch vehicle (carrying the instrument unit, a portion of the shroud, and one



planetary vehicle adapter), a single shroud midsection (carrying the second planetary vehicle adapter), and two separated planetary vehicles. (The conical nose portion of the shroud will have been jettisoned in earth parking orbit, and poses no contamination threat to Mars.)

The two planetary vehicles will perform several propulsive maneuvers before approaching Mars, so the result of the injection procedure is not critical from the point of view of spacecraft impact at Mars. However, the shroud center section does not have any further propulsive capability, so that it is necessary to assure that the injection targeting is biased sufficiently far from an impact trajectory that the probability that this section impacts Mars is held to 2×10^{-6} , as shown in Figure 3-39. While it would be possible to use a retro rocket to assure that the S-IVB stage of the launch vehicle does not impact Mars, a biasing approach, having already been adopted because of the shroud section, will also serve to limit the probability of S-IVB stage impact without depending on the reliability of a retro maneuver.

Depending on the details of the actual separation mechanization, it is possible that other discrete items of hardware would be released at this time. The suballocation of Figure 3-39 provides for two such items. In addition, there is the possibility that the separation mechanization will result in the ejection of a spray of debris. Presumably, mechanizations will be chosen which do not nominally result in such ejection; however, with pyrotechnic separations, there is always the possibility that seals designed to contain all separation products will fail. This topic is reviewed at greater length in Volume 10.

Having adopted a bias of the injection target point as the primary means of satisfying this aspect of the quarantine constraint, it remains to be determined how great this bias should be, and in which direction. The magnitude of bias is dictated by the dispersion of the injection which is caused by inaccuracies in the launch vehicle. In Reference 1, this dispersion is required to be no greater than 5 m/sec, 1σ , expressed as velocity necessary to correct miss and time of arrival, if employed two days after injection. Under the assumption that a spherically distributed dispersion of this magnitude does exist, the following calculation shows that a bias of 3.96σ , or 19.8 m/sec is necessary to keep the probability of impact as low as 2×10^{-6} .

$$\begin{aligned}
 &(\text{probability of impact}) = (\text{cross section of Mars}) \\
 &\quad \times (\text{probability density function})
 \end{aligned}$$

$$p \approx (\pi r_o^2) \times \frac{e^{-(1/2)n^2}}{2\pi\sigma_1\sigma_2}$$

$$2 \times 10^{-6} \approx (\pi \cdot 7100^2 \text{ km}^2) \times \frac{e^{-(1/2)n^2}}{2\pi (125,000 \text{ km}) (40,000 \text{ km})}$$

The solution is $n \approx 3.96$, where

n = the multiple of σ of the bias

$r_o = 7100 \text{ km}$ = radius of impact cross section of Mars,
corresponding approach $V_{\infty} = 3.0 \text{ km/sec}$
and periapsis altitude = 400 km

σ_1, σ_2 = semiaxes of projection of 1σ dispersion
on R-T plane

$$= 5 \text{ m/sec} \times (2500, 800 \frac{\text{km}}{0.1 \text{ m/sec}})$$

$$= 125,000, 40,000 \text{ km}$$

Such a bias imposes a negligible penalty in terms of the spacecraft propellant which will subsequently be used to compensate for the bias and direct the spacecraft closer to the final target point. The reason is that the arrival date-separation maneuver, conducted primarily to establish the desired eight-day separation in times of arrival at Mars for the two planetary vehicles and secondarily to remove the effects of the bias and dispersions of the launch vehicle injection, will be substantially greater in magnitude than the amount of the bias. Figure 3-40 shows a simple incorporation of such a bias in the injection V_{∞} vector, and also shows how the propellant allocation for the arrival date-separation maneuver depends on the three components, separation due to change in arrival date, injection bias, and injection dispersion. The mechanism for sizing propellant, and incorporating this sizing into the overall mission performance is covered in Section 3.3 above.

Thus, the conducting of the injection event so as to comply with the planetary quarantine constraint is essentially a matter of biasing the

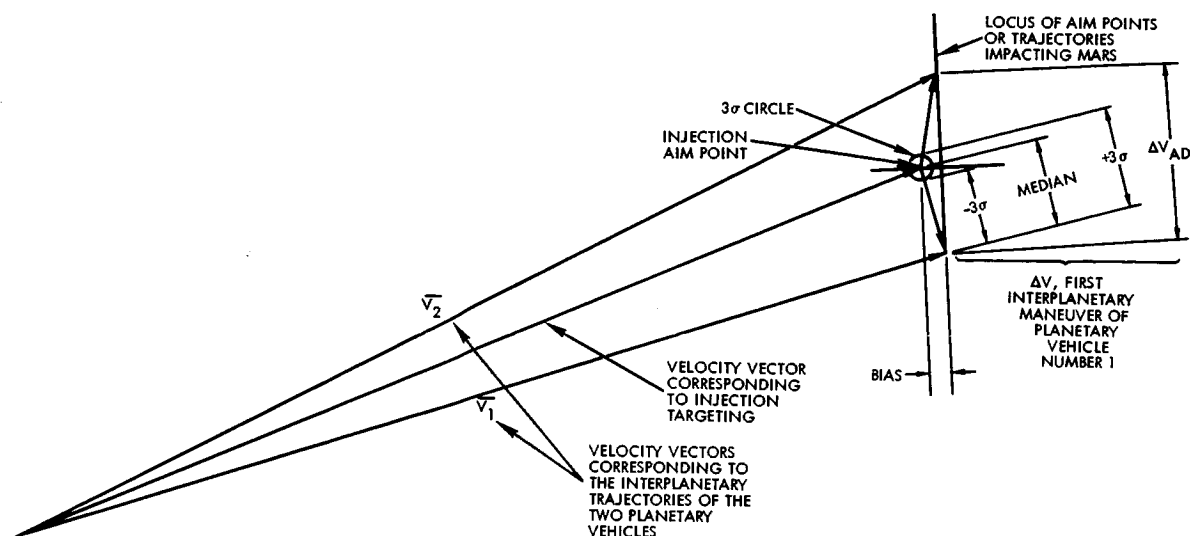


Figure 3-40
BIASING TO AVOID LAUNCH VEHICLE IMPACT AT MARS imposes a negligible penalty on the ΔV requirements for the first interplanetary maneuver. The component due to arrival date separation (ΔV_{AD}) is the principal contribution.

injection target point sufficiently that impact of the planet by the S-IVB stage or shroud section is reduced to a sufficiently low level, accounting for possible dispersions in the injection.

A detailed analysis of the probability of contaminating Mars by means of debris ejected during separation events has not been completed. However, it appears that the larger particles so ejected will have relative velocities no greater than 10 to 20 m/sec, and that only very fine particles will exceed a relative velocity of 30 m/sec. Such particles, if they do exist, could possibly be a contamination threat. However, this threat can be alleviated either by increasing the amount of bias in the injection targeting point—biases of up to 50 m/sec would not impose an undue penalty in spacecraft propellant usage—or by requiring that any surface of a component which might be ejected as a spray of particles as a result of a separation event be painted or anodized black, so that equilibrium temperatures achieved in transit through heliocentric space would be certain to sterilize the particle. In addition, the 10^{-3} reduction due to the low probability of survival and growth on Mars would probably apply.

In Figure 3-39, the probability of contaminating Mars arising from the spacecraft as a result of injection onto the interplanetary trajectory is shown as 0.1×10^{-6} . The planetary vehicle has the same probability of assuming an impact trajectory as the S-IVB stage; however, the probability of failure of subsequent propulsive maneuvers, indicated arbitrarily as 0.05 here, reduces the probability of contaminating Mars by the spacecraft due to off-nominal injections to the insignificant value given.

3.5.3.2 Interplanetary Correction Maneuvers

The philosophy of insuring compliance with the planetary quarantine constraint in the conducting of propulsive maneuvers between injection and arrival at Mars is basically the same as that for injection. That is, the target point of each maneuver is biased sufficiently far from Mars so that the probability of impact is reduced to a sufficiently low value. The control of the planetary vehicle trajectory to Mars as of the time of execution of any propulsive maneuver is subject to these accuracy limitations:

- The accuracy of the estimate of the interplanetary trajectory of the planetary vehicle at the time the maneuver is programmed
- The error in executing the programmed maneuver
- Unpredictable perturbations or noise in the interplanetary trajectory subsequent to the maneuver
- Uncertainty in the location of Mars at arrival, that is, knowledge of the ephemeris of Mars and the length of the astronomical unit

Analysis of these factors determines the dispersion relative to the planet Mars which can be expected as a result of the maneuver, and therefore may be employed to prescribe the magnitude of the bias which must be employed in targeting that maneuver (see Section 3.4.1). As all of these effects decrease progressively as the interplanetary phase proceeds, the magnitude of the bias for each successive maneuver will be reduced. For the final maneuver, the accuracy will be great enough in relation to the final target point and its altitude above Mars that no bias will be required; that is, the target point for the final interplanetary



maneuver will be the intended target point of the approach trajectory; and will be compatible with the desired transfer from the approach trajectory onto the orbit.

The bias need not be chosen so great as to reduce the probability that the resulting trajectory will impact Mars to the allocated allowable probability of contaminating Mars. This is because there is a high probability of being able to ascertain whether an impact trajectory has been attained, and to program a subsequent propulsive maneuver, if necessary, which diverts the planetary vehicle from the impact trajectory. For example, referring to Figure 3-39, a bias may be chosen so that the probability of a resulting impact trajectory is 30×10^{-6} for each of three interplanetary correction maneuvers, probabilities of contaminating Mars are 1×10^{-6} for the first and second maneuvers, reflecting a 0.97 probability of successful performance of a subsequent maneuver, and 2.9×10^{-6} for the third maneuver, reflecting a 0.9 probability of successfully conducting a subsequent maneuver if necessary. This lower confidence for the third maneuver recognizes that it would be performed only 10 to 30 days before encounter, that the orbit determination task after this maneuver to ascertain whether the trajectory will impact Mars is more difficult, and that a subsequent interplanetary maneuver would not normally have been scheduled. These probabilities of 0.97 and 0.9, representing confidences in successful subsequent propulsive maneuvers, are examples only and are not supported by reliability assessments. However, they are not critical to the analysis; substantially lower values could be accommodated within the planetary quarantine constraint by merely requiring greater biases for these maneuvers, thus reducing the probability of attaining an impacting trajectory.

Sample magnitudes of the biases for the interplanetary maneuvers are shown in Table 3-15.

3.5.3.3 Orbit Selection

In the preselection of an orbit to be attained as a result of the orbit insertion maneuver, the primary qualities, from the viewpoint of the planetary quarantine constraint, are the altitudes at the apsides. While the atmospheric drag has its greatest effect at periapsis, the

Table 3-15. Typical Maneuver Magnitudes and Biases

Interplanetary Maneuver	Assumed Time of Execution	Maneuver Magnitude (m/sec)	Bias in Targeting		
			No. of σ	Velocity (m/sec)	Distance in R-T Plane (km)
(Injection)	I	-	4	20	160,000 to 500,000
1st (Arrival date separation)	I + 3 d	150	3.6	2.5	20,000 to 60,000
2nd (First midcourse correction)	I + 30 d	4	4.4	0.08	600 to 2,000
3rd (Second midcourse correction)	E - 30 d	0.5	_____	_____	no bias _____

I = injection
E = encounter



eccentricity of the orbit, determined by also specifying apoapsis altitude, influences the portion of the orbit over which this drag applies. Therefore, the lifetime predictions which may be made in advance of actually establishing the orbit are based on the estimated density of the upper atmosphere of Mars, the size of the orbit, and the ballistic coefficient $M/C_D A$ of the orbiting vehicle. (If major unsterilized components in addition to the spacecraft are to be placed in essentially the same orbit, the component with the lowest ballistic coefficient will govern. This specifically applies to the separated lid of the capsule canister, if a mission profile is chosen in which this lid is jettisoned in orbit about Mars.)

The considerations of the above paragraph specify the minimum orbit size which may be permitted. The results of such calculations are shown in Figure 3-41. This figure is based on a model atmosphere whose density was regarded as "realistic" before the results of the Mariner 4 occultation experiment were obtained (Ref.12), and subsequently regarded as "conservative" by JPL.

In order to verify that the orbit size will not be less than the minimum permitted size, it is necessary to choose a nominal orbit sufficiently large that expected dispersions will not violate the minimum size identified. Components of this dispersion are:

- Uncertainty in the estimate of the approach trajectory at the time the orbit insertion maneuver is programmed
- Errors in execution of the orbit insertion maneuver

Analyses of these contributions to the dispersion of the orbit about Mars is given above in Section 3.4.2.

While the planetary quarantine constraint, as it refers to orbit lifetime, influences the minimum orbit size which may be selected, a more definitive choice of orbit size, the inclination of the orbital plane, and the rotation of the apsidal line within that plane tend to be determined as a result of spacecraft propulsive performance considerations and the scientific objectives of the mission, including experiments to

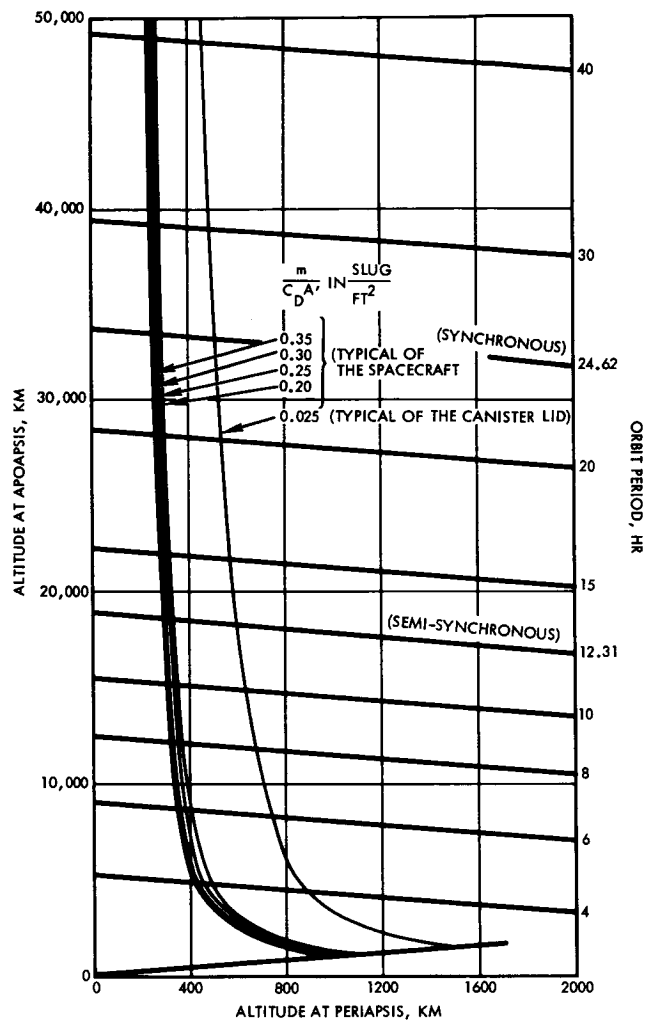


Figure 3-41
ORBITS THAT ASSURE AT LEAST 12 YEARS before decay should keep their periapsis altitude above 500 km, as governed by that component with the lowest ballistic coefficient, although this required altitude will increase sharply for apoapsis altitudes below 8000 or 10,000 km. These curves assume the Martian model atmosphere that preceded the Mariner 4 measurements and are therefore probably quite conservative (i. e., biased toward higher altitudes).

be carried by the orbiter and the delivery of the capsule to a desired landing site.

3.5.3.4 Orbit Insertion

Assuming the nominal orbit has been selected to satisfy the quarantine constraint in recognition of the topics discussed above, the orbit insertion maneuver itself will violate the planetary quarantine constraint only in the event of a malfunction. Figure 3-42 shows, on a scale of vector velocities, an example orbit insertion maneuver. The vector

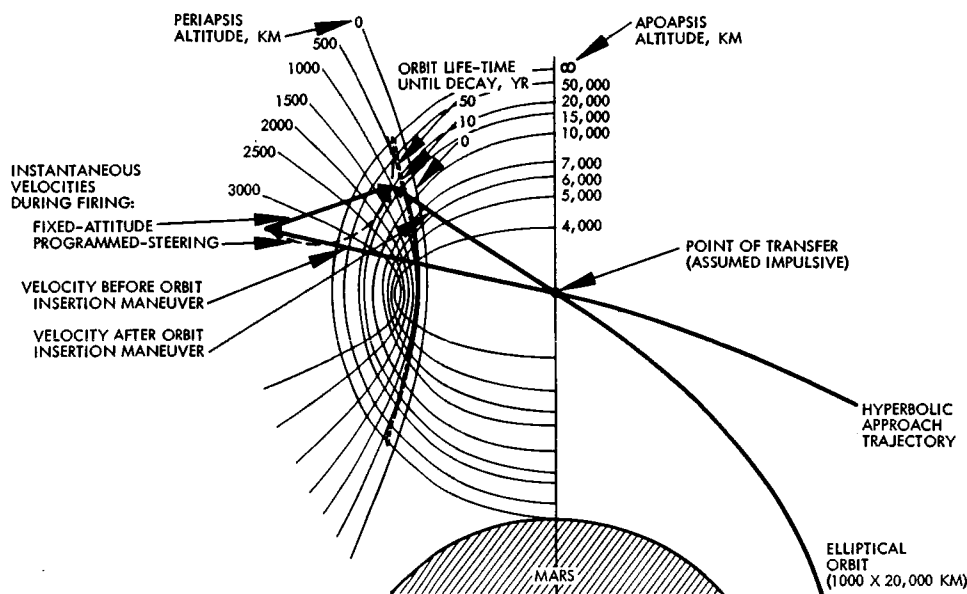


FIGURE 3-42

ORBIT INSERTION MANEUVER changes vector velocity as shown. Grid indicates how instantaneous orbit size parameters change during engine firing. With programmed-steering maneuver, spacecraft is never on impact or early-decay trajectory, even if engine fails to shut off.

velocities before and after the engine is fired are shown, superimposed on a grid which indicates periapsis and apoapsis altitudes as a function of vector velocity.

In the normal orbit insertion maneuver, conducted with the spacecraft in a constant attitude, the instantaneous planetocentric velocity progresses uniformly during firing, along a straight line from the vector velocity "before" to the vector velocity "after" firing.

A malfunction of the orbit insertion maneuver, resulting in the achievement of an unintended final velocity vector may be assessed by reference to this figure, in the context of what the nature of the resulting orbit will be. Obviously, if the malfunction is that no engine firing takes place at all, the spacecraft will merely continue on its hyperbolic path, departing from the planet without having obtained any orbit. While the mission fails, the planetary quarantine constraint has not been violated. Other modes of malfunction would lead to a premature shutdown of the engine, or an inability to turn off the engine at the desired time, resulting in the burning of more than the desired amount of propellant, and the achievement of too great a velocity increment. Figure 3-42 shows

that premature engine shutdown, leading to a deficient ΔV , would not result in impact nor in an orbit of reduced lifetime. On the other hand, if the velocity increment were substantially greater than that intended, due to a failure to shut down the engine, there would be a definite threat of insertion into an orbit which would impact the planet. Therefore the major safeguard required in the orbit insertion process is to guard against failure of the propulsive maneuver to terminate at the desired point. To this end, it is desired to have two independent mechanisms for shutting down the engine and it is proposed that these mechanisms be implemented. For example, the primary shutdown mechanism depends on the use of an integrated accelerometer, as this will give a more accurate indication of completion of the desired velocity increment. A secondary signal, which can serve as a backup, should be provided by a clock on board the spacecraft. The clock would automatically send a command to shut down the engine several seconds after the expected time for the accelerometer to signal the end of firing.

In addition to these two means of commanding the shutdown of the engine, it is possible to interrupt the propellant flow to the engine at more than one spot, so that failure of the ball valves of the engine, the normal mechanism of achieving engine shutdown, would not lead to an impacting trajectory. The slow-acting feed system isolation valves would serve as this backup to the ball valves.

It is possible to consider an orbit insertion maneuver which is not based on the fixed attitude steering law of the nominal mode. Such a maneuver would require a programmed steering rate, and its purpose would be so that if the engine failed to shut down, continuing propulsion with the spacecraft turning at the same rate would not lead to an impact trajectory, nor to an orbit of inadequate periapsis altitude. The locus of instantaneous spacecraft velocities is shown by the dashed line in Figure 3-42, and it can be seen how the goals of this mode are achieved. However, the penalties of this mode are relatively great:

- The total propellant required is increased, because the effective ΔV , as measured by the integral along the dashed line, is greater than the linear ΔV .



- Implementation of this steering law is a complexity in itself, and may lead to pitch or yaw rates above the normal programmed turn capacities.
- While the constant rate steering law would enhance the observance of planetary quarantine in the example illustrated, in which the apsidal line is rotated backwards at insertion, it is not clear that a similar situation could be achieved for zero or positive rotations without an even greater penalty in propellant.

Therefore, this approach is not recommended (thrusting while in a programmed turn) for the normal orbit insertion maneuver and is presented only as a conceivable alternate.

3.5.3.5 Orbit Lifetime Prediction

As a part of the observance of the planetary quarantine constraint, it is necessary to monitor, by tracking the spacecraft in orbit, those influences leading to the decay of the orbit through atmospheric drag. This tracking procedure, scientifically valuable in itself, can lead to a direct measurement of atmospheric densities at the altitude of the orbit, and will serve as a backup to the orbit selection process in that it will detect a situation leading to premature decay and permit corrective maneuvers to be carried out. If the sole influence on orbit lifetime is atmospheric drag, it can be shown (Ref.13) that apoapsis altitude will decay in an almost linear fashion, so that the decrease in the period of the orbit which would accompany premature orbit decay will be noticeable within several days after orbit insertion.

