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VOYAGER

SPACECRAFT Phase B, Task D

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FINAL REPORT

OCTOBER 1967

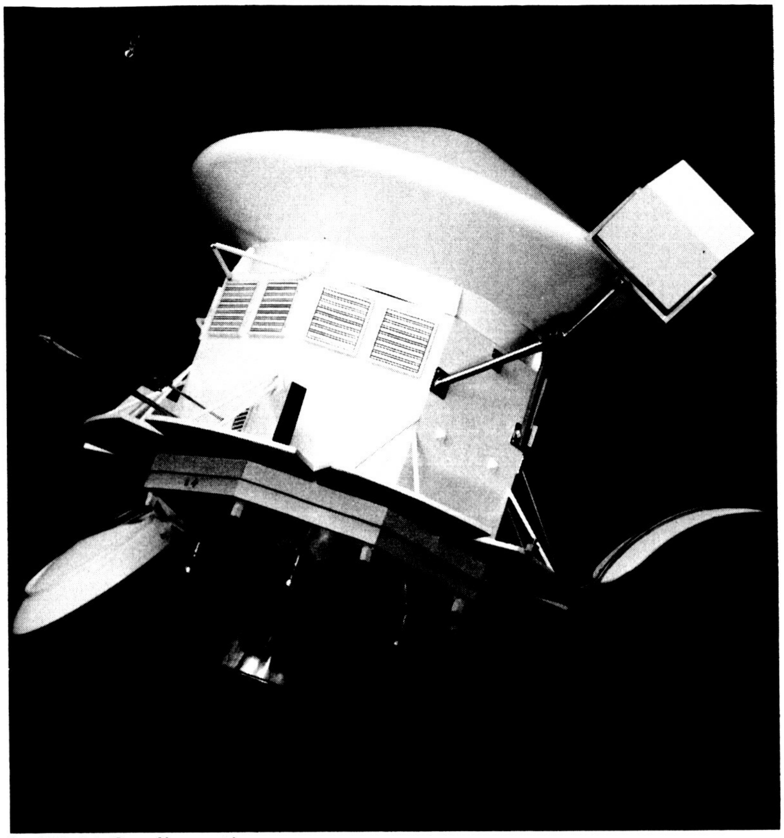
Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

Voyager Operations
Space Vehicles Division

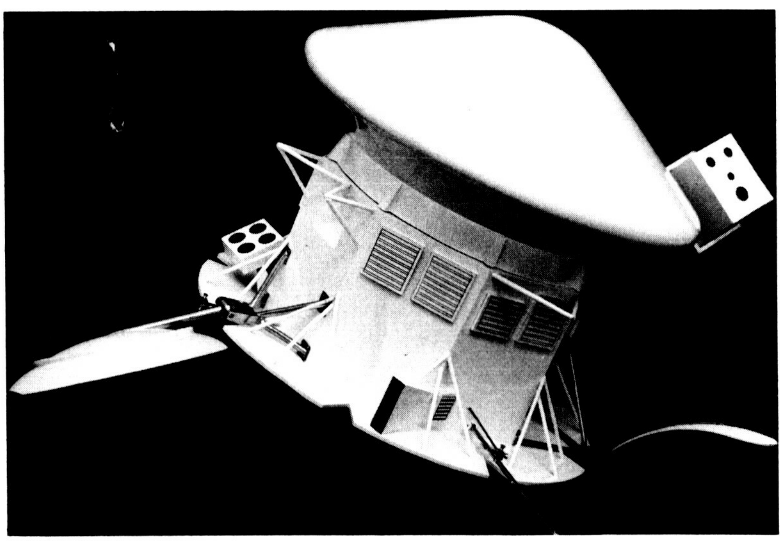
TRW SYSTEMS GROUP

One Space Park, Redondo Beach, California

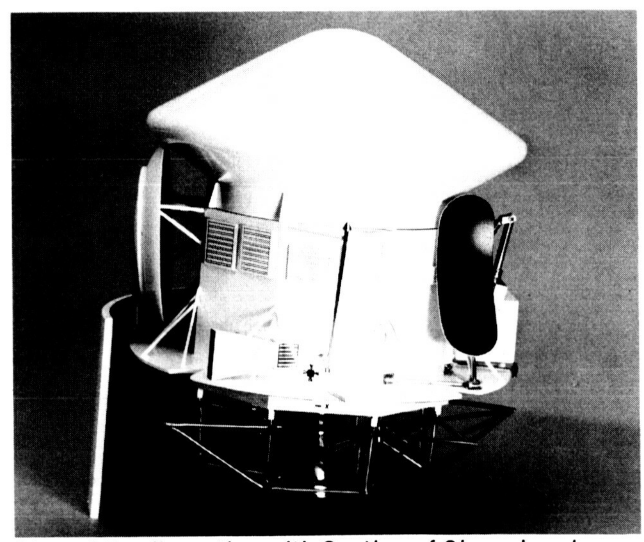
MODEL OF **TRW** RECOMMENDED **VOYAGER SPACECRAFT**



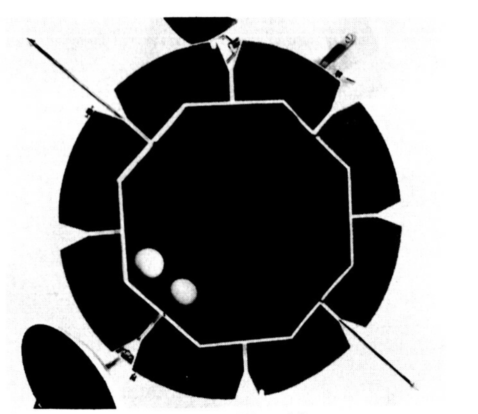
In-Flight Configuration



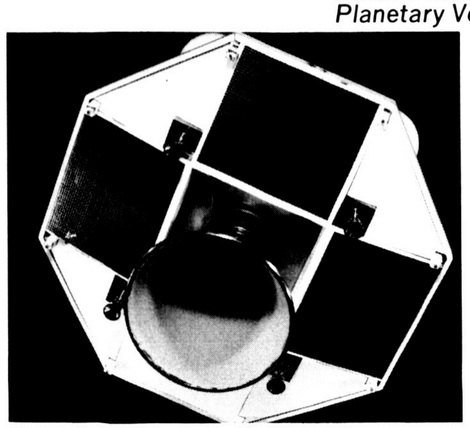
Opposite View In-Flight Configuration



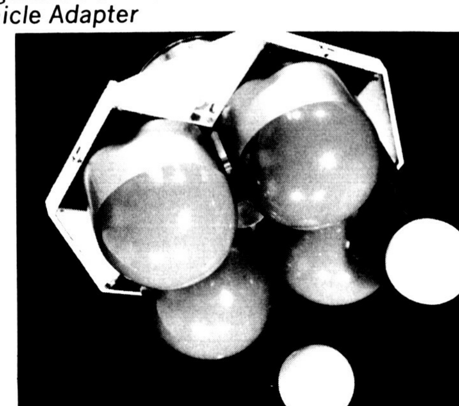
Stowed Configuration with Section of Shroud and Planetary Vehicle Adapter



Propulsion Module, Top View



Propulsion Module, Bottom View



Equipment Module, Bottom View

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VOYAGER

SPACECRAFT Phase B, Task D

FINAL REPORT

Volume 1. Summary

OCTOBER 1967

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

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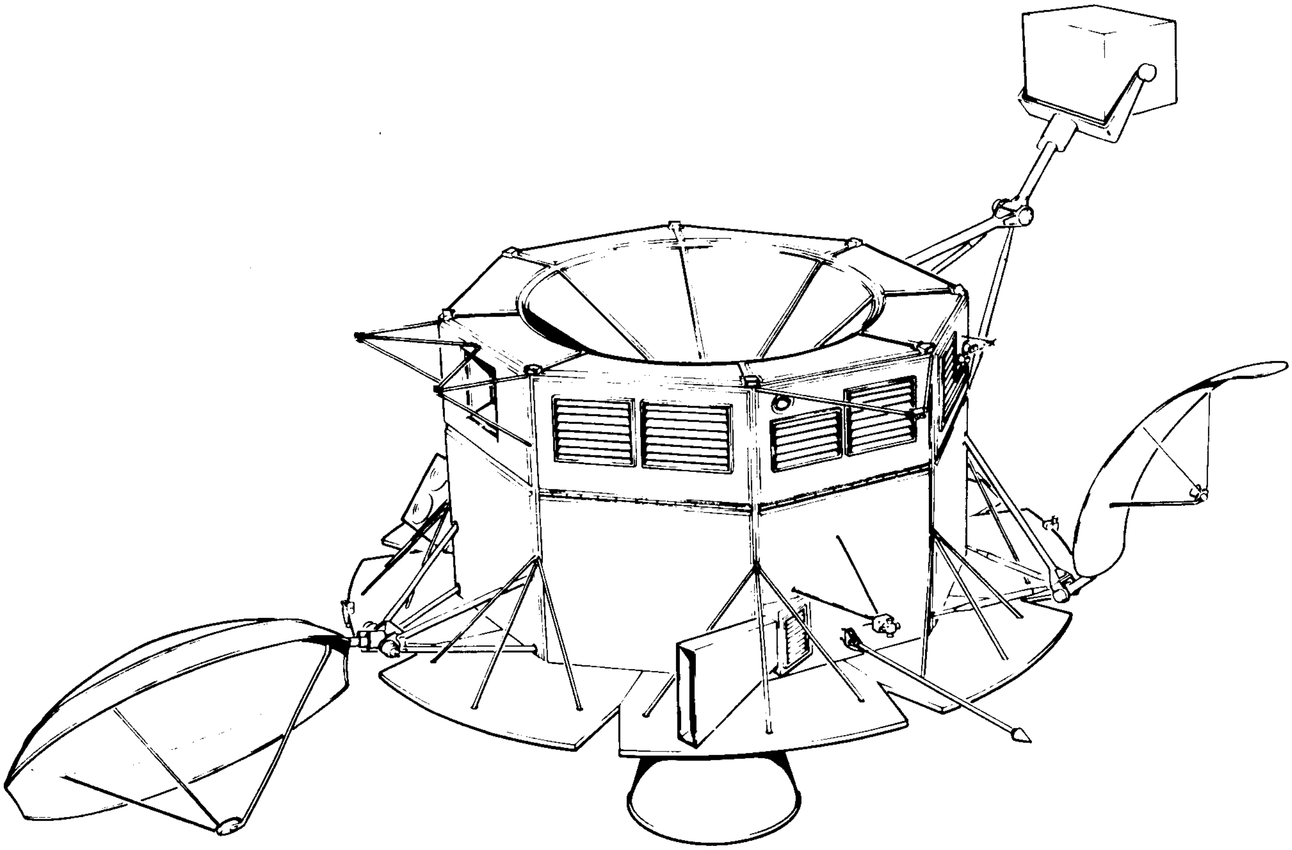
PREFACE

The results of TRW's studies of the Voyager spacecraft system as related to the MSFC Task D requirements are presented in this report, in summary and in detail. Here in Volume 1, the major conclusions are summarized with no attempt to establish the framework of assumptions and definitions within which they fit. In the remaining 10 volumes, however, these conclusions are reached in the context of all of the significant data on which they are based. To aid in relating the objectives and provisions of the report requirements as established by MSFC, the following table locates the response by TRW to the requirements of the contract:

Contract Reference			TRW Report Volume
Article I. Scope			
	Item A		1-7
	Item B,	Paragraph 1	9
		Paragraph 2	10
		Paragraph 3	10
		Paragraph 4	10
		Paragraph 5	11
Exhibit A, Contract Guidelines			
	Section II,	Item A	2, 6
		Item B	2-4
		Item C	6
		Item D	6
		Item E	7
		Item F	9-11
		Item G	5
	Section IV,	Item A	2
		Item B	2
		Item C	2
		Item D	2-4
		Item E	7
		Item F	8
		Item G	7, 8
Appendix A, Report and Visual Aids Requirements			
	Section A,	Paragraph 1	1
		Paragraph 2	2-11

CONTENTS



1. INTRODUCTION

In this report, TRW defines in depth a spacecraft configuration that can be used efficiently for the Martian launch opportunities from 1973 through 1979, without hardware modification, carrying either a 5000- or a 7000-pound capsule. The only difference in the spacecraft for different launch opportunities and different capsule weights is the amount of propellant carried.

In reaching its recommended configuration and evaluating the performance of that configuration, TRW has applied both Case A and Case B as defined by MSFC; Case A calls for a 5000-pound capsule for all four mission years, 1973-79, while Case B calls for a 6000-pound capsule in 1973 and a 7000-pound capsule in 1975, 1977, and 1979. For each opportunity, several combinations of requirements are identified including such mission factors as the length of the allotted launch periods, ellipticity of the Martian orbit, and rotation of the orbit after injection. The recommended spacecraft can accommodate all of these missions with a substantial performance margin, except that in 1975 and 1977 the larger capsule weights will require the more elliptical orbits unless the longer interplanetary trajectories are flown (Type II instead of Type I). With Type II trajectories in 1975 and 1977, a very large performance margin is available. The possibilities for using excess performance capability include:

- Obtaining orbits smaller in size than nominal
- Achieving substantial rotations of the orbit apsidal line
- Carrying a heavier capsule than nominal
- Increasing spacecraft subsystem weights for improving system performance
- Increasing the length of the launch period
- Decreasing transit time between the earth and Mars

Regardless of whether the capsule conforms to Case A or B, the preferable spacecraft design appears to be that which satisfies nominal mission requirements while affording the greatest flexibility and selectivity in the exploitation of the capability of the Saturn V for Voyager. The recommended spacecraft design has been optimized to that end.

The technology of the Voyager spacecraft configuration recommended in this report is entirely within the 1967 state of the art. The design features modularity in the assembly of subsystems and of the spacecraft, permitting an essential flexibility during project implementation and confidence in the project schedule. The design is highly adaptable to changes and to growth. Thus, we can increase the scientific payload by some 200 pounds over the hypothetical payload of the recommended configuration and step up the telemetry rate by a factor of 6, without altering the structure or propulsion or other subsystem configurations. The increase in effective radiated power entailed by this growth is achieved by a larger antenna and transmitter.

FOR A TYPICAL CASE A MISSION in 1973, of the 41,586 pounds of payload (excluding shroud) launched by the Saturn V, each of the two planetary vehicles has expended nearly half of its weight by the time it is in orbit at Mars. These values take into account the 5000-pound project contingency in Saturn V performance.

Condition	Weight (lb) Case A	Case B	
Gross injected weight	20,793	22,852	24,783
Less planetary vehicle adapter	-686	-686	-686
Net injected weight	20,107	22,166	24,097
Less midcourse correction propellant	-1,393	-1,536	-1,670
Less 50 percent usable nitrogen (assumed)	-28	-44	-44
Planetary vehicle at Mars orbit insertion	18,686	20,586	22,383
Less orbit insertion propellant	-7,706	-8,490	-9,231
Planetary vehicle in Mars orbit	10,980	12,096	13,152
Less capsule	-5,000	-6,000	-7,000
Spacecraft in Mars orbit	5,980	6,096	6,152
Less available orbit correction propellant	-564	-627	-680
Less remainder of usable nitrogen	-28	-44	-44
Spacecraft at end of life	5,388	5,425	5,428

2. SPACECRAFT SYSTEM

2.1 GENERAL CONFIGURATION AND PERFORMANCE

The configuration of the recommended spacecraft is summarized on the following two pages. Weight analyses for both Case A and B are presented below, and typical weight histories are given to the left. In this configuration the modular approach is incorporated at the level both of the three major spacecraft modules and at the level of subsystems; in general all components of a given subsystem are mounted together on a single panel, permitting separate assembly and testing on the panel. Propulsion tanks, identical for all mission years, are sized to hold 16,000 pounds of propellant in keeping with the greatest propellant loading anticipated, as determined by the optimization study discussed in Volume 6.

WEIGHT ANALYSES OF THE RECOMMENDED CONFIGURATION shows increases only in propellants and attitude control in going from Case A to Case B

	Weight (lb)		
	Case A	Case B	
Structure	913.0	913.0	913.0
Propulsion	1,568.7	1,568.7	1,568.7
Equipment and Instrumentation	2,276.0	2,340.6	2,340.6
Mechanisms	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
Balance Weights	15.0	15.0	15.0
Contingency	216.1	217.7	217.7
Total Dry Spacecraft	4,988.8	5,055.0	5,055.0
Residuals	455.1	458.2	461.2
Propellant	412.9	416.0	419.0
Pressurant	42.2	42.2	42.2
Total Inert Spacecraft	5,443.9	5,513.2	5,516.2
Usable Propellant	9,663.1	10,652.5	11,580.5
Total Spacecraft at Liftoff	15,107.0	16,165.7	17,096.7
Capsule	5,000.0	6,000.0	7,000.0
Total Planetary Vehicle at Liftoff	20,107.0	22,165.7	24,096.7

PRELIMINARY
Recommended

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

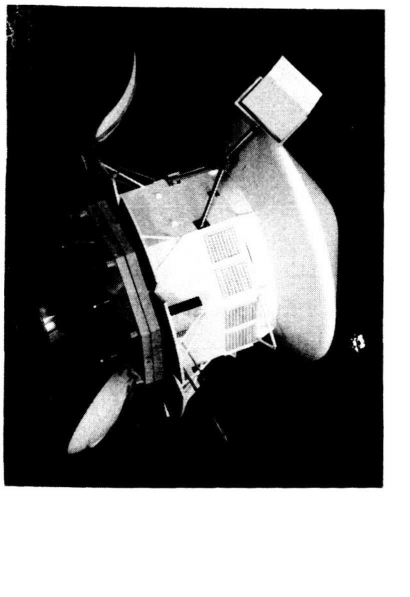
Appendage Mechanization:

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

Capsule	5,000	5,000	7,000
Spacecraft (inert)	5,444	5,513	5,516
Usable propellant *	9,663	10,653	11,581
Total Planetary Vehicle	20,107	22,166	24,097

