VOYAGER SUPPORT STUDY

LM DESCENT STAGE APPLICATIONS

FINAL REPORT

VOLUME V

CONFIGURATION STUDY

TRW
SYSTEMS GROUP

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CONFIGURATION STUDY

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REDONDO BEACH, CALIFORNIA

PREFACE

The LM Descent Stage Applications Task of the TRW Voyager Support Study involves extending and updating selected areas of the TRW Voyager Spacecraft Phase IA Task B preliminary design. The work emphasizes propulsion considerations, and the areas of interest relate to the use of liquid propulsion (specifically utilizing the LM Descent Engine) for the Voyager spacecraft application. Potential problems directly within the propulsion subsystem as well as those relating to integration of propulsion with other subsystems are covered by this task.

The work is made up of five major subtasks which have been carried out and reported individually. The LM Descent Stage Applications Final Report therefore consists of five volumes as follows:

Volume I	Propulsion Studies, February 1967
Volume II	Thermal Control Studies, February 1967
Volume III	Propellant Hydrodynamic Studies, February 1967
Volume IV	Guidance and Control Studies, July 1967
Volume V	Configuration Study, July 1967

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1. INTRODUCTION AND SUMMARY

This document presents results of the Configuration Study Subtask of the TRW Voyager Support Study. It forms a part of the LM Descent Stage Applications Final Report. The basic objective of the study is to develop a gross preliminary design of a spacecraft so as to assess the effect of weight allocation changes from the Phase IA Task B values to those of the current Voyager technical plan. The new spacecraft weight allocation is 2000 pounds less than that for the Task B design so that particular attention in configuration design has necessarily been given to reducing spacecraft weight.

The study has emphasized propulsion aspects of the spacecraft design, using the LM Descent Engine (modified for Voyager) as a basic element and the TRW Task B propulsion approach with a blowdown mode of operation for midcourse maneuvers. Spacecraft modularity has been a design goal, represented by a propulsion module capable of separate assembly and test.

The scope of the study has involved the formulation of several spacecraft configurations to arrive at a gross preliminary design for a preferred approach. The spacecraft subsystems have been described briefly, and any differences from Task B have been indicated. Weight estimates have been developed along with a summary of control moment capabilities and powered flight slosh stability considerations.

The objectives of the study have been met by the preferred approach described herein. Configuration features have been derived that lead to significant weight savings over the Task B design based on the LM Descent Stage structure and tankage. A spacecraft payload margin of over 500 pounds is indicated, which represents a reasonable degree of confidence in being able to satisfy the specified propulsion and capsule support requirements within the allocated weight. A simple design has been achieved which provides the desired propulsion modularity, while allowing the remaining equipment module, containing all nonpropulsion spacecraft elements, to be assembled and tested separately.

2. STUDY CONSTRAINTS AND GUIDELINES

The constraints and guidelines for the study, with comparisons to Task B as applicable, as are follows.

2.1 STANDARD DESIGN

The flight hardware will be standardized to accommodate the 1973, 1975, 1977, and 1979 missions.

2.2 SUBSYSTEM DESIGN

The Task B subsystem designs will be utilized to a maximum extent. In particular, the spacecraft propulsion subsystem is to be functionally the same as for the Task B design. Customized tankage and structure will be used instead of the modified LMDS tankage and structure. The customized tanks will be sized to provide adequate ullage for the blowdown mode of operation for midcourse maneuvers.

2.3 ARRANGEMENT AND ENVELOPE

The planetary vehicle arrangement and dynamic envelope are given in Figure 1. This envelope differs from that specified for Task B. Comparison of the planetary vehicle dynamic envelopes is shown in Figure 2. Consideration will be given to minimizing the spacecraft length along the roll axis.

2.4 WEIGHT ALLOCATIONS

Weight allocations for the current study are given below along with those for Task B.

	Mass (lbm)	
Item	Current Study	Task B
Flight capsule	5,000	3,000
Spacecraft science	400	400
Flight spacecraft with PV adapter	16,600	18,600
Total planetary vehicle with adapter	22,000	22,000