The situation is complicated somewhat by two other influences on the orbit, neither of which is likely to have a strong effect on lifetime by itself, but both of which permit atmospheric drag effects to be magnified by imposing secular variations in periapsis altitude. These effects are the third-body gravitational effects of the sun and small perturbations due to nongravitational forces on the spacecraft, principally those arising from attitude control gas leakage or unbalanced forces. However, in spite of these other perturbing forces it is expected that satisfactory predictions of the orbit lifetime can be made after tracking in orbit for several weeks.

3.5.3.6 Orbit Trim Maneuvers

At the time that orbit trim maneuvers are proposed and programmed, it should be possible to base them on a very refined estimate of the current spacecraft orbit and on an adequate model of the atmospheric drag forces to be expected, both attained through tracking of the spacecraft. Therefore, the primary concern in conducting an orbit trim maneuver is the possibility of malfunction in the execution of the maneuver. The principal fear is that a maneuver which is intended to reduce the orbit size—particularly if it reduces periapsis altitude—leads to an orbit which either impacts the planet or has insufficient altitude if the engine firing fails to terminate. The remedies and safeguards against this type of failure mode are the same as those discussed above for the orbit insertion maneuver: redundant sources of the command to terminate the maneuver, and redundant valves for shutting off the engine. The allocation of a greater allowable probability of impact (4×10^{-6} , compared to 2×10^{-6} for the orbit insertion maneuver) merely recognizes that more orbit trim maneuvers may be programmed than the single orbit insertion maneuver.

3.5.4 Control of Contaminated Efflux and Ejecta

Potential sources of contaminated efflux (fluids) and ejecta (solids) from the spacecraft are illustrated in Figure 3-43. They include external surfaces and subsurfaces of the spacecraft; sealed units which contain free volume, such as tape recorders and drive mechanisms; attitude control gas; propulsion system. Mechanisms whereby contaminated material could be released from these sources have been identified as follows, each with the probability allocation of Figure 3-39:

- Expulsion of contaminated gases by attitude control and propulsion systems and by outgassing 2×10^{-6}
- An explosion of the liquid propellants 2×10^{-6}
- Structural disintegration due to rupture of pressurized tanks 3×10^{-6}

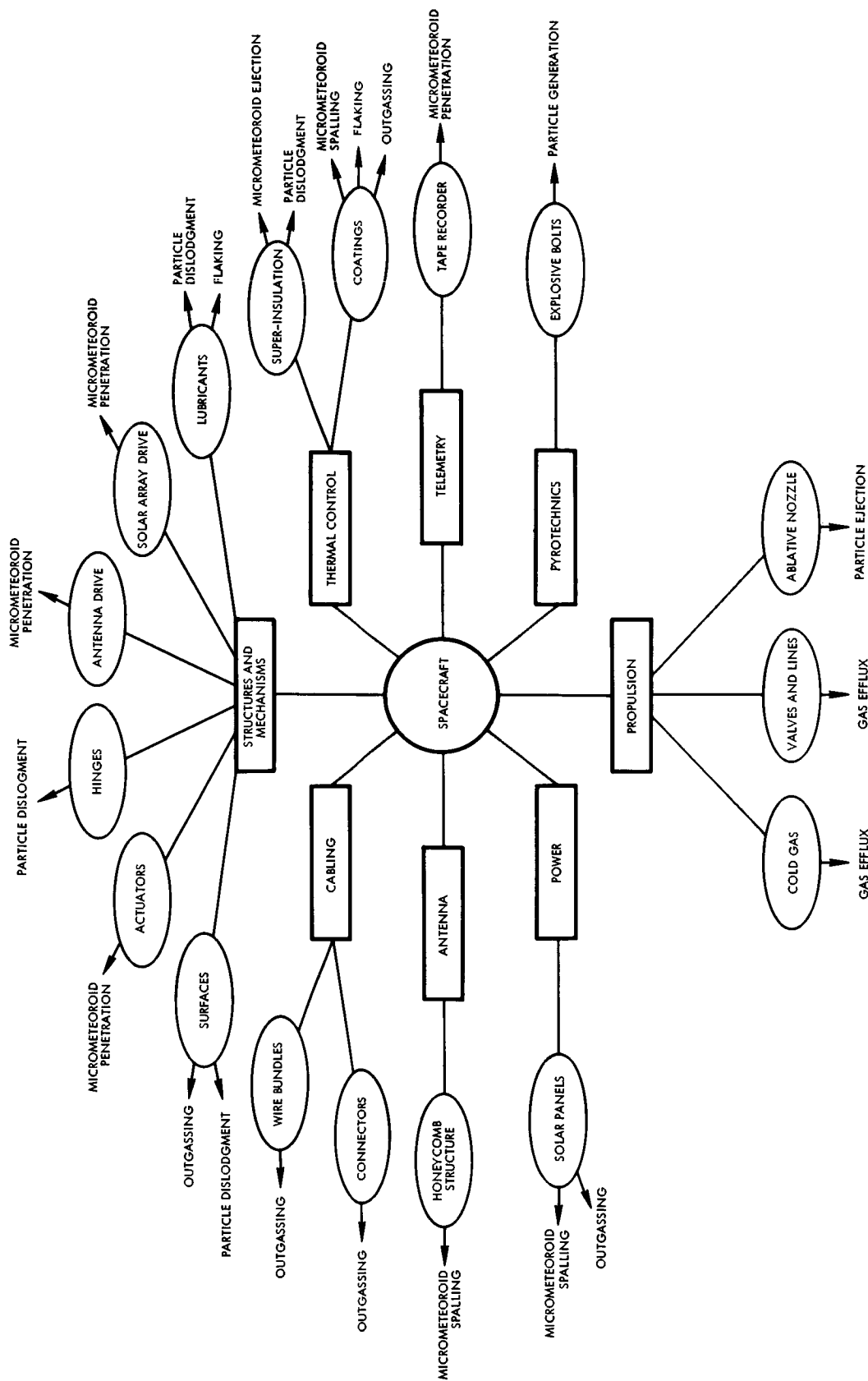


Figure 3-43
POTENTIAL SOURCES OF CONTAMINATED EFFLUENTS on the Voyager spacecraft include these 16 areas and possible mechanisms for dislodgement or ejection.

- Particle dislodgement due to vibration, movement of appendages, surface outgassing
 - Particle release by rupture of sealed units by micrometeoroid penetration
 - Micrometeoroid spalling
- } 3×10^{-6}

Contamination by efflux from the spacecraft is limited by controlling the microbial population of these fluids. Explosions are limited by good propulsion system design. Structural disintegration of pressurized tanks is a complex subject, as it could arise from meteoroid penetration, stress corrosion, or inadequate handling safeguards during assembly. The last three mechanisms above are combatted by limiting the microbial population on and under external surfaces. These various methods are discussed in the following paragraphs.

3.5.4.1 Controlling Microbial Population of Expendable Fluids

Control of viable efflux from the propulsion system is inherent in the design of the system. The propellants, nitrogen tetroxide and a hydrazine derivative, are self sterilizing. Immersion of 10^7 *Bacillus subtilis* spores in nitrogen tetroxide results in a 2 log reduction in viable population after 30 seconds, and no survivors after 8 minutes. D^* values of the same bacteria in hydrazine and Aerozine 50 are 15 and 20 minutes, respectively, with no survivors in both after 24 hours (Ref.14). Thus any part of the propulsion system that is in contact with the propellants for 24 hours will be sterile. Other potential sources of contaminated efflux from the propulsion system are those areas which are not in long contact with the propellants but which are "flushed" by the propulsion exhausts. These are the nozzle and certain valves. The ablative nozzle is formed under pressure and an excess of 400°F ; therefore it is internally sterile. The exterior surfaces which will be swept by propulsion exhausts reach a temperature in excess of $>1000^\circ\text{F}$ at the moment of firing and remain at more than 800°F for more than an hour after firing. Particles from valves are swept into the combustion chamber

*Time to reduce the viable population by 1 log (90 percent).



and are burned with the propellants. Any particles remaining in the valves thereafter are in contact with the propellants and are sterilized.

The attitude control system (ACS) will release pulses of nitrogen or helium gas through nozzles to exert control torques on the vehicle. One operating mode utilizes heated gas. In this mode the gas is heated to 1500°F before exiting, and there is no threat of contaminated efflux. There is also a cold gas operating mode, however, during which time contaminated efflux is possible. Placing a bacteriological filter in the line or sterilization of the ACS are feasible approaches to eliminating this problem. A bacteriological filter in the line will cause a pressure drop and thus complicate the design of the system. Therefore sterilization of the ACS is the recommended approach. The method would be to surface sterilize the assembled hardware with ethylene oxide-Freon 12, then charge the tanks with the gas through bacteriological filters. The filters would be ground equipment and would be removed before launch. More details of the method are given in Volume 9.

3.5.4.2 Explosions

The highest risk of explosions, which would threaten to contaminate Mars by scattering contaminated spacecraft fragments onto a wide range of trajectories, occurs at the ignition of the engine for various propulsive maneuvers. Such a scattering of fragment trajectories could lead to impacts at Mars if it occurred at any of the maneuvers, but particularly at interplanetary trajectory corrections or during orbit trim maneuvers conducted away from periapsis. (Orbit trim maneuvers conducted near periapsis and the orbit insertion maneuver could also lead to impacting fragment trajectories, but the relative ejection velocity of the fragment would have to be greater.)

The most likely cause of such explosions would be an accumulation of fuel in the thrust chamber at the time oxidizer is brought into contact, and could result from inferior propulsion system design or from defective components (valves, etc.). The principal safeguards against explosions lie in the use of a sound, proven design, a comprehensive test program, and quality control procedures which serve to insure that the flight propulsion system will perform as did the tested systems.

3.5.4.3 Structural Disintegration Due to Rupture of Pressurized Tanks

There are several causes which could lead to the rupture of pressurized propellant tanks:

- Defects in the tank affecting its structural soundness, induced by faulty handling during assembly or test
- Reduced tank strength resulting from stress corrosion
- Penetration of the tank by meteoroid impact

The danger of such a rupture is that it would lead (if the tank pressure is above some critical value) to the propagation of cracks, to the shattering of the tank, and, very likely, to major fragmentation of the spacecraft. Thus, the mechanism of contaminating Mars is similar to that following chemical propellant explosions. In this case, however, the energy supporting the fragmentation and ejection is mechanical, derived primarily from the compressed pressurant gas in the tank.

Specific remedies against the above failure modes are:

- Quality control, test, and handling procedures which insure against inducing defects in the tanks, and against the possibility of defective tanks being assembled in flight hardware
- The use of inhibited N_2O_4 oxidizer. Stress corrosion of titanium tanks by nitrogen tetroxide in the past has been solved by recognizing that small quantities of nitrogen oxide inhibit the corrosive action. Test programs of the past and currently under way indicate that stress corrosion can be completely avoided.
- Increasing the general level of meteoroid protection. However, the weight which would be required to insure against a 3×10^{-6} probability of meteoroid penetration would be prohibitive. (For a three-month period at or near Mars, a meteoroid of 1 gram mass would have this probability of impacting an area of 8 m^2 , the average collective cross-section of the propellant tanks.)

A further safeguard which counters all three causes of tank rupture is to keep the tank pressure low except while firing the engine. It is not

clear that this safeguard is necessary, and it is not clear just how low the critical tank pressure is for arresting cracks. If this pressure is low compared to engine operating pressures (95 psi, minimum, in the low-thrust mode; 225 psi, normal, in the high thrust mode), then this depressurization approach requires venting to reduce pressure after each engine firing. The weight penalty to carry sufficient pressurant gas for the required venting and repressurizing operations is rather excessive. Details are given in Volume 3, Section 4.

3.5.4.4 Limiting the Microbial Population of External Surfaces

An analysis has been made of probabilities of contaminating Mars by direct impingement of surface contaminants departing from the spacecraft during interplanetary cruise and planetary orbit (Ref. 10). This analysis concluded that for an initial loading of 10^3 viable spores per square foot, considering an 1100 square foot spacecraft surface area, the probability of such an impingement on Mars by live microorganisms ranged from 10^{-8} to 10^{-2} , depending on the conservatism of the estimate of the processes whereby microorganisms might be transported to Mars. Combining this with the value for probability of growth and spreading recommended at the recent COSPAR meeting,

$$\begin{aligned} & (\text{Probability of Live Impact}) (\text{Probability of Growth and Spreading}) \\ &= (\text{Probability of Contaminating Mars}) \end{aligned}$$

$$\begin{pmatrix} 10^{-8} \\ 10^{-2} \end{pmatrix} \begin{pmatrix} 10^{-3} \end{pmatrix} = \begin{pmatrix} 10^{-11} \\ 10^{-5} \end{pmatrix}$$

Unless the upper limit can be discounted, the above analysis indicates that the external microbial load on the spacecraft at the time of launch must be held to 10^2 viable spores per square foot to meet the quarantine allocation. Two practical methods of accomplishing this control of microbial contamination are "bioclean" manufacture and handling of the spacecraft, and surface decontamination of the spacecraft followed by aseptic handling through launch operations.

"Bioclean" manufacture and handling is conducted by using clean rooms, and shipping and storage containers meeting Fed-STD-209 requirements. Optimum control of contamination without isolating the hardware from human contact can be achieved by the use of laminar flow clean rooms meeting Fed-STD-209 Class 100 standard. Fallout contamination in this type of facility is less than one viable microorganism per sq ft/hr. Electronic modules manufactured in such a facility have contained less than 5 viable microbes per square foot. While achieving the optimum, however, this kind of facility is very expensive to build and maintain, and the time required to fabricate a part in it is about three times longer than required in a conventional manufacturing facility.

A good degree of microbial control can also be obtained by Fed-STD-209 Class 10,000 manufacturing. Measurements of contamination levels on Mariner '67, which was assembled in a laminar flow Class 10,000 facility, showed values of less than 100 spores per square foot. Microbial burden on Lunar Orbiter, also assembled in this type of facility, averaged 131 spores per square foot at the time of system test before shipment to the launch site. Unfortunately, contamination during transportation was not controlled as well, so that the burden rose to 4.5×10^4 per square foot during this period. Prelaunch check-out and fueling was conducted in a Class 100,000 facility. The microbial level during these operations stabilized around 2×10^3 per square foot. These experiences demonstrate that while clean rooms can be used to control microbial contamination, their effect can be nullified by one uncontrolled event.

One further advantage to the use of laminar flow rooms is the phenomenon of "die-off." After a few days in a laminar flow room, microbial population apparently decreases due to die-off of the microorganisms. This phenomenon could be utilized to regain to a certain degree the contamination control lost by an uncontrolled event.

The principal advantage of surface decontamination followed by aseptic handling is that it is not as dependent for success on tightly controlled environments during all spacecraft ground operations. The method of surface decontamination is to enclose the mated spacecraft



and encapsulated lander in the planetary vehicle compartment just prior to launch, filling the compartment with 12% ethylene oxide-88% Freon 12 gas, and after an 11 to 18 hour soak, aseptically purging the compartment with sterile nitrogen. Physical access to the compartment is not allowed after this operation. The planetary vehicle remains enclosed, and protected from recontamination, until earth parking orbit and/or injection. Details of this method are described in Volume 9. The method works best if clean room manufacture, shipping, and storage are utilized, because the use of clean rooms can minimize the viable population, which influences the time to decontaminate, and it can minimize contaminant sources which are difficult to sterilize (such as dirt, grease, corrosion products). Limiting the microbial load to 10^3 microorganisms per square foot should result in effective decontamination with an 11-hour gas soak.

A study of effects of the decontaminating gas on materials and components indicates that the spacecraft can be treated with ethylene oxide-Freon 12 without deleterious effects. Results of this study and analysis of the effects of incorporating this process on program schedule and facility requirements are presented in Volume 9.

From a consideration of the factors involved in the two methods of limiting surface contamination, surface decontamination appears the more feasible. If Class 10,000 handling is chosen as the primary method, either surface decontamination should be available as a backup in the event of failure in contamination control during a ground operations phase, or a laminar flow clean room should be available for storage of the spacecraft for a number of days prior to launch.

3.5.4.5 Avoiding Dispersion of Contaminated Components

The dispersion of contaminated spacecraft components due to violent fragmentation of the entire structure has been discussed in Sections 3.5.4.2 and 3.5.4.3. As noted earlier, other means of dispersing contaminated particles are

- Dislodgement due to vibration and movement of appendages

- Dislodgement or spalling due to micrometeoroid impact
- Particle release by rupture of sealed units (black boxes)

The primary approach to combatting the first means is to limit the number of particles which may be dislodged from various spacecraft surfaces, and this is done by the same procedures indicated in the preceding section for insuring biological cleanliness. There is no remedy against the ejection of particles from the external spacecraft surface by meteoroid impact; the primary defense against contamination by this means is, again, to see that biological contaminants on the surface have been reduced.

The release of contaminated particles by the rupture of sealed units is not of great concern. (These units are primarily those sealed to prevent material deterioration by exposure to vacuum.) These subsystem components are generally located so as to be protected from meteoroid penetration. This location would also serve to trap larger particles which are released, and prevent their escape from the spacecraft. Spontaneously developing leaks in the seals are likely to be small, and would be expected to release mainly the very small particles which are incapable of protecting a live microorganism from solar ultraviolet radiation while transporting it to Mars.

3.5.5 Controlling Cross Contamination of the Capsule

It is assumed that the capsule is within a sealed canister, after having been heat sterilized, when mated to the spacecraft. Prelaunch operations must have as one of their objectives avoidance of violating the integrity of the canister. These operations include:

- Accepting the sterilized, sealed capsule
- Mating the capsule to the spacecraft
- Performing combined system tests
- Enclosing the planetary vehicle within the planetary vehicle compartment (PVC)
- Performing terminal ETO decontamination on the PVC, if required



- Mating two PVC's with each other and with the launch vehicle
- Installing the flight shroud
- Countdown and launch

Once the capsule canister lid is removed during flight, there is a possibility of contaminating the lander surface by efflux from the spacecraft. This efflux is the same as tabulated in Section 3.5.4. The period of concern depends on the sequence for jettisoning the canister. If this is done during the final approach phase, the period may be 5 to 10 days. If it is done in Mars orbit just before capsule separation, it is closer to 2 or 3 hours.

The methods of minimizing the possibility of contaminating Mars directly (i. e., controlling spacecraft surface and subsurface contamination, and sterilizing the attitude control gas (discussed in Section 3.5.4)) also apply to the avoidance of capsule contamination by spacecraft efflux. In addition, use can be made of the planetary vehicle geometry. While the capsule is exposed by discarding the canister lid, the remaining canister half may shield the capsule from direct line of sight of spacecraft surfaces or the attitude control gas nozzle. The TRW-recommended spacecraft design conforms to this geometrical requirement.

A further possible exposure will occur when the capsule separates from the spacecraft, for it then abandons the protection of the remaining canister half. It rapidly achieves the immunity to contamination afforded by distance, however, because it subtends continually smaller angles as it departs at a velocity of some 30 meters per minute.

Some attenuation of possible live microorganisms on the capsule surface will occur during capsule descent, due to solar radiation on surfaces so exposed. Further advantage of this phenomenon can be achieved by use of programmed changes of capsule orientation during descent, so as to expose to solar radiation successively each portion of the exterior surface of the capsule. This exposure would reduce by several orders of magnitude the probability of survival of microorganisms

on the exterior of the capsule. On smooth nonporous surfaces, microorganisms would be almost as sensitive to the ultraviolet component of solar radiation as they are if existing in space unattached, in which case a 20-second exposure attenuates the population by one decade.*

3.6 ACCOMPLISHMENT OF SCIENTIFIC MISSION OBJECTIVES

In brief, the spacecraft scientific objectives for the 1973 Mars Voyager mission are to obtain information for selecting a landing site for the 1973 capsules and for planning and design of subsequent more advanced Voyager missions. This entails experiments on the spacecraft to acquire basic information on the characteristics of the planet's surface and atmosphere and on the planetary environment.

In keeping with these objectives, the following types of observations are to be performed during the 1973 mission from the orbiting Voyager spacecraft:

- Orbital photography of most of the planet overflown at low (1000 m), medium (100 m), and high (10 m) resolution
- Determination of atmospheric constituents, temperatures, density profiles, and measurements of minor constituents and water vapor
- Radiometric determination of surface temperatures
- Determination of the nature of the ionosphere
- Measurement of physical properties of the planet

A more detailed discussion of the scientific objectives and experiments for the Voyager mission is given in Volume 5.

* In vacuum, the most resistant bacterial spores show a sensitivity to ultraviolet radiation which corresponds to a half life of 6 seconds or less exposed to sunlight at 1.4 AU from the sun. (Actual experimental data indicate half lives ranging from 1 to 6 seconds, depending on experimental conditions.) A half life of 6 seconds implies attenuation of one decade in 20 seconds, and 10 decades of 200 seconds. Thus, it is concluded that any process in which all microorganisms are exposed unattached to sunlight for over 20 seconds carries no threat of survival of viable organisms. "Unattached" is used because even very small particles to which the spores are attached offer some protection from the effects of the radiation, and larger ones may more effectively shield them.

3.6.1 Orbital Photo-imaging

One of the primary photo-imaging requirements for the 1973 Voyager mission is fairly complete coverage of the Martian surface at rather low resolution, i.e., surface mapping. This can be accomplished by providing monoscopic photography at a resolution of about 1 kilometer from an altitude of 1000 kilometers in either color or black and white. Color information can be obtained by imaging the same area successively through red, green, and blue filters; the same camera can provide black and white coverage by using a clear filter. The use of this level of resolution is suggested for several reasons:

- a) Photographic observations of the moon from earth are limited to about this resolution by atmospheric inhomogeneities. Such photography has provided a satisfactory starting point for planning and carrying out more detailed photography from lunar spacecraft. On the other hand, 1000-meter resolution pictures would reveal details on the Martian surface 200 or more times smaller than can be observed from earth. Thus, photo-imaging at 1000-meter resolution would provide information significantly better than we now have, known to be useful as a basis for more detailed observations, and at a level with which astronomers have already had considerable experience in lunar work.
- b) Only about 5×10^9 data bits are required to describe the entire surface of Mars in one color at this resolution. This would take about 27 hours total transmission time at 51.2 kbits/sec. Thus, 1000-meter surface resolution imaging is compatible with returning to earth information on broad coverage of Mars during the very early part of the mission when decisions need to be made about the landing site for the capsule, future modes of data gathering, and the like.
- c) After initial rapid mapping has been substantially completed, repeated coverage in monochrome or full color can be obtained with reasonable allocations of transmission time. Color information can provide data useful in deducing rock or soil types. Repeated coverage can give information on seasonal or other temporal changes in the appearance of the Martian surface.