* For minimum ΔV 1.95 Km/sec

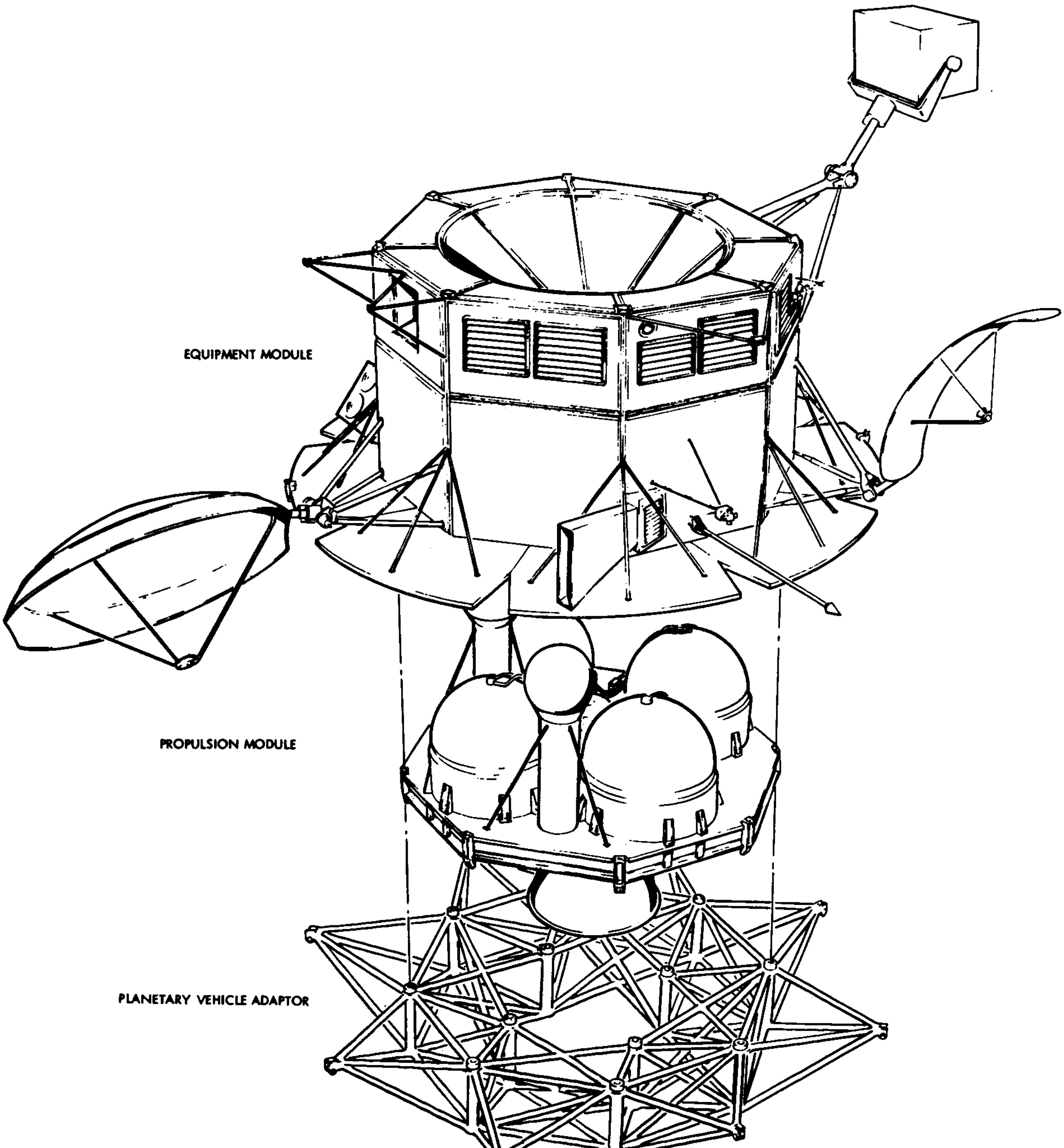


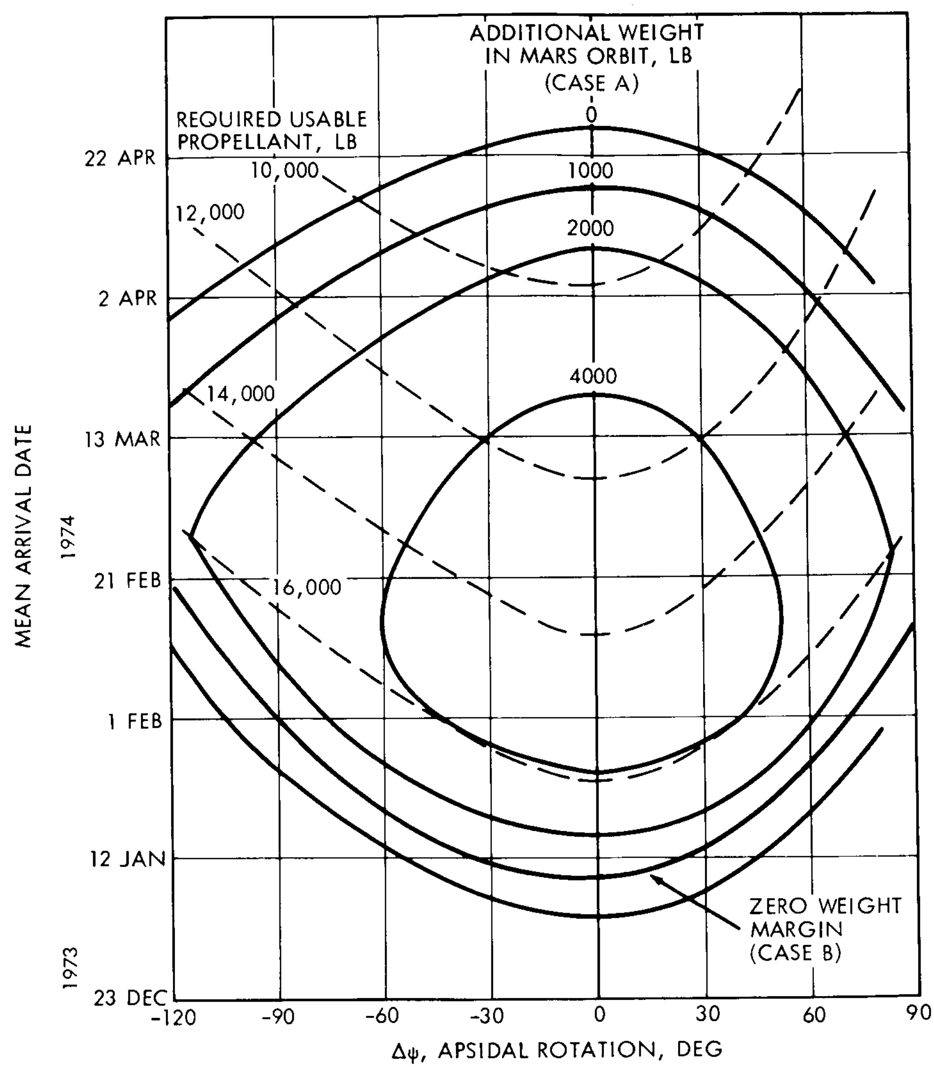
Performance | Characteristics

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 adpatively 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 handling unit plus two remote multiplexer units on Planetary Scan Platform. Six tape recorders. Flexible format generator selects programmed formats or programs new formats.
S-BAND COMMUNICATIONS	
High gain antenna: 34 db 3.2 degrees beam width — beamwidth Medium gain antenna: 28 db 4 x 10 degrees beamwidth — beam width Low gain antenna: 1 db	All digital communications. Redundant 50-watt TWT's. Four receivers, four antennas: high gain (9.5 ft), medium gain; two low gain giving complete 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 synchronizers and decoders demodulate and synchronize commands 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 control; nitrogen heated gas; sun and Canopus references; redundant inertial reference
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. Stoves 512 20-bit words.	Primary sequencer plus accelerometer counters and function generators, with decoder in 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 ±1% Individual DC/DC converters, 400 Hz to appendage drives, 819.2 KHz sync signal to subsystems	226 sq ft fixed solar array; three NiCd batteries (48 amp-hrs)
ELECTRICAL DISTRIBUTION AND PRYOTECHNIC 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 A-50 or MMH Main Engine: Thrust 9850 lbs, I_{sp} 298 sec; 1700 lbs, 289 sec Backup Engines: 400 lbs Total, 292 sec	Modified LMD main engine with tankage identical for all missions; four C-1 engines as automatic backup for Mars orbit insertion to achieve reduced mission; main engine gimballed and back-up engine duty-cycle-modulated for thrust vector control
TEMPERATURE CONTROL	
(see table below)	Multilayer insulation; individually actuated louvers, special surface finishes

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

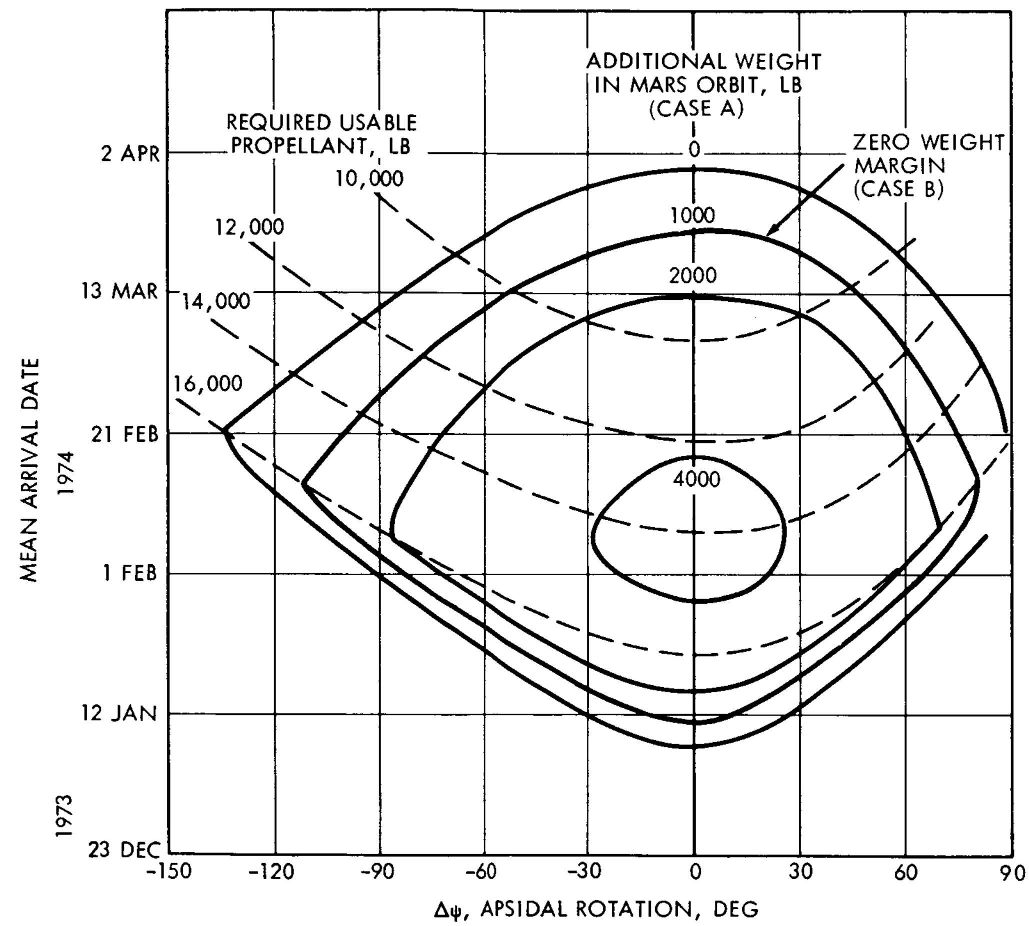
Spacecraft System





THE PERFORMANCE CAPABILITY OF THE RECOMMENDED SPACECRAFT for a Type I trajectory to Mars in 1973 is displayed in terms of six variables that can be adjusted according to the goals of a given mission. On these charts two of the variables are held constant: in both charts the orbit at Mars is assumed to be 1100 by 10,000 km; the chart at the left assumes a launch period of 20 days, that at the right 30 days. The curves then bound the capabilities of the spacecraft in the other four variables: mean arrival date, orbital change capability, required propellant, and payload weight, with payload defined as additional weight on the planetary vehicle beyond the Case A configuration.

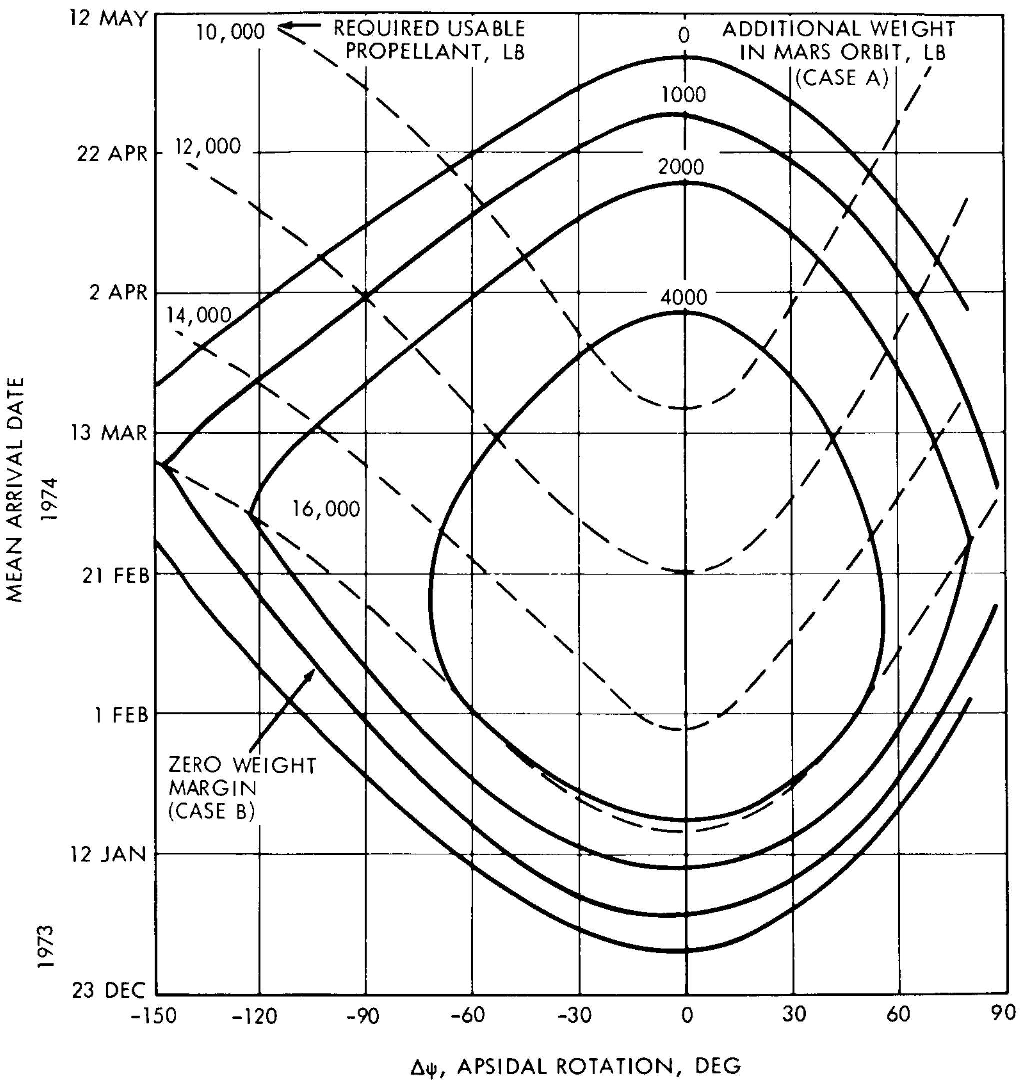
For example, on the left within the solid-line contour labeled 2000 pounds are the options possible with a 7000-pound capsule (5000 + 2000). In this case the position of periapsis of the orbit can be altered 82 degrees through a posigrade rotation (+ $\Delta\psi$) from that provided by the minimum velocity increment at insertion or 112 degrees through a retrograde rotation (- $\Delta\psi$). At the same time the mean arrival date can be selected at any time from January 10 to



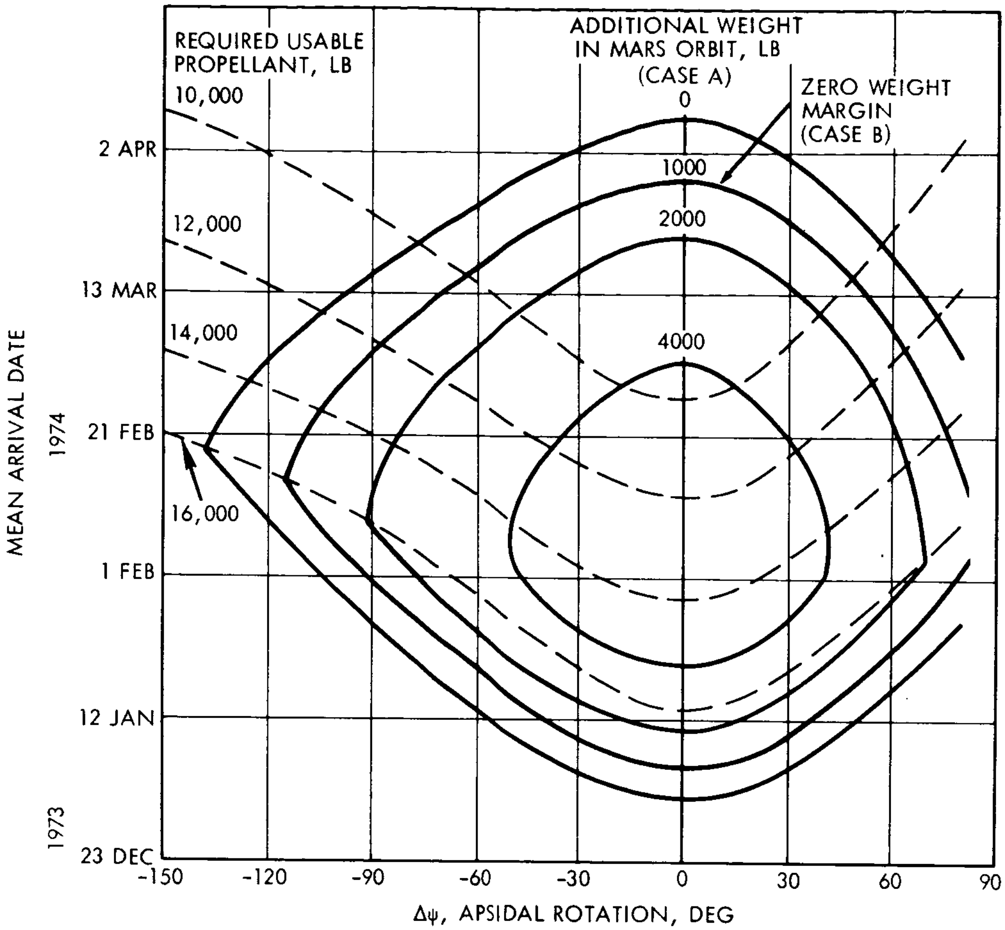
April 8, 1974. Since the 16,000-pound dashed-line curve defines the upper limit of propellant capacity on the recommended configuration, the range of choices below that contour all assume that amount of propellant.

On the other hand, if the mission goal is initially established in terms of a selected arrival date and a desired amount of apsidal rotation, a point on the chart is fixed which defines the payload capability and the consequent amount of propellant required. For example, if one wishes to be able to change the orbit orientation by -60 degrees (or equivalently +50 degrees) and arrive at Mars on March 13, 2800 pounds of additional weight (e.g., a 7800-pound capsule) and 12,825 pounds of propellant are thereby established.

As shown on the right, to plan the mission for the same orbit at Mars but with the greater flexibility of a 30-day launch period reduces the range of the options. With a 7000-pound capsule, the arrival date can be chosen between February 6 and March 12 and the orbit rotated in the range from 68 to -88 degrees.



THE PERFORMANCE CAPABILITY OF THE RECOMMENDED SPACECRAFT is further displayed by showing that if the size of the orbit at Mars is increased, the energy required for orbit injection is reduced and as a consequence more latitude is possible in selecting among other mission variables. As in the previous charts, contours are



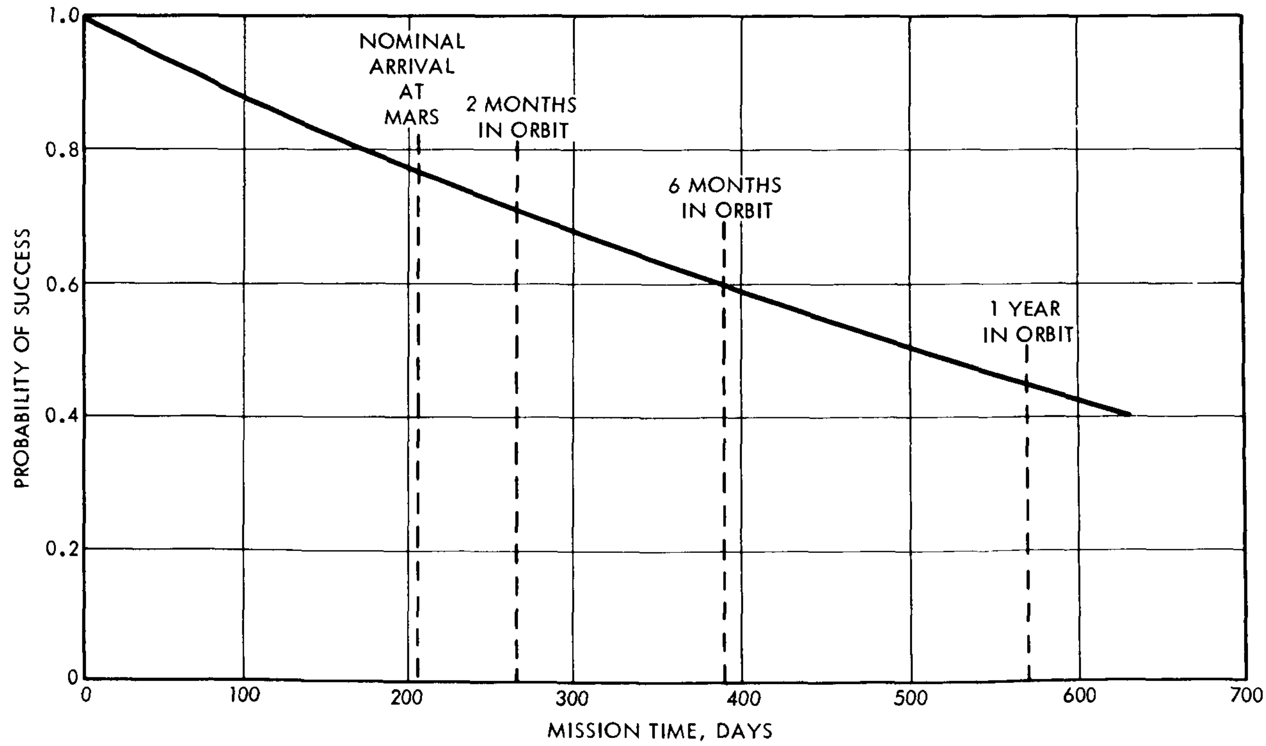
plotted for the two launch periods, 20 and 30 days, and the relationships among the options available in the other four variables are shown.

Given an orbit of 1000 by 20,000 km, a 20-day launch period, and a planetary vehicle carrying a 7000-pound capsule, the apsides of the orbit can be rotated 80 to -122 degrees, keeping the date of arrival at Mars constant around February 27, 1974. Alternately, the capability for altering the orientation of the orbit can be traded for early arrival on January 10 (with 16,000 pounds of propellant).

On the basis of this configuration, the charts on pages 6 through 9 illustrate the type of performance margins available. Contours of additional weight on these figures illustrate the range within which the margin can be devoted to increasing payload weight by reducing the arrival date span and possible rotation of periapsis in the orbit at Mars. For an orbit 1100 x 10,000 km, the maximum additional weight capability of 5480 pounds is achieved with an arrival date of February 11, 1974.

These figures show that selecting an orbit with a higher ellipticity results in a greater spacecraft capability since the velocity increment for the orbit insertion maneuver is reduced. On the other hand, designing for a longer launch period reduces spacecraft capability since both C_3 and V_∞ are necessarily increased (for a fixed mean arrival date). Analyses for 1975, 1977, and 1979, presented in Volume 2, show similar performance margins available in these years.

The analyses include all energy requirements for mission success, including trajectory bias to meet the Martian quarantine objectives. By means of such mission factors, control of the potential sources of



PROBABILITY OF SUCCESS of the recommended configuration is plotted for the 1973 mission. Although the nominal transit time of 205 days is indicated, the reliability calculations are based on the upper-limit transit time of 224 days in 1973, introducing a conservative bias into the results.

contaminated effluents, and well-planned safeguards during spacecraft fabrication and assembly, the probability of contaminating Mars with live terrestrial organisms is kept below the mission allotments, as discussed in Volume 2.

With the recommended configuration, the probability that one of the two tandem-launched planetary vehicles enters the desired orbit at Mars and completes all of its objectives in that orbit is calculated to be 90 percent. As shown to the left this reliability is based on a probability of success of 70 percent calculated for one spacecraft.