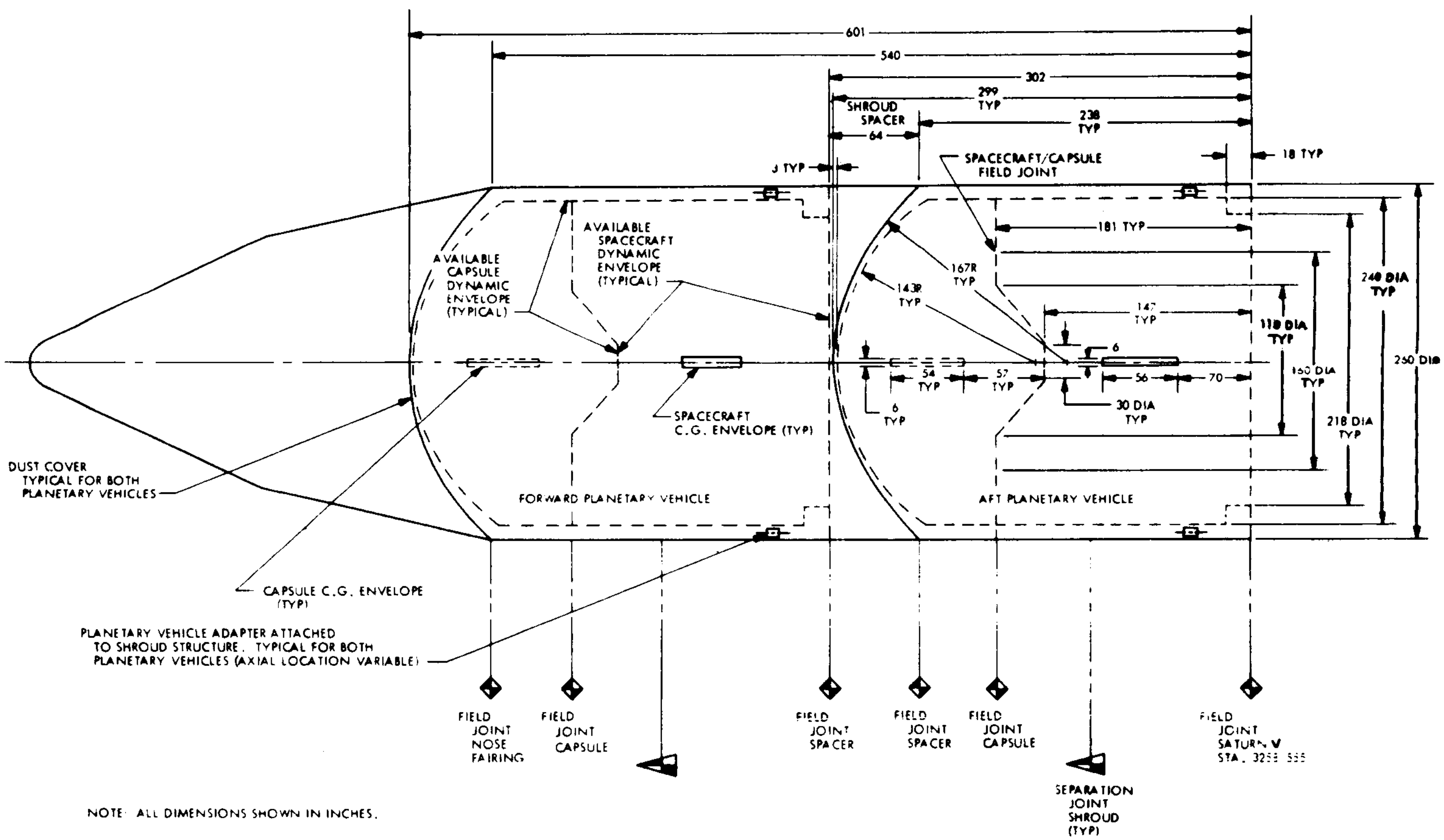


Figure 1. Planetary Vehicle Arrangement and Dynamic Envelopes

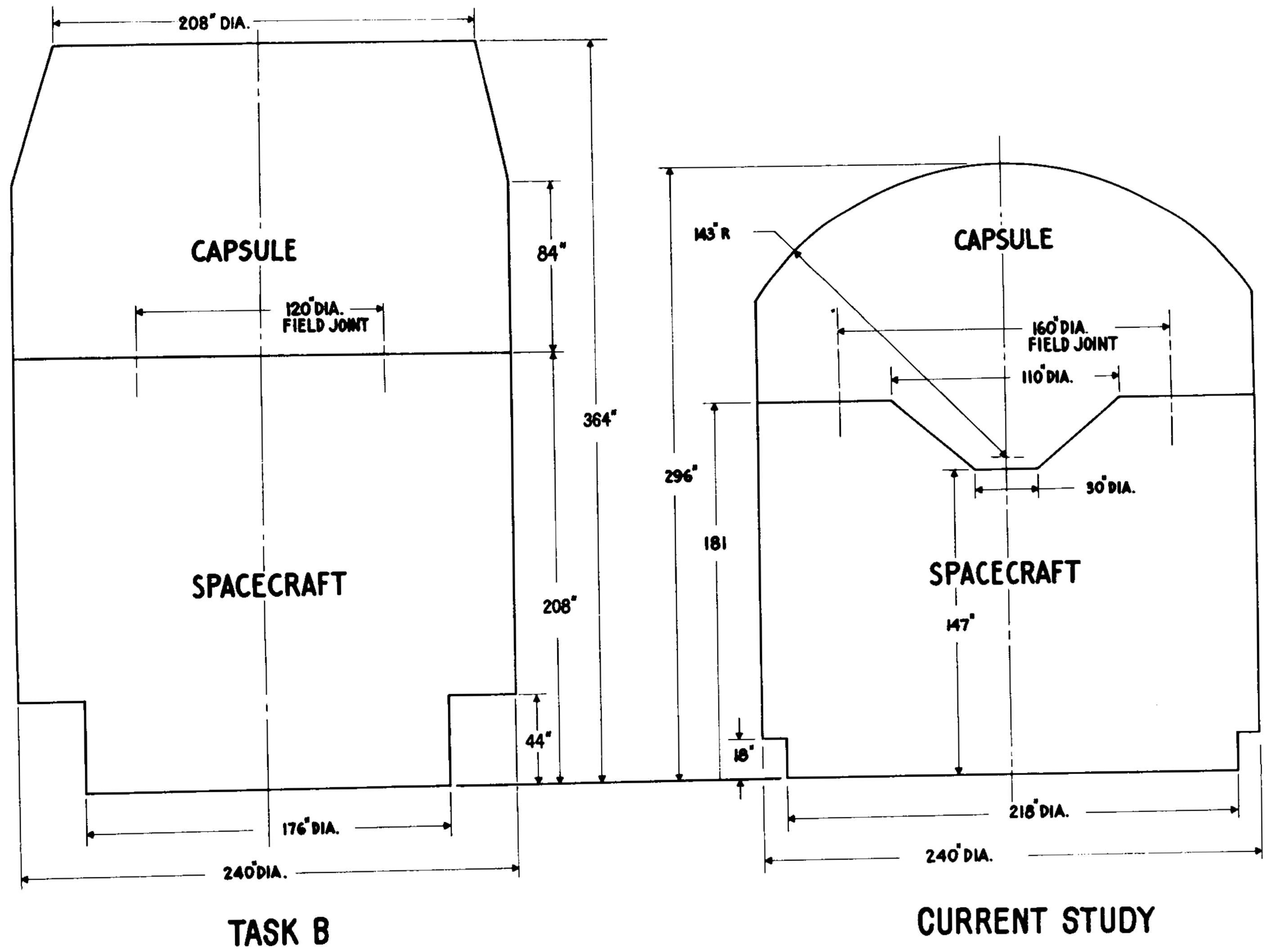


Figure 2. Planetary Vehicle Dynamic Envelope Comparison

2.5 VELOCITY PROFILE AND PROPULSIVE CAPABILITY

Velocity requirements for launches during the 1973-1979 era are as follows.

Velocity Requirements, meters/sec (ft/sec)

Year	Midcourse Correction		Orbit Insertion		Orbit Trim*		Total
1973	210	(689)	1760	(5780)	150	(493)	2120(6950)
1975	75	(246)	1945	(6385)	100	(328)	2120(6950)
1977	75	(246)	1945	(6385)	100	(328)	2120(6950)
1979	110	(361)	1910	(6274)	100	(328)	2120(6950)

The 1973 ΔV requirements shown above represent the critical mission case from a propellant requirements point of view, followed by 1979, because of the low specific impulse during the midcourse corrections. Based upon the 1973 velocity requirements the following velocity profile and mass (propellant) ratios were derived:

Maneuver	Thrust Level (lb)	Required Velocity Increment m/sec (ft/sec)		I_{sp} (sec)	Mass Fraction**
Midcourse correction	1050	210	(689)	285	0.072
Orbit insertion	7750	1760	(5780)	305	0.446
Planetary vehicle orbit trim	1050	150	(493)	285	0.052

2.6 CAPSULE INTERFACE

The flight capsule is mounted forward of the flight spacecraft in accordance with the arrangement shown in Figure 1. The structural interface between the flight capsule and the flight spacecraft will be a structural field joint with the geometry shown in Figure 1.

The in-flight capsule separation joint will be forward of this field joint and will be part of the capsule system. All flight capsule loads are

* Capability with capsule not separated.

** Ratio of consumed propellant to initial mass.

to be transmitted across this field joint to the flight spacecraft, and there will be no other structural support provided to the flight capsule.

The flight spacecraft will provide 200 watts of raw DC power to the flight capsule except during high usage periods such as midcourse maneuvers. Also, 50 watts of power in suitable form will be supplied to spacecraft-mounted capsule support equipment in conjunction with operation of the capsule-to-spacecraft link.

The flight spacecraft weight allocation for spacecraft-mounted capsule support equipment will be 50 pounds.

In the event the flight capsule is not launched with the flight spacecraft or the spacecraft-capsule separation is not performed as initially planned prior to launch, the flight spacecraft will be capable of performing the remainder of its mission.

2.7 LAUNCH VEHICLE INTERFACE

The planetary vehicles are enclosed by the launch vehicle shroud to protect them from the aerodynamic and thermal environment prior to and during launch. Each planetary vehicle is enclosed in a cylindrical shroud section as shown in Figure 1. To provide for separation of the planetary vehicle from the launch vehicle, a circumferential shroud separation joint is utilized as shown in Figure 1. An "over-the-nose" separation scheme is utilized in which the shroud section forward of this joint is jettisoned to clear the way for planetary vehicle separation.

The structural interface between the spacecraft flight hardware and the launch vehicle is to be the attachment surface at which the planetary vehicle adapter is attached to the shroud (see Figure 1). The attachment interfaces of both planetary vehicles will be identical.

3. DESIGN APPROACH

Pertinent sections of the Task B final report and the related requirements documents have been reviewed and compared with the current Voyager Mission Specification (Reference 1) and Environmental Predictions Document (Reference 2) to determine similarities and variations between requirements for the Task B study and those for the current work. Consistent with the objective of the study, only requirements having an effect on gross configuration design have been considered.

The overall approach to configuration design is directly related to the spacecraft weight allocation of Section 2. 4. To reduce the total flight spacecraft/planetary vehicle adapter weight by 2000 pounds requires particular emphasis in configuration design to be given to minimizing inert flight spacecraft weight. Approximately 1. 3 pounds of propellant is required for each pound of inert flight spacecraft weight to meet the mission velocity requirements. Therefore every pound added to the flight spacecraft to achieve a certain performance capability costs the planetary vehicle 2. 3 pounds. Thus, if the same capability can be achieved by adding less than 2. 3 pounds to the planetary vehicle adapter, which remains with the shroud, then a net weight saving results. It therefore appears appropriate to configure a spacecraft and adapter geometry so as to minimize the loads and weight of the flight spacecraft, even at the expense of increased adapter weight. In two of the three configurations developed for this study a planetary vehicle adapter that remains with the shroud is shown instead of the Task B spacecraft outriggers that are integral with the planetary vehicle; in the third configuration the adapter is part of the spacecraft.

The first step in the design procedure is propellant tank sizing to accommodate the required propellant loading plus the necessary ullage for the blowdown operation. The tanks are sized for a total separated planetary vehicle weight of 22,000 pounds less adapter weight. Thus any weight margin represents additional spacecraft payload capacity which may be used to increase the spacecraft science or other planetary vehicle subsystems. Weight statements for the several configurations resulting from this study are given in Section 7.

The roll axis length limitation of the dynamic envelope necessitates a multiple tankage system. Four tanks are used for the propellants, following the Task B study, except they are spherical to minimize weight. Although tanks of other shapes might simplify the associated mounting arrangement, the minimum overall weight appears to correspond to a spherical tank with a cylindrical mounting skirt, which represents the Task B propellant tank installation arrangement.

The loaded propellant tanks present the major loads to be withstood by the spacecraft during launch, which is the governing design condition. The derived spacecraft designs attempt to configure the load paths of these tanks as directly as possible into the adapter, in order to minimize spacecraft loads (and hence weight).

Finally, modularization of the propulsion subsystem for each of the configurations permits more efficient handling and allows its assembly, integration, and testing independent of the other spacecraft subsystems. Accordingly, the equipment module includes all subsystems other than the propulsion subsystem, for the same reason. Spacecraft modularity is illustrated for two of the configurations presented in Section 4.

4. PROPULSION SYSTEM SIZING

During transit to Mars, after separation from the launch vehicle, midcourse corrections have to be applied to the planetary vehicle. This will be accomplished by means of the modified LM descent engine, acting in a low thrust mode. After 5 seconds of start tank operation, propellant will be fed to the engine from the main propellant tanks, which are pressurized prior to launch with gaseous helium at 235 psia. No additional pressurizing gas is used and propellant feed for this stage of operation is by a blowdown mode. The tank pressure reduces to approximately 95 psia at the end of the midcourse maneuver(s) as the ullage gas expands to replace the expended propellant.

Prior to Mars orbit insertion the tanks will be pressurized to 235 psia from an external helium pressurization system; this pressure is maintained until the insertion maneuver is complete.

Orbit trim(s) subsequent to this will again be accomplished by the engine acting in the low thrust mode and utilizing a blowdown approach.

Design data for the propellants, pressurization, system, and propellant tanks are given in Table 1.

Based upon the maximum planetary vehicle weight (including adapter) of 22, 000 pounds, estimates of total propellant requirements have been made. The variable in this instance is the planetary vehicle adapter weight, or alternatively the separated planetary vehicle weight. For the Task B design this was estimated to be 422 pounds while a weight of 1500 pounds was allocated. Assuming these two numbers bracket the adapter design weight, it is possible to derive the required usable propellant range as being 10, 516 to 11, 070 pounds. The ratio of usable propellant to separated planetary vehicle weight is 0. 513.