The low resolution coverage must be augmented by imagery with as much higher resolution as possible. A ground resolution of

10 meters is about the highest obtainable consistent with mission constraints. Capability for getting some information at this level of detail is desirable, but even if the entire transmission capacity for the mission were allotted to this, pictures of only one-half of one percent of Mars surface could be sent to earth. Thus an intermediate resolution level, providing a compromise between coverage and detail, should be included; this was chosen as 100 meters, the geometric mean between the high and low extremes. At this resolution a substantial fraction, but not all, of the surface of Mars can be covered.

In summary, photo-imaging objectives for 1973 consistent with feasible instrumentation can be specified as:

- Color and/or black and white coverage of all the Martian surface overflown having suitable illumination at 1000 meters resolution
- Monoscopic and selected stereo coverage of about 25 to 40 percent of the Martian surface at 100 meters resolution
- Limited high resolution coverage (0.1 to 0.25 percent of the surface) at 10 meters resolution

The extent to which these objectives are met by the hypothetical all-TV photo-imaging system is shown in Table 3-16. The coverage estimates are based on the nominal ground coverage, with a total data link allotment to the photo-imaging system of 2.64×10^{11} bits over a six-month mission.

The coverage obtainable with the recommended film/TV combination of a low resolution color TV camera and a dual-framing film camera for medium and high resolution photography is shown in Table 3-17. The recommended system provides considerably more flexibility in mode of operation than the all-TV system. In particular any different allocation of the data capability is possible; for example, either the medium or high resolution system could obtain enough coverage to use all the transmission time. Thus the medium resolution coverage could be as high as 52 percent (giving no high resolution coverage) or as low as zero (with 0.52 percent coverage of Mars at high resolution).

Table 3-16. Coverage of Mars Obtained with Hypothetical Photo-Imaging System

	Low Resolution TV (color)	Medium Resolution TV	High Resolution TV
Nominal ground resolution	1 km	100 m	10 m
Nominal coverage/frame (at 1000 km), km	500 x 500	150 x 150	10 x 10
Bits/frame (6-bit encoding)	8.58×10^6	77.04×10^6	34.32×10^6
Frames/orbit (maximum)	11 color pictures (33 frames)	27	18
Data/orbit (maximum), bits	2.83×10^8	2.08×10^9	6.17×10^8
Ground coverage/frame (nominal), km ²	2.5×10^5	2.25×10^4	100
Area of Mars, km ²	1.46×10^8	1.46×10^8	1.46×10^8
Data allotment (total data link capacity = 2.64×10^{11} bits/6 mo),* bits	1.52×10^{10}	1.94×10^{11}	0.55×10^{11}
Number of frames	585 (color pictures) 1755 frames	2620	1600
Percentage of Mars area covered	100	36	0.11

*Total data assumed for an average trajectory; actual total data returned depends on arrival date as shown in Section 3.7 of this volume.



Table 3-17. Coverage Obtainable with the Recommended Photo-Imaging System

	Low Resolution TV (color)	Medium Resolution TV	High Resolution TV
Nominal ground resolution	1 km	100 m	10 m
Nominal coverage/frame, km	500 x 500	600 x 600	60 x 60
Bits/frame (6-bit encoding)	8.58×10^6	1.24×10^9	1.24×10^9
Frames/orbit	11 color pictures (33 frames)	10	10
Data/orbit, bits	2.83×10^8	12.4×10^9	12.4×10^9
Ground coverage/frame (nominal) km ²	2.5×10^5	3.6×10^5	3600
Area of Mars, km ²	1.46×10^8	1.46×10^8	1.46×10^8
Data allotment (total data link capacity = 2.64×10^{11} bits/6 mo)* bits	0.16×10^{11}	1.24×10^{11}	1.24×10^{11}
Number of frames	585 (color pictures) 1755 frames	100	100
Percentage of Mars area covered	100	26	0.26

* Total data for average trajectory; see Section 3.7 for total data vs encounter date



3.6.2 Other Orbital Scientific Experiments

The basic core group of hypothetical experiments listed in Section 3.1.2.6 for the 1973 mission accomplish the following scientific objectives:

The high resolution infrared spectrometer measures atmospheric constituents such as carbon dioxide, water vapor, and infrared inactive elements such as argon and nitrogen. It also provides data on temperature variations in the atmosphere and reflectivity of the surface in the range of 1.5 to 2.0 and 3.0 to 4.0 microns. The high resolution IR spectrometer will also detect the presence of high clouds and will measure surface or cloud top temperatures.

The broadband infrared spectrometer measures atmospheric constituents in the range of 4 to 15 and 1.5 to 6 microns. The broadband IR spectrometer can be used to measure atmospheric constituents such as methane, ethylene, acetylene, various oxides, nitrogen, ammonia, carbon dioxide, carbon monoxide, and oxygen. Hydrogen peroxide and water can also be qualitatively identified and quantitatively determined by the spectrometer; surface composition can also be determined from background reflection spectra, and surface temperature and albedo can be measured.

The UV spectrometer measures characteristic emission and absorption from minor constituents of the atmosphere and measures characteristics of the Martian aurora and the presence of aerosols. The UV spectrometer can also measure possible atmospheric species. Some of the elements which can be sought are: H(1216 Å), O(1304 Å), N(1200 Å), N_2 (3914 Å), CO (4264 Å), NO(2150 Å), CO(2160 Å), C_2N_2 (3876 Å). Absorption can be used to search for ozone in the region around 2500 to 3000 angstroms and perhaps for other minor constituents as well. By viewing both vertically and at layers near the limb the scale heights can be determined for those species which it is possible to detect and identify. In addition, the UV spectrometer measures rayleigh scattering in the lower atmosphere (acting as a multi-wavelength photometer) and the UV reflectivity of the planet's surface.

The infrared radiometer is useful for measuring the surface temperatures and provides a backup over wide energy bands for the spectrometry measurements. The chief functions of the radiometer are to obtain overall atmospheric temperature measurements and to carry out detailed temperature mapping of the planet's surface which can be correlated with the photo-imaging results.

The radio occultation experiment is performed to measure the scale heights of the Martian atmosphere and ionosphere, and to supplement the other experiments described above in the determination of such properties as atmospheric composition, temperature, and density. As described in Volume 5 the simplest occultation experiment placing the least demands on the spacecraft system utilizes the spacecraft downlink communication signal. Phase shifts in opposite directions are produced during transmission through an ionosphere and through a neutral atmosphere.

3.7 SPACECRAFT COMMUNICATION PERFORMANCE

The communication subsystem has been designed so that data rates and modes are available to meet all the requirements listed in Section 3.1.2.7. The quality with which commands and telemetry can be communicated and metric measurements made, however, varies with the range as well as spacecraft and ground antenna selection. Table 3-18 lists the maximum range over which selected functions can be performed at the maximum acceptable error rate. These ranges have also made allowance for a margin equal to the sum of the negative performance tolerances for each element of the system. Thus, if nominal performance is achieved, each range can be extended to between 1.5 and 2 times its listed value. Entries have been made for various combinations of both spaceborne and ground antennas. Thus, performance in both normal and backup modes can be assessed. The largest range listed is 395×10^6 km which equals the conjunction range between Mars and earth for 1973 and will never be exceeded.

The maximum range data has been used to establish total video data capability for in-orbit operation for the 1973 mission. Figure 3-44



Table 3-18. Voyager Performance Given as Maximum Range for Selected Functions and Antenna Combinations

Operation	High Gain Antenna	Medium Gain Antenna	Low Gain Antenna
210 ft Dish			
Range rate determination	> 395	> 395	> 190
Range measurement	> 395	> 395	-
Command reception	> 395	> 395	> 395
Engineering data transmission (8 bits/sec)	> 395	> 395	240
Scientific data transmission (51.2 kbits/sec)	205	103	-
Scientific data transmission (12.8 kbits/sec)	293	147	-
Scientific data transmission (3.2 kbits/sec)	> 395	293	-
85 ft Dish			
Range rate determination	> 395	205	-
Range measurement	> 395	200	8
Command reception	> 395	> 395	>395

Notes: All ranges are given in millions of kilometers.

All calculations are based on minimum acceptable bit error rate and on a margin equal to the sum of negative tolerances.

indicates the cumulative data as a function of time after encounter for three encounter dates. The changes in slope of the curves indicate when a reduction in data rate is required by the increasing range. The earliest and latest dates correspond to the extremes of trajectory capability. The mid date corresponds to the time of optimum launch from a payload point of view. From the data point of view, however, it can be seen that the earlier launches are considerably more desirable although all launch dates permit collection of nearly twice the data specified in the lifetime design goal of six months.

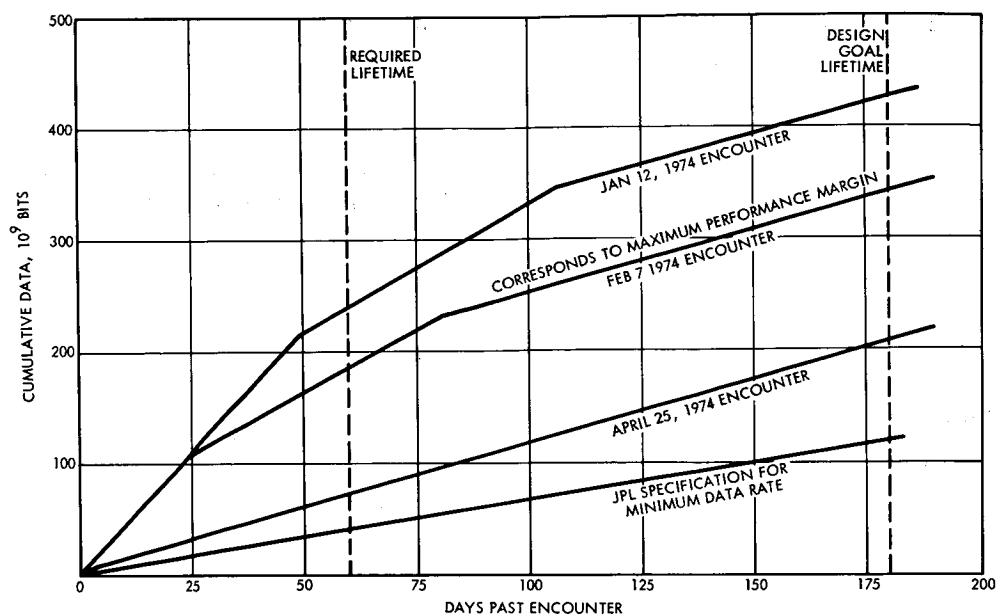


Figure 3-44
POST ENCOUNTER CUMULATIVE DATA is shown for the 1973 LAUNCH for three different arrival dates. The spacecraft's high-gain antenna is utilized to provide 51.2, 25.6, or 12.8 kbits/sec depending on range.

Section 3.6 deals with the scientific requirements for the mission and mentions that a very satisfactory mission can be performed with 2.64×10^{11} bits of data over a 6-month period. Figure 3-44 indicates that for all except the very latest encounter, this is possible and in fact for an early encounter about half again as many bits can be transmitted. For an early launch the assumed 2.64×10^{11} bits could be communicated in slightly over the specified lifetime of two months.

3.8 PROBABILITY OF SUCCESS

3.8.1 Summary

3.8.1.1 Reliability Estimates

The estimated probability of success for the recommended Voyager spacecraft performing the 1973 Mars mission is 0.692 for a single spacecraft and 0.905 for at least one of two spacecraft. The mission time for this analysis is 6800 hours, which includes the maximum 1973 interplanetary transit time of 224 days and 60 days in Mars orbit. The success estimates for the system represent the contribution of the basic spacecraft subsystems, but not including the science



package, launch vehicle, or capsule. Table 3-18 presents the reliability estimates for each spacecraft subsystem.

Table 3-18. Voyager Spacecraft Reliability Estimate

Subsystem	Probability of Success for 6800-Hour Mission
Guidance and control	0.958
S-band radio	0.916
Command and sequencing	0.976
Electrical power	0.985
Telemetry and data storage	0.921
Electrical distribution	0.999
Propulsion	0.966
Structures	0.972
Thermal control	0.994
Planetary scan platform	0.955
Total Spacecraft	0.692

These estimates are conservative since they do not take into account ground-command backup of spacecraft failures or decrease failure probabilities associated with some equipment (such as the TWT's) during periods of nonoperation. The estimates also presuppose maximum interplanetary transit time. This conservatism is consistent with the philosophy underlying the use of reliability estimates during the conceptual design phases; specifically, to determine for each subsystem the areas which may limit reliability and to recommend improvements.

3.8.1.2 Analytic Approach and Assumptions

The reliability estimates of the spacecraft subsystems were prepared in the following sequence. First, determine the mission duration to be used for reliability analyses. Then, divide the mission

into a logical sequence of mission phases (Table 3-19) and determine those phases during which particular equipment is in service. Now, establish equipment operating profiles by relating the capabilities of each to performance of specified functions within the phase.

Table 3-19. Mission Phases and Durations
(Hours) for Reliability Analyses

Launch		Injection		Acquisition		Inter- planetary Transit		Interplanetary Trajectory Corrections	
Boost	Cruise	Boost	Cruise	Boost	Cruise	Boost	Cruise	Boost	Cruise
0.20	0.40	0.10	0.10	-	2.00	-	5350.	0.30	6.00

Mars Orbit Insertion		Mars Orbit Trim		Capsule Separation		Mars Orbital Operations	
Boost	Cruise	Boost	Cruise	Boost	Cruise	Boost	Cruise
0.10	4.00	0.20	8.00	-	4.00	-	1425.

The next series of steps begins with failure modes and effects analyses to determine the potential modes of equipment failure and their impact on subsystem success. Now, we relate the subsystem failure modes to the mission functions to determine their impact on mission success. And as a next step, we generate mathematical models to represent the relationship between subsystem success and equipment-component failure and, where applicable, prepare reliability block diagrams of the models. Next, determine appropriate failure rates and stress factors for use in the reliability analyses and comparisons (Appendix E). These values have been derived from standard sources, and experience gained on TRW spacecraft such as Pioneer, OGO, and Vela.

For all subsystems other than structures, computer programs were used to combine the failure rates and stress factors with the reliability



models to provide the estimates. For structures, the estimate was derived using a stress-strength technique which evaluates the probability of an environmental stress exceeding the designed margin of safety. A sensitivity study was performed to determine the items in each subsystem which limit the reliability level, and the relative magnitude of the limitation.

Recommendations were made as to means of reducing the impact of the limiting reliability items. Tradeoff studies were conducted to determine the applicability of various redundancy implementations for increasing system reliability subject to specified system weight and cost constraints (see Volume 6, Section 3.3).

The principal assumptions underlying the reliability predictions are:

- Failures among items are in a statistical sense independent.
- For items other than one-shot devices, the times to failure are exponentially distributed. (This assumption appears adequate for the present study.) In succeeding program phases, emphasis will be placed upon determining appropriate distributions based upon the data received from TRW satellites operating in orbit.
- Finally, a set of success criteria has been drawn up for the spacecraft subsystems (Table 3-20).

3.8.2 Structures Subsystem Reliability Assessment

The structure subsystem consists of the basic Voyager spacecraft body and frame, the planetary vehicle adapter, and the separation mechanism. The basic functions of the subsystem are to: maintain the structural integrity of the spacecraft and the adapter, separate the spacecraft from the boost vehicle, and provide meteoroid protection.

The reliability equation and block diagram for the structure subsystem is shown in Figure 3-45. A description of the analysis techniques employed, failure modes analyses, and detailed computations are presented in Appendix E.

Table 3-20. Subsystem Success Criteria

Subsystem	Criteria
Guidance and control	Provide three-axis stabilization during flight, gyro control for maneuvers and orbital operations, and velocity increment measurements during engine firings
S-band radio	Transmit data to earth via high-gain or medium-gain antenna and provide for command reception employing at least one of the four antennas on board the spacecraft
Command and sequencing	Receive and process ground commands, provide command sequencing to the various subsystems, and verify memory content
Electrical power	Supply electrical power to the spacecraft from the solar array or, when the array is dark, from at least two batteries
Telemetry and data storage	Process and transmit no less than 80 percent of all engineering data, no less than 80 percent of all low rate and medium rate science data, and store and transmit TV pictures using at least three of four TV tape recorders
Electrical distribution	Distribute electrical power to subsystem areas, and arm and distribute firing signals to all ordnance devices
Propulsion	Power all maneuver corrections, de-boost into Mars orbit, and perform orbit trim
Structure	Support all spacecraft components and protect components against meteoroids
Thermal control	Protect all spacecraft components from excessive heat or cold
Planetary scan platform	Mount scientific equipment and orient it to perform experiments



$$\begin{aligned}\text{RELIABILITY} &= R_{ss} \times R_A \times R_{SM} \times (R_M) \\ &= (0.99753) (0.99714) (0.999603) (0.977) \\ &= 0.9715\end{aligned}$$

WHERE R_{ss} IS THE SPACECRAFT STRUCTURE RELIABILITY
 R_A IS THE ADAPTER STRUCTURE RELIABILITY
 R_{SM} IS THE ADAPTER SEPARATION MECHANISM RELIABILITY
 R_M IS THE PROBABILITY OF NO MISSION FAILURE DUE TO MICRO-METEOROID IMPACT

Figure 3-45
STRUCTURE SUBSYSTEM RELIABILITY MODEL.

3.8.3 Propulsion

The propulsion subsystem performs three basic functions: interplanetary midcourse corrections, insertion into Mars orbit, and orbit trimming during Mars orbit.

For a reliability analysis the propulsion subsystem divides into the pressurization and feed equipment, propellant acquisition equipment, and engine. The feed equipment consists of hardware that supplies helium at regulated pressures to the propellant tankage. The propellant acquisition equipment contains the tank, bellows, valving, and other plumbing for supplying propellant to the engine. The modified LMD engine consists of the thrust chamber and the associated valving. For engine backup, the spacecraft incorporates four 100-pound thrust C-1 engines. In the event of LMD engine malfunction, the four C-1 engines power maneuvers for orbit insertion. It is assumed that the C-1 engines can produce an orbit which permits accomplishment of 75 percent of the orbital objectives. This 0.75 factor is used to modify the probability of mission success using the C-1 engines.

The reliability estimate for the propulsion subsystem, including the four C-1 engines as backup, is 0.9656. The reliability expression for the propulsion subsystem is:

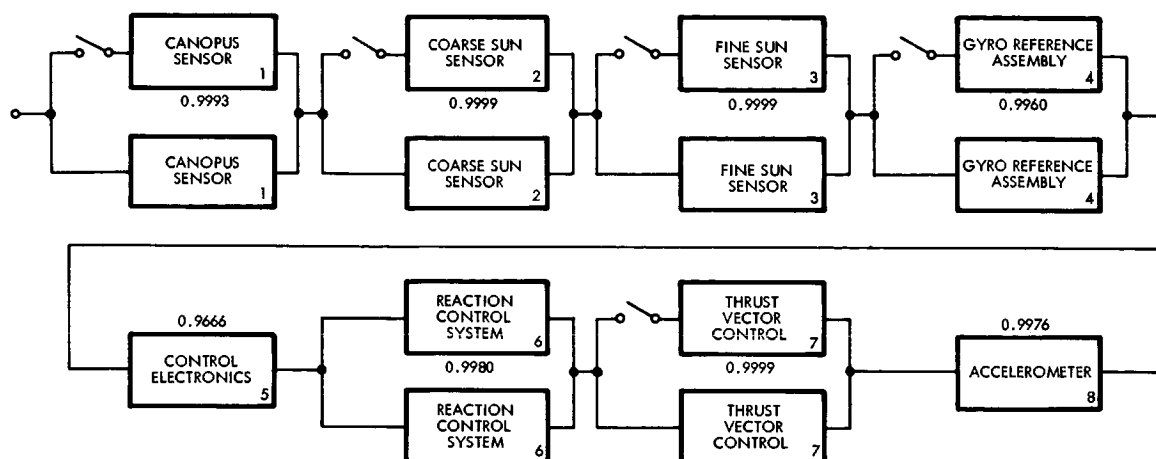
$R_{\text{Propulsion}}$	=	$[R_{\text{F\&P}}] \{1 - [1 - R_{\text{LMD}}] [1 - (R_{\text{C-1}} \cdot P_o)]\}$
$R_{\text{Propulsion}}$	=	Reliability of propulsion subsystem
$R_{\text{F\&P}}$	=	Reliability of feed pressurization and propellant acquisition equipment
R_{LMD}	=	Reliability of LMD engine
$R_{\text{C-1}}$	=	Reliability of four C-1 engines
$P_o = 0.75$	=	Assumed fraction of the mission goals achievable from the looser orbit obtained when the C-1 engines are used for orbit insertion

The detailed sequence of operation for the two types of subsystem firings (pressurized-high or low thrust, and nonpressurized low thrust) and all applicable component data are presented in Appendix E.

3.8.4 Guidance and Control

The guidance and control subsystem provides three-axis attitude control of the spacecraft at all times after separation from the launch vehicle, provides thrust vector control of the engine, and measures vehicle acceleration during propulsion operations. During interplanetary cruise, the spacecraft yaw and pitch axes are stabilized with respect to the sun and Canopus, maintaining the solar array surface normal to the sun. Roll stabilization is maintained by celestial references. Upon command from the ground or the computing and sequencing unit, the guidance and control subsystem reorients the spacecraft to attitudes required for velocity adjustments or capsule separation. Attitude reference is provided by redundant sets of three single-axis gyros. Thrust vector control during engine firing is provided by gimbaling the engine and controlling engine position using electrical actuators for the yaw and pitch axes and pneumatics for the roll axis.