2.2 SPACECRAFT DESIGN

The recommended configuration provides modularity, good load paths, fixed solar-cell panels, and convenient, structurally sound mounting and control of appendages. The octagonal cross-section accommodates the propellant tanks efficiently while retaining structural simplicity and direct load paths. On the equipment module structure, panels for meteoroid protection form the structural sides of the spacecraft and serve as mounting surfaces for the astrionics equipment.

PRELIMINARY SPECIFICATION

Structural Subsystem

Purpose

Provides structural integration, support and environmental protection for the spacecraft subsystems and mounting provisions for the flight capsule.

SUBSYSTEM CHARACTERISTICS

The subsystem is composed of an equipment module structure and propulsion module structure plus support and release mechanisms for appendages.

Characteristics

LOAD FACTORS (LIMIT)	LONGITUDINAL		LATERAL	
Primary structures	Static	Dynamic	Static	Dynamic
1st stage burnout	+5.0		±1.0	
1st stage cutoff		2.0		±1.0
Retrofire	+2.0		±0.3	

FACTORS OF SAFETY	Yield	Ultimate
General Structure	1.00	1.25

METEOROID PROTECTION	
Spacecraft surface area	650 square feet
Mission time	284 days
Probability of zero penetration	0.87
Mission reliability	0.97

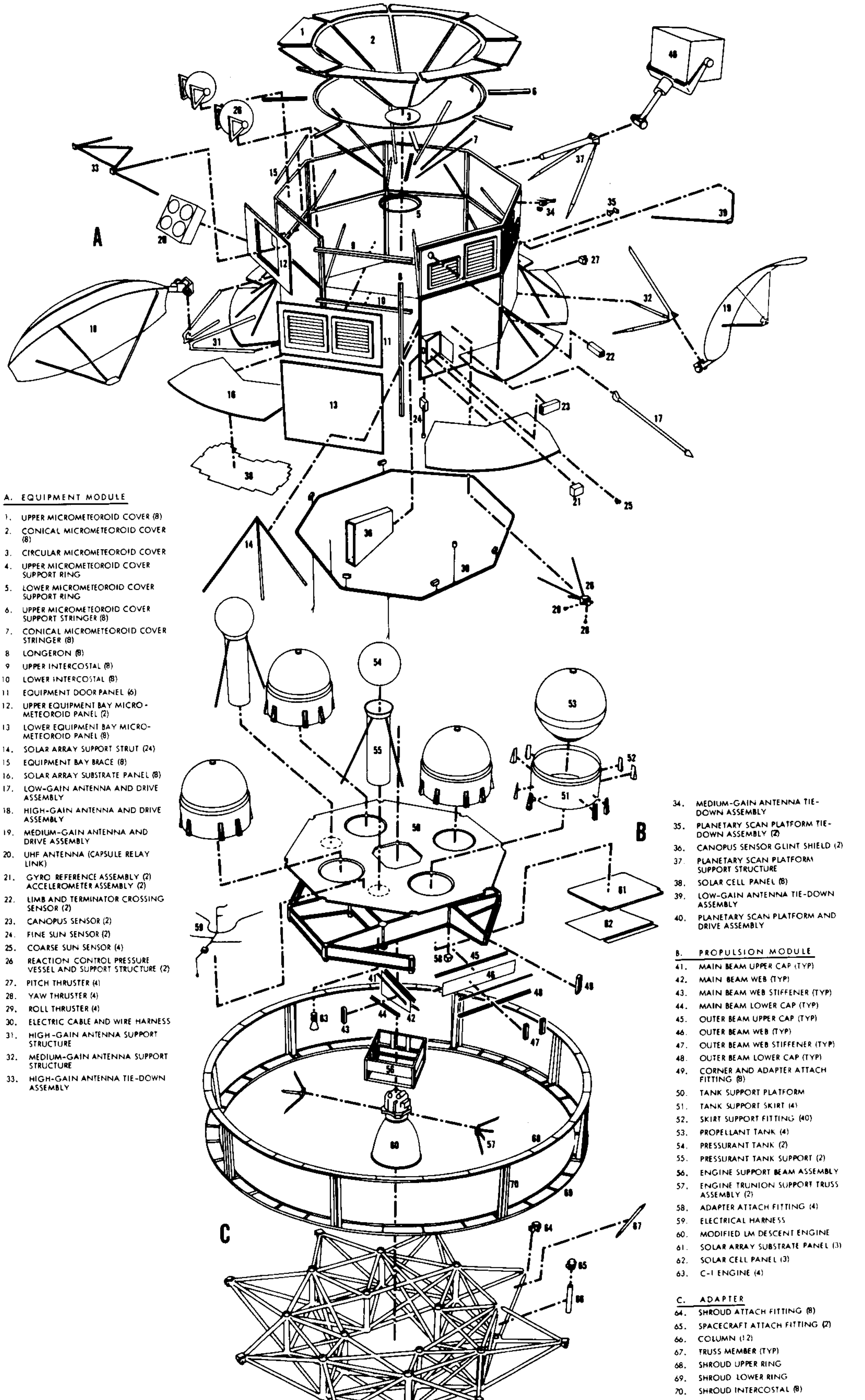
Performance Characteristics

CONFIGURATION

COMPONENT	OVERALL DIMENSIONS	WEIGHT	MATERIALS AND CONSTRUCTION
Propulsion module structure	158 in. across flats octagon taper to 57 in. square x 22 in. high	484 lb	7075 Al built up beams plus honeycomb deck
Equipment module structure	158 in. octagon x 100 in. high	1008 lb	7075 Al-semi-monocoque plus meteroid panels
Subsystem		1492 lb	

Interfaces FLIGHT CAPSULE: 8 equally spaced bolts on a 160 in. diameter bolt circle.

PLANETARY VEHICLE ADAPTER: 12 points, 8 equally spaced on 160 in. diameter bolt circle
4 equally spaced on 80 in. diameter bolt circle



A. EQUIPMENT MODULE

1. UPPER MICROMETEOROID COVER (8)
2. CONICAL MICROMETEOROID COVER (8)
3. CIRCULAR MICROMETEOROID COVER
4. UPPER MICROMETEOROID COVER SUPPORT RING
5. LOWER MICROMETEOROID COVER SUPPORT RING
6. UPPER MICROMETEOROID COVER SUPPORT STRINGER (8)
7. CONICAL MICROMETEOROID COVER STRINGER (8)
8. LONGERON (8)
9. UPPER INTERCOSTAL (8)
10. LOWER INTERCOSTAL (8)
11. EQUIPMENT DOOR PANEL (6)
12. UPPER EQUIPMENT BAY MICRO-METEOROID PANEL (2)
13. LOWER EQUIPMENT BAY MICRO-METEOROID PANEL (8)
14. SOLAR ARRAY SUPPORT STRUT (24)
15. EQUIPMENT BAY BRACE (8)
16. SOLAR ARRAY SUBSTRATE PANEL (8)
17. LOW-GAIN ANTENNA AND DRIVE ASSEMBLY
18. HIGH-GAIN ANTENNA AND DRIVE ASSEMBLY
19. MEDIUM-GAIN ANTENNA AND DRIVE ASSEMBLY
20. UHF ANTENNA (CAPSULE RELAY LINK)
21. GYRO REFERENCE ASSEMBLY (2) ACCELEROMETER ASSEMBLY (2)
22. LIMB AND TERMINATOR CROSSING SENSOR (2)
23. CANOPUS SENSOR (2)
24. FINE SUN SENSOR (2)
25. COARSE SUN SENSOR (4)
26. REACTION CONTROL PRESSURE VESSEL AND SUPPORT STRUCTURE (2)
27. PITCH THRUSTER (4)
28. YAW THRUSTER (4)
29. ROLL THRUSTER (4)
30. ELECTRIC CABLE AND WIRE HARNESS
31. HIGH-GAIN ANTENNA SUPPORT STRUCTURE
32. MEDIUM-GAIN ANTENNA SUPPORT STRUCTURE
33. HIGH-GAIN ANTENNA TIE-DOWN ASSEMBLY
34. MEDIUM-GAIN ANTENNA TIE-DOWN ASSEMBLY
35. PLANETARY SCAN PLATFORM TIE-DOWN ASSEMBLY (2)
36. CANOPUS SENSOR GLINT SHIELD (2)
37. PLANETARY SCAN PLATFORM SUPPORT STRUCTURE
38. SOLAR CELL PANEL (8)
39. LOW-GAIN ANTENNA TIE-DOWN ASSEMBLY
40. PLANETARY SCAN PLATFORM AND DRIVE ASSEMBLY

B. PROPULSION MODULE

41. MAIN BEAM UPPER CAP (TYP)
42. MAIN BEAM WEB (TYP)
43. MAIN BEAM WEB STIFFENER (TYP)
44. MAIN BEAM LOWER CAP (TYP)
45. OUTER BEAM UPPER CAP (TYP)
46. OUTER BEAM WEB (TYP)
47. OUTER BEAM WEB STIFFENER (TYP)
48. OUTER BEAM LOWER CAP (TYP)
49. CORNER AND ADAPTER ATTACH FITTING (8)
50. TANK SUPPORT PLATFORM
51. TANK SUPPORT SKIRT (4)
52. SKIRT SUPPORT FITTING (40)
53. PROPELLANT TANK (4)
54. PRESSURANT TANK (2)
55. PRESSURANT TANK SUPPORT (2)
56. ENGINE SUPPORT BEAM ASSEMBLY
57. ENGINE TRUNION SUPPORT TRUSS ASSEMBLY (2)
58. ADAPTER ATTACH FITTING (4)
59. ELECTRICAL HARNESS
60. MODIFIED LM DESCENT ENGINE
61. SOLAR ARRAY SUBSTRATE PANEL (3)
62. SOLAR CELL PANEL (3)
63. C-I ENGINE (4)

C. ADAPTER

64. SHROUD ATTACH FITTING (8)
65. SPACECRAFT ATTACH FITTING (2)
66. COLUMN (12)
67. TRUSS MEMBER (TYP)
68. SHROUD UPPER RING
69. SHROUD LOWER RING
70. SHROUD INTERCOSTAL (8)

Attached to the upper face of the grid beams of the propulsion module are honeycomb sandwich panels forming a platform on which the tank skirts are supported. On the rectangular areas of the lower surface of the grid assembly are mounted solar-cell arrays, with four triangular areas available for growth in the power supply.

Activated by pyrotechnic release devices at the attachments to the adapter, a spring system imparts a velocity differential to the planetary vehicle for separation from the adapter. The net perturbation velocity off axis is statistically small, even including c.g. offset effects, such that simple rails and followers are adequate to guide the spacecraft from the shroud. Pyrotechnic releases are also employed for deploying the four antennas and the planetary scan platform.

The spacecraft is designed so that there can be no unstable slosh mode in the propellant tanks at any time in the mission, even if the capsule must be separated before arrival at Mars.

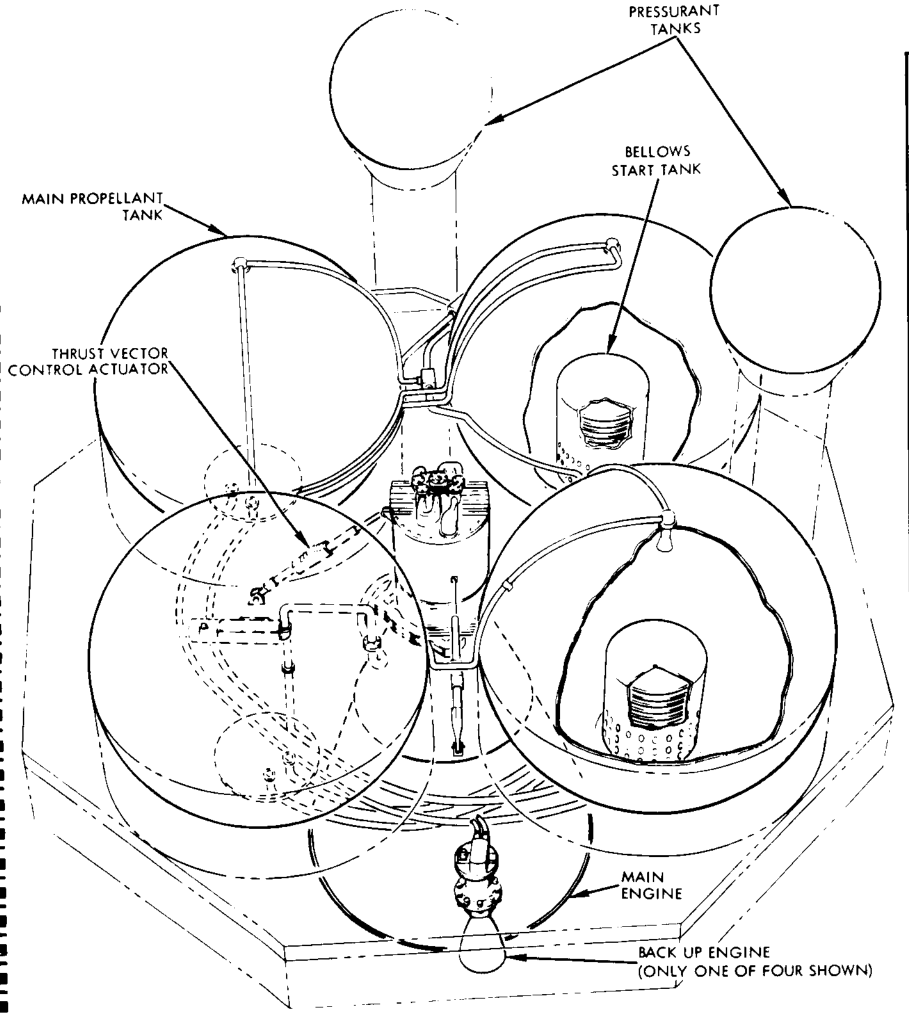
The Lunar Module Descent Engine, modified for two discrete thrust levels, 1700 and 9850 pounds, is incorporated in the main propulsion subsystem. The lower thrust is for interplanetary trajectory corrections and for orbit adjustment at Mars; the higher thrust is for orbit insertion at Mars. A cluster of four C-1 engines is also included, which, as discussed in Volume 3, can place the planetary vehicle in a highly elliptical orbit of typically 1000 x 80, 000 km, permitting thereby a means of saving the Voyager mission if the main engine fails.

The guidance and control subsystem provides three-axis attitude control of the planetary vehicle and spacecraft at all times after separation from the launch vehicle. It also measures acceleration during engine firing and senses the crossing of the Mars limb and terminator. In the guidance and control subsystem a proven design is used with full redundancy of all critical components. In addition, the system can operate in various alternate modes if a component fails.

Immediately following separation of the planetary vehicle from the S-IVB, the guidance and control subsystem reduces tumbling rates to a low level (approximately 0.01 deg/sec) to facilitate and prepare for communication and prepare for operations following emergence from eclipse.

PRELIMINARY

Propulsion



Physical Characteristics

WEIGHTS, LB

Dry	1565.2
Burnout without helium)	1998.4
Pressurant (helium)	42.2
Fuel (50-50 UDMH/N_2H_4)	6153.8
Oxidizer (N_2O_4)	9846.2

PRESSURE, PSIA

Helium storage	4000
Regulator outlet	249
Propellant tank operating	235
Engine inlet	220
Engine chamber	
High thrust	100
Low thrust	18

MISCELLANEOUS

Maximum engine gimbal angle	± 6°, 2 axis
Nozzle area ratio	47.5:1
Response, signal to 90% thrust	0.25 sec
Engine mixture ratio (oxidizer/fuel)	1.6:1

Purpose

To provide thrust at levels and times as required to accomplish the planetary arrival date separation maneuver, interplanetary trajectory corrections, orbit insertion, and trimming of the attained orbit.

Performance Characteristics

Total impulse:	4.82×10^6 lb/sec
Thrust levels	
High:	9850 lb
Low:	1700 lb

PERFORMANCE

Maneuver	Nominal Thrust	Nominal I_{sp}
Planetary arrival date separation	1,700	298
Orbit insertion, start	9,850	305
Orbit insertion, end	10,000	303
Orbit trim	1,700	289

THERMAL REQUIREMENTS

Propellant temperature	Bulk temperature 70 ± 20°F ΔT between unlike propellants, 5°F ΔT between like propellants, 2°F
Feed system temperature	70 ± 20°F
Engine head end valve temperature	120°F max, 20°F min
Engine internal surfaces exposed to solar heat during cycle	200°F max

ENGINE FLOW RATES, LB/SEC

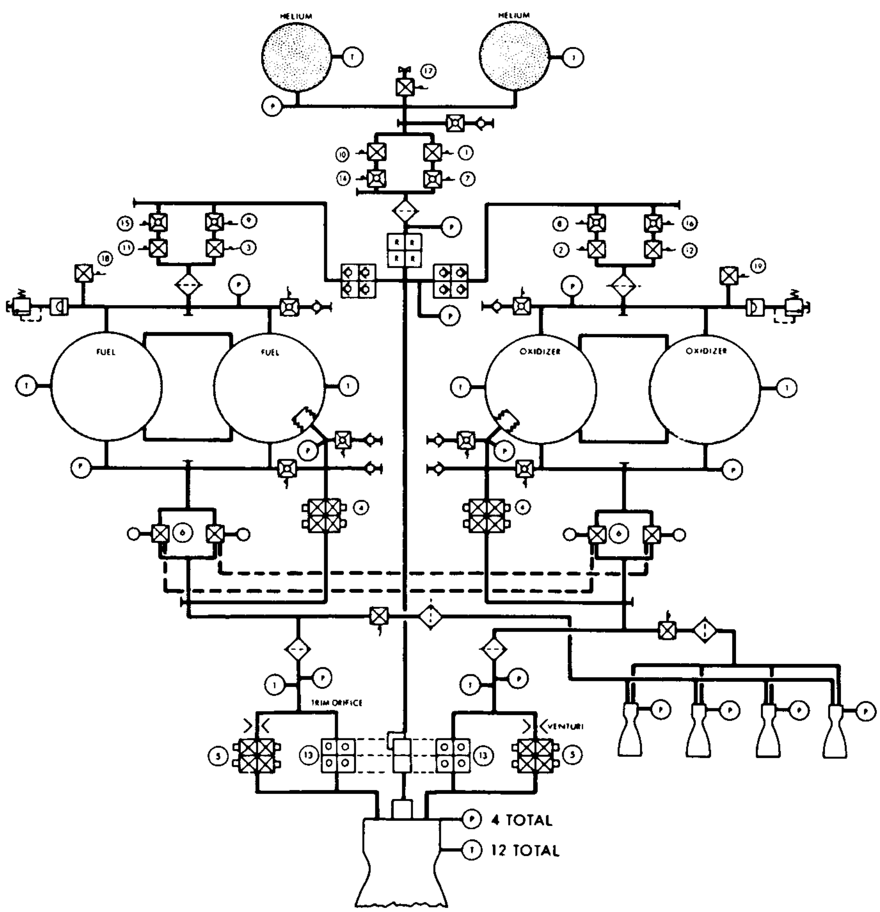
Maneuver	Nominal Thrust	Flow Rate
Planetary arrival date separation	1700	5.7
Orbit insertion, start	9850	32.3
Orbit insertion, end	10,000	33.0
Orbit trim	1700	5.9

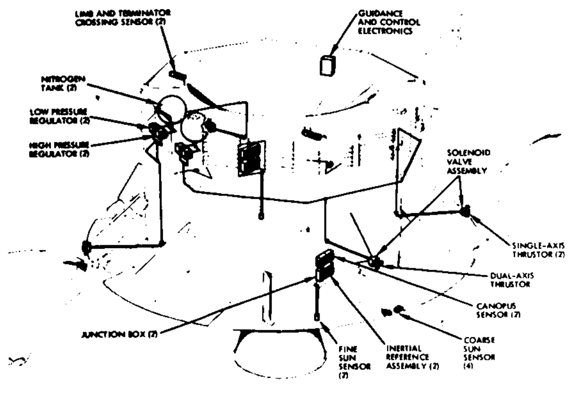
SPECIFICATION

Subsystem

ITEM	QUANTITY	WEIGHT (LB)	SYMBOL
PRESSURIZATION SYSTEM			
Pressurant tank	2	264.0	
Fill and vent coupling (helium)	1	0.3	
Vent coupling (propellant)	2	0.9	
Explosive valve, normally closed	11	5.0	
Explosive valve, normally open	13	7.0	
Filter	3	0.9	
Quad pressure regulator	1	12.0	
Quad check valve assembly	2	1.8	
Burst disc and relief valve assembly	2	1.6	
Pressure transducer	3	1.6	
Miscellaneous hardware and lines	2	18.0	
Temperature transducers		0.5	
		312.8	
PROPELLANT FEED SYSTEM			
Propellant tank	4	292.0	
Start tank assembly	2	50.0	
Fill and drain coupling	4	1.8	
Prevalve	4	20.0	
Start tank control valve	2	10.0	
Pressure transducer	2	0.5	
Temperature transducer	4	1.0	
Miscellaneous hardware		4.3	
Fuel lines		13.0	
Oxidizer lines		18.0	
Filters	4	4.6	
Electrical harness		5.0	
Junction box		5.0	
		425.2	

ITEM	WEIGHT (LB)	SYMBOL
ENGINE ASSEMBLY		
Combustion chamber assembly	202.5	
Chamber heat shield	8.0	
Seal	2.0	
Nozzle insulation	26.5	
Nozzle extension	36.0	
Hardware	5.0	
Injector	29.3	
Propellant lines and ducts	13.0	
Control valve - high thrust	17.0	
Control valve - low thrust	13.8	
Hardware	0.8	
Trim orifices	0.5	
Electrical harness	6.0	
Junction box	4.0	
Hardware - J.B.	3.0	
Instrumentation	26.1	
Gimbal assembly	5.5	
Pintle actuator	4.0	
	405.0	
TOTAL DRY WEIGHT	1156.8	





Entering sunlight, the pitch and yaw axes stabilize with solar array normal to the sun. The sun is also a celestial rèference for maneuvers. The roll axis stabilizes with respect to Canopus. This orientation is maintained throughout interplanetary cruise,

PRELIMINARY SPECIFICATION

Guidance and Control Subsystem

Purpose

The guidance and control subsystem provides continuous three axis attitude control of the planetary vehicle using either a sun-canopus celestial reference system or an on-board inertial reference system. Capability is provided for reorientation in all three axis and for the measurement of incremental velocity in the thrust axis. An indication of the crossing of the Mars dawn and evening terminators is provided.