For a given separated planetary vehicle weight, the tanks are sized to accommodate the total propellant load (allowing 3 percent non-usables) and adequate ullage volume for the midcourse blowdown operation. For the specified initial and final pressure levels, the increase in volume corresponding to expended midcourse propellant allows the initial volume

Table 1. Propulsion Design Data

Propellants	
Oxidizer	N_2O_4 at 90.1 lb/ft^3 at 70^{o}F
Fuel	Aerozene 50 at 56.3 lb/ft^3 at 70^{o}F
Mixture Ratio	1.6 at 70^{o}F
I_{sp}	285 sec at F = 1050 lb
	305 sec at F = 7750 lb
Pressurization System	
For Midcourse Maneuvers and Orbit Trim	
Type	Blowdown
Pressurizing medium	Gaseous helium
Initial pressure	235 psi
Final pressure	95 psi
For Mars Orbit Insertion	
Type	Regulated pressure
Pressurizing medium	Gaseous helium
Number of pressurant tanks	Variable, from 1 to 4 spherical tanks
Material	Titanium 6A1-4V
Ultimate tensile strength	160,000 psi
Design factor of safety*	2.2
Minimum gage	0.020 in
Nonoptimum factor	1.10
Storage pressure	3000 psia
Final tank pressure	335 psia
Polytropic helium gas exponent	1.14
Propellant Tanks	
Type	Spherical
Number	2 oxidizer, 2 fuel
Size	Mixture ratio adjusted such that all tanks are same diameter
Material	Titanium 6A1-4V

Table 1. Propulsion Design Data (Continued)

Propellant Tanks (Continued)	
Ultimate tensile strength	160,000 psi
Tank pressure	270 psia maximum
Factor of safety*	2.2
Minimum gage	0.020 in
Nonoptimum factor	1.10

*Ratio of design ultimate pressure to limit pressure

and corresponding weight of helium to be determined. A polytropic helium gas expansion with an exponent of 1.14 is assumed, based on LM experience. The corresponding propellant and pressurization tankage weights shown in Figures 3 and 4, respectively, include a 10 percent increase over the theoretical weight to account for welds, bosses, and other nonoptimum factors.

The external pressurization system is sized to maintain a constant 235 psia in the propellant tanks. Account is taken of the fact that at the time the tanks are pressurized for the orbit insertion maneuver, a pressure of 95 psia exists in the tanks from the previous blowdown operations. This corresponds to a partial pressure of approximately 17 psia when the orbit insertion maneuver is completed. Consequently the equivalent of 218 psia of helium is to be provided additionally from the external tank(s).

Orbit trim propellant requirements can be assumed as being obtained with the propulsion system again operating in the blowdown mode.

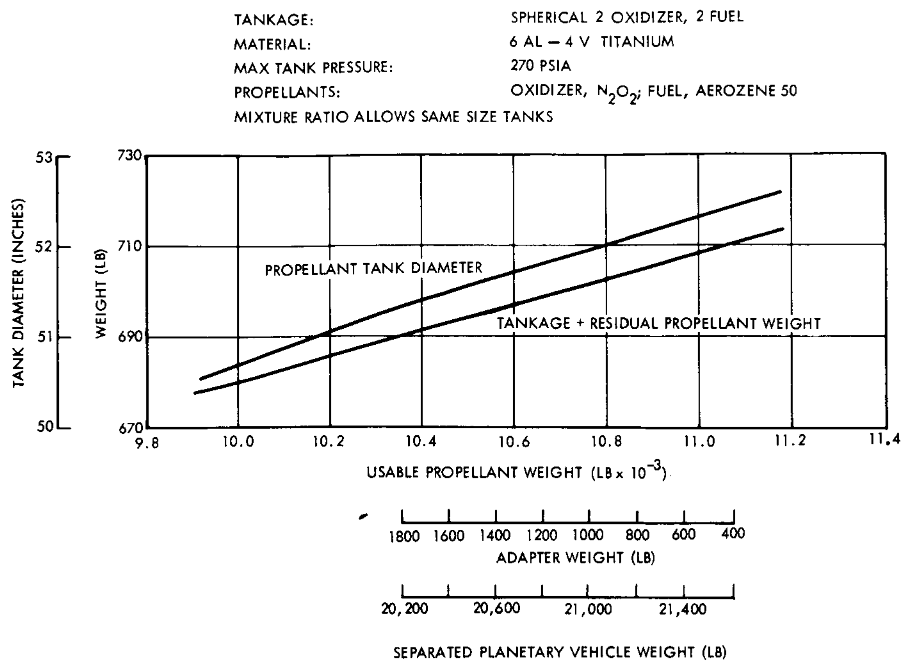


Figure 3. Voyager Propulsion System Propellant Tank Sizing

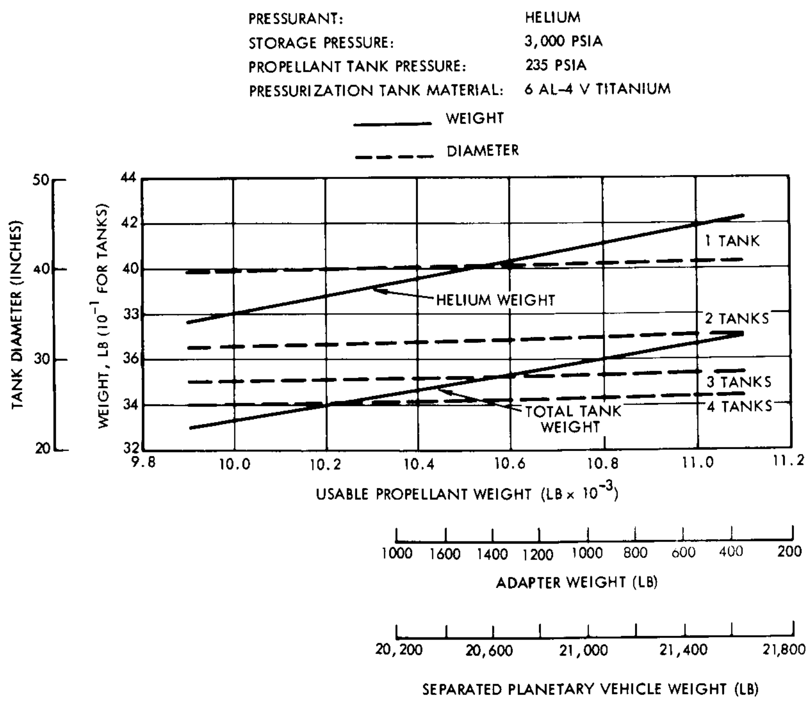


Figure 4. Voyager Propulsion System Pressurization Tank Sizing

5. SPACECRAFT CONFIGURATION

Three configurations meeting the study constraints and satisfying the design approach and propulsion sizing discussed above are presented in this section. Several criteria have been selected in addition to the presented study constraints to be imposed upon such designs, and these are given below:

- Engine gimbal point/spacecraft center of mass moment arm of at least 32.0 inches, to provide adequate control moment
- Spacecraft longitudinal center of mass to be forward of the propellant tank center (for spherical tanks) to minimize propellant sloshing instability problems
- Sufficient solar array area, stowage volume for deployable equipment, and in general sufficient areas and volumes to satisfy subsystem requirements

The design procedure followed consists of the following gross steps:

- Given the required propellant and pressurization tank volumes, solar array areas, and deployable equipment sizes, a configuration arrangement is formulated.
- Based on a set of design loads and factors, load paths within the spacecraft are defined; forces in the structural elements are then determined, and the basic sizes of the structure are derived.
- The weight and center of mass for the configuration are computed, and the results tested to see if the criteria above for slosh stability and gimbal point moment arm are met.

For this study three such configurations were conceived and the above steps performed. Only one of the three configurations, however, meets both the slosh stability and gimbal arm criteria. Each has a satisfactory weight margin demonstrating the feasibility of meeting the study objective of developing a spacecraft preliminary design which is 2000 pounds lighter than the Task B design.

Configuration Characteristics

Configuration		Spacecraft Weight Margin (lb)	Gimbal Moment Arm (in)	Slosh Stable*
Preferred	(AD7-122)	713.3	48.6	Yes
Semimonocoque Alternate	(AD7-100A)	511.6	15.0	No
Trussed Alternate	(AD7-125)	101.6	17.1	Yes

*Worst case, assuming an emergency capsule separation prior to the midcourse maneuvers

Each of the configurations will now be described briefly. The preferred configuration meeting the study objectives is AD7-122; the two other configurations do not meet the above criteria. Slosh stability can be provided where it does not exist by the addition of baffles in the propellant tanks; this is discussed in Section 6.2. The gimbal moment arm length can be increased only by moving the engine aft with respect to the spacecraft which violates the presented spacecraft envelope.