The reliability estimate for the guidance and control equipment is 0.958. Figure 3-46 shows the applicable reliability equation and block diagram. In addition to the indicated component redundancy, there are modes of operation which can result in mission success. For example, the course sun sensors could be used as a backup for the fine sun sensors.



$$R = [R_1 (1 + \lambda_1 T)] [R_2 (1 + \lambda_2 T)] [R_3 (1 + \lambda_3 T)] [R_4 (1 + \lambda_4 T)] R_5 [1 - (1 - R_6)^2] [R_7 (1 + \lambda_7 T)] R_8 = 0.958$$

Figure 3-46

GUIDANCE AND CONTROL RELIABILITY DIAGRAM shows extent of equipment redundancy and component reliability assessment. Mathematical reliability expression is also given.

The backup gyros are in a standby mode and can be activated when required. In the event of failure of one gyro, the redundant set can be employed regardless of the celestial references that were activated.

Appendix E lists the guidance and control subsystem equipment, the failure rates, the probability of success for the mission per unit, and the probability of success for the redundant configurations.

3.8.5 Command and Sequencing Subsystem

The command and sequencing subsystem is comprised of two major units; a command unit, and a computer and sequencing unit. The command unit receives digital commands from the ground via the S-band radio link and provides for (1) routing these commands to the computer and sequencer unit (e. g. , spacecraft maneuver parameters) or directly to specified subsystems, and (2) verification of command reception and execution of the spacecraft's operational status.

The computing and sequencing unit provides timed output commands for sequencing of the flight spacecraft operations and scientific experiments during all mission phases, engine cut-off signals upon attainment of the required velocity increments during engine firing maneuvers, and high- and medium-gain antenna pointing angles. The timing and sequencing parameters within the sequencer may be altered or adjusted

by ground command. Likewise stored program commands may be enabled or inhibited by ground command. All stored programs are verified by readout via the telemetry link prior to execution. Both the command and the computer and sequencing units are required throughout the entire mission.

The reliability for the command and sequencer subsystem is 0.976. The reliability equation and block diagram for the command unit is shown in Figure 3-47 and for the computer and sequencer unit in Figure 3-48. Appendix E lists the command unit equipment, the component failure rates, and the component probability of success for the mission.

The major components in the computer and sequencing unit are redundant; however, failure of a component in the on-line equipment (memory, timers, decoders, and drivers) causes a complete switch-over to the standby units. Failure of the accelerometer counter is not a system failure, as the sequencing unit issues backup engine turnoff commands in the event the accelerometer counter fails. The failure of the function generator will not result in system failure since the sequencer also can issue backup antenna pointing commands to the high- and medium-gain antennas. The reliability estimate does not include the command events register since its function is to count the number of commands issued,

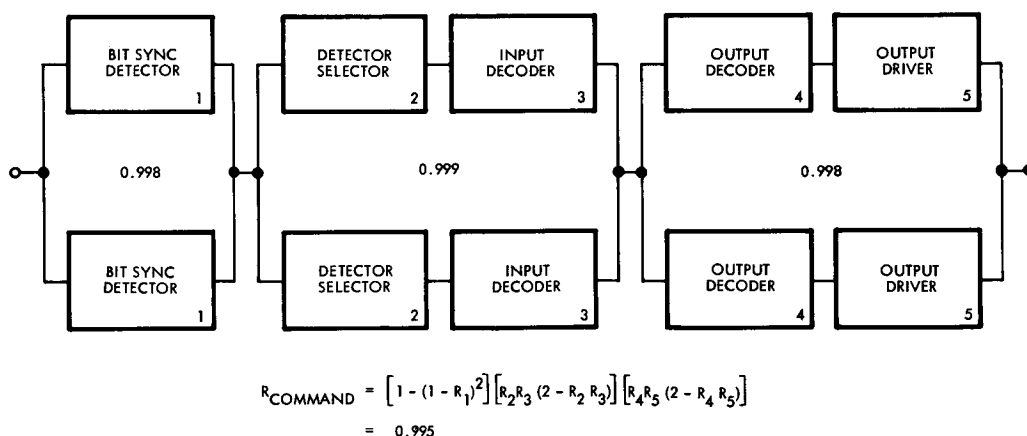
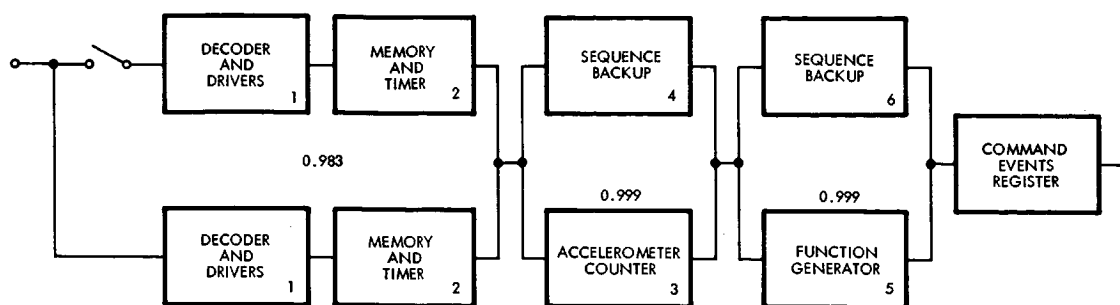


Figure 3-47

COMMAND UNIT RELIABILITY DIAGRAM gives reliability assessment for each major section. Note complete redundancy and additional cross-strapped success paths.



$$\begin{aligned}
 R_{\text{SEQUENCER}} &= \left[R_1 R_2 (1 + (\lambda_1 + \lambda_2) T) \right] \left[R_3 R_4 (2 - R_3 R_4) \right] \\
 &\quad \left[R_5 R_6 (2 - R_5 R_6) \right] \\
 &= 0.981
 \end{aligned}$$

Figure 3-48

COMPUTING AND SEQUENCING UNIT RELIABILITY DIAGRAM shows unit redundancy and cross stripping, plus assessment components or section.

and provide ground control with this information. This register is effectively a piece of test or verification equipment and failure is not considered to be a system failure. Appendix E lists the computing and sequencing unit equipment, the component failure rates, and the component probability of success for the mission.

3.8.6 S-Band Radio

The S-band radio subsystem performs the following functions:

- Receives RF signals transmitted to the spacecraft from the ground stations
- Coherently translates received RF signals
- Demodulates the received RF signal and transmits signals to the command and sequencing subsystem
- Transmits modulated RF signals to the ground
- Provides appropriate antenna patterns for transmit receive

The reliability estimate for the S-band radio subsystem is 0.916.

Figure 3-49 presents the reliability block diagram of the subsystem, and the reliability equation. The assumptions underlying this estimate are:

- a) The primary mode of operation for high-bit-rate transmission and reception is the high-gain antenna and associated

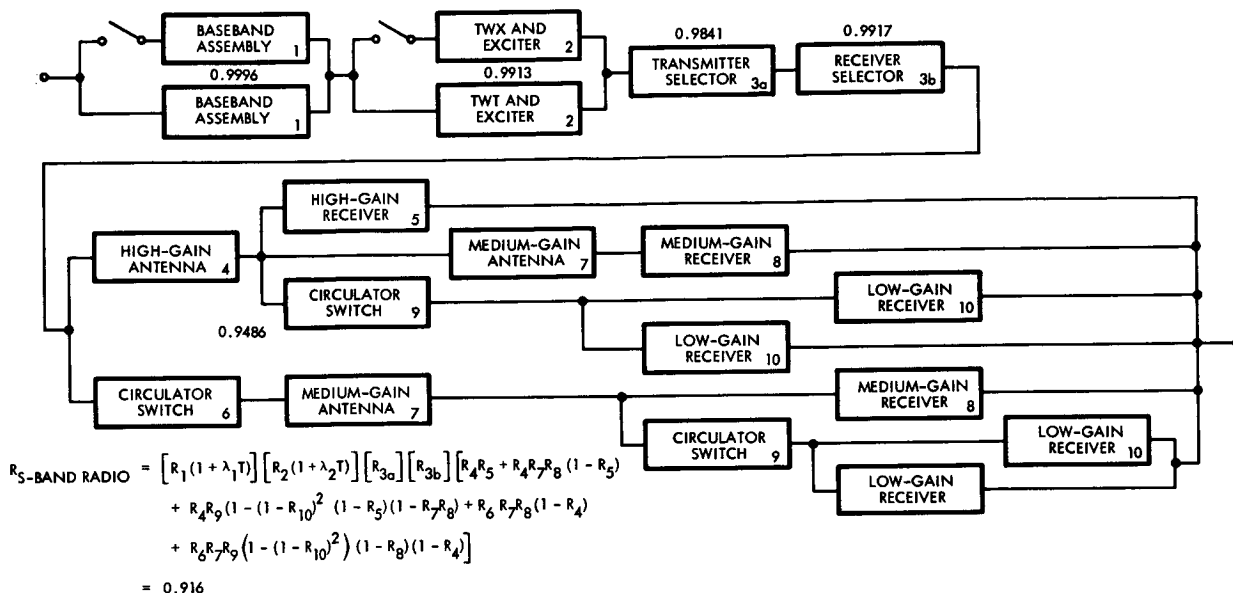


Figure 3-49
S-BAND RADIO RELIABILITY DIAGRAM indicates that application of unit redundancy and the backup capability which exists in the antenna-receiver section.

receiver. If the receiver associated with the high-gain antenna fails, the high-gain antenna is used for transmission and the medium-gain antenna and receiver are used for reception. If both the receivers associated with the high- and medium-gain antenna fail, the appropriate low-gain antenna and receiver are used for reception, at the appropriate bit rates.

- b) If the high-gain antenna fails, the medium-gain antenna is used for transmission and the medium-gain antenna and receiver are used for reception. If this receiver should fail, the appropriate low-gain antenna-receiver link would be used for reception.
- c) If both the high- and medium-gain antennas fail, it is judged that the low-gain transmission rate of 8 bits/sec is too low to effectively transmit TV pictures, film, and medium-rate science data. Thus, this is not considered a success mode.
- d) The baseband assembly is standby redundant with automatic switching upon failure of the on-line unit.
- e) The 50-watt TWT and its associated exciter are standby redundant.

The subsystem components and associated failure rates and probabilities are listed in Appendix E.

3.8.7 Telemetry and Data Storage Subsystem

The telemetry and data storage subsystem receives signals from the scientific experiments, surface scan sensors, spacecraft subsystems, spacecraft transducers, and the capsule (before and during descent). The input signals are sampled, digitized, and organized into a data format according to the mission phase and the type of data required. Synchronization signals are added to the data format and the data is fed to tape recorders for later playback or fed directly to the S-band radio subsystem for real-time transmission. A tape recorder is provided for storage of engineering data and one for low-rate and medium-rate science data. The subsystem is capable of handling data from film cameras, color television cameras, and tape cameras. Four tape recorders are utilized with the TV system.

The basic functions of the subsystem are:

- Format and transmit real-time engineering data
- Store engineering data during maneuvers and occultations for delayed transmission
- Read out computer and sequencer memory prior to maneuvers
- Format and transmit real-time science data
- Format and store science data for delayed transmission
- Relay real-time capsule engineering data during capsule descent
- Store capsule TV data during capsule descent for delayed transmission

The reliability estimate for the telemetry and data storage subsystem is 0.921. The equation and block diagram appear in Figure 3-50.

The subsystem incorporates considerable redundancy and alternate modes of operation. The hard wired programmers (1 and 2) act as a backup for the flexible format generator (3). If the flexible format generator fails, data can be sequenced in a given preprogrammed order rather than in variable order, the capability that exists when the flexible

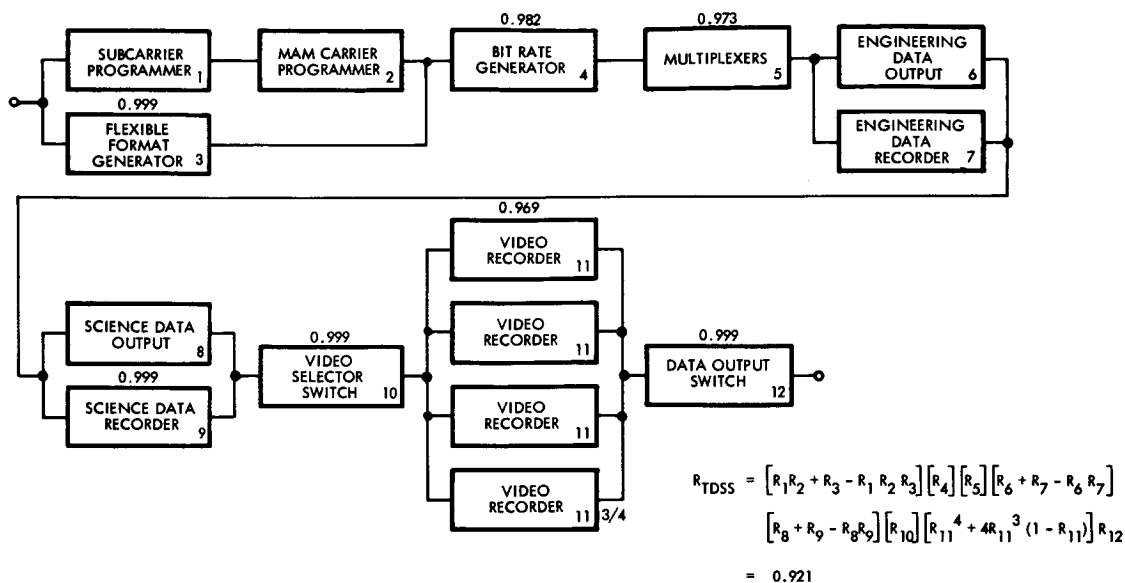


Figure 3-50
TELEMETRY AND DATA STORAGE RELIABILITY DIAGRAM shows reliability assessment for each component or section. Multiplier (5) includes quad redundancy.

format generator is operating properly. Engineering data can be transmitted in real-time (6), or can be stored (7) and transmitted when more convenient. Likewise science data can be transmitted in real-time (8), or can be stored (9). TV data must be stored and then retransmitted. Four recorders (11) are provided for this purpose and mission success is assured if at least three of four recorders are operational.

In addition to the redundancy shown in Figure 3-50, the multiplexer (5) incorporates quad-redundant circuitry. The failure rate shown for this equipment is the effective component failure rate including this quad redundancy. Appendix E lists the components and applicable data included in the subsystem.

3.8.8 Electrical Power Subsystem

The electrical power subsystem distributes power in suitable forms for distribution to the electrical equipment on board the spacecraft and to the capsule until its separation. Primary power is derived from the sun by means of silicon photovoltaic cells mounted on a fixed solar array. Nickel-cadmium batteries are used to energize the spacecraft whenever the solar array is incapable of supporting loads, as during launch, maneuvers, and eclipses. Controls are incorporated to insure proper bus



voltages and battery charge rates, and provide necessary power conditioning. The power subsystem must function continuously throughout the mission.

The reliability estimates for the electrical power subsystem is 0.985. Figure 3-51 presents the reliability equation and block diagram. The solar array and shunt elements incorporate considerable inherent

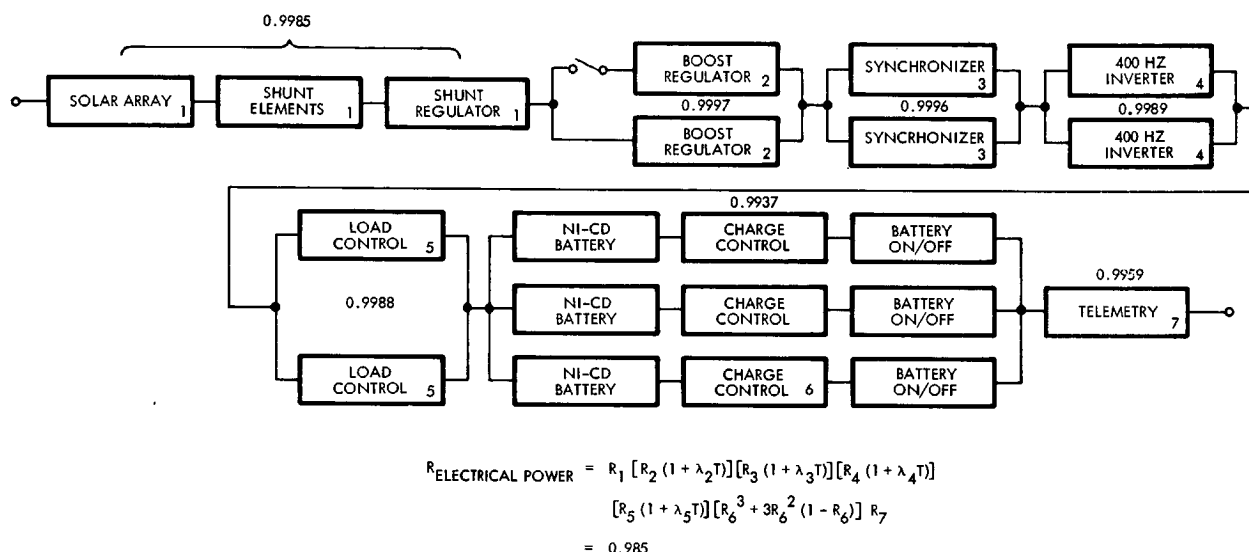


Figure 3-51

ELECTRICAL POWER SUBSYSTEM DIAGRAM incorporates a considerable degree of redundancy.

and component redundancy such that failures in these areas will only slightly degrade subsystem capability. Automatic detection of subsystem failure and switching of the standby units is provided. The battery equipment includes three nickel-cadmium batteries, and the associated charge and on-off control. During maximum eclipse time in Mars orbit, all three batteries are required to provide power to the active equipment. During periods of lesser duration, two of three batteries are required to provide spacecraft power. In the event one battery fails, and battery power is required, the spacecraft can be operated with reduced battery loads or the depth of battery discharge can be increased. In this analysis, it is assumed that two of the three battery chains are required for mission success. Appendix E lists the power subsystem components.

3.8.9 Electrical Distribution Subsystem

The electrical distribution subsystem provides (1) electrical inter-connections between various elements of the spacecraft, the spacecraft and launch vehicle, and the spacecraft and the capsule by the use of harnesses, junction boxes, and umbilical connectors, (2) a source distribution point for primary power, (3) a safe/arm function for all ordnance devices, and (4) a pyrotechnic control for all ordnance devices.

The reliability estimate is 0.9998. The reliability equation and block diagram appear in Figure 3-52. Group 1 equipment consists of the

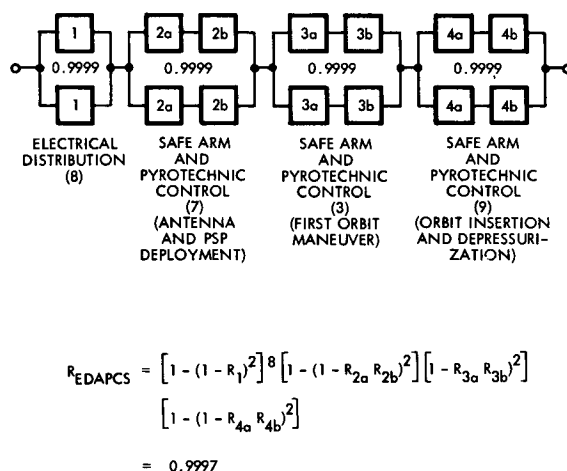


Figure 3-52
ELECTRICAL DISTRIBUTION SUBSYSTEM RELIABILITY DIAGRAM
shows redundant elements.

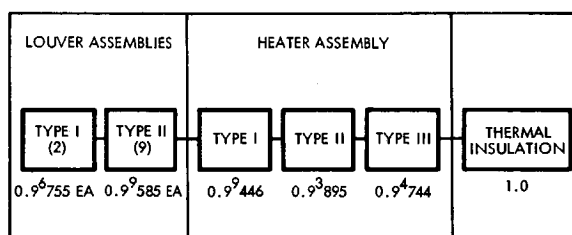
pyrotechnics required to release the antenna booms. Group 2 consists of the pyro equipment required to actuate engine valves for the mid-course corrections. At the time of Mars orbit insertion, the third group of pyro devices are activated prior to deboost engine firings. After orbit trim, the pyro devices for fuel tank depressurization are fired.

Appendix E lists the subsystem components. Since the pyrotechnic equipment is required to operate up to a specific time, four time intervals are considered: Group 1 is required to operate up to 240 hours, Group 2 up to 720 hours, Group 3 up to 5360 hours; and Group 4 up to 5500 hours. The electrical distribution equipment is required to operate for the full 6800-hour mission.

3.8.10 Temperature Control Subsystem

The subsystem ensures that the spacecraft components are maintained at the required temperatures during the mission. This is accomplished by the use of thermal actuated, heat conducting louver assemblies, heater elements controlled by thermostats or relays, thermal insulation, and special coatings and finishes.

The reliability estimate is 0.9943. Figure 3-53 presents the temperature control reliability equation and block diagram.



$$R_S = R_{\text{LOUVERS}} \times R_{\text{HEATERS}} \times R_{\text{INSULATION}}$$

WHERE:

$$= R_{\text{LOUVERS}} = [R_{\text{TYPE I}}]^2 \times [R_{\text{TYPE II}}]^9$$

$$= R_{\text{HEATERS}} = [R_{\text{TYPE I}}] \times [R_{\text{TYPE II}}] \times [R_{\text{TYPE III}}]$$

$$= R_{\text{INSULATION}} = R_{\text{INSULATION}}$$

THEREFORE:

$$R_S = [0.9^6755]^2 \times [0.9^9585]^9 \times [0.9^3895] \times [0.9^446] \times [0.9^4774] \times 1.0$$

$$R_S = 0.9943$$

Figure 3-53
TEMPERATURE CONTROL DIAGRAM assesses reliability of the three subsystem elements.

The assumptions underlying this analysis are:

- Type I louver assemblies are comprised of 18 blades and associated bimetallic spring actuators. At least 15 of the 18 assemblies must perform during the mission.
- Type II louver assemblies are comprised of 30 blades and associated bimetallic spring actuators. At least 25 of the 30 assemblies must perform during the mission.