Performance

ATTITUDE CONTROL

Maintains pointing accuracy of ± 0.6 deg, 3σ each axis with respect to celestial reference frame.

Orients to desired direction, provides thrust vector pointing accuracy of ± 0.43 deg, 3σ each axis with respect to the celestial reference frame.

Photo-Imaging

Maintains spacecraft pointing accuracy of ± 0.35 deg, 3 each axis and angular rates less than 10 deg/hr.

Intertial Hold

Provides for attitude control using on-board inertial reference with attitude drift less than 0.5 deg/hr.

VELOCITY MEASUREMENT PERFORMANCE

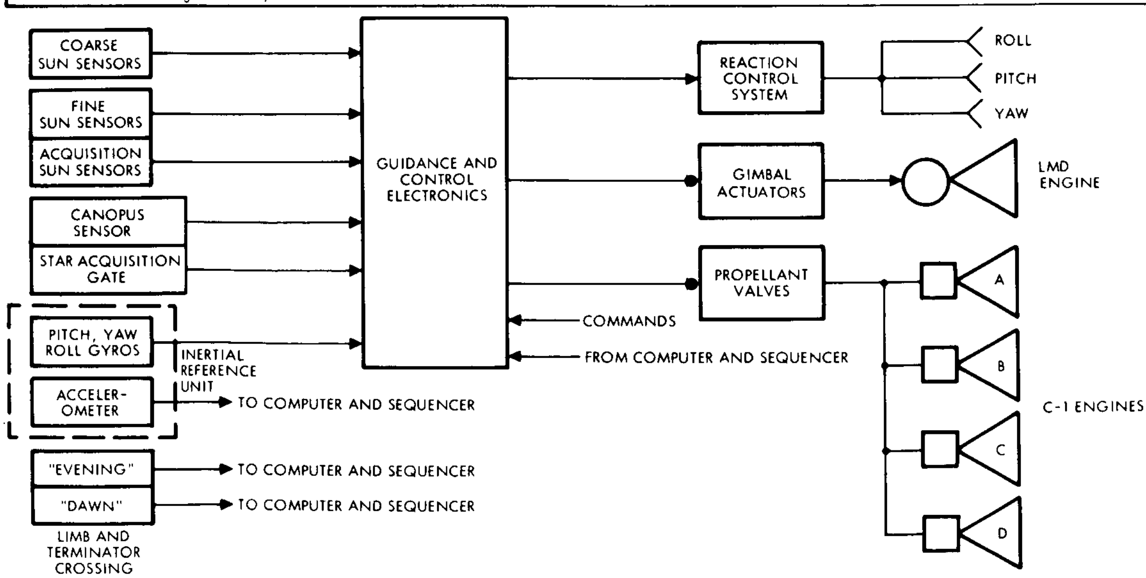
Measures velocity increments of up to 3000 meters/sec with an accuracy of 0.1 percent and with a resolution of 0.005 meters/sec.

TERMINATOR CROSSING INDICATION PERFORMANCE

Provides an indication of the passage of the sunlit portion of the Mars disc through a cone angle of 90 degrees in the spacecraft coordinate system.

Physical Characteristics

Weight:	279 lb
Power:	205 watts (average during maneuvers)



except during trajectory correction maneuvers. Before the main engine is fired, the spacecraft is oriented correctly in accordance with ground commands previously stored in the computer and sequencer and the attitude is verified by achieving communications on the high-gain antenna, the antenna having been positioned before the maneuver. The attitude reference for these operations is an inertial reference unit, the elements of which are initially aligned with the celestial reference frame. After each maneuver the cruise mode of operation is re-established. In Mars orbit the guidance and control subsystem maintains three-axis stabilization of the vehicle using again the sun-Canopus reference. Two Canopus sensors are carried, on opposite sides of the spacecraft, to allow rapid roll orientation at Mars irrespective of whether the direction of the spacecraft in orbit is northward or southward. Should the sun or Canopus be occulted, control is switched to the inertial unit until the celestial references are again visible.

The commands and control data for a complete mission are stored in the sequencer before launch. The subsystem can then program the mission independently, using sensor outputs for control or generation

PRELIMINARY SPECIFICATION

Command Subsystem

Purpose

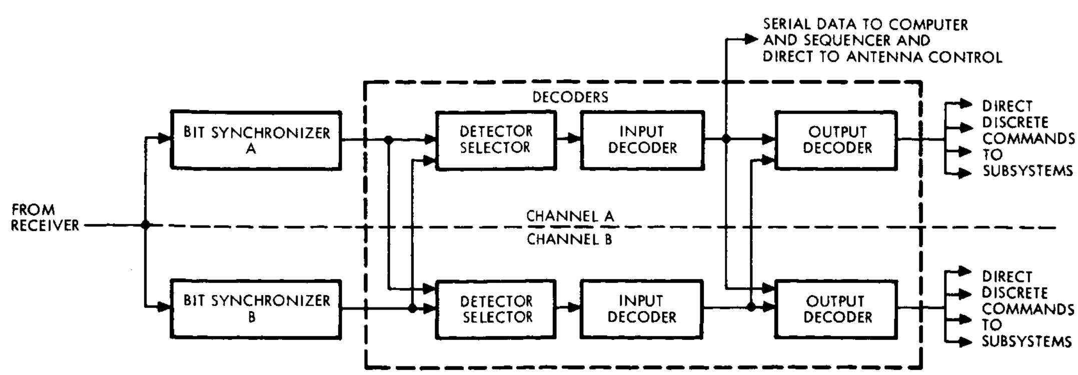
The command subsystem receives commands from the S-band radio subsystem and routes them to spacecraft subsystems and the capsule as directed by addresses in the command data.

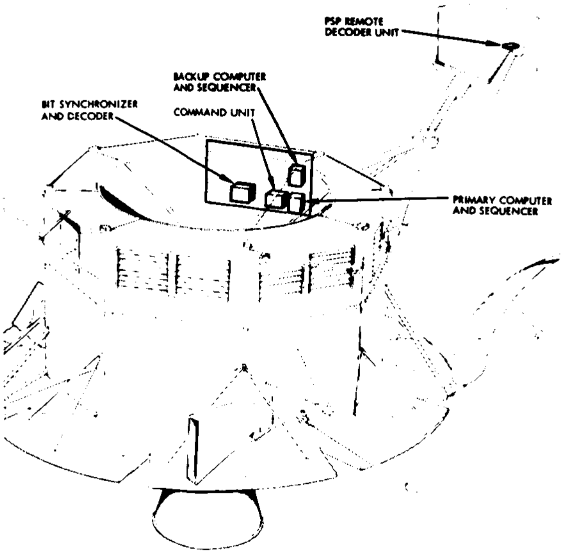
Performance Characteristics

Decoding:	256 discrete commands 13 addresses for serial messages 12-bit serial words to antenna 32-bit serial words to computer and sequencer

Physical Characteristics

Total volume:	380 cu in.
Total weight:	10.5 lb
Total power:	12.4 watts



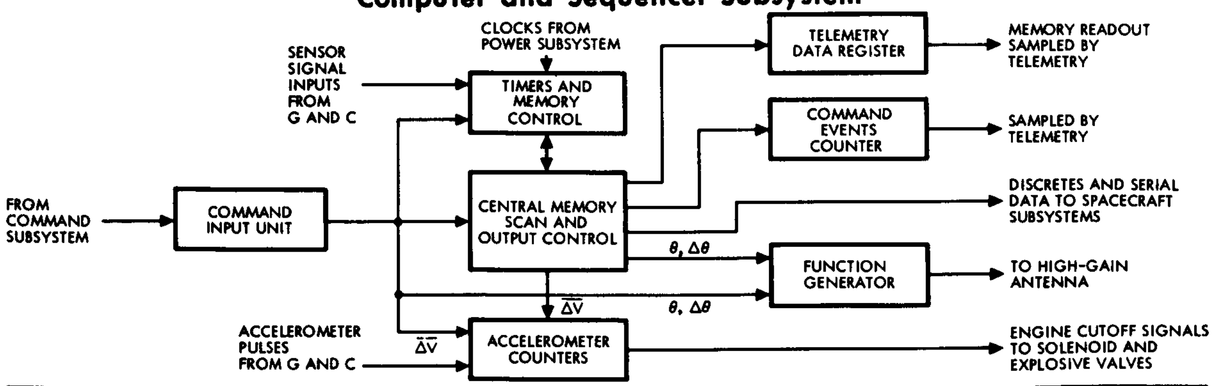


of stored command sequences, although all of the operations can be modified by ground command. In particular, specialized science sequences can be commanded from the earth to adapt the sequence to each orbit. For reliability a completely redundant and fully programmed sequencer is included. The computer and sequencer subsystem sends its commands directly to spacecraft subsystems, not through the command subsystem decoder.

In the S-band radio subsystem each of four antennas and diplexers is connected to separate phase-locked receivers which synchronously demodulate the uplink and provide a coherently related output carrier frequency to drive the downlink transmitters. The command data is biphase

PRELIMINARY SPECIFICATION

Computer and Sequencer Subsystem



Performance Characteristics

Output voltage:	10 volts
Output current:	25 MA
Pulse width:	50 MS

Physical Characteristics

	PRIMARY	BACKUP
Weight:	18 lb	16 lb
Power:	20 watts	18 watts
Volume:	460 in.^{3}	400 in.^{3}
Reliability:	0.966	
Memory size:	512 words	512 words
Word size:	20 bits	20 bits

SEQUENCING CAPABILITY

Up to 4 simultaneous sequences in operation

Capacity for 32 sequences

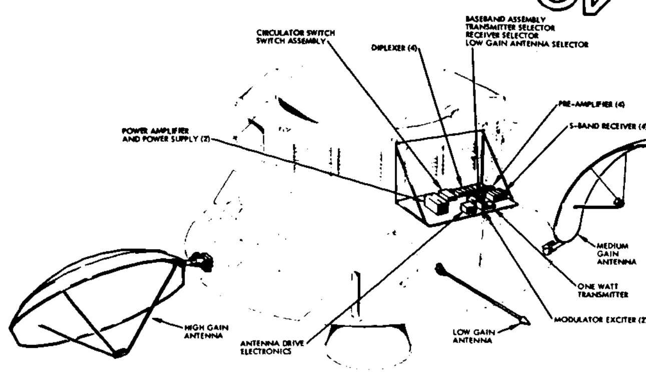
Sequences may be recycled from other sequences

Capacity for storage of command sequences for entire mission

Sequences alterable by ground command

Sequences synchronized to variety of physical events

Sequences may be enabled or inhibited by ground command modulated onto a 512-Hz subcarrier, while the ranging data is transferred to a modulation combiner where it is filtered and combined with the downlink data.



PRELIMINARY SPECIFICATION

S–Band Radio Subsystem

Purpose

The S-band radio subsystem provides reception and demodulation of command and ranging data from the DSN ground stations and transmission of telemetry data and ranging from the spacecraft to the ground station. The subsystem also provides for a doppler frequency shift measurement by transmitting a carrier frequency coherently related to the received uplink frequency.

Physical Characteristics

Weight:

S-band electronics	68.6 pounds
Antennas and drives	135.6 pounds

Power:

S-band electronics	180 watts max
Antennas and drives	60 watts peak

Performance Characteristics

RECEPTION:

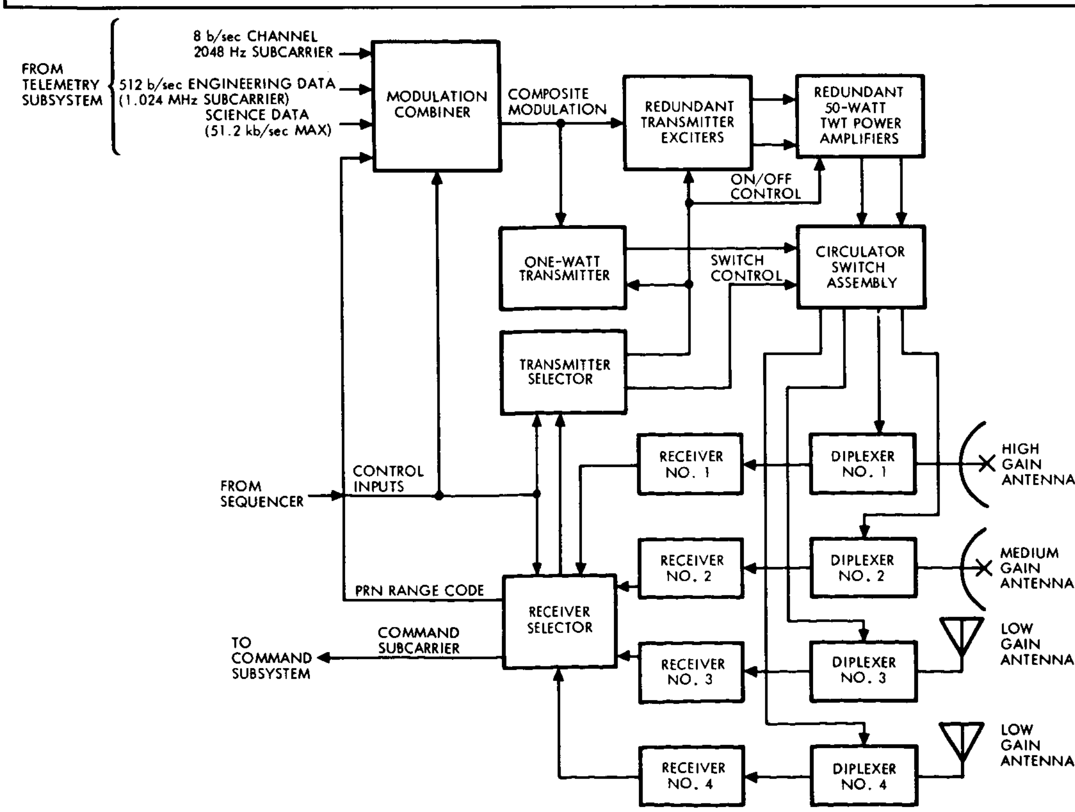
- Number of channels: Four (2 low gain, 1 medium gain and 1 high gain)
- Frequency: 2110 - 2120 MHz
- Receiver sensitivity: -149 dbm
- Noise figure: 5 db maximum
- Phase lock bandwidth: 32 Hz
- IF Bandwidth
  - wideband channel: 2.5 MHz
  - narrowband channel: 1 KHz
- Receiver dynamic range: -149 to -50 dbm

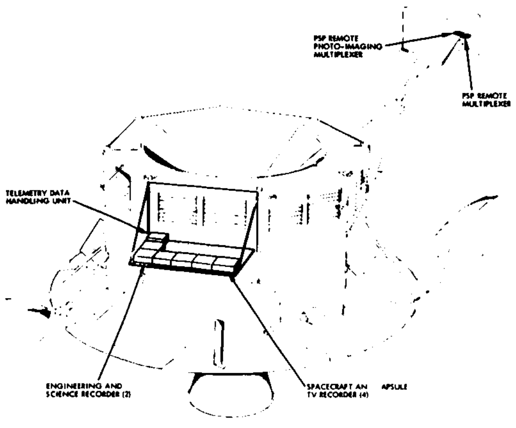
TRANSMISSION:

- Number of transmission channels: three (1 low gain, 1 medium gain and 1 high gain)
- Frequency: 2290 to 2300 MHz (coherent with uplink or optional internal crystal control)
- High power transmitter output: 50 watts minimum
- Low power transmitter output: 1 watt minimum
- Transmission bandwidth: 3 MHz

ANTENNAS:

- High Gain
  - Type: double gimballed circular paraboloid
  - Gain: 34 db
  - 3 db beamwidth: 3.2 degrees
- Medium Gain
  - Type: single gimballed elliptical paraboloid
  - Gain: 28 db
  - 3 db beamwidth: 4 degrees by 10 degrees
- Low Gain
  - Type: four arm conical spiral
  - Gain: +1 db
  - 3 db beamwidth: 180° minimum





The link supports 8 bits/sec command data through the omnidirectional antennas with adequate margin during the entire mission. The command link is an adaptation of the proven Pioneer system offering simpler hardware, shorter acquisition times, and higher bit

PRELIMINARY SPECIFICATION

Telemetry and Data Storage Subsystem

Purpose

Formats and codes data from the science payload and measuring points in the spacecraft for digital transmission by the S-band communications system. Store this data during critical operations or when data rate is higher than communication link can handle.

Performance Characteristics

BIT RATE

(Main Carrier)
Split phase — 51.2 kb/sec, 25.6 kb/sec, 12.8 kb/sec, 6.4 kb/sec, 3.2 kb/sec

(Subcarrier)
Normal — 512 b/sec
Emergency — 8 b/sec

ENCODING ACCURACY

Science — 8 bits
Engineering — 6 bits
Video — 6 bits

DATA FORMATS

Programmable for engineering and science
Fixed frame size for video

TYPES OF DATA INPUT

Analog (0 to 5 volts)
Discrete
Digitals

OUTPUT

Biorthogonal block code
Modulation
Biphase (Subcarrier)
Split phase (Main Carrier)

STORAGE

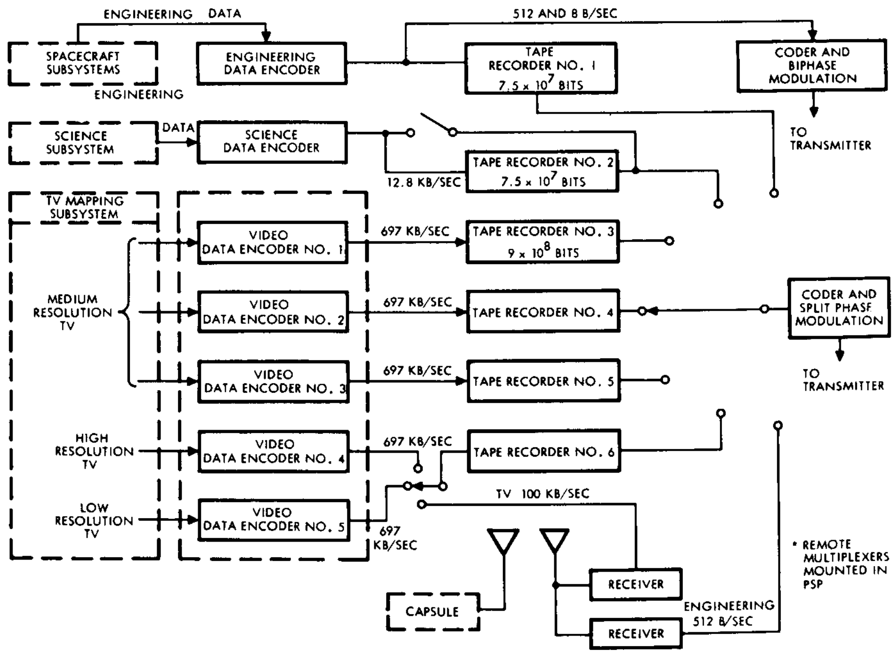
Video data recorders 3.6×10^9 bits (4 recorders)
Engineering and science recorders 1.5×10^8 bits (2 recorders)

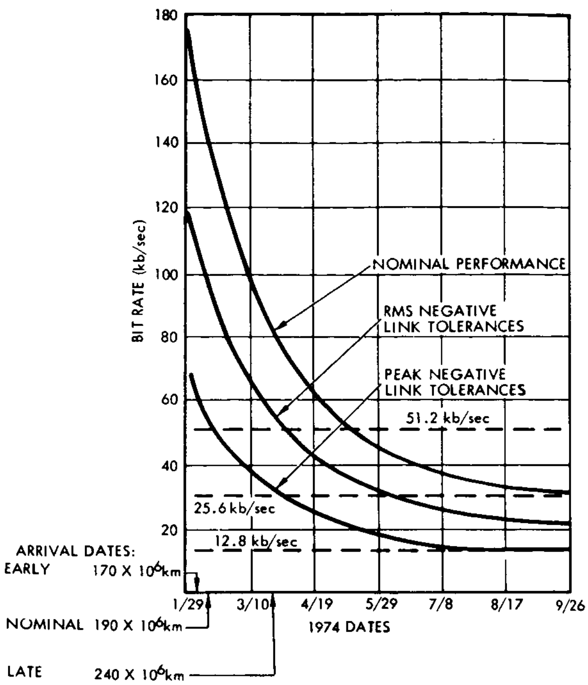
SIZE, WEIGHT AND POWER (EXCLUDING TAPE RECORDERS)

Central telemetry data handling unit
Size 480 in.^{3} (10 x 8 x 6 in.)
Weight 11 lb
Power 6 watts

Remote science multiplexer (located on PSP)
Size 20 in.^{3} (4 x 5 x 1 in.)
Weight 2 lb
Power 500 mw

Remote photo-imaging multiplexer (located on PSP)
Size 40 in.^{3} (4 x 5 x 2 in.)
Weight 2 lb
Power 2 watts

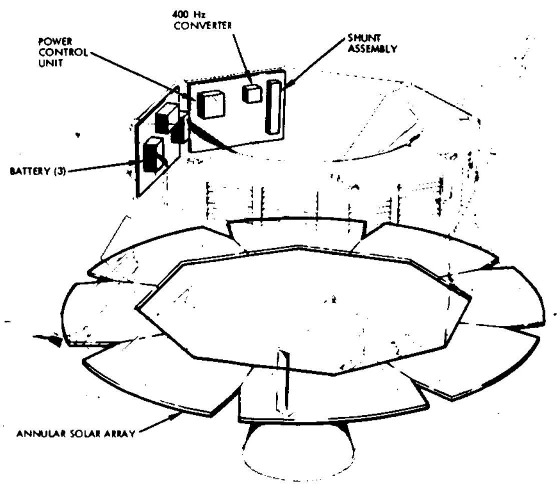




DATA RATE VERSUS TIME shows variation in data rate capability as a function of time for the 1973 mission. The three curves indicate the achievable data rates for: (1) nominal link performance, (2) the RMS of the negative link tolerances, and (3) the sum of the negative link tolerances. The horizontal lines indicate the available transmission rate as a function of time. The communication distances for an early, a nominal and a late arrival at Mars are shown on the abscissa.

rates than the older Mariner system. Any transmitter can be connected to any antenna through circulator switches, providing a versatile means for adapting to any component failure. The (32, 6) biorthogonal code adopted for communications is selected, over normal modulation, since it more than doubles the link capacity while still relying on proven components. As shown above, the communication link can support an average data rate in excess of 60, 50, 30 kbits/sec depending on range at encounter. The 8 bits/sec downlink is normally required only during reorientation maneuvers when the high- and medium-gain antennas do not point at earth. The link can also be used for low-rate engineering telemetry to ranges that encompass late encounter distances.