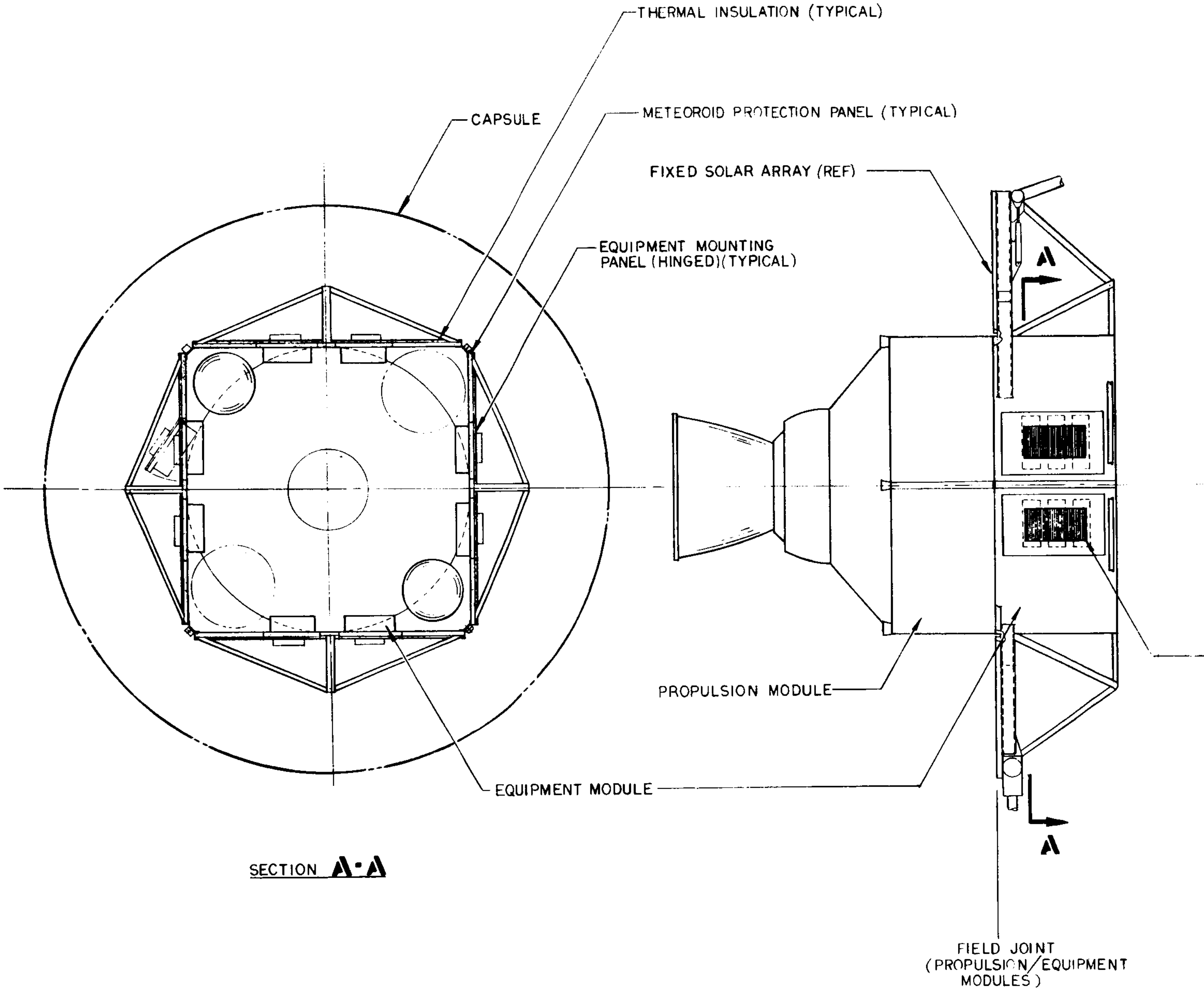
The three configurations use substantially the same subsystems; these are briefly described in Section 6.

5.1 PREFERRED CONFIGURATION

The preferred configuration is illustrated in Figure 5; its modularity is depicted in Figure 6. Some further detail of its propulsion module is shown in Figure 7.

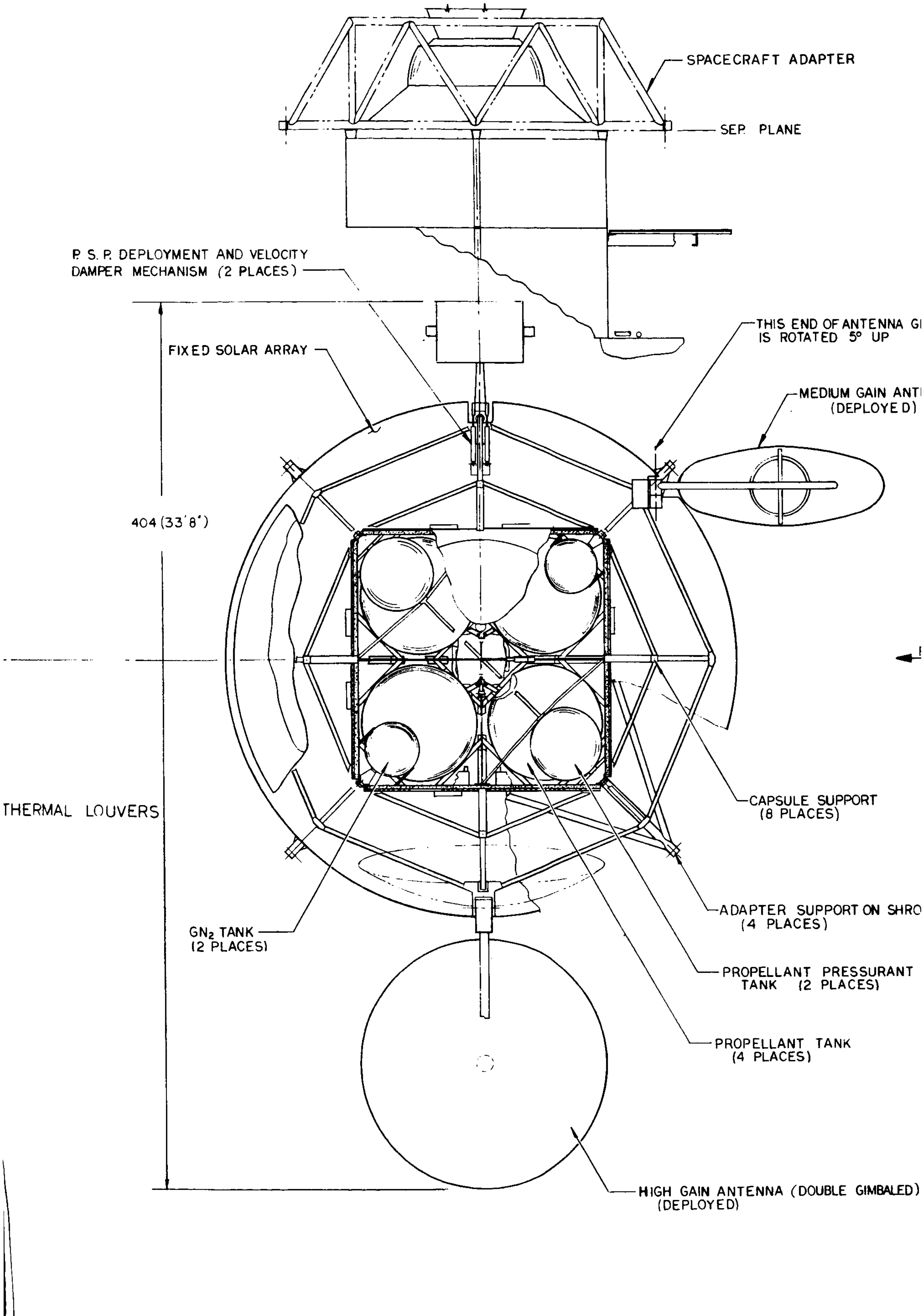
The adapter is composed of a group of truss beams arranged as a square, with attachment to the shroud occurring as diagonal beam extensions at the corners of the square. Lateral stabilization is provided by bracing each shroud attachment point to the midpoint of each side of the square. The corners of the square and the midpoints of each side are the eight hardpoints used for spacecraft attachment. Four alternate points are tension joints; shear and compression are transferred at all eight points.

The propulsion module consists of a semimonocoque structure whose geometry matches the square form of the adapter. The structure's



SECTION A-A

1/20 AD7-134


SPACECRAFT ADAPTER
SEP. PLANE
P. S. P. DEPLOYMENT AND VELOCITY DAMPER MECHANISM (2 PLACES)
FIXED SOLAR ARRAY
THIS END OF ANTENNA GI
IS ROTATED 5° UP
MEDIUM GAIN ANT
(DEPLOYED)
404 (33'8")
THERMAL LOUVERS
CAPSULE SUPPORT
(8 PLACES)
ADAPTER SUPPORT ON SHRO
(4 PLACES)
GN2 TANK
(2 PLACES)
PROPELLANT PRESSURANT TANK (2 PLACES)
PROPELLANT TANK
(4 PLACES)
HIGH GAIN ANTENNA (DOUBLE GIMBALED)
(DEPLOYED)