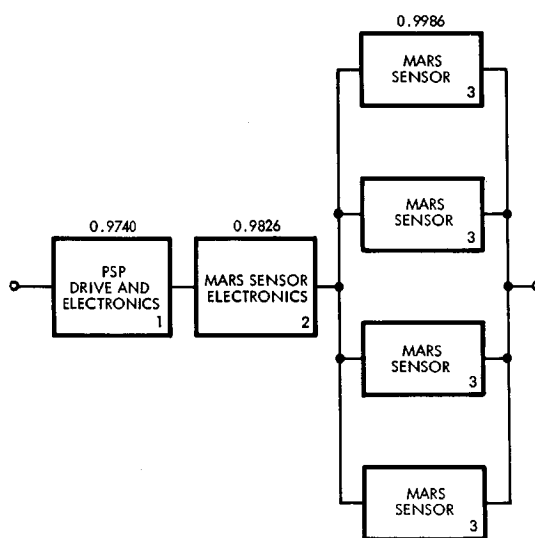
- There are two Type I and nine Type II louver assemblies in the spacecraft, all of which must perform throughout the mission.
- The louver mission time is 2500 actuation cycles.
- Type I heater assemblies contain eight internal heaters located in the equipment compartment, each activated by a thermal switch. At least seven of the eight heater assemblies must perform during the mission.
- Type II heater assemblies contain four external heaters located on the appendages, each activated by a ground actuated relay. All of the appendage heater assemblies must perform during the mission.
- Type III heater assemblies contain four internal heaters located in the propulsion module each activated by a thermal switch. At least three of the four heater assemblies must perform during the mission.
- The heater mission time is 6800 hours with a 50 percent on-off duty cycle. The heater failure rate during the off period is 10 percent of the failure rate during the on period.
- The inherent design reliability of the insulation is assumed equal to 1.0.
- The principal failure mode has been determined from OGO test data to be failure of the blade to open.

Detailed calculations are presented in Appendix E.

3.8.11 Planetary Scan Platform

The planetary scan platform provides electrical connections, structural support, a thermally controlled environment, and the required micrometeoroid protection for the planet-oriented experiments, sensors, and associated electronic equipment. The PSP is considered to consist of four principal groups of equipment; the mounting platform, gimbal and deployment drives, drive electronics, and the Mars sensors.

The reliability estimate for the PSP equipment is 0.954. The reliability block diagram and equation for the equipment is given in Figure 3-54.



$$R_{PSP} = R_1 R_2 R_3^4 + 4R_3^3 (1 - R_3) = 0.955$$

Figure 3-54
RELIABILITY BLOCK DIAGRAM OF PSP EQUIPMENT.

	Page
4. THE 1975, 1977, and 1979 MISSIONS	4-1
4.1 MISSION DEFINITION	4-1
4.2 SPACECRAFT MODIFICATIONS	4-1
4.2.1 Medium-Gain Antenna Modification	4-1
4.2.2 Science Integration	4-2
4.3 OTHER VOYAGER ELEMENTS	4-4
4.4 INTERPLANETARY TRAJECTORY AND MARS ORBIT CAPABILITY	4-4
4.4.1 Characteristics of Earth-Mars Trajectories, 1975, 1977, and 1979 Opportunities	4-4
4.4.2 Performance Capability, 1975 Opportunity	4-11
4.4.3 Performance Capability, 1977 Opportunity	4-14
4.4.4 Comparison of Type I and Type II Trajectory Characteristics	4-17
4.4.5 Performance Capability, 1979 Opportunity	4-18
4.5 TRAJECTORY ACCURACY CAPABILITY	4-20
4.6 OBSERVANCE OF THE PLANETARY QUARANTINE CONSTRAINT	4-23
4.7 ACCOMPLISHMENT OF SCIENTIFIC MISSION OBJECTIVES	4-25
4.7.1 Orbital Photo-Imaging	4-25
4.7.2 Other Orbital Scientific Experiments	4-26
4.8 SPACECRAFT COMMUNICATION PERFORMANCE	4-28
4.9 PROBABILITY OF SUCCESS	4-32



4. THE 1975, 1977, AND 1979 MISSIONS

This section discusses the applicability of the recommended spacecraft to the 1975, 1977, and 1979 missions. Because of the guideline to design the recommended spacecraft for minimum modification, the same capabilities presented in the previous section are generally applicable. These are discussed below for the same performance areas as covered in Section 3 for the 1973 mission.

4.1 MISSION DEFINITION

The basic mission profile and spacecraft functions as described in Section 3.1.1 for the 1973 mission apply to the later missions as well. The changes associated with these missions relate to longer lifetime requirements (as given in Section 3.1.2.2), the variation in solar system geometry, and the possible incorporation of additional science experiments. Some individual changes in requirements such as arrival date separation and launch azimuth limits have been covered in Section 3.1 along with the 1973 requirements.

4.2 SPACECRAFT MODIFICATIONS

One of the basic guidelines for the study is to design the spacecraft for the 1973 mission so as to be applicable to the 1975 through 1979 missions with minimum modification. The recommended spacecraft meets this requirement. The addition of new equipment items is associated only with science payload changes. One basic spacecraft hardware item, the medium-gain antenna, will undergo a fitting modification. In all other respects the spacecraft remains unchanged.

4.2.1 Medium-Gain Antenna Modification

From Mars, the earth traces out a locus in cone and clock angles as shown in Figure 4-1. For the 1973 mission, the medium-gain antenna's single hinge axis is along the 0 to 180 degrees clock angle line at a cone angle of 90 degrees (see Figure 2-7). By rotating about this axis to track the earth, its ± 5 degrees fan beam irradiates earth. The beamwidth along the direction of rotation is 4 degrees to provide adequate gain yet allow for pointing errors. For the 1975 mission

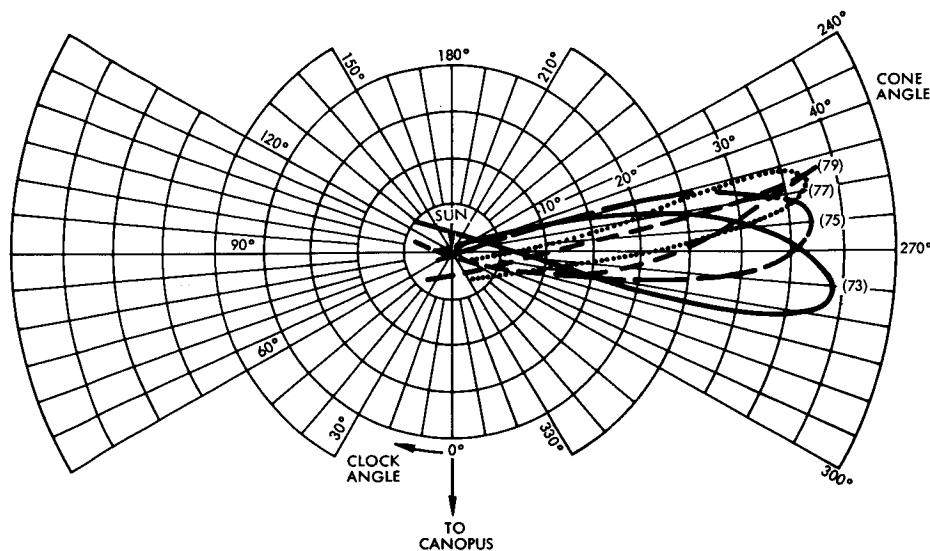


Figure 4-1
DIRECTION OF EARTH (in cone and clock angle) as seen from Mars.

the earth's locus as seen from Mars rotates slightly counterclockwise. This would place it outside the medium-gain antenna beam if the antenna installation remained the same as for the 1973 mission. This is taken care of, however, by a simple modification of the antenna hinge fitting so as to rotate the hinge axis counterclockwise about 5 degrees. For the 1977 and 1979 missions an additional rotation is required.

Although the above approach calls for a modification to the 1973 spacecraft for later missions, it is considered a simple one. Thus, it is preferred over broadening the fan beam to the point that antenna gain would be severely reduced.

4.2.2 Science Integration

The recommended spacecraft contains a combination film-TV photo-imaging system consisting of a 1000-meter color TV camera and a Lunar Orbiter type film system with 100 and 10-meter resolution capabilities for photography at 1000 km altitude. The recommended spacecraft also contains a core group of scientific experiments which consists of a high resolution IR spectrometer, a broadband IR spectrometer, a IR radiometer, and a UV spectrometer. These experiments and photo-imaging systems are described in more detail in Volumes 5 and 11.



The basic configuration of the recommended spacecraft is such that additional experiments can be added given the appropriate weight and power allocations. Since the spacecraft performance can accommodate more experiments, these are considered as additions to be carried on the recommended spacecraft in 1975-1979 (as well as 1973 if desired).

The planetary scan platform box for the recommended spacecraft has been sized to accommodate the addition of the following three experiments: gamma ray spectrometer, meteor formation detector, and polarimeter. The remote data multiplexers, switching assembly, and decoders located in the PSP have been designed to accommodate these additional experiments. However, to incorporate these experiments in the PSP will require the addition of viewing ports and mounting hardware.

To add experiments to the spacecraft body or to add boom-mounted experiments requires the simple addition of an experiment electrical integration assembly. With this electrical assembly, the additional boom or body-fixed experiments interface with the spacecraft telemetry data storage, power, and command subsystems in the same manner as those experiments mounted on the PSP. As described in Volume 5, two additional instruments are suggested for mounting on booms. They are the magnetometer and neutron albedo experiments. In addition the following experiments are suggested for mounting to the spacecraft body as fixed mounted experiments:

- Atmospheric mass spectrometer, solar occultation, cosmic ray
- Micrometeor impact detector, solar plasma, solar x-ray
- Celestial x-ray

These experiments and their estimated weight and power requirements are also defined in Volume 5.

The film-TV photo-imaging system can also be modified to take advantage of the longer mission life in 1977 and 1979 by the simple addition of a larger film cassette and the associated film processing web and the additional radiation shielding. With this growth in mind

the basic film pressure vessel would be sized initially to accommodate the addition of the larger film supply.

4.3 OTHER VOYAGER ELEMENTS

The other Voyager system elements will probably undergo evolutionary changes for performance improvement over the long time span being considered for the later missions. For example, Saturn V payload capability will probably improve during this period. However, such changes have not been identified at this time, and so have not been considered in the study. Also, the general Voyager project approach has been standardization in all areas where requirements and capabilities can be established. This applies particularly for items affecting intersystem interfaces. Therefore, the characteristics of the other Voyager elements, as described in Section 3.2 for the 1973 mission, have been considered applicable to the later missions for purposes of the study.

4.4 INTERPLANETARY TRAJECTORY AND MARS ORBIT CAPABILITY

4.4.1 Characteristics of Earth-Mars Trajectories, 1975, 1977, and 1979 Opportunities

Figures 4-2 through 4-7 display the characteristics of the Type I and Type II earth-Mars trajectories for the 1975, 1977, and 1979 opportunities. The C_3 and V_∞ contours indicate regions of interest; DLA contours serve to constrain these regions. The ensuing sections define the performance capability of the recommended spacecraft. This capability has been evaluated on the basis of the following DLA constraints (Table 4-1). Type II trajectories have not been evaluated for the 1979 opportunity because the capability attainable with the Type I trajectories appears to be quite adequate and does not limit performance over the four opportunities under consideration.

Planetary vehicle gross weight has been determined from the Saturn V launch vehicle performance data given in Figure 3-15 and discussed previously in Section 3.3.4. These data are based on a launch azimuth of 115 degrees. The use of these data are conservative as long as the required launch azimuth is in the range of 90 to 115 degrees.

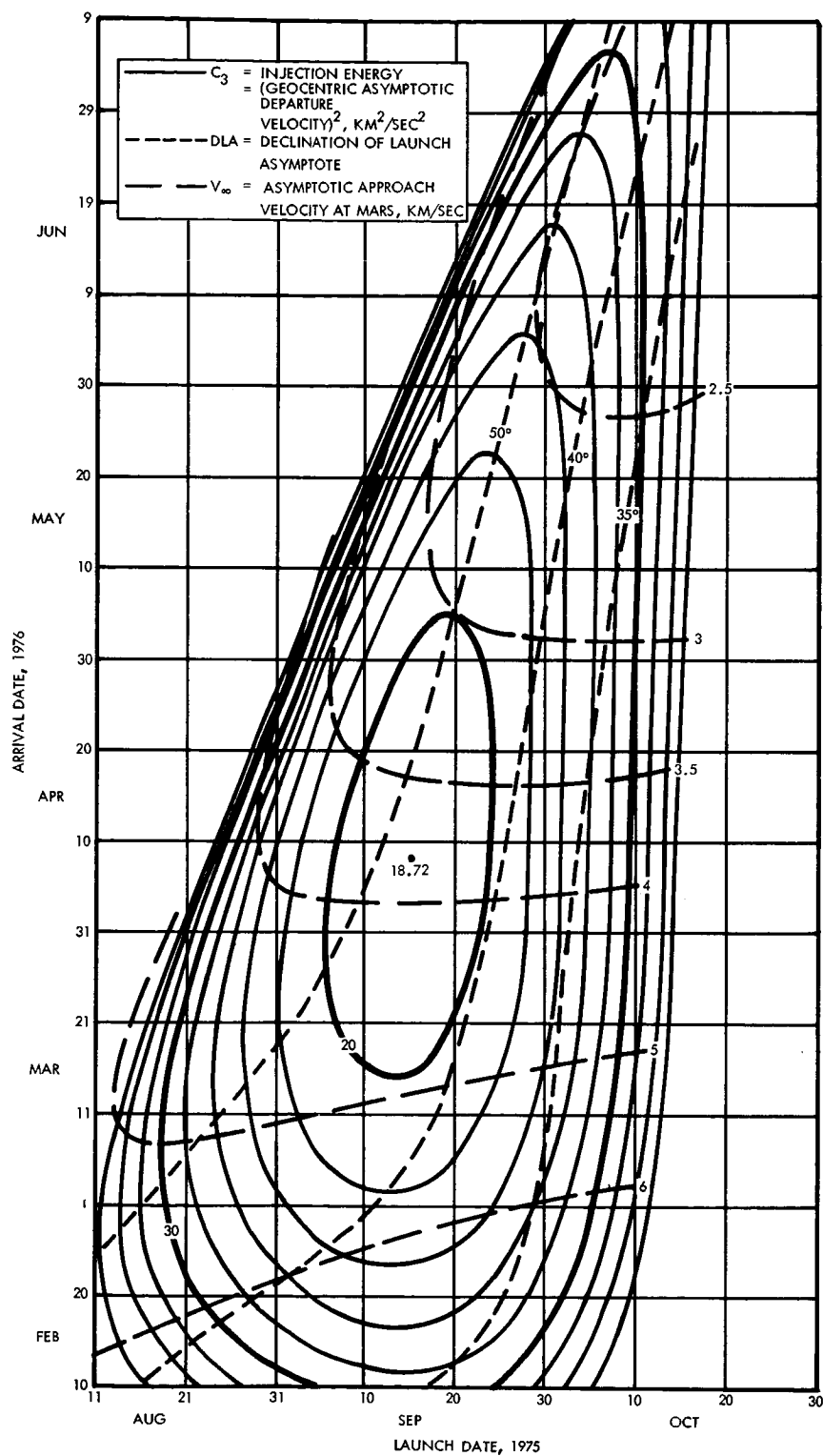


Figure 4-2
THE CHARACTERISTICS OF A TYPE I TRAJECTORY IN 1975 for earth-Mars transit include variations in injection energy, declination of launch asymptote, and asymptotic approach velocity at Mars with changes in launch and arrival dates.

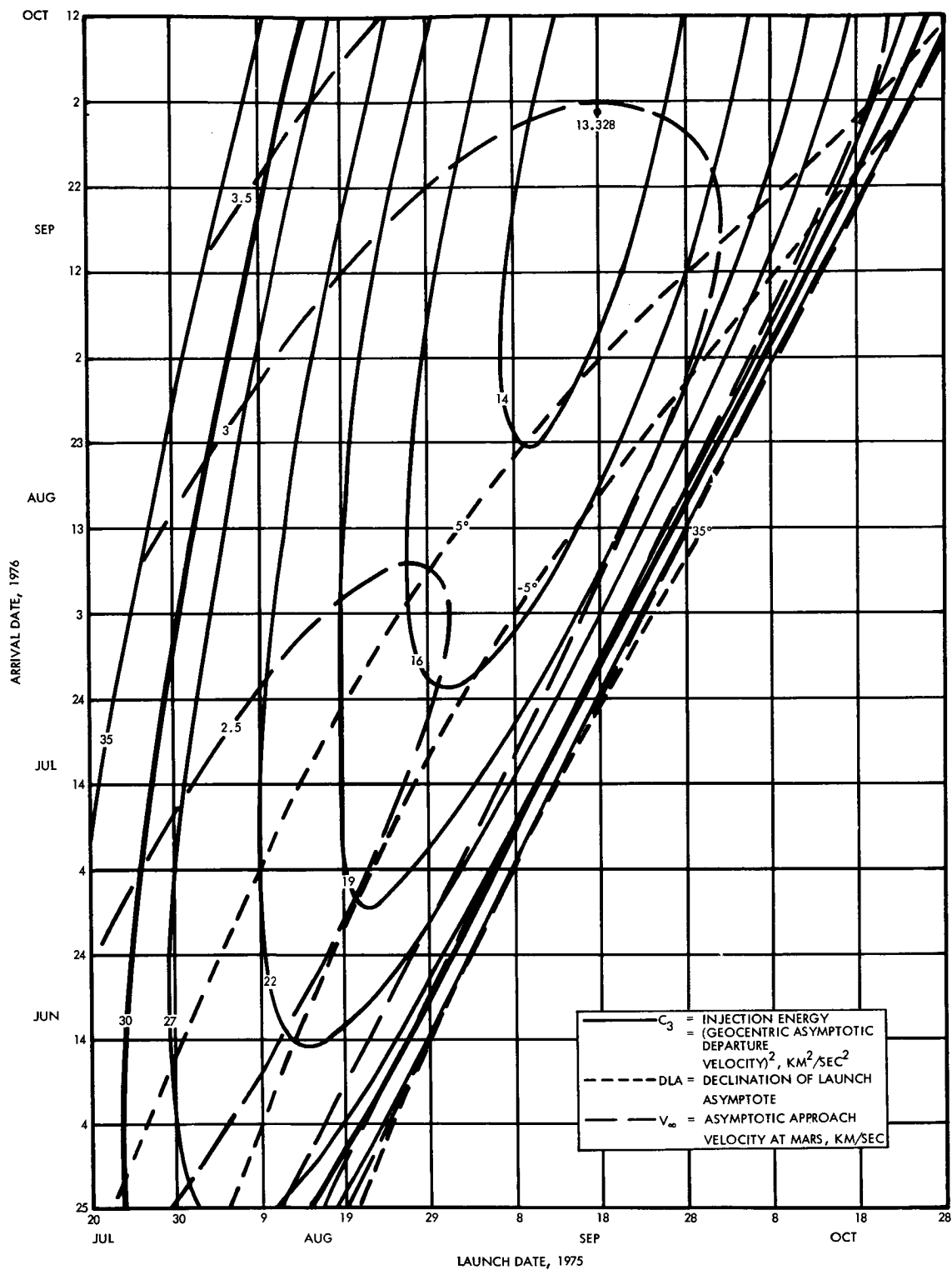


Figure 4-3
THE CHARACTERISTICS OF A TYPE II TRAJECTORY IN 1975 for earth-Mars transit include variations in injection energy, declination of launch asymptote, and asymptotic approach velocity at Mars with changes in launch and arrival dates.

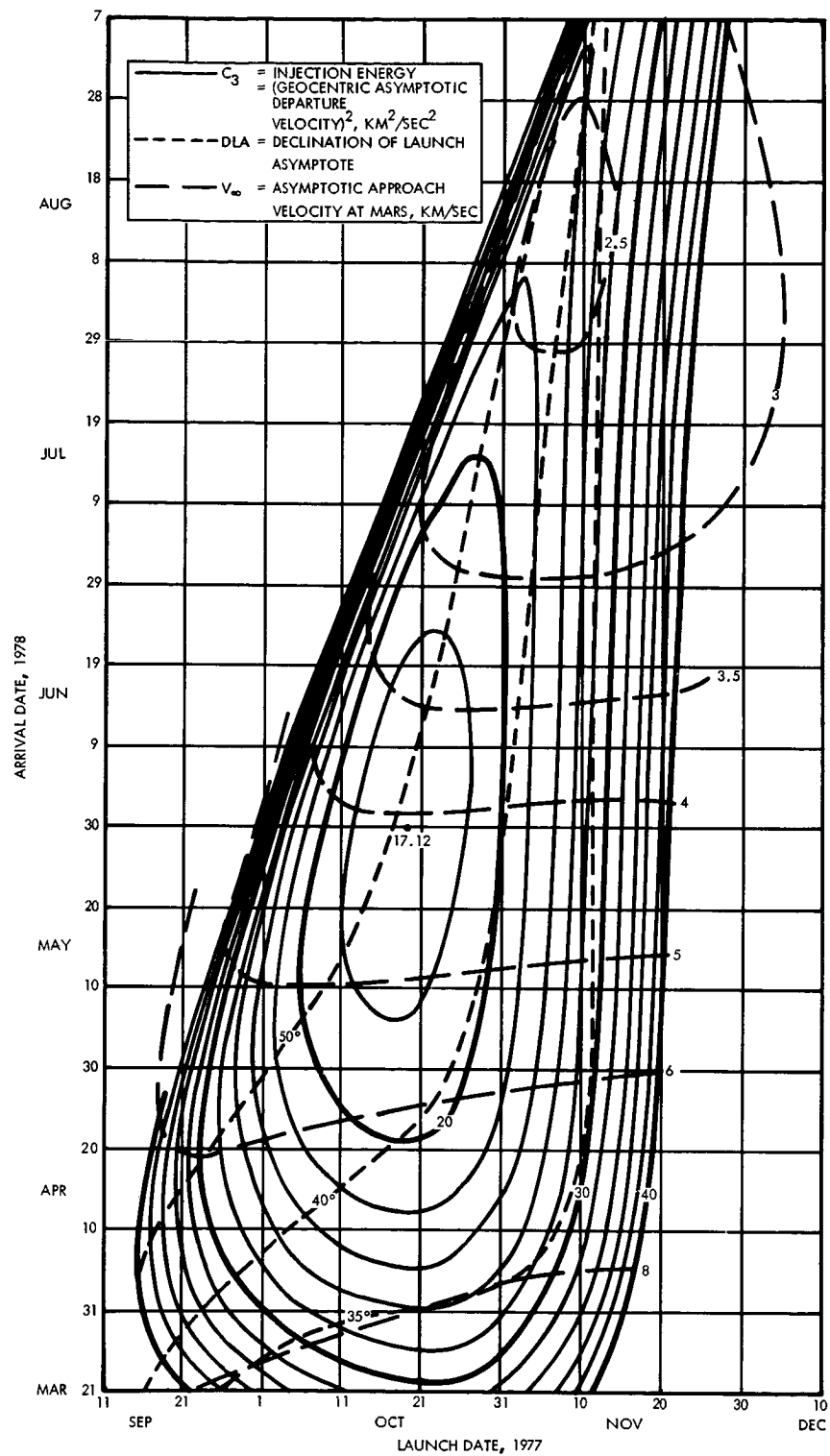


Figure 4-4
THE CHARACTERISTICS OF A TYPE I TRAJECTORY IN 1977 for earth-Mars transit include variations in injection energy, declination of launch asymptote, and asymptotic approach velocity at Mars with changes in launch and arrival dates.