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 digital form, to be transmitted in real-time or stored on tape recorders for later playback. Tape storage is used as a buffer whenever data is generated faster than the communications rate. During descent of the capsule, its formatted TV and engineering data is received via the UHF link and either stored or directly retransmitted to earth. Programmable data format and seven data transmission rates are furnished. The versatile centralized telemetry data handling unit is supplemented by two multiplexers on the planetary scan



platform. The central unit is readily programmed for changing data requirements.

The maximum power demand for the recommended configuration is calculated to be 808 watts, in Martian orbit at 1.62 AU. At that time the solar array can deliver 836 watts, as shown on page 23. A power control unit, specifically adapted to the Voyager missions, regulates solar array and battery power. The control unit boosts line voltage early in the mission and reduces it slightly later on so that the array operates near maximum efficiency during the critical phases of the mission, at the greatest ranges from the sun. Conversion losses are accepted when greater power is available near earth, in trade for the greatest efficiency at Mars. Nickel-cadmium batteries are also suited to the Voyager mission because of their greater ability to recover from the discharge-charge cycles encountered during Martian eclipse seasons.

PRELIMINARY SPECIFICATION

Electrical Power Subsystem

Performance Characteristics

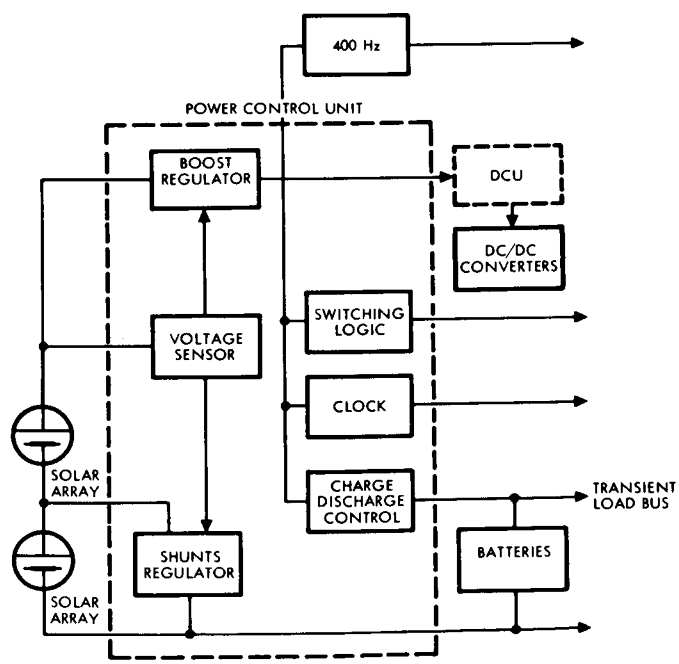
Main Bus Voltage	37 to 50 volts DC
Solar Array Power	1.67 au 787 w 1.62 au 836 w 1.00 au 1426 w
Battery Energy	1968 watt-hours
400 Hz Two-Phase for Antenna and Planetary Scan Platform Drives	50 v RMS ± 2%
Synchronization signal (Clock)	819.2 kHz

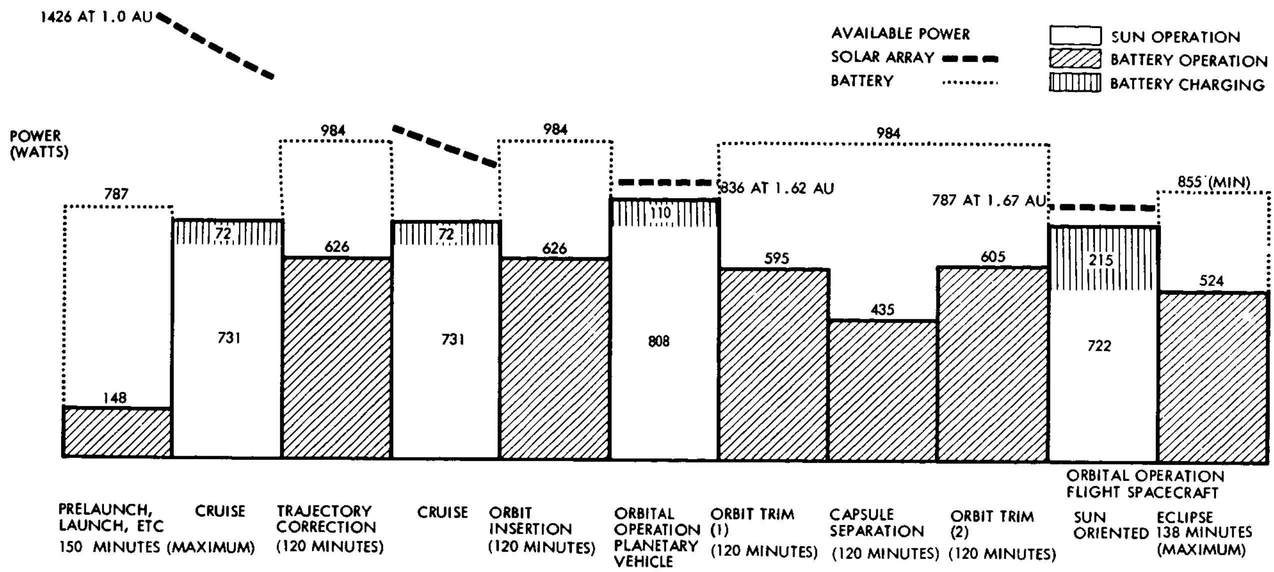
Physical Characteristics

Subsystem weight	443 pounds
Solar Array Area	226 square feet

Components

44,640 N-ON-P Solar Cells
Three 16 Ampere-Hour Nickel-Cadmium Batteries
Power Control Unit
400 Hz Inverter
DC-DC Converters

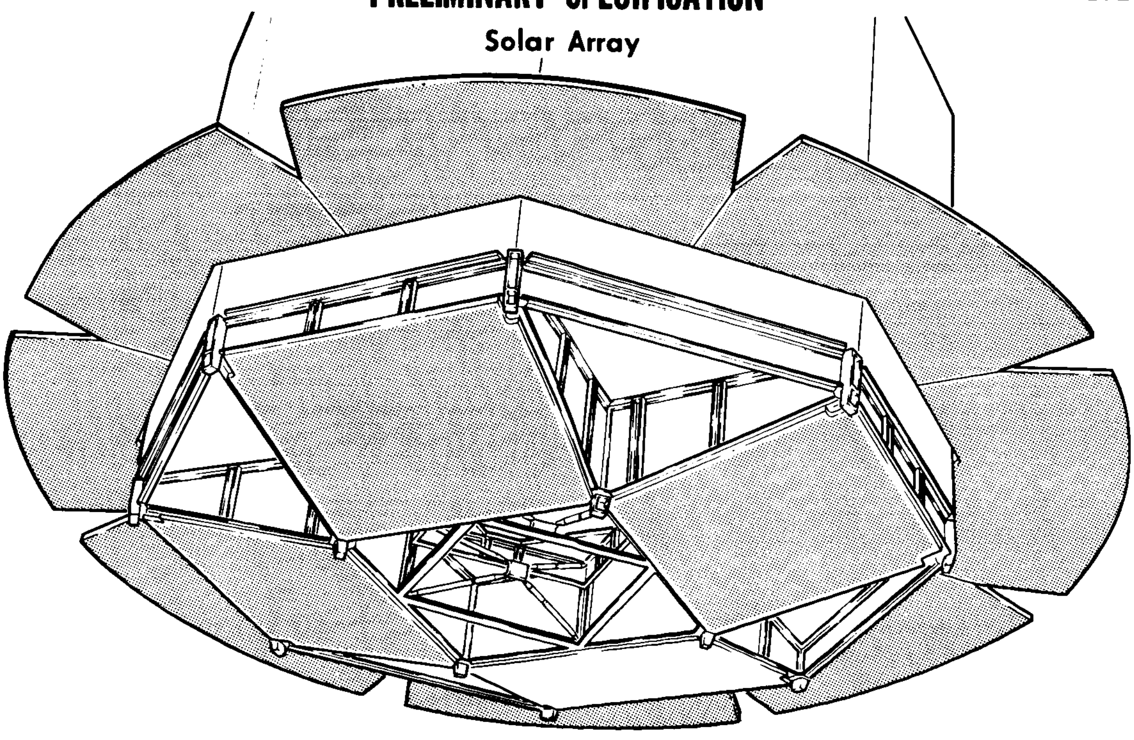




POWER REQUIREMENTS for operation of equipment and battery charging are matched by power available, with substantial margin, throughout the mission.

PRELIMINARY SPECIFICATION

Solar Array

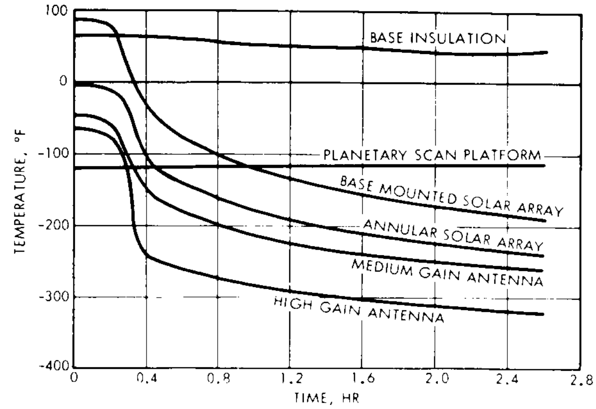


Purpose

The solar array fits well within the launch vehicle shroud and provides growth potential of about 300 watts at 1.62 AU by filling empty panel areas.

Performance Characteristics

Power output at 1 AU	1426 watts
Power output at 1.67 AU	787 watts
Total area	226 sq ft
Total weight	217 lb
Voltage at 1.67 AU	52.0 volts
Cell type	N- on-P, 2x2 cm 7 to 14 ohm-cm



THE COLDEST TEMPERATURES DURING THE VOYAGER MISSION occur at the end of the 2.6 hr Martian eclipse.

The coldest temperatures in the mission occur during solar eclipse at Mars, the longest of which is anticipated to last 2.6 hours. In eclipse for that long, the temperature of the most cold-sensitive elements remains within safe limits. Engine firing produces the highest temperatures; the first midcourse correction is the upper bound. Here again, all heat-sensitive components are kept well within their respective limits except for the base-mounted solar array, which briefly overheats 50°F above its nominal upper limit of 250°F. TRW's considerable experience with solar arrays has shown that such brief thermal excursions are harmless.

2.3 SCIENCE

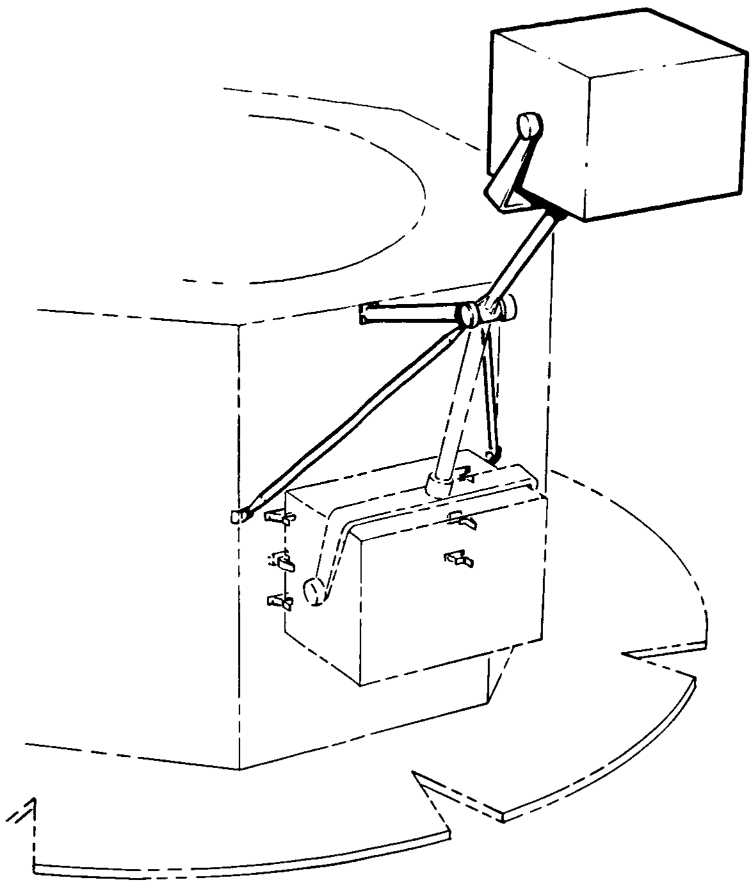
For the recommended spacecraft all of the sensors in the science payload are contained in the planetary scan platform. This platform is designed for flexibility, adapting to a theoretical 400-pound payload, the 500-pound hypothetical payload defined by MSFC guidelines, or even a larger payload carrying additional instruments. Moreover, the spacecraft can accommodate sensors mounted on the equipment module or on booms deployed from that module to provide additional flexibility in orientation and reduce the possibility of effects of the spacecraft on sensors.

The planetary scan platform can be controlled in two basic modes: a closed loop mode controlled by error signals from the Mars tracker, and an open loop mode commanded by the spacecraft sequencer. The error analysis (discussed in Volume 5) indicates that the platform can be pointed with an accuracy of ± 0.45 degree during open-loop.

Electrical cabling between the instruments and their supporting equipment in the spacecraft is reduced by two multiplexers in the planetary scan platform. One multiplexer, under the control of the flexible

PRELIMINARY SPECIFICATION

Hypothetical Science Payload



STRUCTURE

Weight: 183 lb

Strength: Designed to withstand Mars orbit insertion G loads with PSP deployed

Front face area: 45 in. x 37 in.

Volume: 34 cu ft

Deployment: electric hinge axis drive, explosive release from stowed position

THERMAL CONTROL

Insulation: Foam within meteroid protection honeycomb panels

Thermostats and heaters: For experiments and drive mechanisms

Coatings: Black Cat-a-lac paint except white IIT RI Z-93 on cooling radiators

Radiation: 3/16 aluminum plates for IR Spectrometers

Cyrogenic refrigerator: For broadband IR Spectrometer

POINTING AND CONTROL

Pointing accuracy: 0.3° (Mars sensors – closed loop mode)

Control rates: ±10 mr sec ±0.17 mr sec

Reliability: 0.955

EXPERIMENTS

High resolution IR Spectrometer

Broadband IR Spectrometer

IR Radiometer

UV Spectrometer

Photo-imaging System

INTERFACES

Spacecraft structure: 3 support struts

Data handling: Through remote multiplexers in PSP

Computer and sequencer: Through remote decoders in PSP

Power: Through remote switching assembly in Planetary Scan Platform

34.6 watts for noneclipse orbit

55.43 watts for eclipse orbit

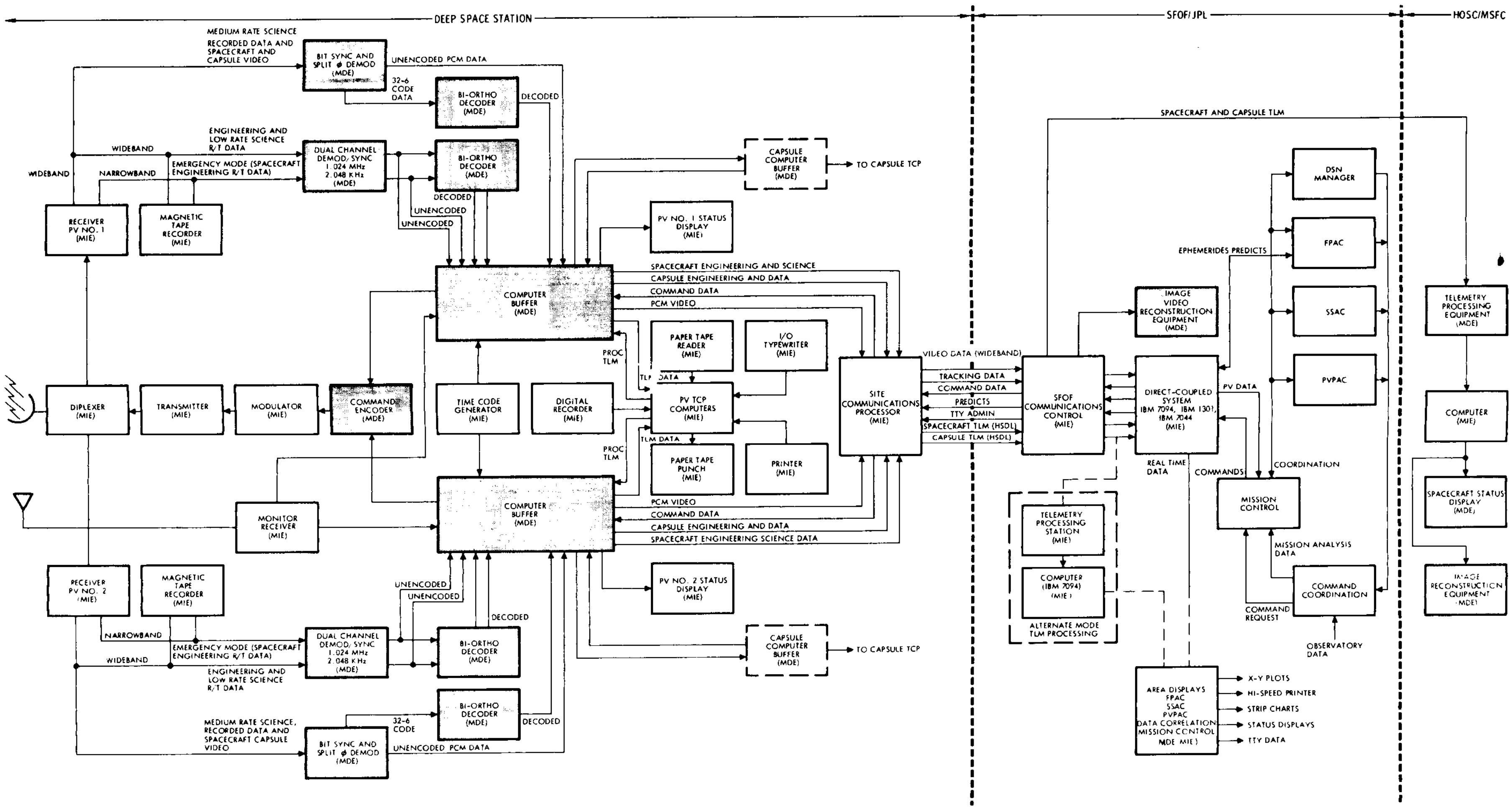
format generator, samples science data. The other samples and formats photo-imaging data.

The sensor for tracking Mars is a modified version of the horizon sensor used on the Orbiting Geophysical Observatories, with a spectral passband at 18 to 35 microns. Three drives on the shaft of the platform provide three degrees of freedom, although in one plane gimballing is limited by the body of the spacecraft to an arc somewhat less than 180 degrees. One drive is used for initial deployment and insofar as possible to allow positioning the main shaft perpendicular to the plane of the orbit. The other two orthogonal gimbal drives point the sensors at the planet, one drive rotating the shaft axis and the other the yoke axis. The same wabble gear drive is used on the shaft axes of the planetary platform as on the antennas to provide the developmental and testing benefits possible with commonality of equipment.