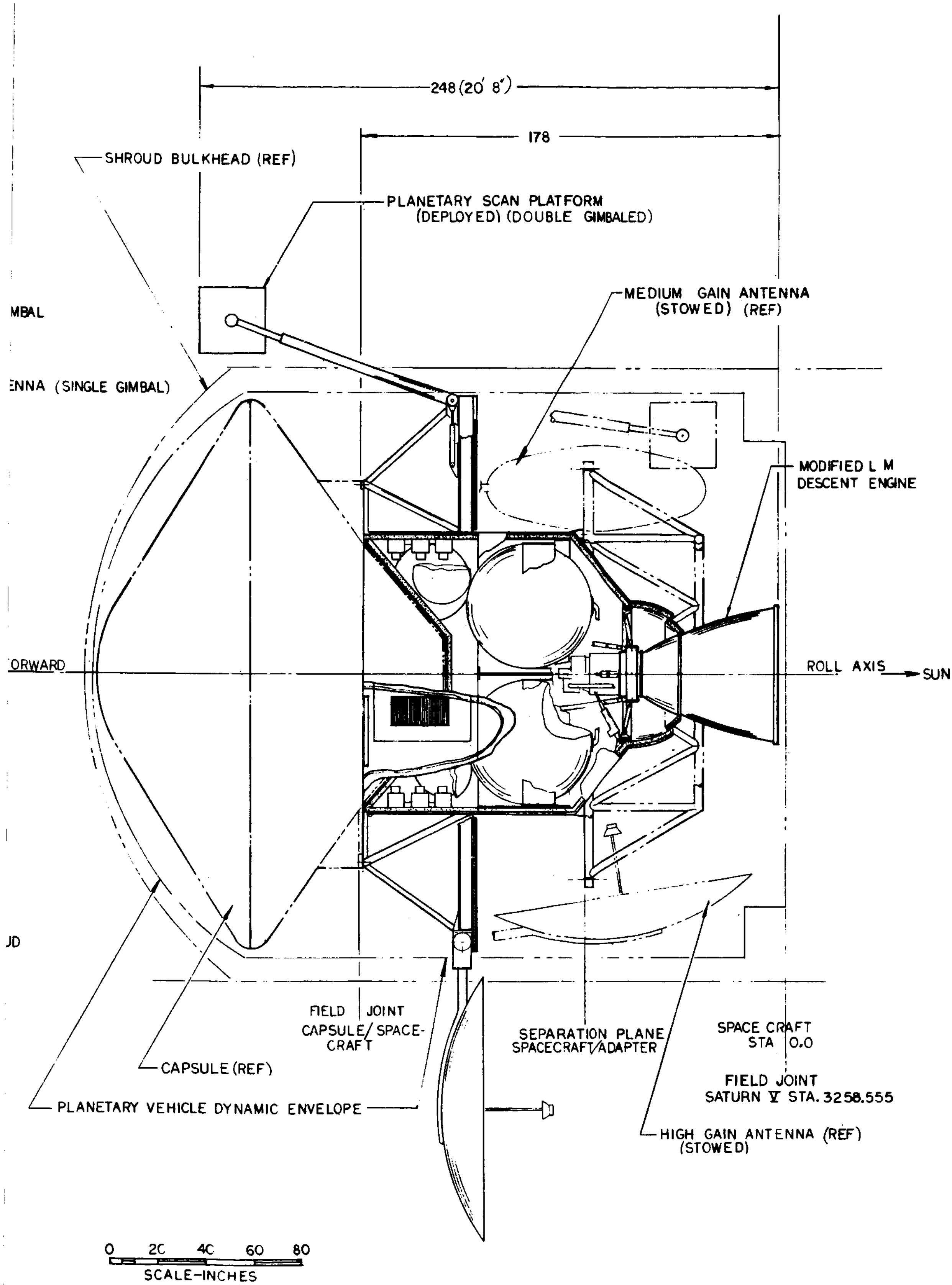


Figure 5. Preferred Configuration

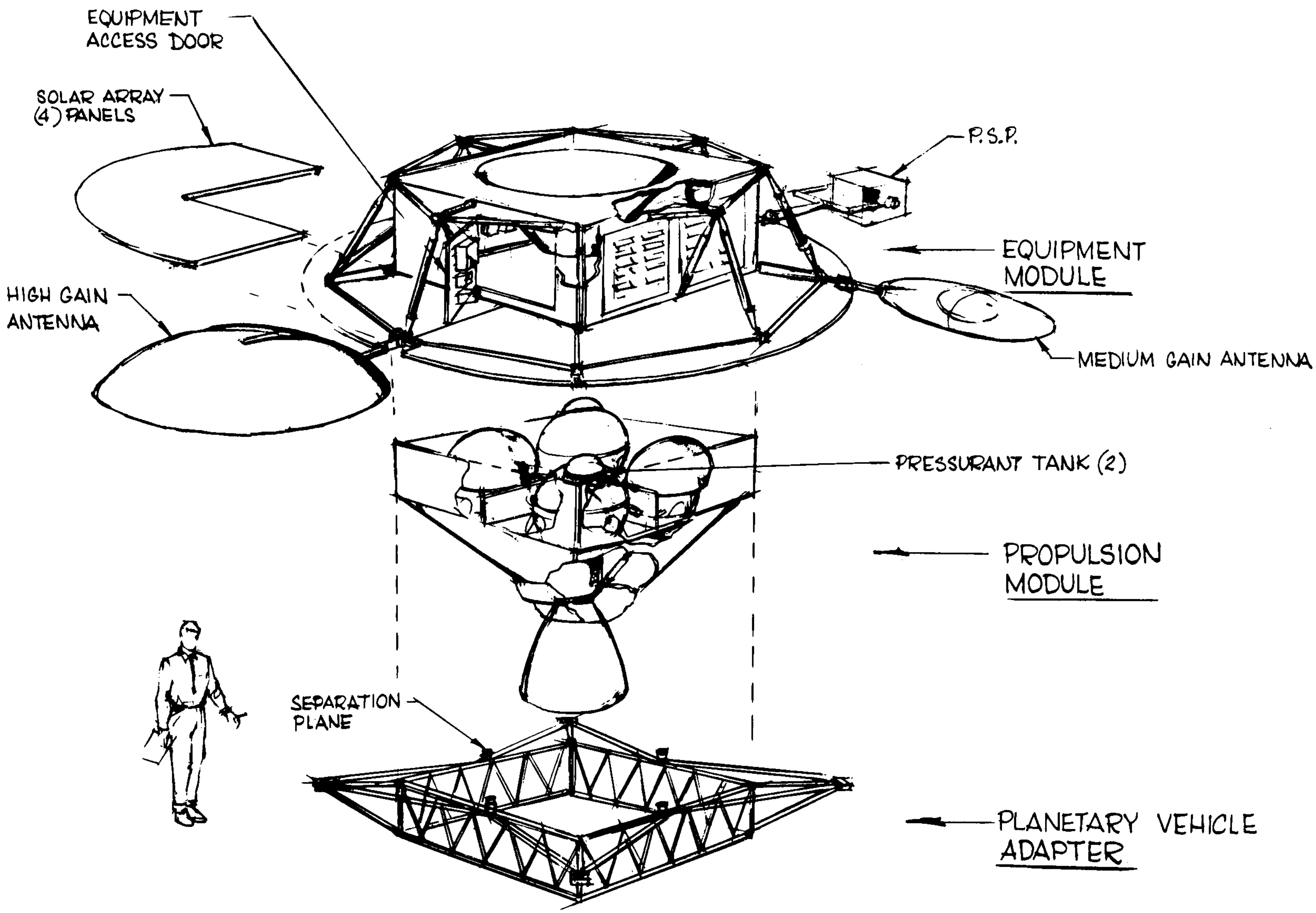


Figure 6. Modularity of the Preferred Spacecraft Configuration

sandwich panels provide the micrometeoroid shielding required by the tankage and other propulsion subsystem elements. Two shear resistant beams are arranged in a cruciform that form four identical compartments within the module. Within each compartment a propellant tank is mounted using an installation scheme that is similar to the LM descent stage. The engine is mounted below the propellant tankage array, protected by micrometeoroid shielding. Two helium tanks are used to keep the diameters of the spheres small enough to fit into available volumes. They are mounted using a two-point suspension, one of which is fixed in space while the other point has one-degree-of-freedom normal to the tank to accommodate tank expansion.