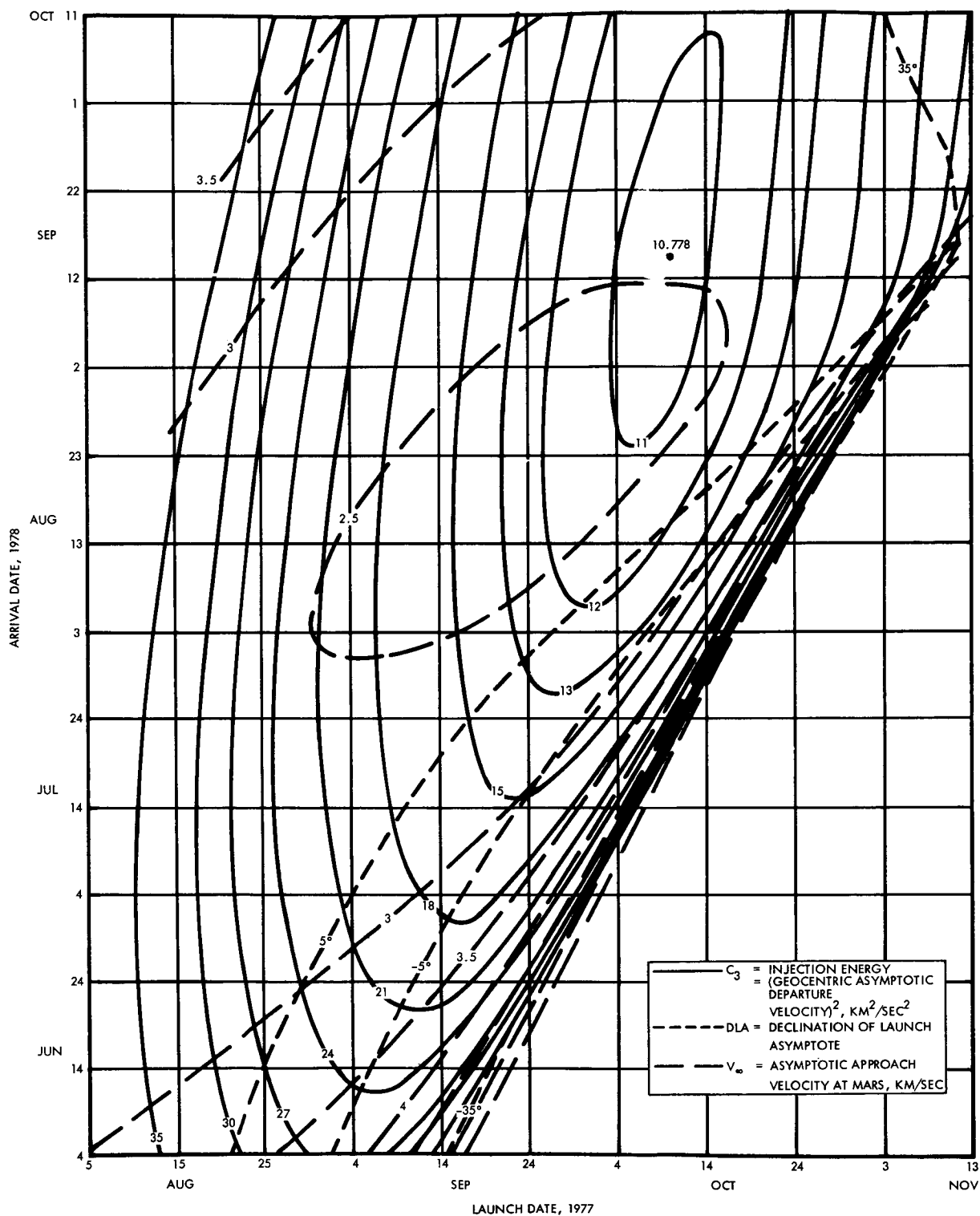


Figure 4-5
THE CHARACTERISTICS OF A TYPE II TRAJECTORY IN 1977 for earth-Mars transit include variations in injection energy, declination of launch asymptote, and asymptotic approach velocity at Mars with changes in launch and arrival dates.

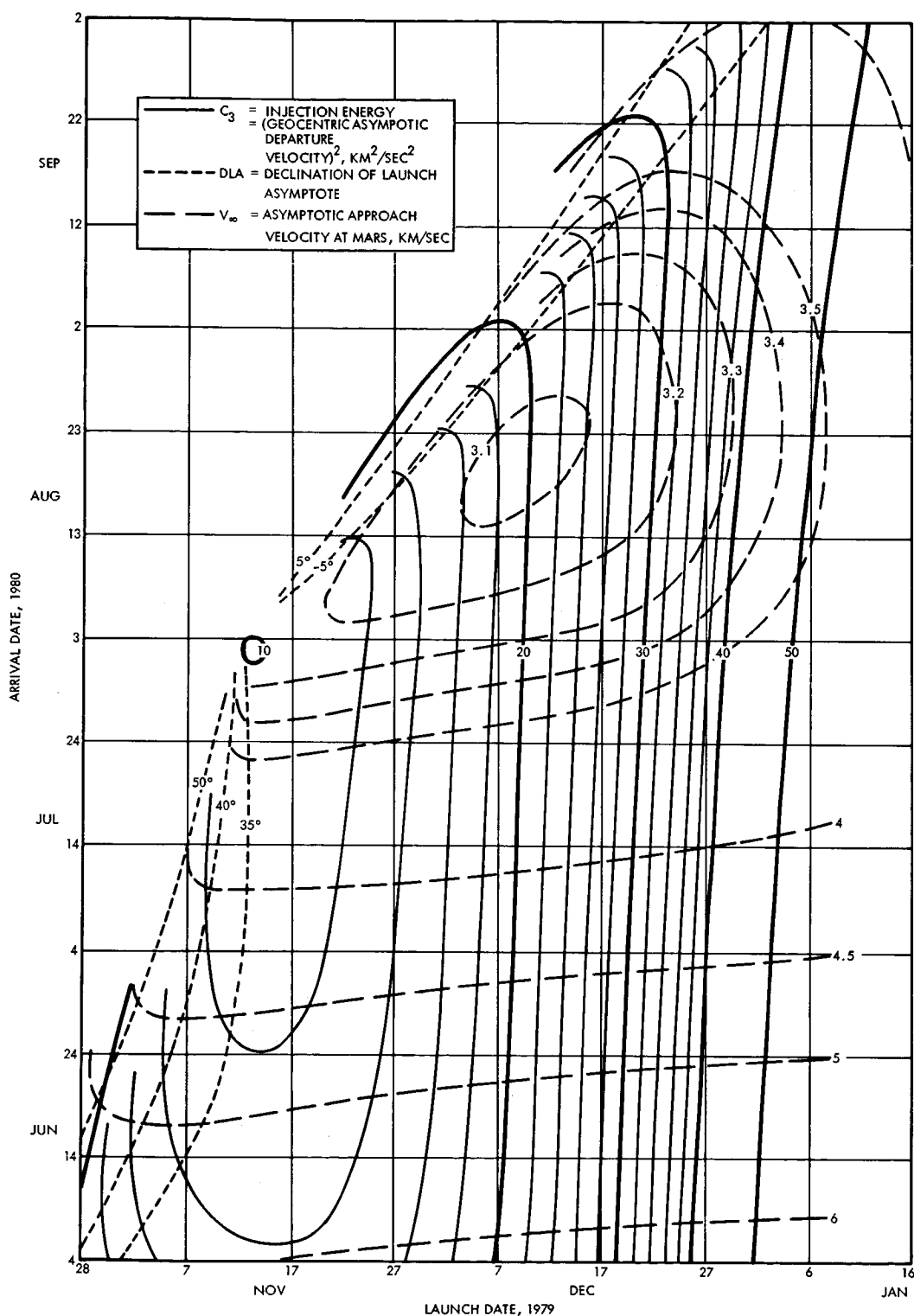


Figure 4-6
THE CHARACTERISTICS OF A TYPE I TRAJECTORY IN 1979 for earth-Mars transit include variations in injection energy, declination of launch asymptote, and asymptotic approach velocity at Mars with changes in launch and arrival dates.

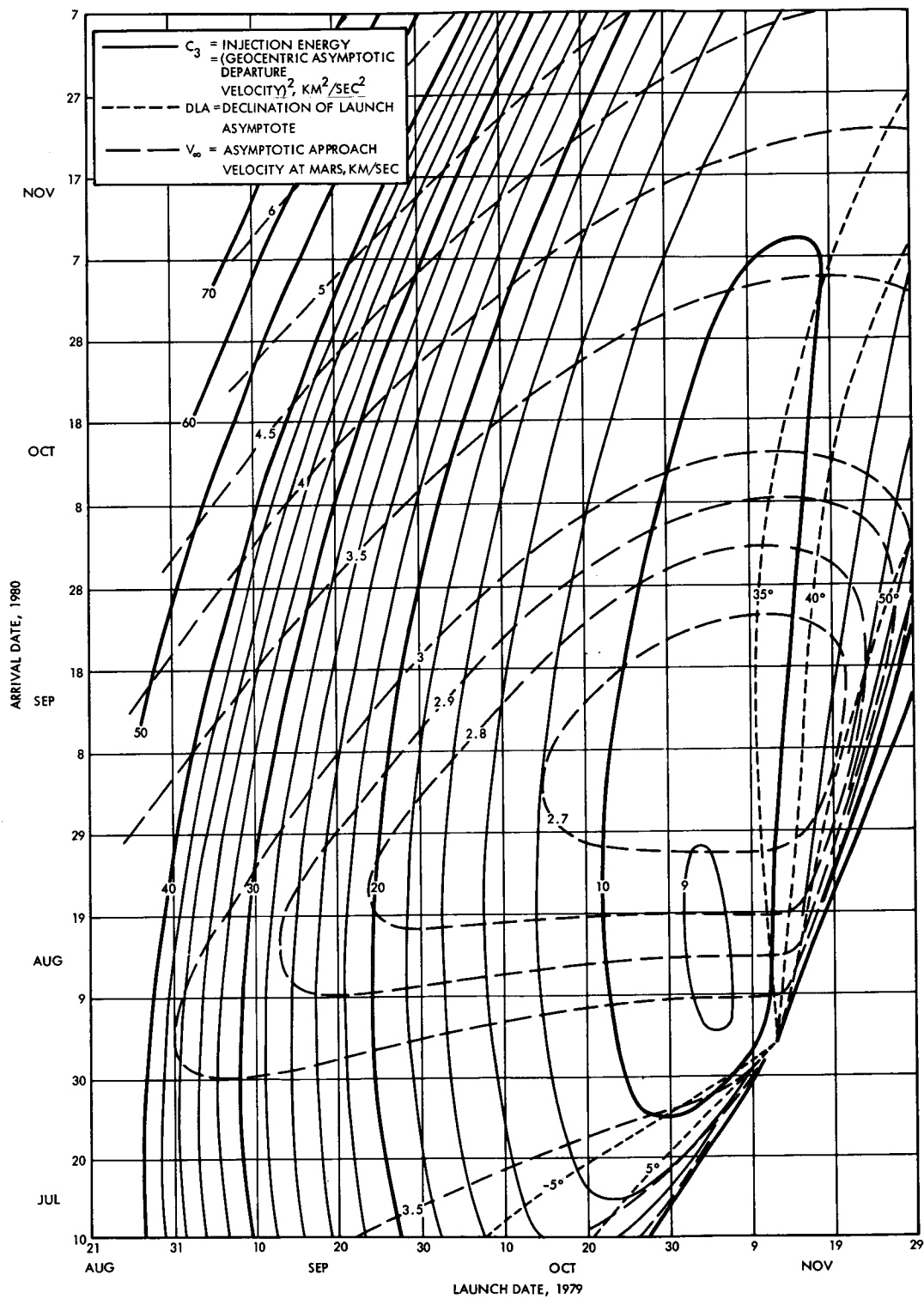


Figure 4-7
THE CHARACTERISTICS OF A TYPE II TRAJECTORY IN 1979 for earth-Mars transit include variations in injection energy, declination of launch asymptote, and asymptotic approach velocity at Mars with changes in launch and arrival dates.



Table 4-1 . Maximum Allowable DLA Limits

Mars Opportunity	Trajectory Type	Maximum DLA (deg)
1975	I	+50
1975	II	+36
1977	I	+50
1977	II	+36
1979	I	+36

For the 1975 and 1977 Type I trajectories, however, a maximum DLA of +50 degrees has been employed, corresponding to a launch azimuth range of 45 to 115 degrees. It is certain that some arrival dates during those two mission opportunities will require launch azimuths in the range of 45 to 65 degrees. However, in practical cases, this does not imply a launch vehicle capability which is less than that which has been assumed, because generally the extreme launch azimuths occur only early in the launch period, when the C_3 requirement is well below its maximum values.

The impulsive velocity requirements associated with these opportunities are the same as those for the 1973 opportunity, with the exception of the orbit trim maneuver; a 0.100 km/sec ΔV allocation is assumed for this maneuver. The arrival date separation is assumed to be ± 2 days.

4.4.2 Performance Capability, 1975 Opportunity

Figures 4-8 and 4-9 summarize the performance capability for mission Case A (see Section 3.1) during the 1975 Type I opportunity. Mars orbit sizes of 1100 x 10,000 and 1000 x 20,000 km are postulated together with a 20-day launch period. As noted previously, the larger orbit size permits greater capability, since the orbit insertion velocity increment requirements are lower. Additional weight contours for mission Case B are simply those for Case A, reduced by 2000 pounds. For

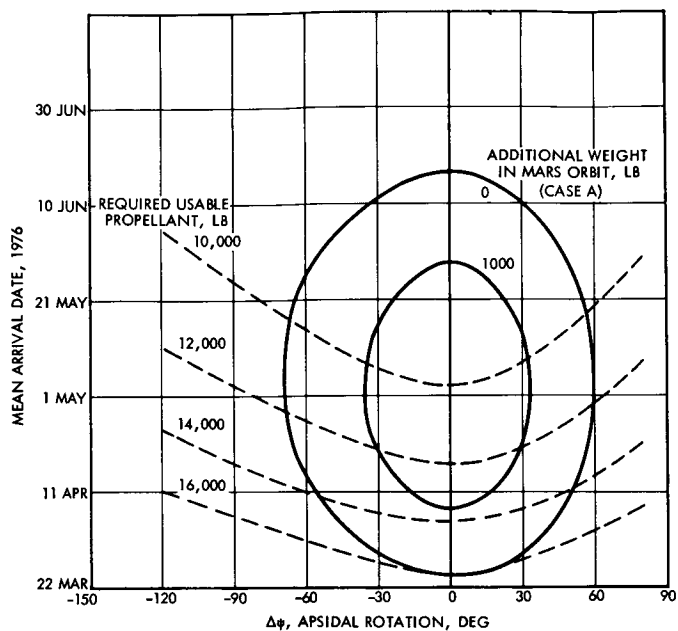


Figure 4-8
PERFORMANCE CAPABILITY and propellant requirements for the 1975 Type I mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1100 x 10,000 kilometers.

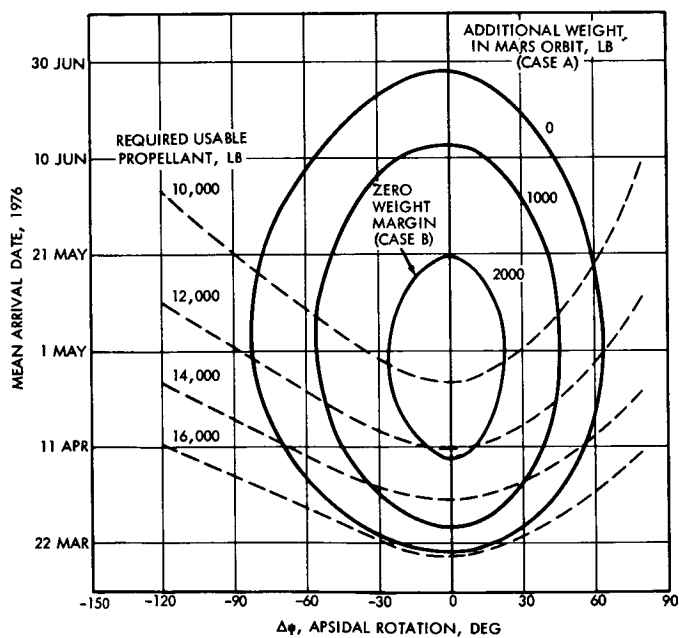


Figure 4-9
PERFORMANCE CAPABILITY and propellant requirements for the 1975 Type I mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1000 x 20,000 kilometers.

example, the 2000-pound contour on Figure 4-9 corresponds to zero additional weight for mission Case B. Thirty-day launch periods have also been evaluated. It was found that this larger launch period did not permit any additional spacecraft or capsule weight capability above that which is included in the recommended spacecraft design.

Figures 4-10 and 4-11 show the increased performance capability which may be achieved if Type II trajectories are utilized in the 1975 opportunity. These Type II trajectories also exceed the 1973 Type I performance, but at the expense of a larger transit time. A comparison of the additional spacecraft plus capsule weight capability and the transit times for the optimum launch periods is given in Table 4-2 for these opportunities.*

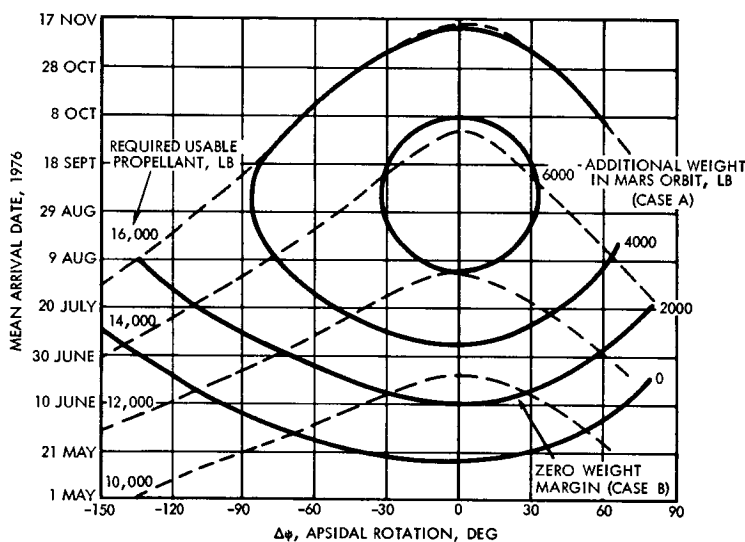


Figure 4-10

PERFORMANCE CAPABILITY and propellant requirements for the 1975 Type II mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1100 x 10,000 kilometers.

An additional transit time of 147 days is required if the maximum weight capability is sought for both the 1975 Type I and 1975 Type II mission opportunities. However, the 1975 Type II transit times may be reduced by utilizing earlier mean arrival dates and smaller increases in weight capability. For example, an additional weight capability of

* An "optimum launch period" is defined here as that combination of launch date interval-mean arrival date which yields the maximum weight in Martian orbit for that launch period duration and mission opportunity.

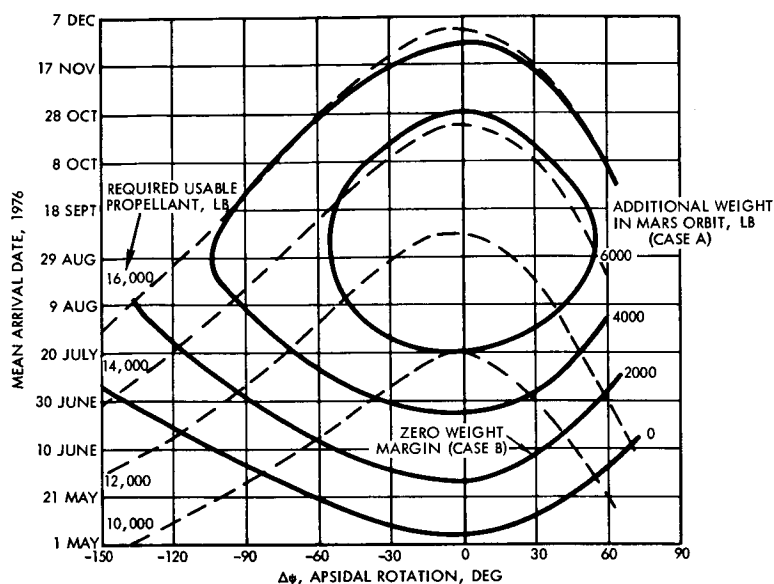


Figure 4-11

PERFORMANCE CAPABILITY and propellant requirements for the 1975 Type II mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1000 x 20,000 kilometers.