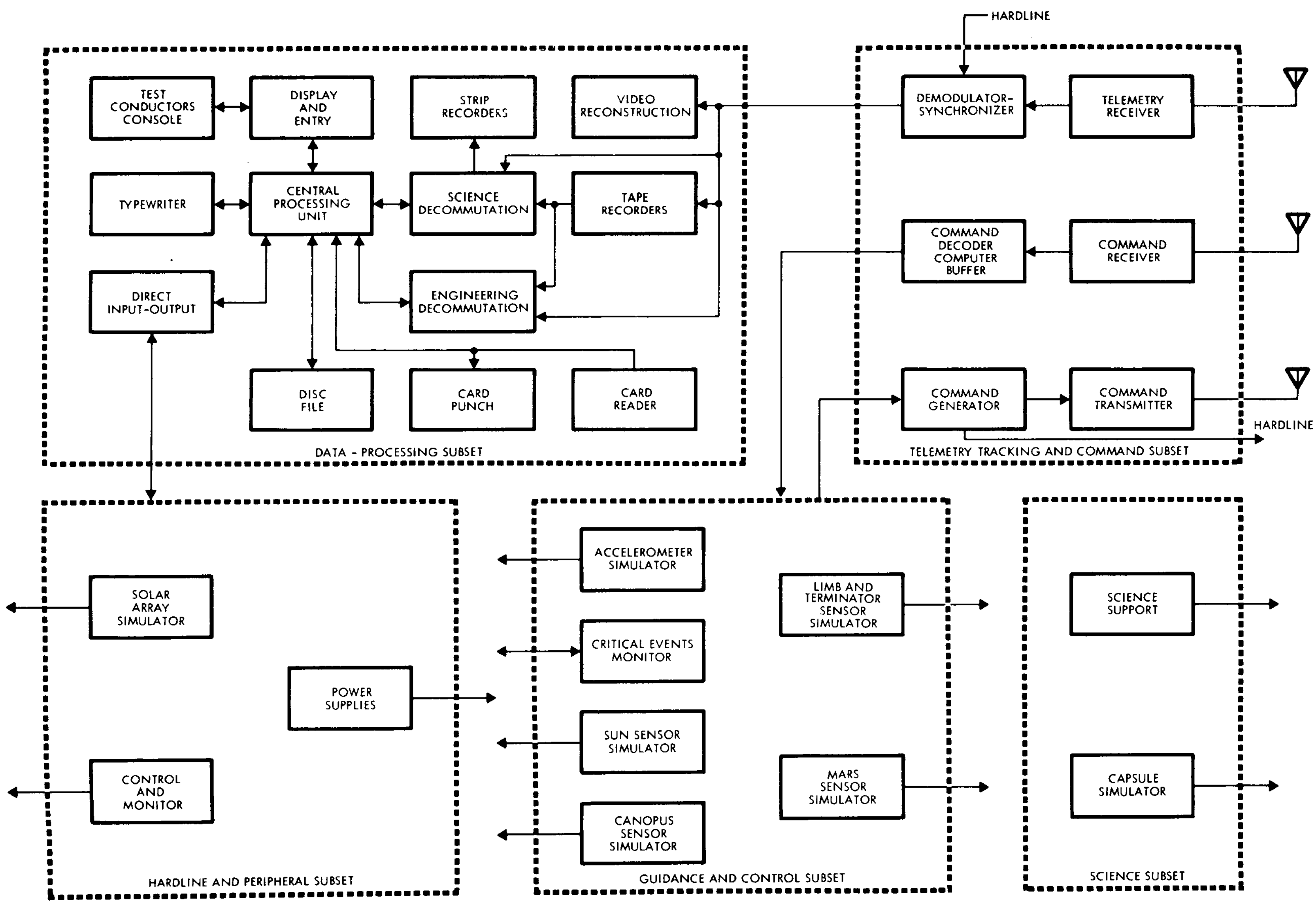
The platform and all supporting equipment within the spacecraft have been designed in the expectation that the experiments carried will not necessarily be identical to the payload considered here. To assure flexibility a variety of other experiments has been evaluated, conceived as mixed with the hypothetical set or as substitute experiments. The adaptability of the spacecraft to nine other instruments is discussed in Volume 5.

2.4 GROUND EQUIPMENT

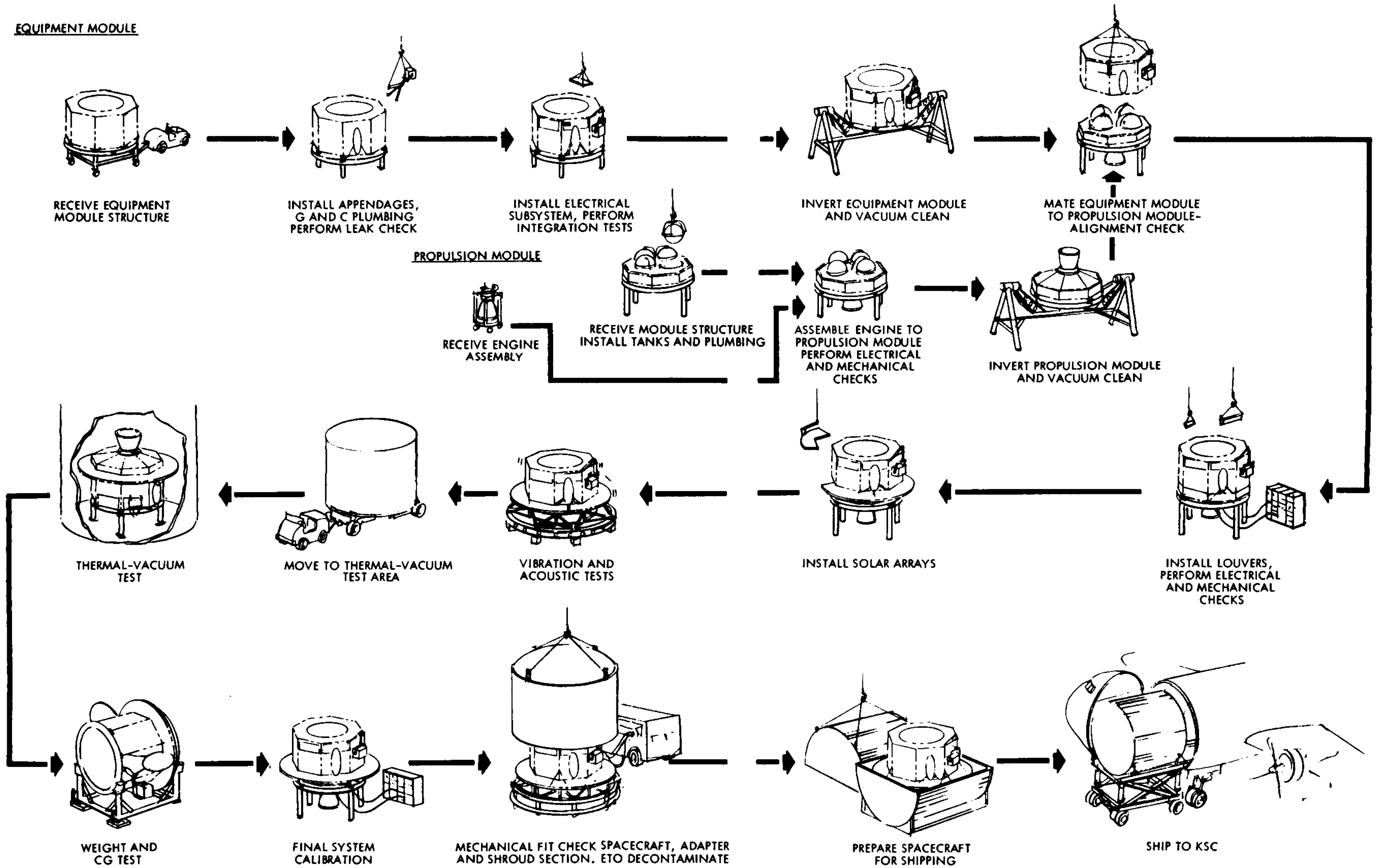
Three types of ground equipment have been designed as an integral part of the spacecraft system design process: the mission-dependent equipment, the electrical, and the mechanical operational support equipment. By defining in precise terms the substantial overlap between the requirements of the ground network supporting Voyager and the electrical checkout gear, six major items have been identified as common to the two uses: the command generator, the power supplies, demodulator-synchronizer, biorthogonal decoder, data format generator, and the video reconstruction equipment. As the design of the spacecraft system is refined, it is possible that additional items of mission-dependent equipment can be defined as identical to those in the system test complex. It also appears desirable to have commonality or at least program compatibility



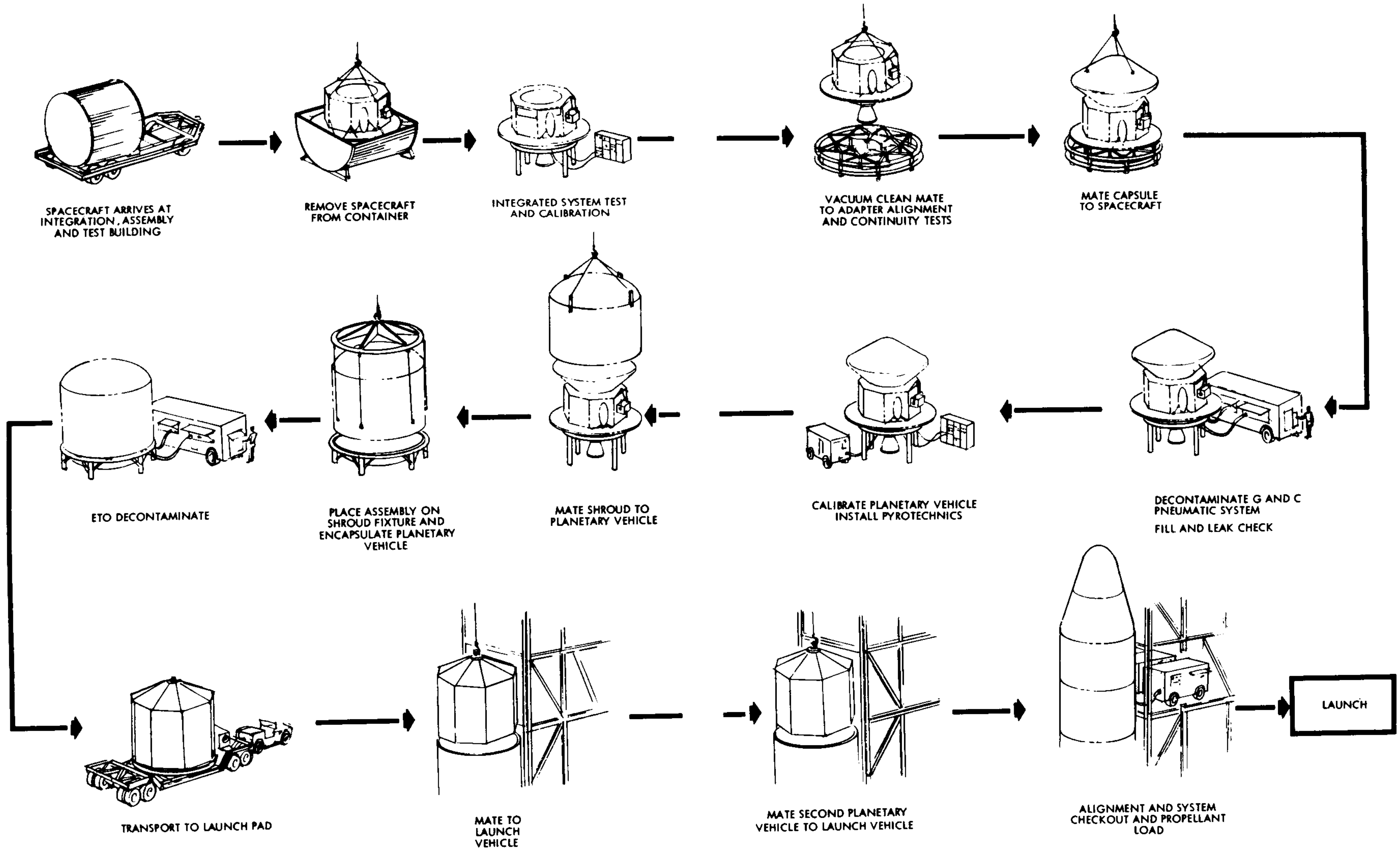
VOYAGER'S MISSION-DEPENDENT EQUIPMENT includes bit synchronizer and split-phase demodulator, dual-channel demodulator-synchronizer, biorthogonal decoder, computer buffer, and command encoder at the various Deep Space Stations; video reconstruction equipment at the Huntsville Operations Support Center; and telemetry processing equipment at the Huntsville center.



THE SYSTEM TEST COMPLEX is divided into five subsets for ease of location and cabling. Hardline connections are with the spacecraft, but system tests can be conducted entirely over the RF link.



MECHANICAL SUPPORT EQUIPMENT during assembly and test at the spacecraft fabrication facility is needed principally for mobility and support during vibration, thermal-vacuum, and weight tests.



AT THE LAUNCH SITE mechanical support equipment is needed for assembly and test operations and decontamination cycles if these prove necessary.

between the computer in the system test complex and the computers in the Deep Space Network.

The design assumes that the mission-independent equipment at the ground stations and the equipment for the Voyager capsule and surface laboratory are shared, constituting an integrated program capability. The spacecraft mission-dependent equipment is sized for simultaneous operation of two planetary vehicles, providing a ground operational system capable of supporting all of the tracking, telemetry, data processing, and display and command generation and verification requirements. The plan for incorporating Voyager-peculiar equipment into the ground systems is shown on page 27.

The mission-dependent computer programs are designed to permit efficient monitoring of the performance of the spacecraft; processing and correlating of information for engineering evaluations and operational decisions; evaluating the implications of alternate courses of action; and implementing the selected course of action. The programs, defined in Volume 7, are also designed to be utilized for preflight mission planning and feasibility studies, for training, and for operational readiness tests and system checks.

The support equipment for system level testing of the spacecraft is embodied in the system test complex, a simplified block diagram of which is presented on page 28. As discussed in Volume 7, the identification and design of the mechanical support equipment has occurred in conjunction with the analysis of logistics for the Voyager spacecraft system. All major items for mechanical support have been defined; their application is exemplified on pages 29 and 30.

2.5 DESIGN AND OPERATIONAL ALTERNATIVES

In reaching decisions concerning both the general approach to the recommended configuration and the specific components and assemblies incorporated in the subsystems, tradeoff evaluations of competitive alternatives have been completed. The rationale behind the choices in the recommended design is presented throughout the report where the specific choice is defined, and Volume 6 evaluates the alternate approaches that have been studied for the overall configuration. The means for adapting to an altered mission profile are also shown.

Beyond these, however, the study has brought into focus four major design alternatives which clearly warrant further study. It is concluded that a photo-imaging system modeled after that of the Lunar Orbiter has advantages over the all-television system, advantages both in planetary coverage and in data storage requirements. This alternative is discussed broadly in Volume 5 and in detail in Volume 11. It appears further that the operating alternatives of the C-1 engine cluster either as a redundant propulsion subsystem or as a means of enhancing the versatility of that subsystem need additional study. Additional study is needed to evaluate the level of pressurization maintained in the propellant tanks as a factor in the possible violation of the Martian quarantine through meteoroid-induced explosion of the tanks. Finally, it is concluded that a centralized, general-purpose computer in the spacecraft has potentially far-reaching implications on mission success. The operational capabilities of the spacecraft and the potential for enhancing the inherent flexibility of the mission by means of such a computer warrant thorough study. In Volume 4 is presented the first step toward evaluating the implementation of a computer, the investigation of digital control systems.

The Agena and Transtage engines have been evaluated for the Voyager spacecraft with respect to mechanical fit. To that end the Agena engine (Bell Model 8533) and Transtage engine (Aerojet Model AJ 10-138) were reviewed with respect to the following:

- Mechanical interface
- Clearance of engine gimbal and head end assembly
- Propellant line interface and line routing
- Structure to distribute loads from engine to spacecraft
- Fit within the sterilization canister
- Electrical interface
- Thermal control of engine

It is shown that the Agena will fit within the spacecraft. The only modification is to the spacecraft truss members that distribute the load from the engine interface to the main module structural members. To keep the gimbal location the same as that of the LMDE, minor repackaging of the Agena accessories is required around the chamber. The Transtage layout study indicates that it can also fit within the Voyager spacecraft. As with the Agena, the only modification is to the truss members that distribute the loads from the engine to the spacecraft structural members. The gimbal is located 10 inches aft of that of the LMDE, to permit the most efficient engine supporting structure, but remains within the envelope of the spacecraft and sterilization canister.

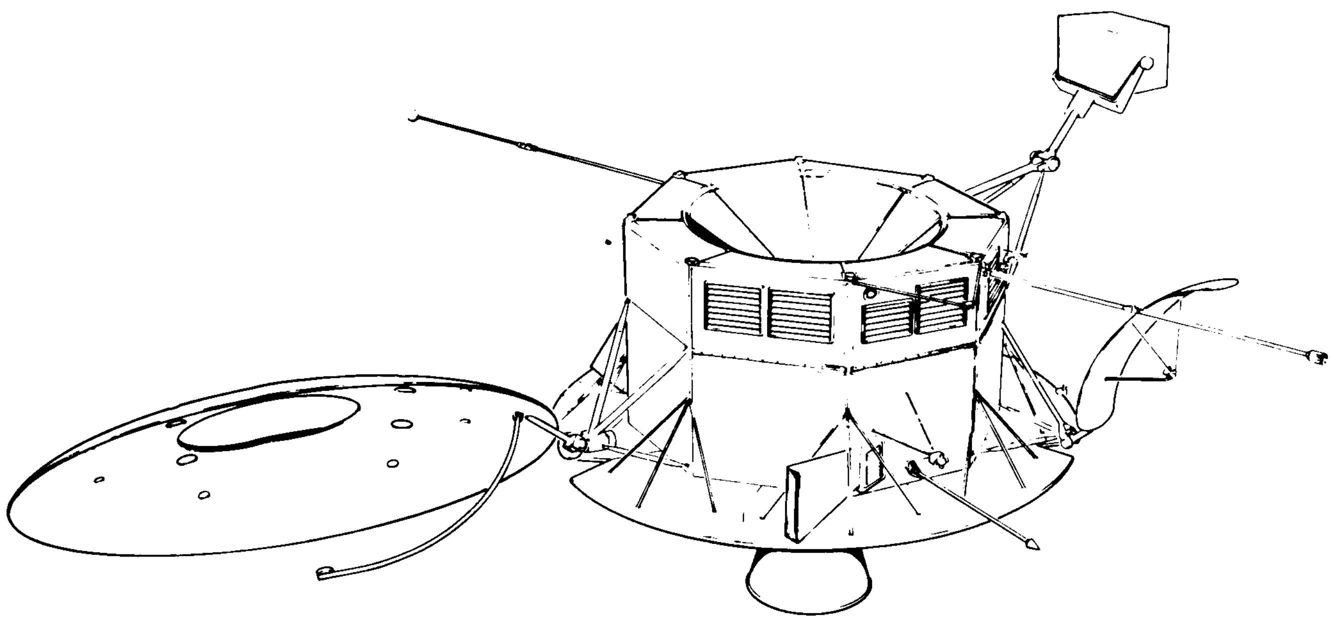
2.6 ADAPTABILITY AND GROWTH POTENTIAL

One of the principal criteria applied in designing the spacecraft for the 1973 mission has been its adaptability for other missions and its potential for growth to larger payloads and more demanding performance. As discussed in Volume 6, the recommended configuration is adaptable without change to capsule weights of 5000, 6000, and 7000 pounds and to the energy requirements of the launch opportunities through 1979. It can equally well carry a complement of experiments calling for sensors mounted on booms or the surface of the spacecraft as well as in the planetary scan platform and can mount a subsatellite for particles and fields measurements.

The potential for growth in the configuration has been evaluated by posing an upgraded spacecraft capable of carrying up to an 8000-pound capsule and 600 or more pounds of science payload, as outlined on the following two pages. The amendments to meet these added requirements, while retaining the adaptability for different mission years, proved to be relatively minor. The additional electronic equipment is mounted on a panel in the equipment module which is empty in the recommended configuration. Solar-cell panels are added to the four triangular surfaces adjacent to the areas on which panels are mounted on the recommended configuration.

To obtain higher data rates, the 9.5-foot antenna is replaced by a parabolic reflector with a 238-inch diameter, with a hole cut from its center to permit stowage without interference with the engine nozzle, and 100-watt traveling wave tube amplifiers are substituted for the 50-watt amplifiers. Since the narrower beam of the enlarged antenna

PRELIMINARY
Upgraded



Physical Characteristics

Structure: 3 modules: equipment module, propulsion module, and planetary vehicle adapter (not shown). 7075 aluminum framing and honeycomb deck; semimonocoque; meteroid panels.