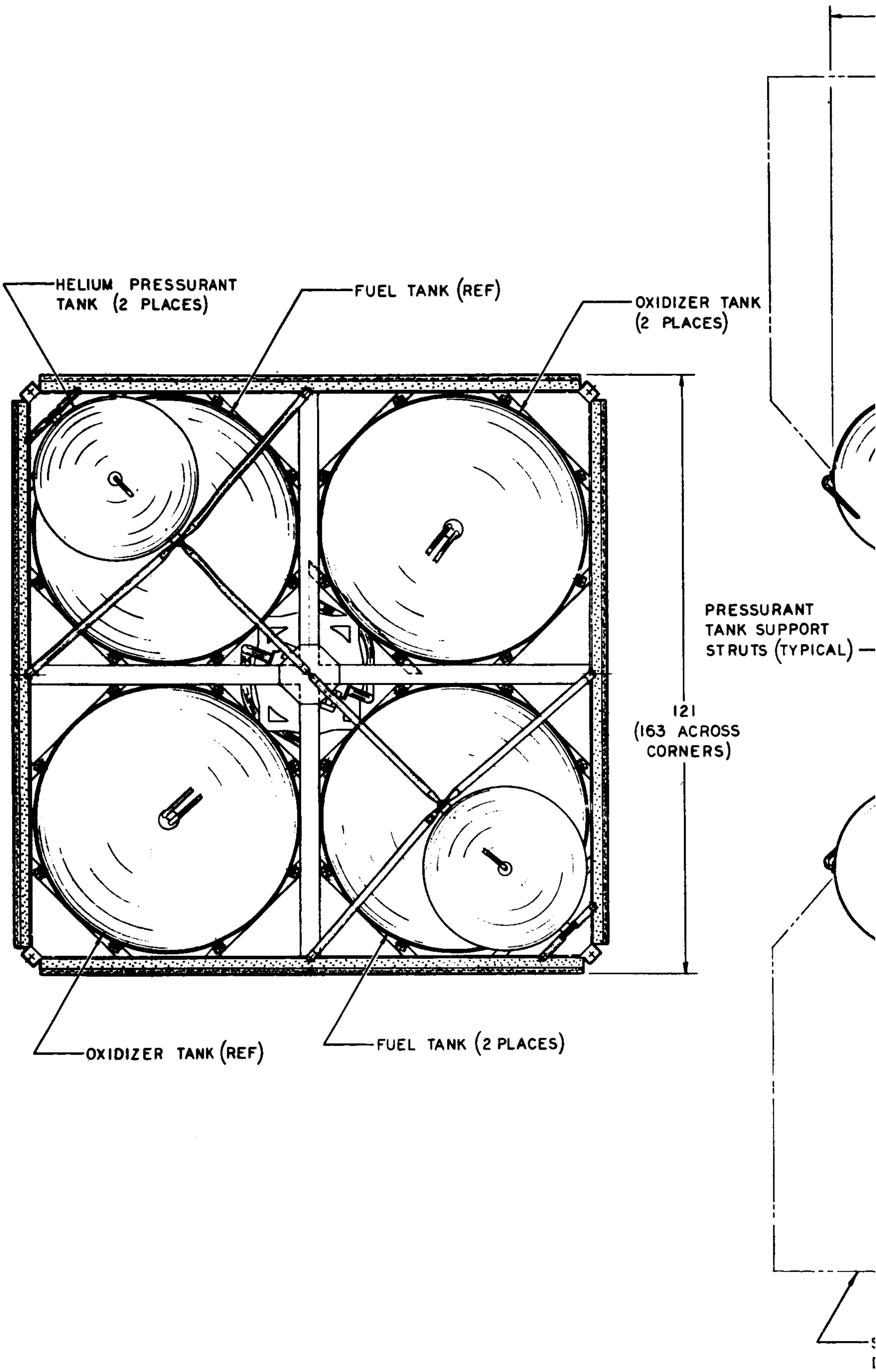
The equipment module consists of a matching square monocoque structure, which accommodates equipment panels for most of the electronic equipment, and truss elements which support the solar array, the deployable antennas, and planetary scan platform (PSP), and provides additional hardpoints for capsule support. Again, the external sandwich panels provide the required micrometeroid shielding. The area aft of the capsule is similarly protected for after capsule ejection.

The capsule is supported at eight equally spaced hardpoints, and its loads are transmitted through the equipment and propulsion modules into the adapter.

The deployable equipment is stowed aft of the solar array between the mounting elements of the adapter.

Although shroud separation requirements have not been studied in detail, an over-the-nose separation scheme may require some guidance for the planetary vehicle as it flies out of the shroud, since the axial station of the maximum capsule diameter may be as far as 90 inches from the station of the maximum spacecraft diameter, which is at the solar array.

This configuration has a substantial weight margin, meets the slosh stability requirements relative to the center of gravity location, and has a gimbal moment arm significantly in excess of the minimum requirement. The semimonocoque configuration realizes weight economies in the additional use of structure for micrometeoroid protection.



HELIUM PRESSURANT TANK (2 PLACES)
FUEL TANK (REF)
OXIDIZER TANK (2 PLACES)
PRESSURANT TANK SUPPORT STRUTS (TYPICAL)
121 (163 ACROSS CORNERS)
OXIDIZER TANK (REF)
FUEL TANK (2 PLACES)

AD7-135

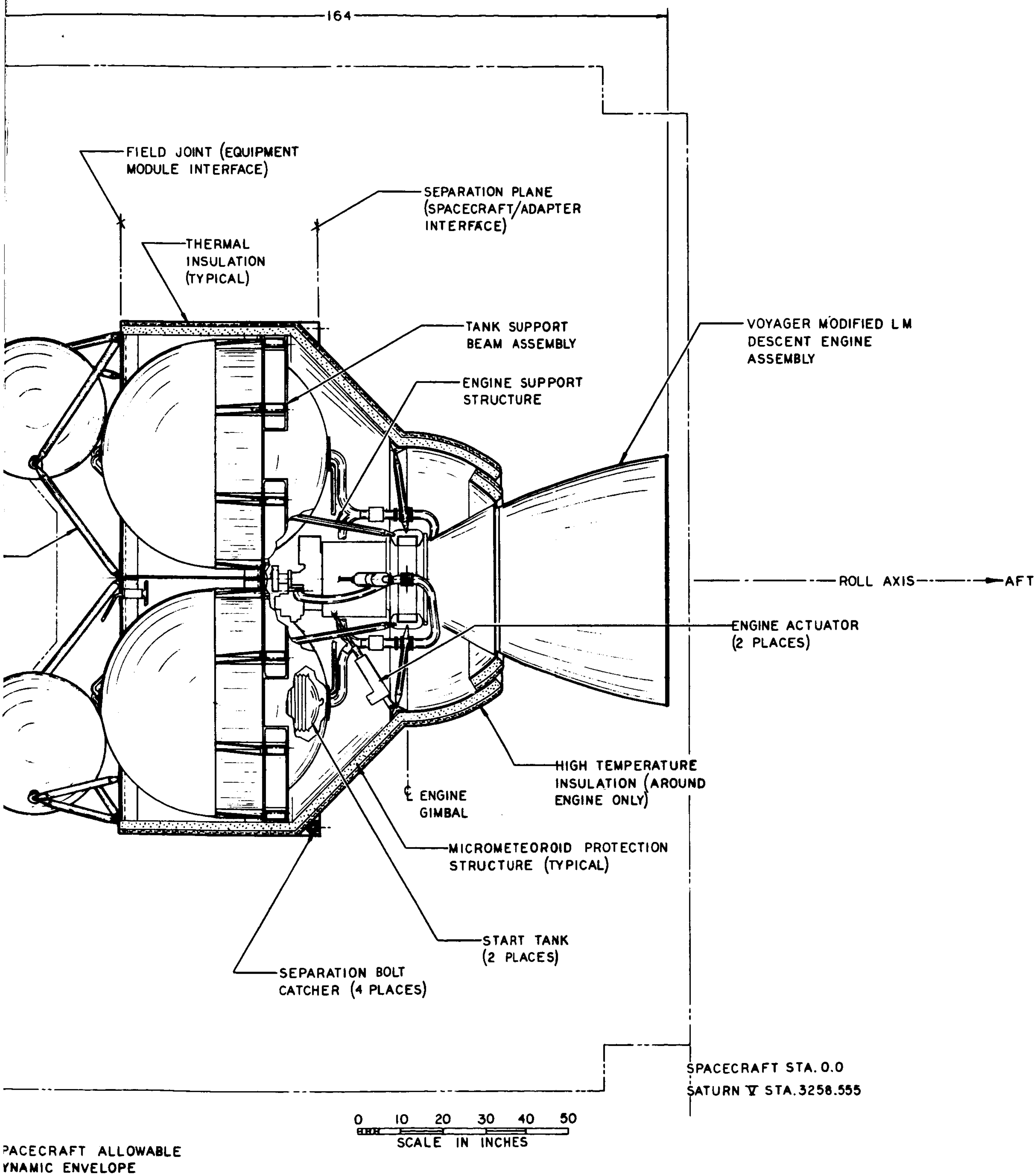


Figure 7. Propulsion Module of Preferred Configuration

This is seen by comparing the weight margin of this configuration with that of the second alternate configuration which is a truss arrangement; in the latter the non-structural micrometeoroid panels significantly increase the weight of the structure.

5.2 FIRST ALTERNATE CONFIGURATION (SEMIMONOCOQUE)

Figure 8 illustrates a planetary vehicle configuration in which the adapter takes the form of four truss beams in a double cruciform geometry. The basic spacecraft structure of this design is composed of four shear resistant beams also arranged in a double cruciform. It is essentially the same design concept as that used in the LM descent module of the Task B design, except that the beam height has been reduced to 36 inches. This structure, together with the tankage and engine which it supports, constitutes the propulsion module. The shear panel stiffeners, located at the eight beam extremities and at the four beam intersections, provide 12 hard points, all of which are located directly forward of the adapter beams. Column members, extending between the two structures and located at these 12 hard points, transfer the propulsion module boost loads to the adapter. In this manner the load paths within the propulsion module structure are minimized.

Modularity of this configuration is shown in Figure 9. The capsule is supported by a sculptured interstage which conducts the capsule loads directly to the eight propulsion module web stiffening members located at the beam extremities. With this geometry the capsule loads are transmitted directly without inducing bending moments on the propulsion module structural members. In the Task B design all capsule, tankage, and engine loads were carried in bending by the LM propulsion module beams and through the outriggers before unloading at the shroud. An overall weight saving compared to Task B results for this design because of the significantly shorter load paths within the propulsion module structure.