Table 4-2. Comparison of Optimum 1973 and 1975 Trajectories - 20-day launch period, 1100 x 10,000 km Mars Orbit

Mission Opportunity	Additional Weight Capability (Case A) (lb)	Longest Transit Time (days)
1973 Type I	5480	214
1975 Type I	1534	227
1975 Type II	6826	374

2000 pounds may be obtained with a mean arrival date of 10 June 1976 and no apsidal rotation. The longest transit time associated with this 1975 Type II mission is 322 days.

4.4.3 Performance Capability, 1977 Opportunity

The performance capability which may be achieved during the 1977 opportunity is summarized in Figures 4-12 through 4-15. This opportunity is very similar to the 1975 opportunity. No performance margin

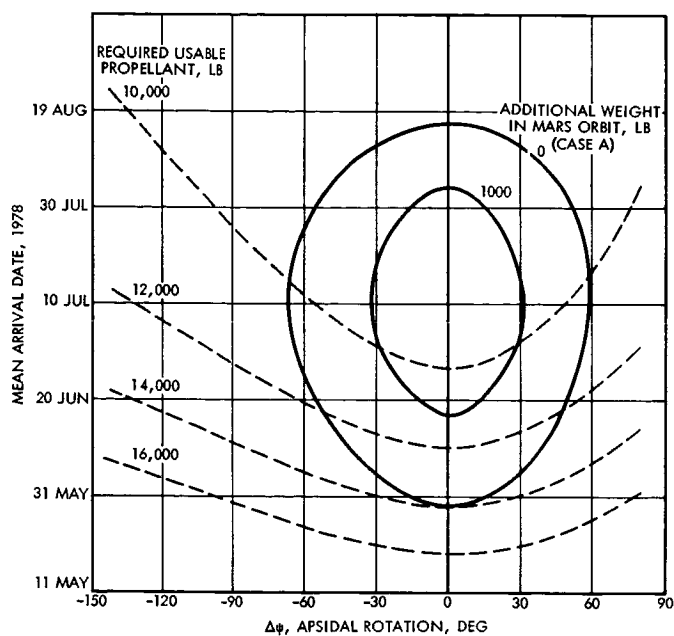


Figure 4-12

PERFORMANCE CAPABILITY and propellant requirements for the 1977 Type I mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1100 x 10,000 kilometers.

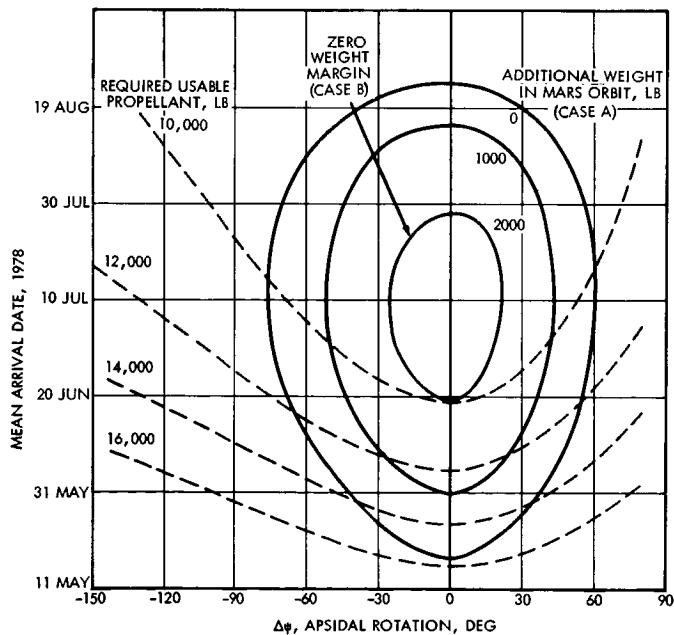


Figure 4-13

PERFORMANCE CAPABILITY and propellant requirements for the 1977 Type I mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1000 x 20,000 kilometers.

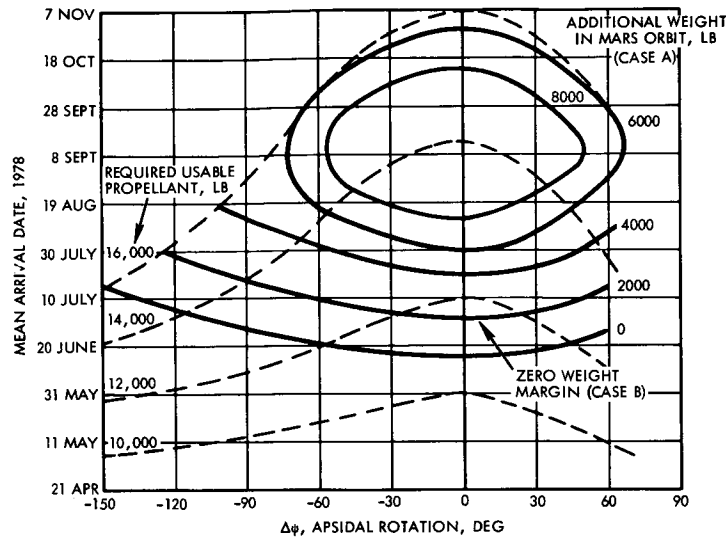


Figure 4-14

PERFORMANCE CAPABILITY and propellant requirements for the 1977 Type II mission opportunity are displayed as a function of apsidal rotation and arrival date for a 30-day launch period and a Mars orbit size of 1100 x 10,000 Kilometers.

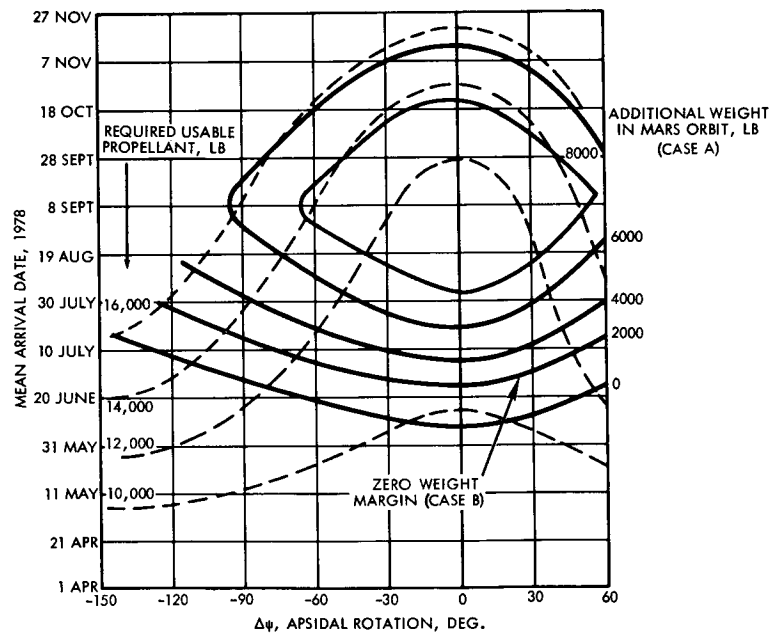


Figure 4-15

PERFORMANCE CAPABILITY and propellant requirements for the 1977 Type II mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1000 x 20,000 Kilometers.



may be obtained with a 30-day launch period and the Type II trajectories provide considerably greater performance margin than the Type I trajectories. Additional weight capability and transit time comparisons are given in Table 4-3 below for the optimum 1977 Type I and Type II 20-day launch periods.

Table 4-3. Comparison of Optimum 1977 Type I and Type II Trajectories - 20-day Launch Period, 1100 x 10,000 km Mars Orbit

Mission Opportunity	Additional Weight Capability (Case A) (lb)	Longest Transit Time (days)
1977 Type I	1499	258
1977 Type II	9006	348

Once again, the "optimum" Type II missions require greater transit times than do the "optimum" Type I missions. As with the 1975 missions, however, the 1977 Type II transit times may be reduced by utilizing earlier mean arrival dates and smaller increases in weight capability. An additional weight capability of 2000 pounds may be obtained with a transit time of 311 days, a mean arrival date of 2 July 1978, and no apsidal rotation.

4.4.4 Comparison of Type I and Type II Trajectory Characteristics

Section 4.4.2 and 4.4.3 have compared the performance capabilities and transit times of the Type I and Type II trajectories for the 1975 and 1977 mission opportunities. Additional considerations which include launch azimuth or DLA requirements, communications distances, uniform Mars arrival geometry, arrival date flexibility, and range of apsidal rotation, must also be evaluated. These considerations, which are by no means a complete compilation of factors affecting the selection of a Type I or Type II trajectory, are discussed briefly below.

The 1975 and 1977 Type II trajectories were constrained to values of $DLA \leq 36$ degrees. This constraint was relaxed to 50 degrees for the Type I trajectories. Thus, northeast launch azimuths as low as

45 degrees will be required for some of the Type I trajectories. This creates problems with range safety and tracking during the launch and near-earth phases. Relative to the Type II trajectories, Type I trajectories are characterized by shorter transit times and reduced communications distances during the first several months after Mars encounter.

Uniform Mars arrival geometry requires relatively constant arrival dates throughout any given Type I launch period. Variable arrival dates are permitted during the Type II launch periods if uniform Mars arrival geometry is to be obtained, tending to relax a constraint on performance. However, the primary advantage of the 1975 and 1977 Type II missions is the increased weight capability that they provide over their Type I counterparts. These Type II missions also permit a much greater flexibility in both Mars arrival date and apsidal rotation.

4.4.5 Performance Capability, 1979 Opportunity

Figures 4-16 through 4-19 summarize the 1979 Type I performance which may be realized with both 20- and 30-day launch periods and Mars orbit sizes of 1100 x 10,000 and 1000 x 20,000 km. The ranges of apsidal rotation and mean arrival date are comparable to those which may be achieved during the 1973 opportunity.

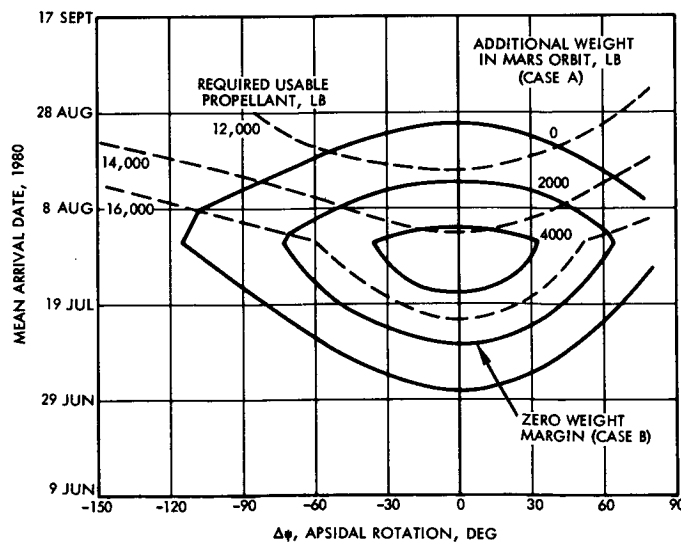


Figure 4-16
PERFORMANCE CAPABILITY and propellant requirements for the 1979 Type I mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1100 x 10,000 kilometers.

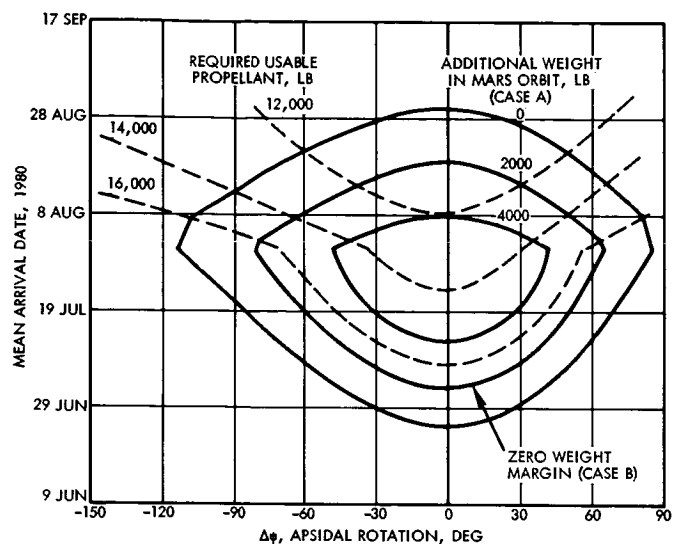


Figure 4-17

PERFORMANCE CAPABILITY and propellant requirements for the 1979 Type I mission opportunity are displayed as a function of apsidal rotation and arrival date for a 20-day launch period and a Mars orbit size of 1000 x 20,000 kilometers.

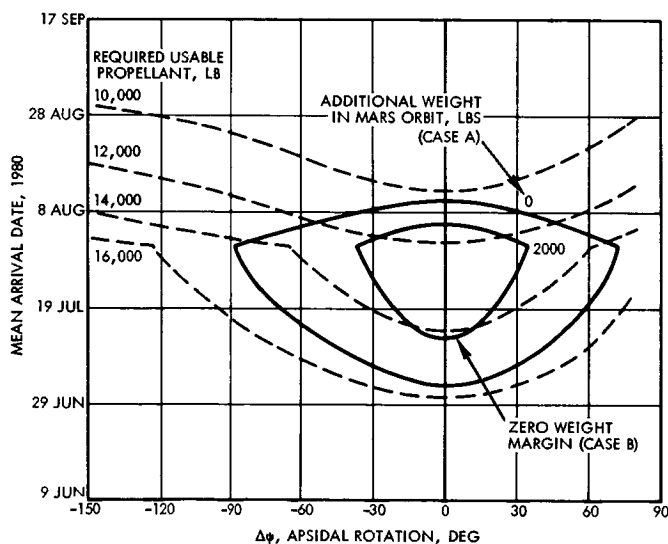


Figure 4-18

PERFORMANCE CAPABILITY and propellant requirements for the 1979 Type I mission opportunity are displayed as a function of apsidal rotation and arrival date for a 30-day launch period and a Mars orbit size of 1100 x 10,000 kilometers.

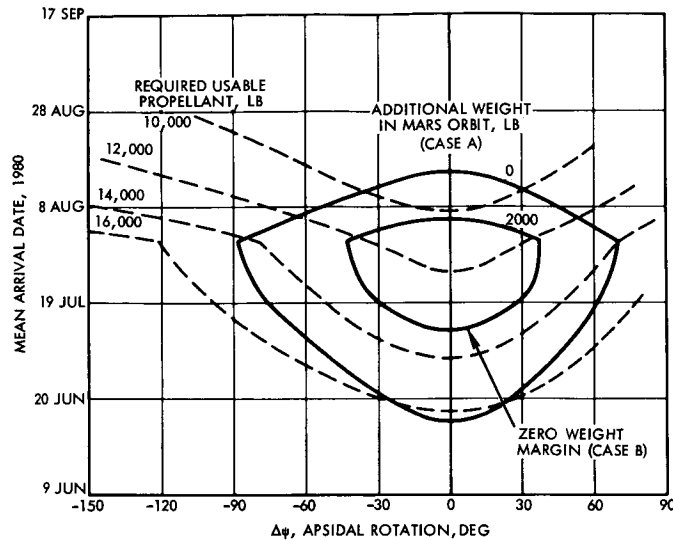


Figure 4-19

PERFORMANCE CAPABILITY and propellant requirements for the 1979 Type I mission opportunity are displayed as a function of apsidal rotation and arrival date for a 30-day launch period and a Mars orbit size of 1000 x 20,000 kilometers.

4.5 TRAJECTORY ACCURACY CAPABILITY

The accuracy requirements for Martian mission in 1975 and 1977 have been examined in a qualitative manner and the results indicate that the proposed spacecraft will satisfy these requirements.

Section 3.4 shows that the recommended spacecraft design satisfies all the accuracy requirements for Type I trajectories in 1973. It also shows which factors are most influential in establishing the significant guidance characteristics of the mission, as illustrated in Figure 4-20. A secondary factor, influencing the number of maneuvers, is the dispersion at Mars per unit velocity error soon after injection.

The vehicle maneuver execution errors will not change from 1973 to the later missions, as the spacecraft design is unchanged. The other two factors, arrival date separation ΔV and dispersion of trajectories at Mars per unit velocity error near earth, are compared in Table 4-4.

This table shows that the total propellant required for all interplanetary maneuvers will be reduced in 1975-1979, compared with 1973. (The performance analyses of the preceding section incorporate this reduction.) It also indicates that the dispersion of trajectories at Mars,

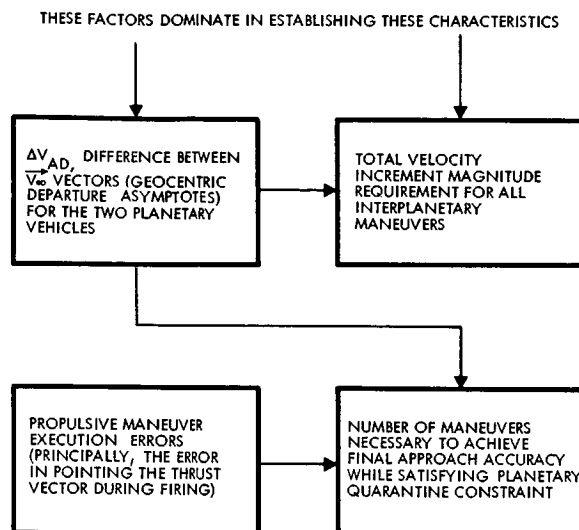


Figure 4-20
DOMINANT FACTORS in establishing the significant guidance characteristics for the interplanetary phase of the Voyager mission.

Table 4-4. Qualitative Evaluation of Factors Which Affect Guidance Characteristics

Year	Trajectory Type	Arrival Date Separation			Relative Dispersion in R-T Plane per Unit Velocity Error (Major Axis)
		Number of Days	ΔV Per Day (m/sec)	ΔV , Each PV (m/sec)	
1973	I	8	20-44	80-176	1 (Base)
1975	I	4	10-32	20-64	1.2
	II	4	24-40	48-80	4
1977	I	4	5-26	10-52	1.3
	II	4	19-38	38-76	3.5
1979	I	4	31-56	62-112	~ 1

as a result of the first maneuver, is reduced for Type I trajectories, especially those in 1975 and 1977, but may be enlarged by about a factor of two for Type II trajectories in those years.

Figure 4-21 shows the dispersion ellipses in the R-T plane resulting from a spherical error distribution whose standard deviation is 0.1 meter per second for the years 1973, 1975, and 1977. (The error distribution may arise from tracking uncertainty or maneuver execution errors.) The errors are taken two days after injection. Two ellipses appear for 1975 and 1977, one for Type I trajectories, and one for Type II trajectories. For these Type II trajectories only the major axes of the dispersions at Mars (after the first maneuver) are enlarged; the minor axes are reduced.

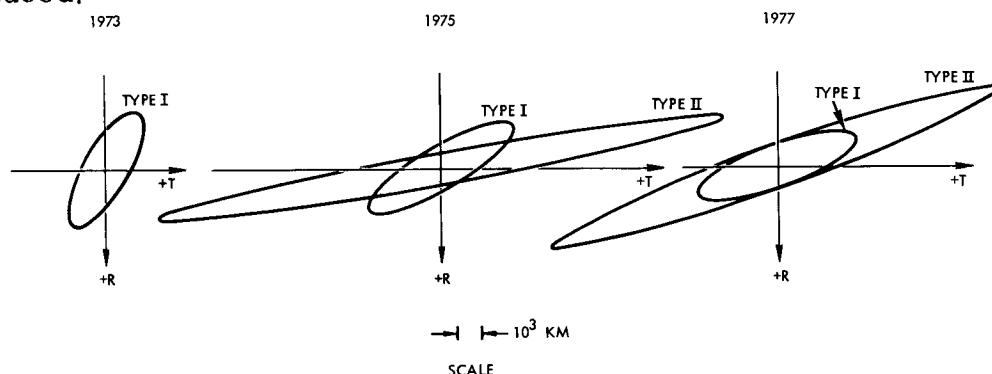


Figure 4-21
DISPERSION ELLIPSES FOR VARIOUS YEARS represent the propagation of a 0.1 meter/second spherical velocity error several days after launch to the impact-parameter plane (R - T plane) at Mars.

Thus Type II trajectories, resulting in longer but narrower dispersion ellipses, may call for a change in aim point selection. The strategy discussed previously for 1973 Type I trajectories called for the aim point for the separation maneuver and all subsequent maneuvers to lie along a line containing the center of Mars and the final aim point. Let us define this line as $\bar{M}$. This strategy maximized the probability of being within the final success zone after the separation maneuver, subject to planetary quarantine constraints, and minimized the probability of needing additional interplanetary corrections. However, if the angle between $\bar{M}$ and the major axis of the dispersion ellipse is small, the strategy may be revised to take advantage of the narrower ellipses. The new strategy would require the aim points of only the final one or two midcourse maneuvers to be along $\bar{M}$ and the selection of the aim point for the separation maneuver would be optimized by other techniques. This revision in strategy would very likely avoid a requirement for a third midcourse maneuver.



It was demonstrated in Section 3.4 that 70 percent of the final dispersions were due to tracking uncertainty before the final midcourse correction. Tracking accuracy is a function of equipment capabilities of the DSIF and the control center, the reliability of the on-board transponder, and the knowledge of the AU and the ephemeris of Mars. The equipment capabilities should be equal or superior in later years, and the uncertainty in the AU and the Martian ephemeris should be reduced as a result of interim interplanetary missions.

Therefore it is concluded that the proposed spacecraft system will meet the trajectory accuracy requirements for the 1975 to 1979 missions, whether Type I or Type II trajectories are employed.