Size: (Folded) 14 feet, 1 inch long x 19 feet, 10 inches wide
(Deployed) 21 feet, long x 43 feet, 4 inches wide

Science Payload: 600 lb (in PSP, on booms, and body-mounted)

Capsule Capacity: Accommodates 8,000 lb for all missions, 1973 through 1979

Hardware and Tankage: Identical for all 1973-1979 missions

Appendage Deployment: PSP high and medium gain antennas deployed by wobble gear mechanisms

System Reliability: 0.66 for 6800-hr mission (224 days interplanetary, 60 days Mars orbit)

Weight: Weight: (8000 lb capsule)

Equipment module	3,095.6 lb
Propulsion module	18,330.4
Planetary vehicle adapter	685.5
Total	30,761.5 lb

SPECIFICATION

Spacecraft

Performance	Characteristics
TELEMETRY AND DATA STORAGE	
2 tape recorders store 7.5×10^6 each Transmitted in 16/5-bit biorthogonal code Video transmitted adaptively between 307.2 and 19.2 kb/sec Engineering data at 512 bits/sec continuous	Central telemetry data handling unit plus two remote diplexers on PSP and one on Equipment Module. Flexible format generator selects programmed formats or programs new formats.
S-BAND COMMUNICATIONS	
High gain antenna: 39.9 db Medium gain antenna: 28 db	All digital communications. Redundant 100 watt TWT's. Four receivers, four antennas: high gain (20 ft dia), medium gain; two low gain giving complete spherical coverage.
COMMAND	
Rate: 8 bits/sec. Accepts up to 256 commands, processes serial data for 11 on-board destinations Acquisition time: 2 min	Redundant bit synchronizers and decoders demodulate and synchronize commands from S-band radio.
GUIDANCE AND CONTROL	
Limit Cycle: ±0.2 axis (cruise and orbit) ±0.43 degree per axis (maneuvering thrust vector) ±0.20 degree per axis (photo-imaging)	Fully-stabilized, 3-axis control; nitrogen heated gas; sun and Canopus references; redundant inertial reference
COMPUTER AND SEQUENCER	
Issues discrete commands and serial messages for automatic on-board control. Computes pointing angles for high gain antenna and PSP. Integrates velocity meter outputs during engine burn.	Primary sequencer plus velocity meter counters and function generators, with decoder in PSP. Backup sequencer.
ELECTRIC POWER	
300 sq ft array generates 1099 watts at 1.62 AU. 37 to 50 VDC (unregulated) to individual DC/DC converters. 400 Hz to appendage drives; 4.1952 MHz sync signal to subsystems	300 sq ft fixed solar array; three NiCd batteries
ELECTRICAL DISTRIBUTION AND PYROTECHNIC CONTROL	
Distribution control unit feeds DC power to all subsystems. Pyrotechnic control unit operates all electro-explosives	All harnesses removable Flexible harness to experiment booms
PROPULSION	
LMDE: Total impulse: 4.82×10^6 lb-sec Thrust level: 9850 lb (high), 1700 lb (low) Fuel: N_2O_4 and UDMH C-1: Specific impulse: 292 sec Thrust level: 100 lb ea Fuel: N_2O_4 and UDMH	LMD main engine with four C-1 engine backups. Tankage identical for all missions.
TEMPERATURE CONTROL	
Solar Array Temperatures (Average), Near Earth °F / Near Mars °F: Annular array: 125 / -7 Base array: 246 / 86 Equipment Panel: 90 / 50 PSP Interior Temp: 84 / 35	Multilayer insulation; individually actuated louvers, special finishes

calls for tighter limit cycles on attitude control, to keep the earth always within the beam, increased consumption of reaction control gas is required. The only changes to the thermal subsystem are increases in the thermal radiation areas and the size of the thermal louvers; the total insulated area decreases from 494 to 471 square feet, the total area for louver-controlled radiation increases from 35 to 46 square feet, and the average power for heaters increases from 21 to 33 watts.

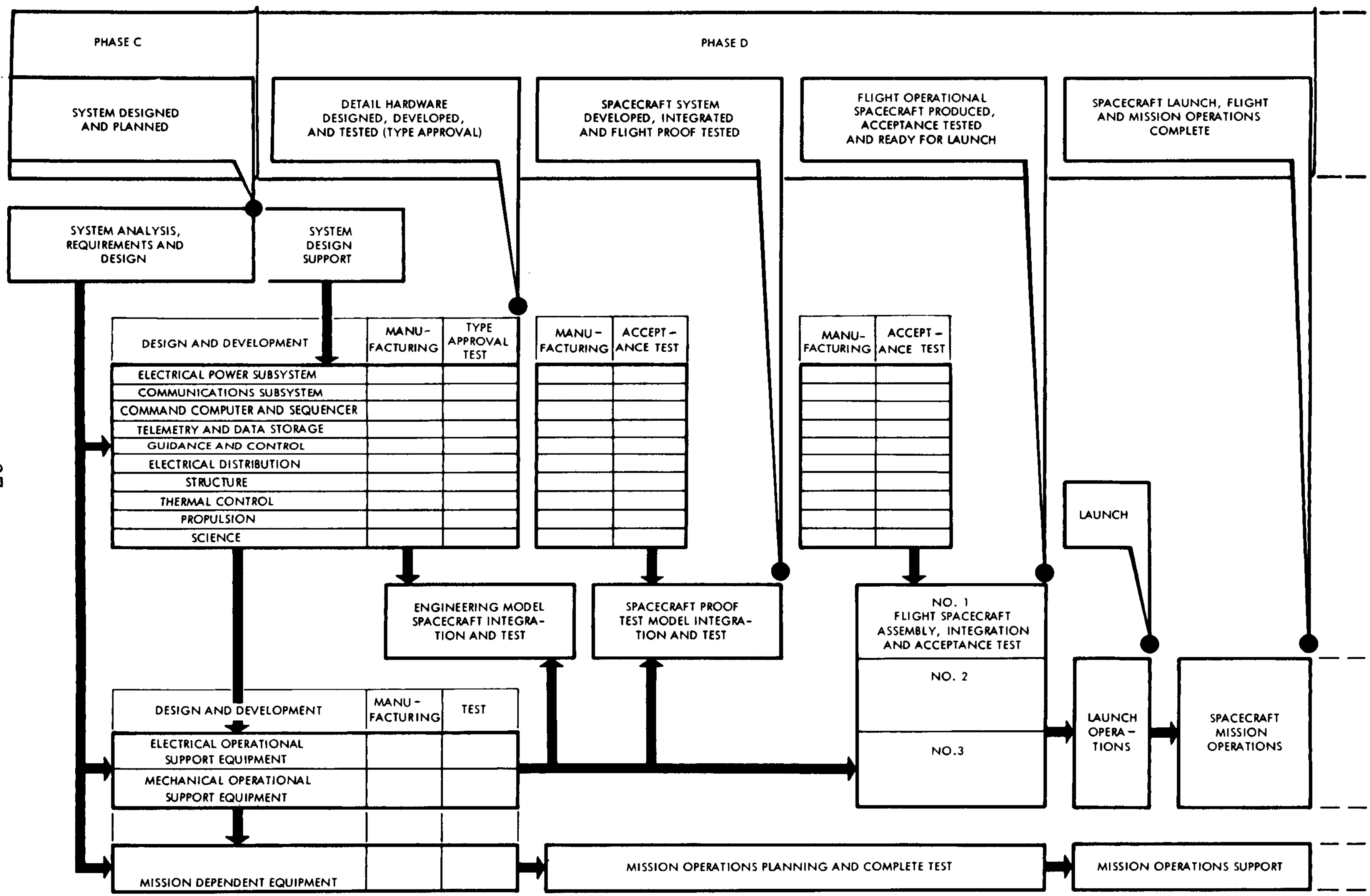
The effects on system reliability of these changes are negligible except for the fact that the greatly increased information rate, approximately 300 kbits/sec, makes it no longer practical to conceive of the medium-gain antenna as backup for the high-gain.

3. IMPLEMENTATION

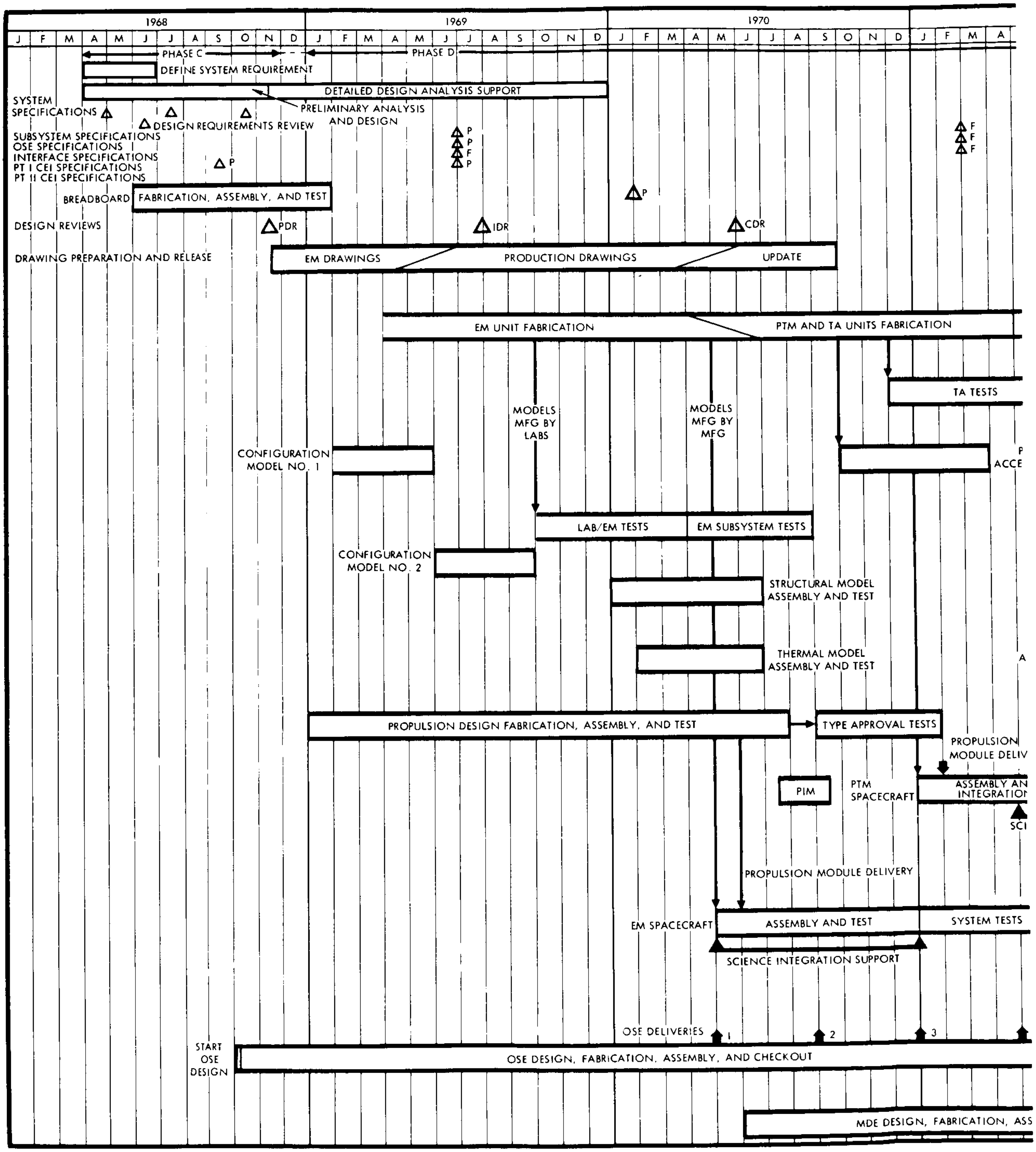
In general, the approach to the implementing activities for the spacecraft is to arrange them so that the adequacy of the design is confirmed as early as possible and reasonable time is allowed for unpredicted development problems. The relationship of the major activities is summarized on the next page, and their schedule is shown on the following pages. Design reviews, system type approval, and mission acceptance review are anchor points in this schedule, as well as the launch date. Key features in the scheduling are:

- Early design data from development test is obtained by completing environmental tests and integrating the engineering model units before final drawing release.
- Early reliability data is obtained from engineering model and type approval tests before proof test model testing. In addition, spacecraft life testing is conducted first on the engineering model spacecraft and subsequently on the proof test model.
- Final design is verified by proof test model tests six months before flight-article spacecraft are committed to environmental tests.

The success of the project implementation described in Volume 8 rests strongly on the carefully planned and intelligent use of engineering



SUMMARY FLOW OF MAJOR IMPLEMENTATION ACTIVITIES shows concurrent subsystem development, followed by sequential activities phased to verify progress before committing to next step.



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1969
1970
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PHASE D
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DETAILED DESIGN ANALYSIS SUPPORT
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PRELIMINARY ANALYSIS AND DESIGN
DESIGN REQUIREMENTS REVIEW
SUBSYSTEM SPECIFICATIONS
OSE SPECIFICATIONS
INTERFACE SPECIFICATIONS
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PT II CEI SPECIFICATIONS
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F
BREADBOARD
FABRICATION, ASSEMBLY, AND TEST
DESIGN REVIEWS
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MODELS MFG BY MFG
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LAB/EM TESTS
EM SUBSYSTEM TESTS
CONFIGURATION MODEL NO. 2
STRUCTURAL MODEL ASSEMBLY AND TEST
THERMAL MODEL ASSEMBLY AND TEST
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TYPE APPROVAL TESTS
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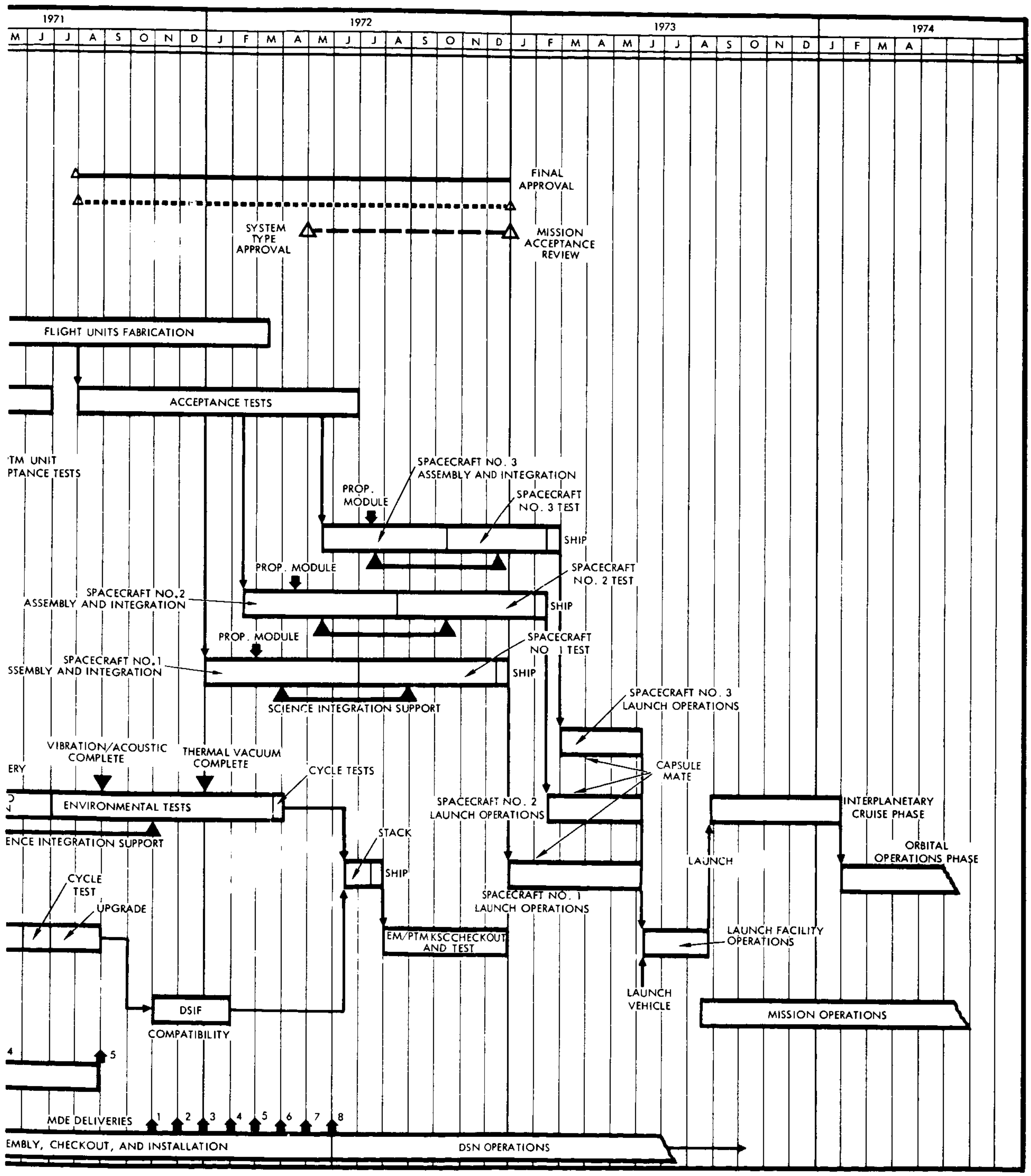
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FINAL APPROVAL
SYSTEM TYPE APPROVAL
MISSION ACCEPTANCE REVIEW
FLIGHT UNITS FABRICATION
ACCEPTANCE TESTS
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SPACECRAFT NO. 3 ASSEMBLY AND INTEGRATION
PROP. MODULE
SPACECRAFT NO. 3 TEST
SHIP
PROP. MODULE
SPACECRAFT NO. 2 TEST
SPACECRAFT NO.2 ASSEMBLY AND INTEGRATION
SHIP
PROP. MODULE
SPACECRAFT NO. 1 TEST
SPACECRAFT NO.1 SSEMBLY AND INTEGRATION
SHIP
SCIENCE INTEGRATION SUPPORT
SPACECRAFT NO. 3 LAUNCH OPERATIONS
VIBRATION/ACOUSTIC COMPLETE
THERMAL VACUUM COMPLETE
CYCLE TESTS
ERY
CAPSULE MATE
ENVIRONMENTAL TESTS
SPACECRAFT NO. 2 LAUNCH OPERATIONS
INTERPLANETARY CRUISE PHASE
ENCE INTEGRATION SUPPORT
STACK
LAUNCH
ORBITAL OPERATIONS PHASE
SHIP
CYCLE TEST
UPGRADE
SPACECRAFT NO. 1 LAUNCH OPERATIONS
EM/PTM KSC CHECKOUT AND TEST
LAUNCH FACILITY OPERATIONS
LAUNCH VEHICLE
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COMPATIBILITY
MISSION OPERATIONS
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5
MDE DELIVERIES
1 2 3 4 5 6 7 8
EMBLY, CHECKOUT, AND INSTALLATION
DSN OPERATIONS

models. In addition, the modularity of the configuration permits the flexibility for incorporating additional developmental time, through parallel activities, if it is needed. Treating the engineering models (structural, thermal, and full spacecraft engineering models) as a major step toward the flight model entails the application of configuration control disciplines as soon as laboratory fabrication of these models begins. The full approval cycles that are applied later are not applied at this stage of development, but drawings, test data, and other design information as it is revised is fed immediately into the configuration-control cycle so that all changes can be evaluated at the hardware level.

At this time, as well, the controlled engineering models are used with the initial developmental models of the ground support equipment so that the two proceed toward final hardware as a system and so that when flight models are checked the support equipment is already known to be satisfactory. Moreover, in this engineering-model period, decontamination processes and requirements are instituted at appropriate points so that by the time flight units are produced the ETO processes are smoothly coordinated.

Confidence in the schedule rests substantially on the fact that the project proceeds through parallel steps to its goals, up to assembly and test of flight hardware. The ability to continue to move forward while problems are uncovered and solved is thereby built into the project. Even when sequential steps are unavoidable, as in assembly and test of a flight spacecraft, work on the individual flight models is staggered and located in different areas of the assembly building so that isolated difficulties on one model can have no effect on the others.

Establishing and checking of mission-operations equipment, personnel, and procedures is scheduled to begin early in the project so that problems that may be uncovered in this area cannot jeopardize the first launch.

These elements of the implementation plan have been instituted and evaluated now to assure that the schedule is realistic. Specific decisions in many areas of the project need further study, however, although none of these are likely to affect schedules. Integration and checkout of the planetary vehicle is shown to occur at Kennedy Space Center, but it is possible that program benefits may accrue from scheduling this activity at the Huntsville facilities. The plan as now prepared is sufficiently versatile to be modified in many such ways without altering its validity or significantly changing the flow of events.

Assuming the configuration of this report, early development and test programs need to be undertaken with respect to the cooling equipment for a cryogenically-cooled IR spectrometer and the test procedures bearing on ETO decontamination. As the spacecraft development program proceeds subsystem design provides data for manufacturing planning, test planning, and the planning of spacecraft assembly and integration operations. Engineering model units, fabricated within the engineering laboratories, confirm the subsystem design. A second set of engineering models is assembled into a spacecraft by the manufacturing organization to confirm tooling and manufacturing processes. The final step in subsystem development is the manufacture of test articles to flight configuration for type approval test of the electrical units and the corresponding design verification testing and qualification of propulsion, mechanical, and structural hardware. This final step of type approval takes place before flight unit fabrication begins.

Flight configuration articles are manufactured and acceptance tested for integration into the proof test model spacecraft, before assembly of the three spacecraft. After flight acceptance test, the three spacecraft are delivered sequentially to the launch site for integration into the planetary vehicle. To support the mission operations eight sets of mission-dependent equipment are manufactured at a rate of approximately one set per month. Activation for Voyager of a station in the Deep Space Network begins nearly a year before the first launch.

4. ENGINEERING STUDIES

4.1 ETO DECONTAMINATION

If conservative assumptions are made, the current Martian quarantine requirement for the Voyager program appears to call for a limit of 100 viable microorganisms per square foot of spacecraft surface. To achieve that goal appears in turn to call for decontamination procedures based on the use of ethylene oxide-Freon 12 (ETO) as a biocidal agent. Such procedures would significantly affect the program, calling for relatively large increases in manpower, for longer schedules during fabrication and assembly to allow for training and the ETO cycles, and for certain facilities in the program that otherwise would not be needed.

Although relatively little test data is available on the compatibility of ETO with the specific materials and components planned for Voyager under the conditions applicable to Voyager, extrapolating the available data indicates that there should be no serious problem in material and component selection. In most cases, the use of sensitive materials and components can be avoided and where this is not possible units can be sealed to prevent contact of the sensitive component with ETO during terminal decontamination.

Materials and components for Voyager that have been found to be degraded by ETO have been identified and evaluated as follows:

Materials and Components of Potential Concern	Conclusions
Propellants	Sealed from contact with ETO
Lubricants	Alternates available for lubricants that are degraded
Wire insulation	Alternates available for insulation that is degraded
Conductive adhesive	Alternates available
Capacitors	Certain types of capacitors must be avoided
Magnetic tape	Sealed from contact
Photographic film	Sealed from contact

The fact that the majority of the materials and components expected to be used on Voyager have not yet been tested for compatibility with ETO in Voyager-defined environments need not burden the project excessively since qualification tests must be conducted for Voyager in any event.

Clean assembly is required to keep biological contamination of the spacecraft acceptably low. Such assembly represents a major facility requirement. Also required are ETO facilities for evaluation and test and terminal decontamination. Class 10,000 clean rooms appear to be desirable for all operations from unit assembly through terminal decontamination. The requirements do not exceed to any major extent what would normally be required for particulate contamination control. The principal equipment needed for terminal decontamination are a mobile ETO cart and an aseptic purge-gas cooling cart.