In Task B the high-gain antenna was stowed forward of the aft equipment module. In order to eliminate high-gain antenna stowage and fly-out problems in conjunction with the adapter structure in this design,

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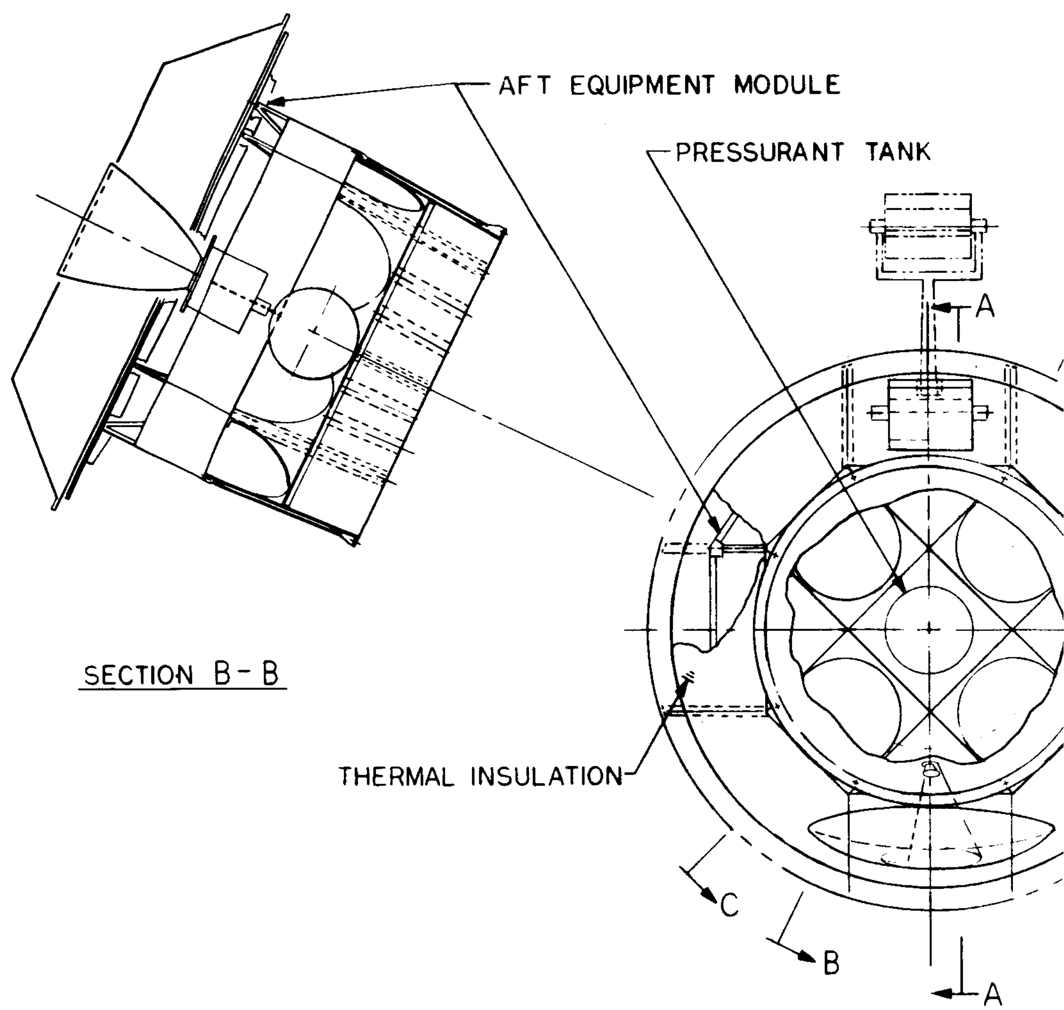


AD7 100A
1/40

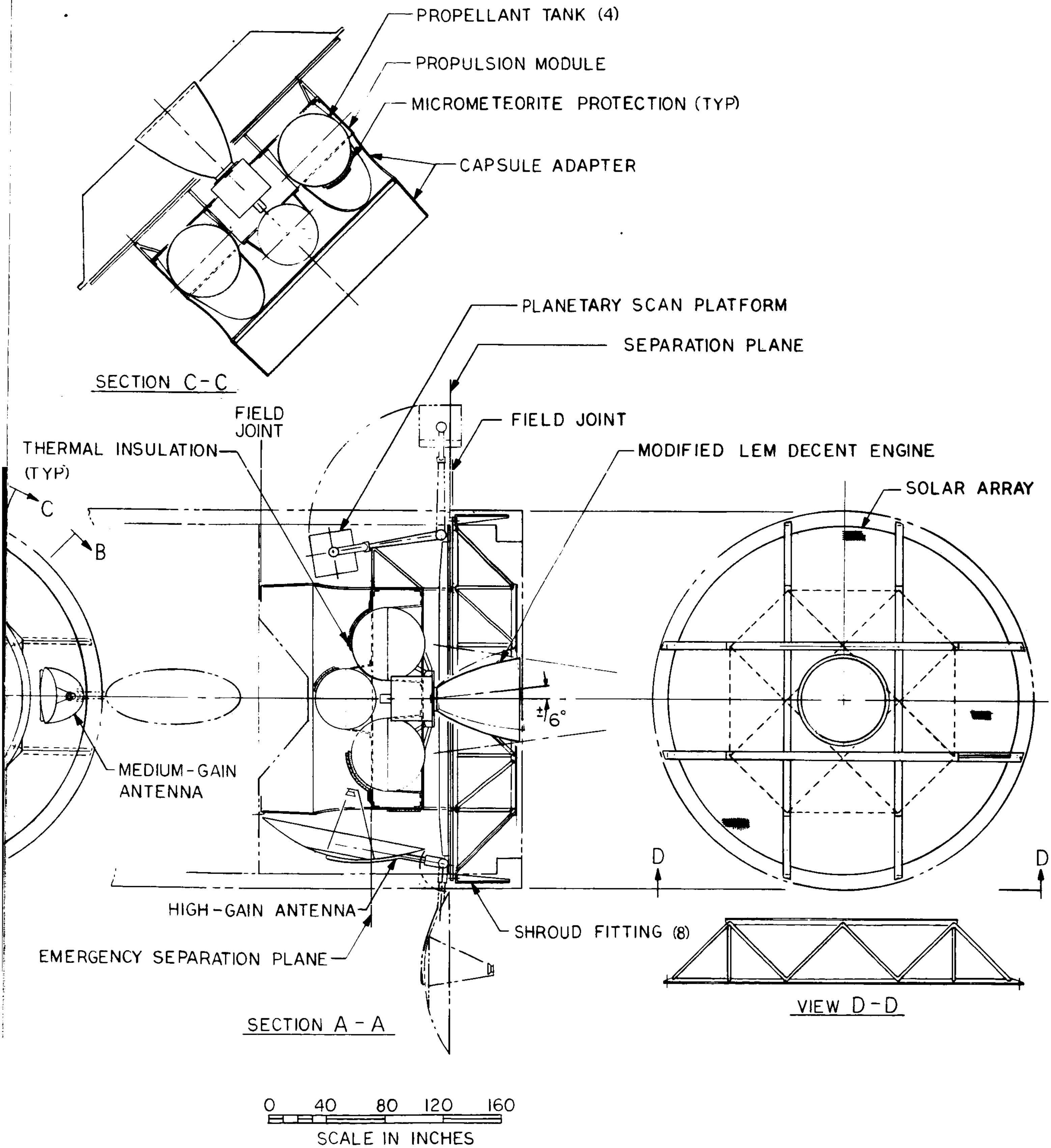


Figure 8. First Alternate Configuration (Semimonocoque)

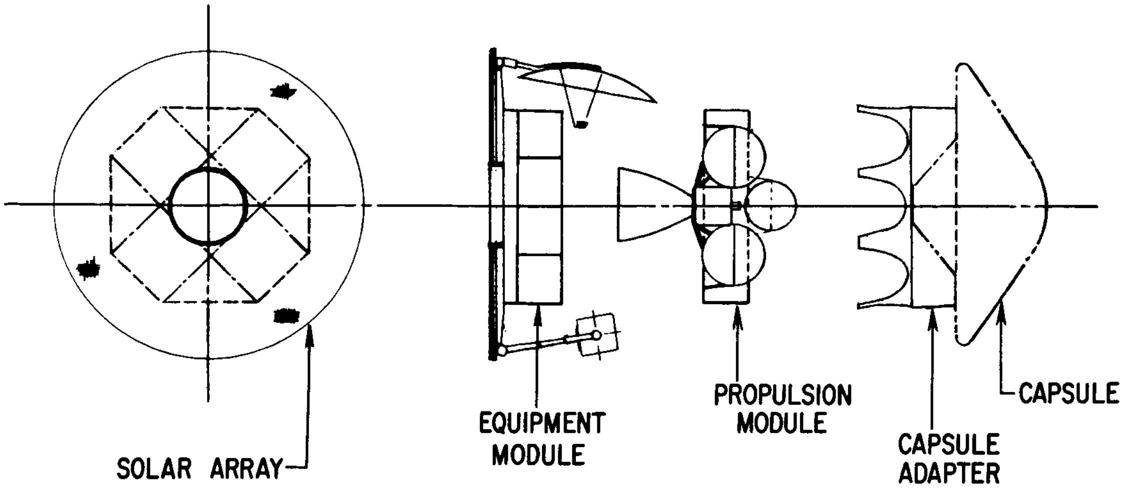


Figure 9. Modularity of First Alternate Configuration (Semimonocoque)

the antenna is shown stowed forward of the aft equipment module. This design incurs the penalty of a relatively long spacecraft-capsule interstage as well as tending to increase the overall planetary vehicle length along the roll axis, although still within the allowable envelope.

For over-the-nose shroud separation it may be necessary to change the stowed positions of the high- and medium-gain antennas and the PSP from those shown on the drawing. Controlled shroud separation may be feasible or, alternately, a combination of "clamshell" and over-the-nose shroud separations may be utilized. This latter would require one extra separation sequence and so would represent an in-line reliability penalty. Unrestrained spacecraft separation from the adapter provides an 8.5 degree fly-out angle of the engine bell with respect to the adapter structure. Initial control at separation or some modification of the adapter structural geometry may be desirable to increase this fly-out angle allowance.