The accuracies of entering into and of estimating the orbit about Mars are not very sensitive to the characteristics of interplanetary trajectories. Therefore, these accuracies are not expected to vary appreciably from those resulting in the 1973 mission, as long as orbits of similar size and orientation are used. Results similar to those indicated in Section 3.4.2 will apply in the 1975 to 1979 missions.

4.6 OBSERVANCE OF THE PLANETARY QUARANTINE CONSTRAINT

An assessment of the ability to observe the planetary quarantine constraint in the 1975, 1977, and 1979 missions, as compared with the 1973 mission, is concerned with the status of the planetary quarantine plan and on available information concerning the environment of Mars. Three sets of assessment conditions can be postulated, as follows:

- The present planetary quarantine plan and present information on the Martian environment
- The present planetary quarantine plan and improved information on the Martian environment
- A revised planetary quarantine plan with revisions based on data from the landed 1973 experiments

An assessment based on the present planetary quarantine plan and on present information on the Martian environment leads to three principal differences to be accommodated to satisfy the quarantine constraint in the later missions:

- a) The latter missions call for reduced separation in the arrival dates of the two planetary vehicles and utilize trajectories that generally have different sensitivities of miss at Mars per unit velocity error near earth, particularly considering the possible use of Type II trajectories in 1975 and 1977. These changes result in different magnitudes of biasing for injection and interplanetary maneuvers, and possibly a different number of such maneuvers. However, such differences will not call for any significant change in the trajectory control concepts which are applied to restrict the threat of contamination by impact resulting from injection or from interplanetary maneuvers. The same low probabilities of such contamination can be achieved in the later missions.
- b) Orbits of slightly lower altitudes may be selected for the later missions, because the required orbit lifetime through 1985 corresponds to a reduced elapsed time. However, the procedures to conduct the orbit insertion and orbit trim maneuvers to satisfy the constraint remain the same as for 1973.
- c) The control of contaminated efflux and ejecta from the spacecraft will be the same as in 1973.

If the present planetary quarantine plan remains in effect, but interim information is received on the Martian environment (from the 1973 Voyager mission, from other missions, or from earth-based investigations), then observance of the quarantine may lead to the following two mission differences:

- a) A better knowledge of the density of the upper Martian atmosphere could lead to a firmer (and presumably reduced) minimum orbital altitude requirement. This would influence orbit selection and could significantly enhance photo-imaging resolution from orbit.
- b) A refined model for the meteoroid environment at Mars could call for revisions in certain design aspects of the spacecraft and in mission procedures. Such revisions might apply to the requirement for preserving low contamination levels on exterior surfaces, and to the propellant tank pressure-venting procedures (discussed in Section 3.5). (Possible revisions in these areas may also occur as the result of interim research efforts, and may affect the 1973 mission as well as later ones.)



The most speculative assessment is based on a revised planetary quarantine plan. The significant revisions would probably be based on data from the initial surface biological experiments of Voyager 1973. While such revisions could have a far-reaching effect on the course of subsequent missions, the possible form of the effects can only be a guess. Because the current plan is relatively stringent, major revisions would likely be in the direction of relaxed constraints, and would therefore be capable of being fulfilled by the present spacecraft design.

To sum up, the observance of the planetary quarantine constraint is probably most critical for the 1973 mission, with no reduction in the spacecraft capability in this regard expected in the later missions.

4.7 ACCOMPLISHMENT OF SCIENTIFIC MISSION OBJECTIVES

4.7.1 Orbital Photo-Imaging

Photo-imaging data from the 1973 flight will allow changes in operation and configuration of the photo-imaging equipment to be used for the 1975 flight; in the same manner, each flight will affect the subsequent flights. The emphasis will shift from lower to higher resolution, a greater fraction of the imaging will be devoted to stereoscopic coverage, and possibly equipment modifications permitting resolution better than 10 meters on the ground will be added.

The camera giving 1000-meter ground resolution will be used to complete the planetary coverage for areas missed on earlier missions if they were inaccessible for good photography. Recording of seasonal changes such as the advance and retreat of the polar caps will probably continue to be valuable during these missions, and areas will be rephotographed to look for non-seasonal changes. Other temporal changes such as variations in the blue haze and occurrence of yellow clouds will also warrant color coverage at relatively low resolution. Overall, however, the amount and importance of the low resolution coverage can be greatly reduced.

Filling-in missing areas in the 100-meter ground resolution monoscopic coverage will be necessary for several missions because no initial

choice of orbital parameters provides complete coverage of the planet with satisfactory distance and illumination conditions. More important, however, will be the gathering of stereoscopic information for areas which previous coverage has shown to be of special interest.

High resolution photography will assume increasing importance as time goes on. The requirement to devote most of the transmission time to general coverage will no longer exist as that coverage will have been obtained. Moreover, previous information will indicate areas for which high resolution photography is desirable, eliminating the necessarily somewhat uncontrolled coverage of the first mission or two. As is the case with medium resolution photo-imaging, stereoscopic coverage will, for the same reasons, receive increasingly greater emphasis.

It is possible that some modest gains in the highest resolution obtainable can be made in later missions. The focal lengths of the optics, and hence the ground resolution, cannot be increased a great deal without an increase in weight allotment. Either the apertures must be increased to keep focal ratio about the same, or greater attention must be given to spacecraft stability and image motion compensation. However, a factor of two or so might be achievable with an acceptable weight increase. Also, it may be possible to reduce the orbital altitude as discussed in Section 4.6.

This improved resolution would permit highly selective coverage at greater detail than available previously, and could provide valuable information on the character of the Martian surface. This coverage would aid greatly in correlating orbital data with the surface investigations so as to allow global extrapolation of the latter.

4.7.2 Other Orbital Scientific Experiments

The recommended spacecraft described in Section 2 of this Volume supports either an all TV photo-imaging system or a combination film/TV system, as well as a basic "core" group of scientific experiments. This equipment and the experiments which they accomplish are described in Section 3.6. Detailed descriptions of the science experiments and objectives are given in Volume 5 (Science Integration) and Volume 11 (Photo-Imaging).



The recommended spacecraft is also designed to accommodate additional experiments with minimum impact on interfacing spacecraft subsystems. A list of additional experiments, which can also be carried on the recommended spacecraft, is given in Table 4-5 and the later missions should allow these to be accommodated. The experiments are listed under categories related to their mounting locations on the spacecraft.

The interfacing spacecraft electrical subsystems have been designed to allow addition of these additional experiments in the same manner as they interface with the basic core group of experiments. With the simple addition of an electrical integration assembly all of the body- and boom-mounted experiments can be accommodated.

Table 4-5. Additional Science Experiments

Planetary Scan Platform Mounted Experiments

- Polarimeter
- Meteoroid flash detector
- Gamma ray spectrometer

Boom-Mounted Experiments

- Magnetometer
- Neutron albedo

Body-Mounted Experiments

- Atmospheric mass spectrometer
 - Solar occultation
 - Cosmic ray
 - Micrometeor impact detector
 - Solar plasma
 - Solar X-ray
 - Celestial X-ray
-

The three additional experiments that are mounted in the planetary scan platform are interfaced through the PSP remote multiplexers, switching assemblies, and decoders designed for the core experiments. The computer and sequencing subsystem is also capable of handling the additional commands and providing the required sequencing of experiments in its recommended configuration.

The PSP box for the recommended spacecraft has also been sized to allow incorporation of the three additional PSP-mounted experiments without change in the basic box outer dimensions, without change in the yoke and shaft size and design, and without change in the PSP control system and gimbal drives. The addition of these experiments does not alter the previous requirement for PSP pointing accuracy or for image motion compensation (needed for the high resolution film portion of the photo-imaging system only).

An extensive discussion of the objectives accomplished by these additional experiments is given in Volume 5 and therefore is not repeated here.

4.8 SPACECRAFT COMMUNICATION PERFORMANCE

Section 3.7 dealt with the performance of the communication subsystem for a 1973 launch. The general comments and the range data presented in Table 3-18 are equally applicable for later launches. Figures 4-22 through 4-26 give the cumulative scientific data conveyed to earth after encounter for 1975, 1977, and 1979 launches. These figures result from the application of the ranges listed in Table 3-18 to early mid and late launch dates for Type I and Type II trajectories for each year.

The scientific role of post 1973 missions will differ from the 1973 mission to an extent depending on the success of the 1973 mission. For example, a completely successful 1973 mission will yield gross resolution photographs of much of the planet and leave to later missions the role of obtaining more detailed pictures and more stereoscopic and color images. The scientific implications of data communicated to earth are discussed further in Section 4.7.

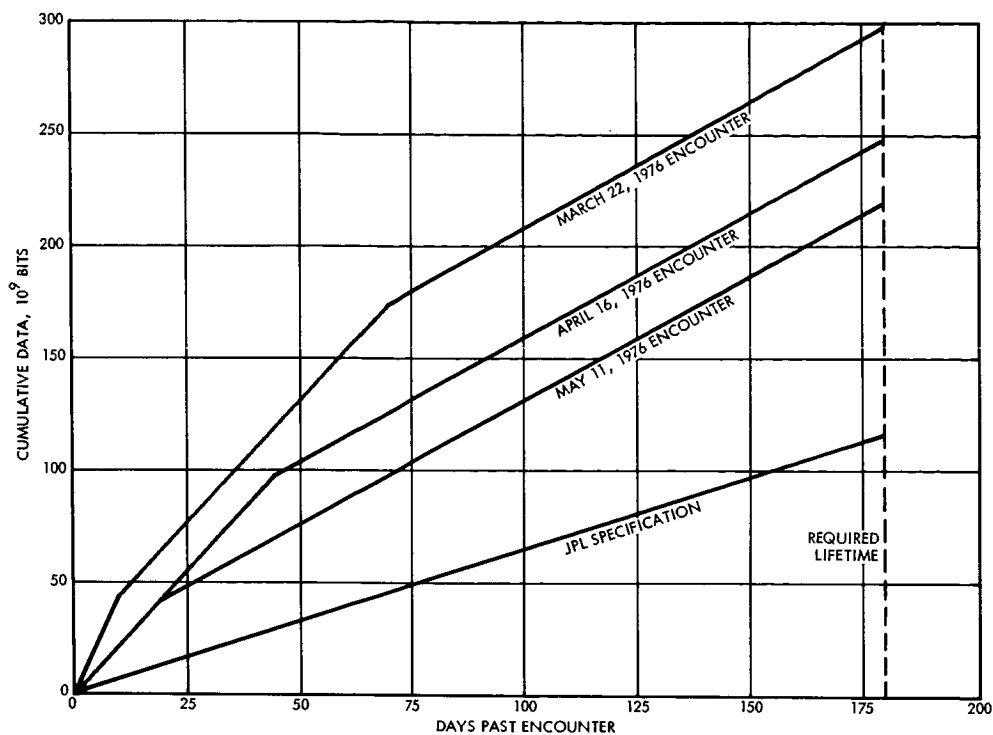


Figure 4-22
POST ENCOUNTER CUMULATIVE DATA for Type I trajectory for the 1975 launch

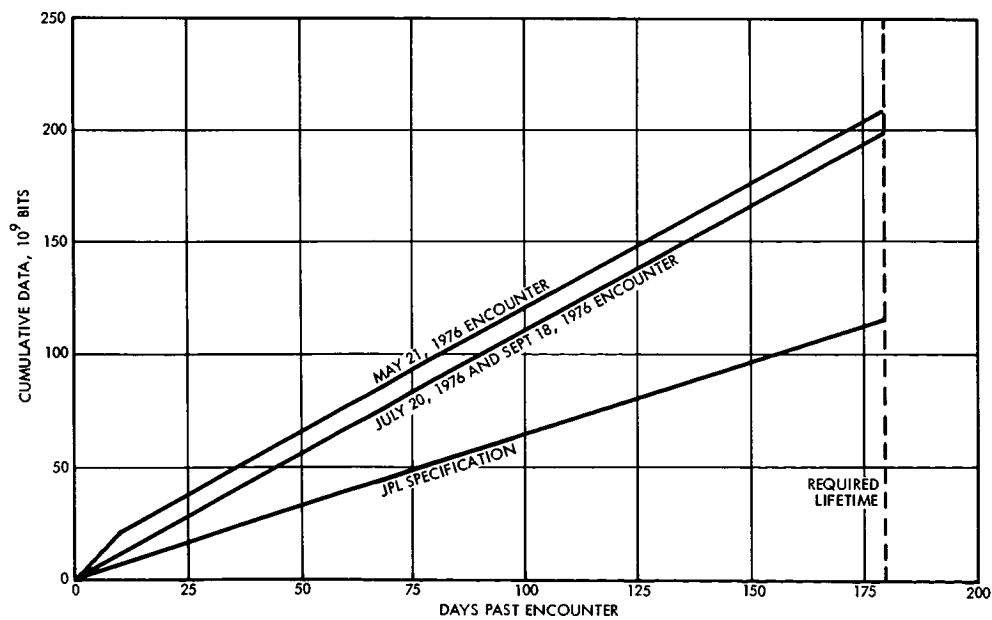


Figure 4-23
POST ENCOUNTER CUMULATIVE DATA for Type II trajectory for the 1975 launch

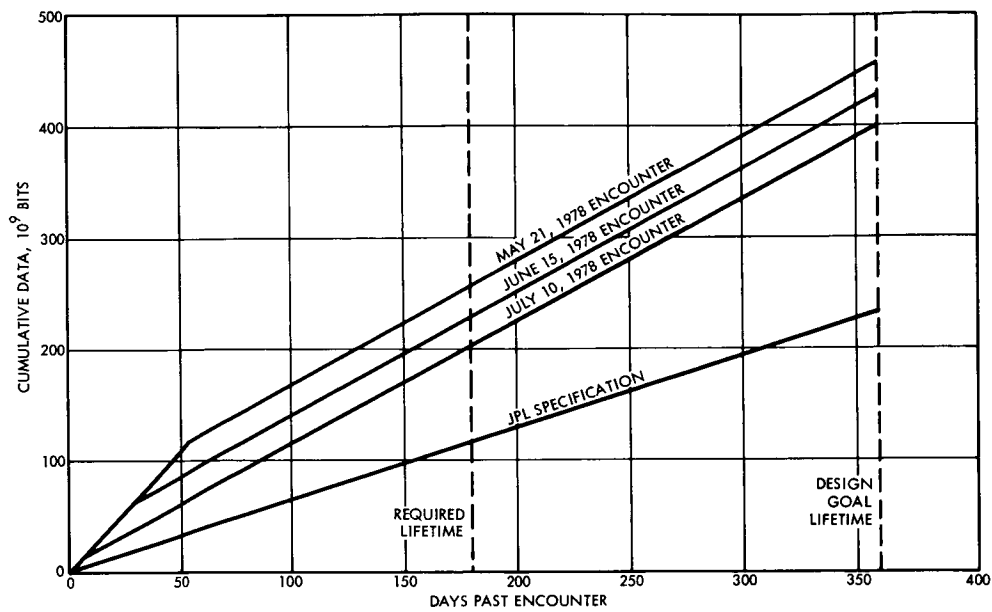


Figure 4-24
POST ENCOUNTER CUMULATIVE DATA for Type I trajectory for the 1977 launch

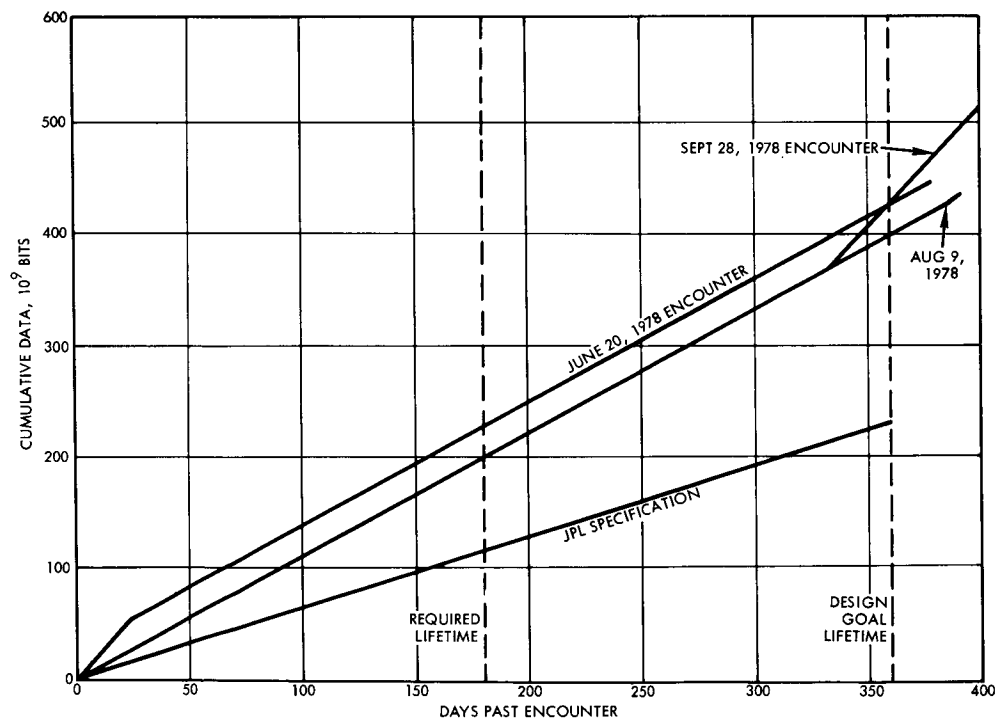


Figure 4-25
POST ENCOUNTER CUMULATIVE DATA for Type II trajectory for the 1977 launch

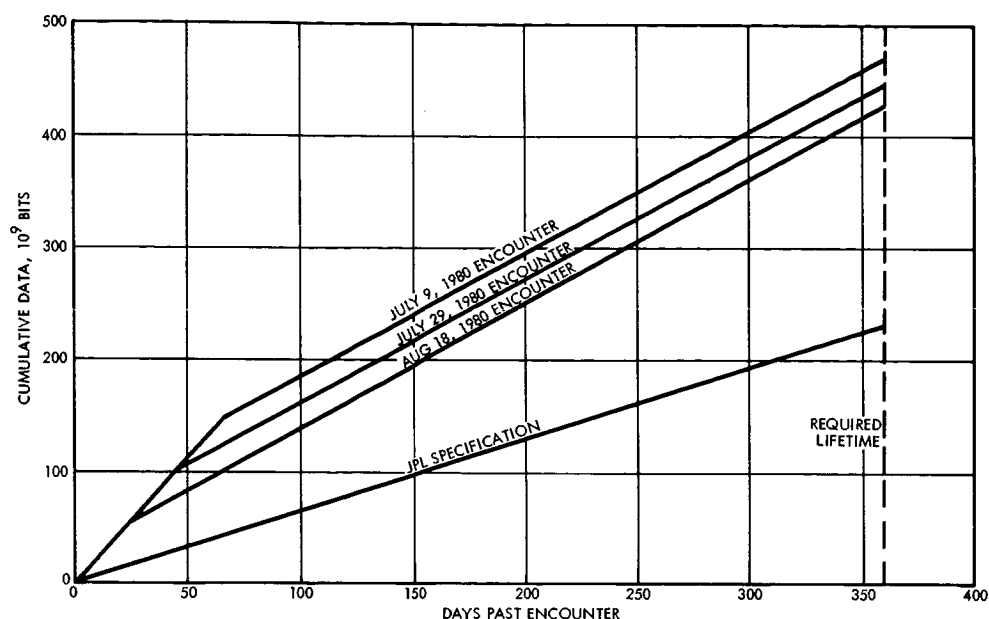


Figure 4-26
POST ENCOUNTER CUMULATIVE DATA for Type I trajectory for the 1979 launch

As for the 1973 launch period all missions exceed the specified minimum by a factor of nearly two or more over a six-month period. The total data received, however, is somewhat less than for a 1973 launch. The reason for this is that the ranges at encounter are substantially greater and consequently the higher data rates drop out more quickly. These reduced data rates in later missions are offset however, by the longer mission lifetime. For the 1977 Type II launch shown in Figure 4-22, all but the earliest encounters produce ranges so large that only the 12.8 kbit/sec rates are possible. In the case of an extremely late 1977 Type II launch, however, the range will have decreased sufficiently near the end of the design goal life of one year to permit a return to the 25.6 kilobit rate.

It can be seen by comparing the Type I and Type II trajectories for the same years that a serious data penalty results from the choice of a Type II trajectory and this factor must be included in evaluating which trajectory to select. The reader is referred to Volume 6 for the discussion of an upgraded version of this spacecraft and how the data rates could be improved.

4.9 PROBABILITY OF SUCCESS

A comparison between the spacecraft reliability levels for the 1973-1979 Mars opportunities is presented in Table 4-6. For this study, the spacecraft configuration, subsystem success criteria, reliability data, and analysis assumptions are the same as described for the 1973 mission (Section 3.8).

The differences in the success estimates reflect the impact of increased operating time on the accomplishment of mission objectives. The analysis does not account, however, for the reliability improvements which could be realized due to improvements in the state of the art, or manufacturing and flight experience gained on successive missions.

Table 4-6. Subsystem Reliability
for Mars Missions

Subsystem	1973 Mission	1975 Mission	1977 Mission	1979 Mission
Guidance and control	0.958	0.926	0.928	0.906
S-band radio	0.916	0.832	0.834	0.781
Command and Sequencing	0.976	0.929	0.932	0.899
Electrical Power	0.985	0.964	0.965	0.949
Telemetry and Data Storage	0.921	0.830	0.835	0.778
Electrical Distribution	0.999	0.994	0.994	0.993
Propulsion	0.966	0.966	0.966	0.966
Structures	0.972	0.937	0.938	0.922
Thermal control	0.994	0.993	0.993	0.990
Planetary scan platform	<u>0.955</u>	<u>0.919</u>	<u>0.921</u>	<u>0.901</u>
Single Spacecraft	0.697	0.470	0.488	0.371
At least 1 of 2 Spacecraft	0.905	0.720	0.733	0.604
Total Mission Time (hr)	6,800	12,200	12,000	15,100
Interplanetary Transit (days)	224	330	320	270
Mars Orbit (days)	60	180	180	360



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