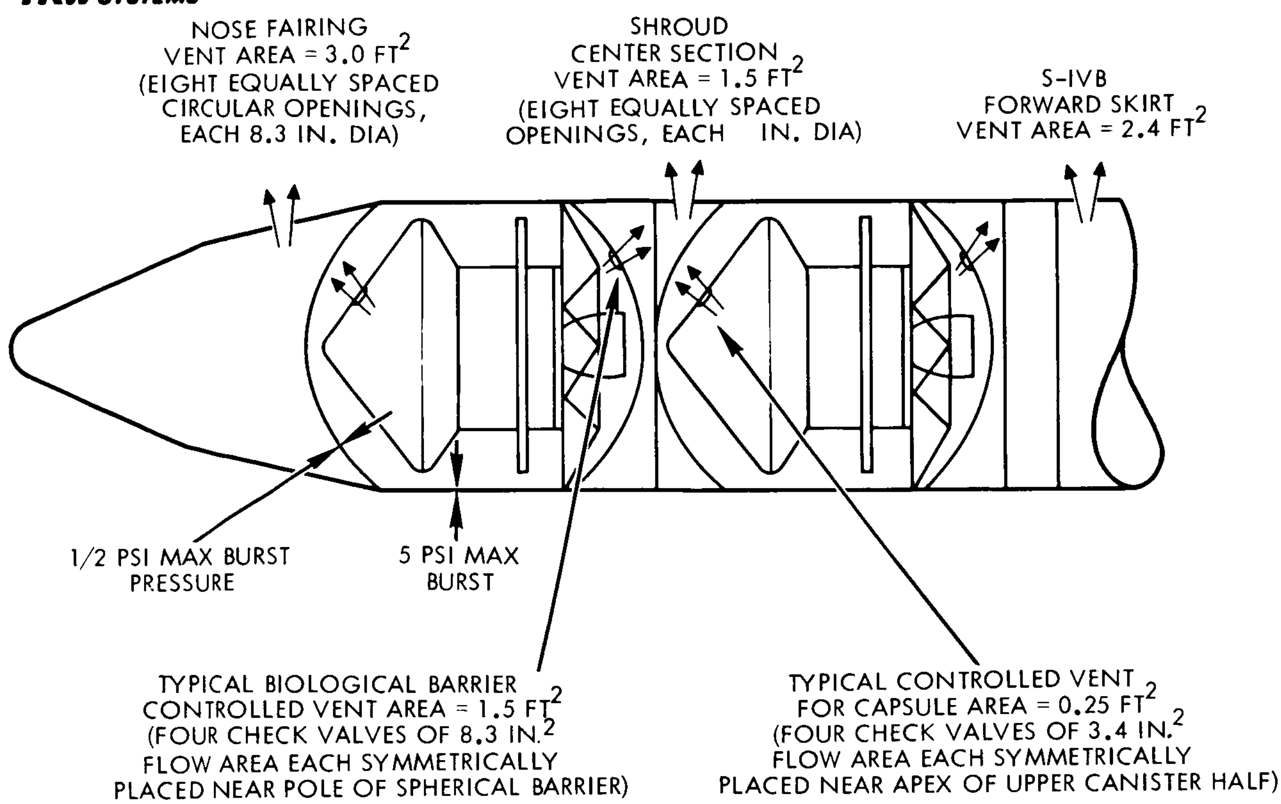
Design requirements for the planetary vehicle compartment so that it is suitable for terminal decontamination are the following:

- Maintenance of positive pressure in the compartment with sterilized gas until launch, calling for one-way aseptic pressure-relief valves
- A smooth surface to minimize pockets of contamination and to aid in final surface cleaning
- Sealed one-way ports for ETO sampling and contamination monitoring

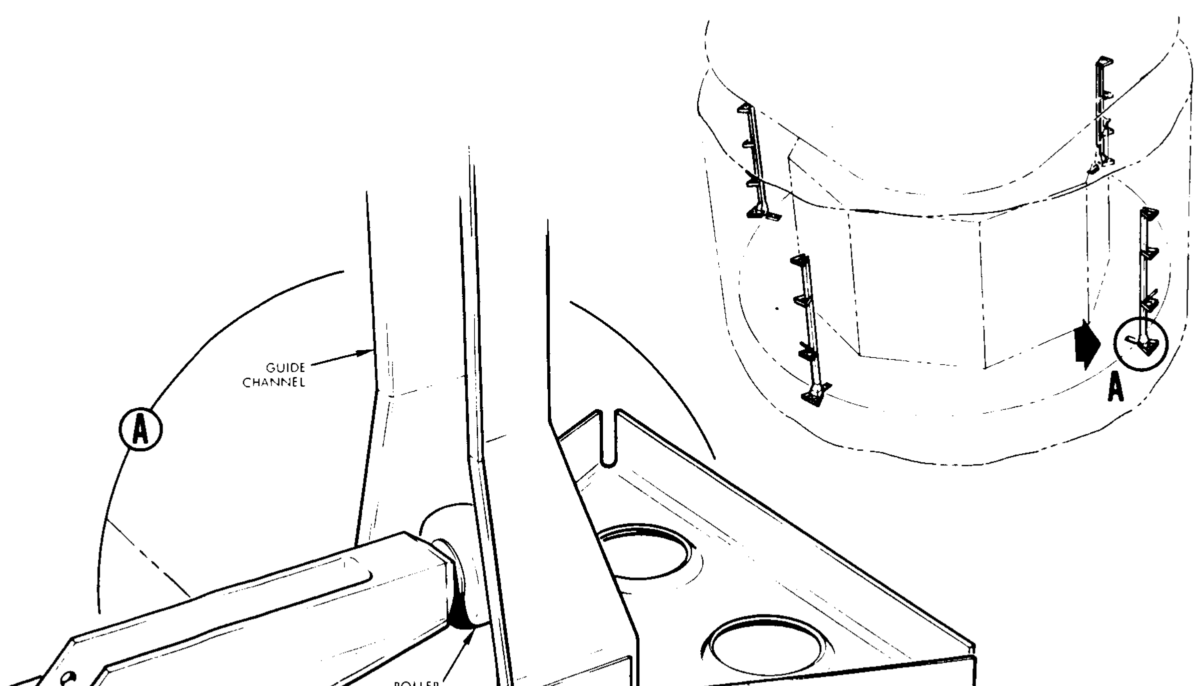
Most of the existing facilities for ETO exposure have inherent deficiencies, primarily in the areas of instrumentation and control functions. It is therefore essential that suitable procurement specifications be developed to satisfy the requirements of the Voyager program if the stringent decontamination goals are to be met.

4.2 SHROUD SEPARATION

Venting of the shroud during launch can be handled readily for Voyager, and the probabilities of shroud elements colliding with the planetary vehicles when they separate can be kept below 10^{-4}, without placing any of these elements on a collision course with Mars. The recommended set of vents for the different elements of the configuration is sketched at the top of page 44, based on allowable pressure differentials determined by strength and contamination considerations. The planetary vehicles need to be guided during their forward separation following a scheme such as sketched here.



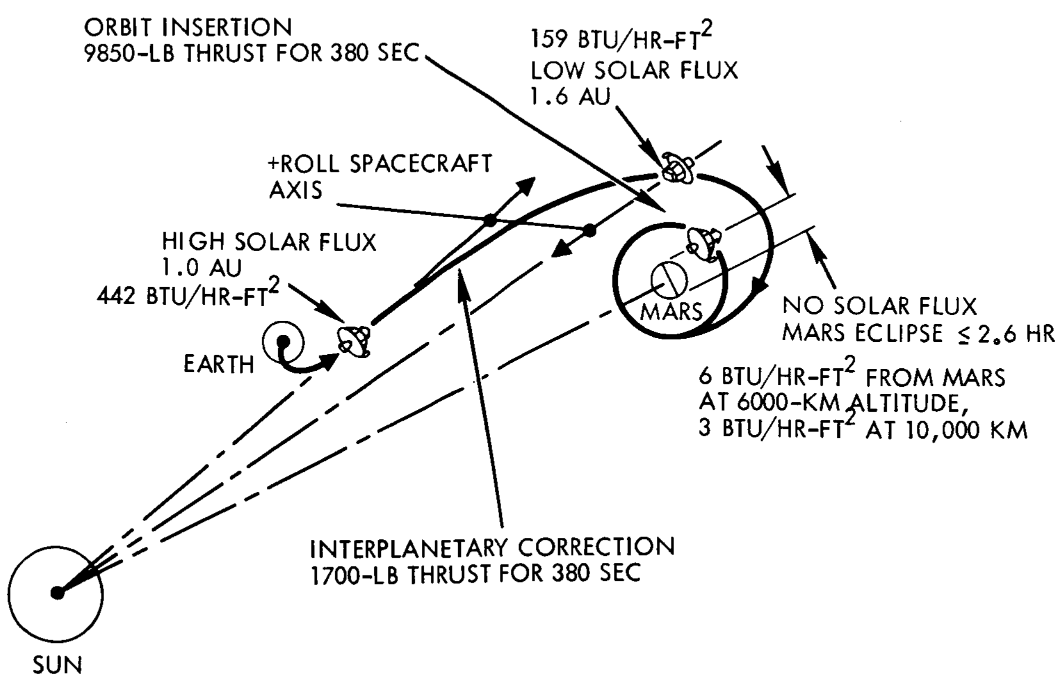
SHROUD VENTS are selected to maintain the minimum overpressure required to prevent contamination during ascent without exceeding the allowable pressure differential.



RAILS ON THE SHROUD ARE MATCHED BY FOLLOWERS ON THE SPACECRAFT to guide the planetary vehicle during over-the-nose separation.

4.3 TEMPERATURE CONTROL

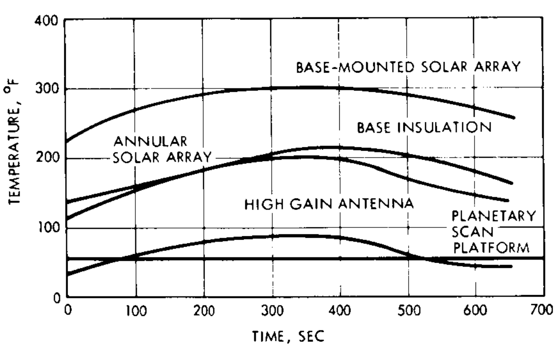
The review of combinations of insulated and uninsulated enclosures and passive and active techniques for radiation control has led to the selection of a temperature control subsystem for the Voyager application that combines passive and active approaches using the totally insulated enclosure concept, thermal louvers, thermostatically controlled heaters, thermal finishes, and varying degrees of structural thermal coupling. Bimetallic-actuated louvers were selected as the preferred louver system because of reliability, lower weight, simplicity, and proven flight performance. Various types of insulation (i.e., crinkled aluminized Mylar, Dimplar, Kapton) were considered, including attachment, overlapping, interleaving, and blanket size and contours. Multilayer crinkled aluminized Mylar was selected because it provides the best thermal performance with minimum weight. The selected method of attachment is Velcro tape.



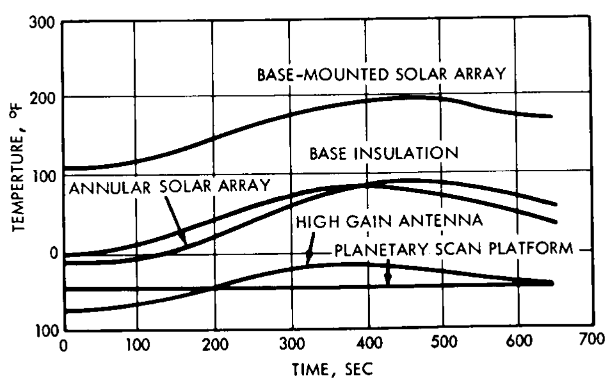
THERMAL CONDITIONS during the Voyager mission vary substantially as solar energy decreases with range, the engine is fired under varying circumstances, and the spacecraft enters eclipse.

4.4 PLUME HEATING

Analysis of the temperature and pressure dynamics for the LM descent engine aboard Voyager shows that there are no areas of plume impingement on the spacecraft and that the first interplanetary trajectory correction is the upper bound hot condition. Since the near-earth, steady-state temperature of the base-mounted solar array is only a few degrees under its 248°F limit, during the 380-second engine firing the heat flux from the engine plume causes a short-duration excursion to 300°F, as shown below. Temperatures to 300°F have been tested in the course of the Vela satellite program, demonstrating that the integrity and functional characteristics of the array can be maintained. During Mars orbit insertion, temporary temperature excursions are also experienced during the 380-second engine firing, as shown below. But since near Mars the steady-state temperatures are relatively low, the rises during engine firing are well within the allowable temperature ranges.



FIRST TRAJECTORY CORRECTION



MARS ORBIT INSERTION TEMPERATURES

MAXIMUM TEMPERATURES RESULTING FROM ENGINE FIRINGS are experienced at the first interplanetary firing since initial temperatures are higher; although the thrust level is greater at Mars orbit insertion, the reduced solar heat leads to cooler temperatures during the firing.

4.5 PHOTO-IMAGING SYSTEM

Because of limitations of currently acceptable tape recorders, it appears that for Voyager a mapping and reconnaissance system using film is preferable to a system relying exclusively on television. Real-time transmission of TV-sensed imagery is generally impossible and once the requirement for data storage is introduced, the advantage of TV systems in providing video signals disappears. Rather those systems having an inherently large storage capacity are advantageous, leading to a preference for storing the information on film.

In the study, three general types of systems were considered. The first is an all-TV system, conforming closely to that incorporated by NASA in the hypothetical science payload. The second is that recommended by TRW, a dual-framing film camera based on Lunar Orbiter concepts for medium and high resolution coverage, along with a TV camera for broad low resolution mapping in color or black and white. The third consists of a dielectric tape camera for medium resolution

PRELIMINARY SPECIFICATION

Photo-Imaging System

Purpose

To provide complete photographic coverage of the planet Mars at low resolution (1 kilometer) in color, mapping of large areas of Mars at medium resolution (100 meters), and imaging of selected areas at high resolution (10 meters).

LOW RESOLUTION CAMERA

Sensor - RCA 1-1/2 inch vidicon with slow scan target
Field of View - 29 x 29 degrees
Ground coverage - 500 x 500 kilometers (per frame)
Nominal ground resolution - 1 kilometer
Number of frames per orbit (maximum) - 33 frames (11 color pictures)
Photography time/orbit - 48 min (maximum)

MEDIUM AND HIGH RESOLUTION CAMERA

Sensor - Eastman Kodak 70 mm film camera

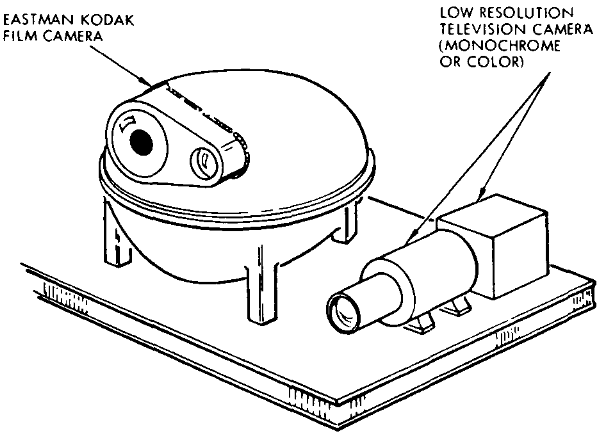
	Medium	High
Lenses- f.l.	100 mm (4 inches)	1000 mm (40 inches)
Format	60 x 60 mm	60 x 60 mm
Ground resolution	100 meters	10 meters
Frames per orbit	10	10
Ground track length per orbit (maximum)	7550 kilometers	
Photography time/orbit	48 min (maximum)	48 min (maximum)

CAMERA EQUIPMENT MOUNTED ON PSP

Weight 220 lb
Size 14.7 ft^3
Power 50.6 watts (average)

TOTAL SYSTEM INCLUDING ONE TAPE RECORDER

Weight 238 lb
Power 58.6 watts (average)



coverage along with the low resolution TV color camera which is common to all three configurations. No high resolution camera is included in the third system. The dielectric tape camera can in principle be fitted with an imaging system of sufficiently long focal length to provide high resolution ground coverage, but this is not an attractive approach because of a resulting long exposure time and consequent severe image motion compensation requirements. A high resolution TV camera as in the baseline system could of course be added.

PRELIMINARY SPECIFICATION

Recommended Film Camera Systems

Purpose

To provide medium and high resolution photographic coverage of the Martial surface. Contains integral film processor and data readout system.

Performance Characteristics

SENSOR

Eastman Kodak film camera modified to incorporate 40 inch and 4 inch focal length lenses, increased radiation shielding, improved "Bimat" film and processor.

FILM FORMAT

70 mm (60 x 60 mm) Type SO-243, length 100 feet (usable)

OPTICAL SYSTEM

	Medium Resolution	High Resolution
Lens	Refractive	Catadioptic
E.F.	100 mm f2.8	1000 mm f.4.0
Exposure	10-140 msec	10-40 msec
Frame Rate	1 frame/2 sec maximum	1 frame/2 sec maximum
IMC	None	5 percent

PROCESSING

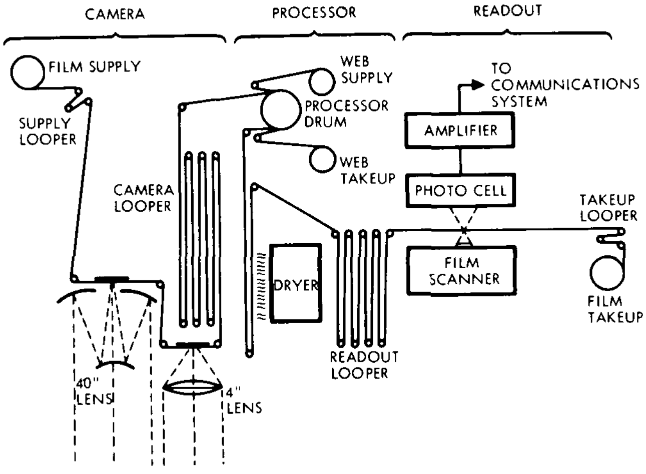
Eastman (new) "Bimat", web application transfer type. 120 feet, approximately 2.5 inches/min speed.

READOUT

Flying spot scanner with optical and mechanical scan. Rate changeable as bit rate is changed. GRE compatible.

Physical Characteristics

Size 14.1 ft^3
Weight 200 lb (including radiation shielding)*
Power 49.2 watts (average)



*NOTE: The weight estimate includes 30 gm/cm^2 of radiation shielding for the film casette, a total of 65 pounds. This shielding reduces the effect of solar cosmic rays to a low level, in the order of 2 or 3 rad per year. The predominant radiation will be galactic cosmic radiation of high energy level, against which shielding is not effective. The level of this radiation is estimated to be from 6 to 20 rads per year. 100 rad causes 50 percent fog level on SO-243 film.

Although TRW recommends the dual-frame photo-imaging system described in the accompanying preliminary specifications, the spacecraft described in Section 2 has been designed to accommodate the hypothetical, all-TV system. Since that system places a heavier demand on the spacecraft, a more flexible spacecraft results which can carry whatever type of photo-imaging equipment is selected.

CONCLUSIONS AND RECOMMENDATIONS

The Voyager spacecraft configuration described in this report is both technically feasible and readily implemented. Launched in tandem by a Saturn V, it can carry capsules weighing up to 8000 pounds into orbit about Mars and rapidly survey the surface. It can alter the period, the line of apsides, or the plane of its orbit as needed to assist in reconnaissance of the planet. No changes in hardware are needed for any of the launch opportunities in the 1970's.

Its communication link can transmit 50,000 bits/sec during its initial weeks in orbit, and this information can be devoted to mapping, high-resolution pictures, the relay of measurements by the capsule, rapid surveys of experimental results, or any combinations of these, as commanded from the ground.

The configuration adapts easily to growth either of the capsule or the equipment in the spacecraft. With a 20-foot antenna substituted for the 9.5-foot high-gain antenna, an information rate of 300 kbits/sec can be achieved.

These conclusions are made possible by the approach that the spacecraft be designed initially in view of the requirements of adaptability to the launch opportunities beyond the first and of growth in performance capability as the program matures. Two significant effects on the design have resulted from this approach: inclusion of propellant tanks of sufficient volume for the most demanding of the mission circumstances that may occur in the 1970's and structuring the spacecraft to support the weight of the heaviest capsule that may be flown in that decade. The success during the study of applying this approach without in any way jeopardizing one set of mission circumstances in favor of another leads TRW to recommend the approach strongly, as a means of providing the most versatile and cost effective Voyager program.

Tradeoff studies documented in the later volumes have led us to favor the following specific approaches:

- Modular construction with distinct equipment, propulsion, and adapter modules for parallel development, assembly, and test

- A photo-imaging system modeled after the dual-frame film camera of the Lunar Orbiter, to ease the data-storage problem

- Integrated approach to ground support that places a great deal of identical equipment in the system test complex and the stations of the Deep Space Network

- Biorthogonal coding of the telemetry, to permit more rapid data rates at given signal-to-noise ratios

- Multiplexers in the planetary scan platform to avoid complicating the cabling that crosses the gimballed interface of the PSP with the spacecraft

- Power regulation that provides the greatest conversion efficiency at Mars

Before the details of the spacecraft configuration are selected, it also appears desirable to conduct further studies in three areas:

- The feasibility and application of a centralized, general-purpose computer in the spacecraft

- Evaluation of the operational applications of a cluster of four C-1 engines in the propulsion subsystem

- The quantitative effects on possible violation of the Mars quarantine requirement of the level of pressurization maintained in the propellant tanks