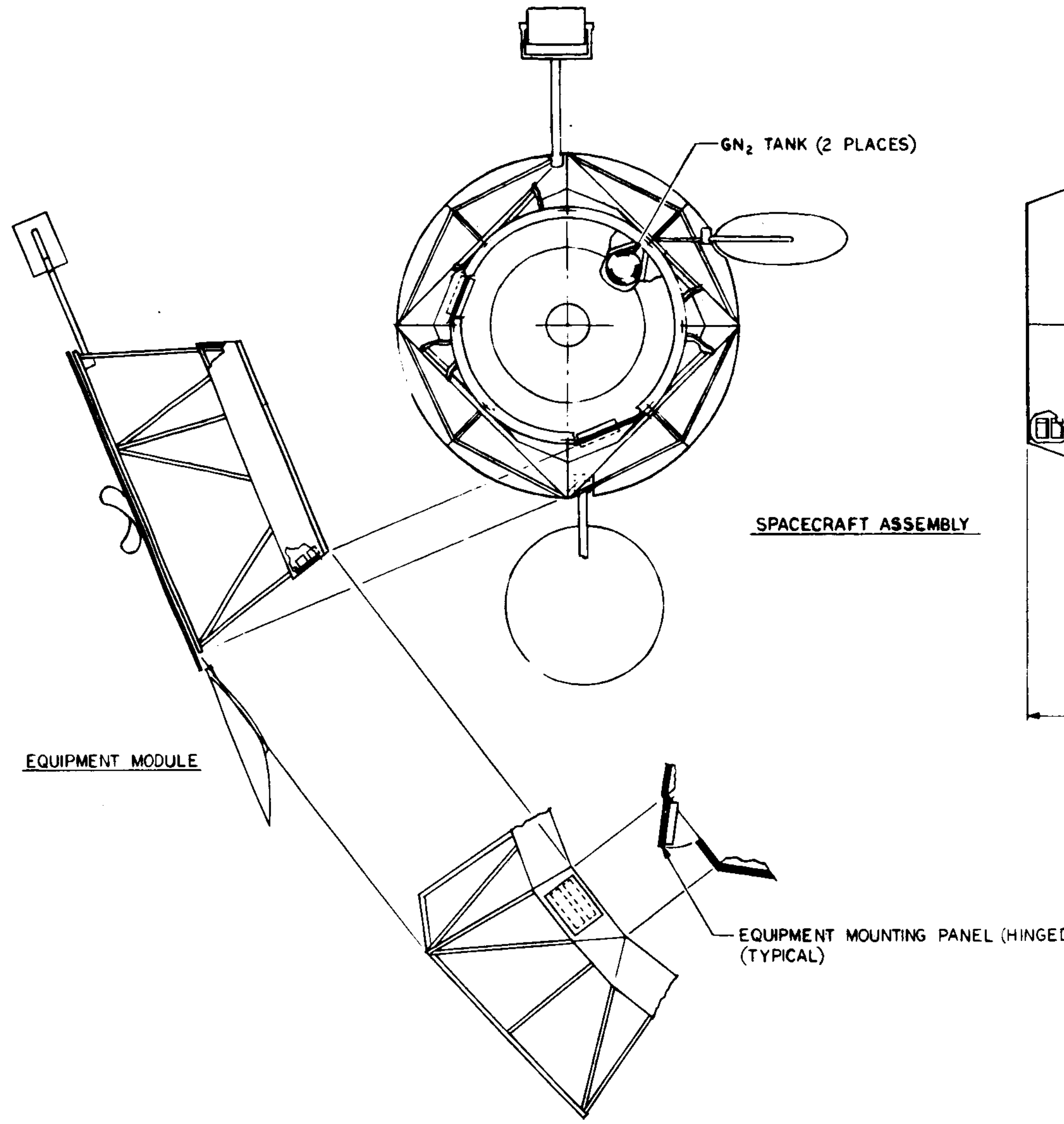
This configuration meets neither the slosh stability criterion (unless baffling is added to the propellant tanks) nor the gimbal moment arm requirement (unless the engine is moved aft).

5.3 SECOND ALTERNATE CONFIGURATION (TRUSS)

The second alternate configuration is illustrated in Figure 10. It differs from the two previous concepts by utilizing a space truss arrangement as the basic load carrying elements instead of a semimonocoque structure. No adapter is used; the truss has elements which support the planetary vehicle directly to the shroud.

One of the major problems with a truss configuration is the support of the propellant tanks. Conical tank skirts are shown in the drawing which put the axial component of each tank load into the truss at one of its joints. Additional support of the tanks is required to take out lateral components, and this adds complexity to the design. Finally, the addition of nonstructural micrometeoroid panels is required to protect the spacecraft equipment; this imposes a significant weight penalty.

Eight equally spaced hard points are provided for capsule support, as in the preferred configuration. This configuration satisfies the slosh stability criteria but not the gimbal arm requirement.


GN$_2$ TANK (2 PLACES)
SPACECRAFT ASSEMBLY
EQUIPMENT MODULE
EQUIPMENT MOUNTING PANEL (HINGE
(TYPICAL)

1/40 AD7-132

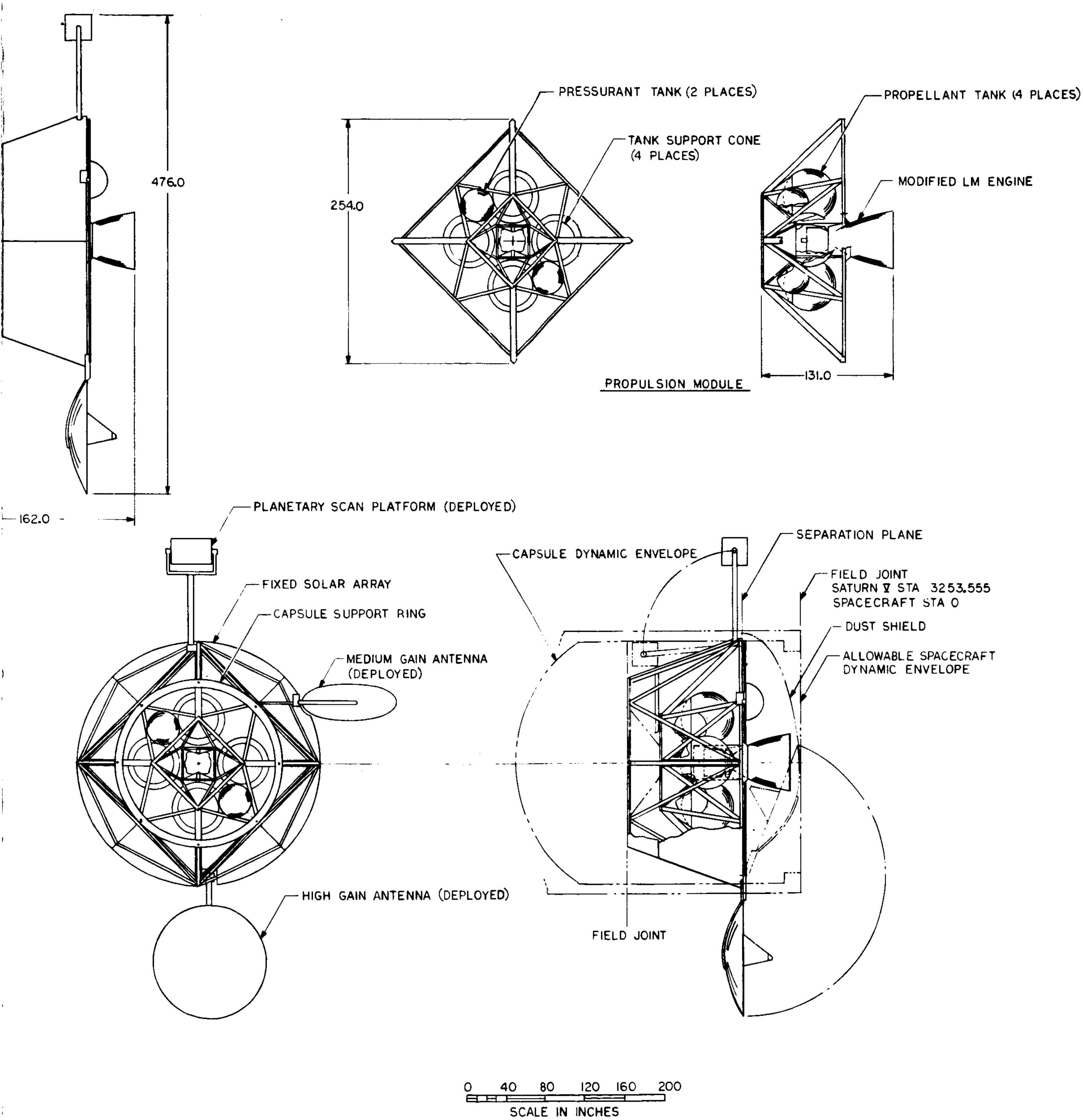


Figure 10. Second Alternate Configuration (Truss)

6. SUBSYSTEM DESCRIPTION

The elements of the flight spacecraft are summarized in Table 2. The subsystems of the preferred configuration are essentially the same as those defined by the Task B preliminary design except where particular requirements of this study have required changes. The functional requirements and characteristics imposed upon the subsystem by system considerations are described briefly below. Tables 4 and 5 list the major hardware elements of each subsystem. With the exception of the propulsion subsystem, all subsystem elements are located within the spacecraft equipment module.

6.1 SPACECRAFT PROPULSION

The spacecraft propulsion subsystem provides velocity increments for interplanetary trajectory corrections, Mars orbit insertion, and orbit trim both before and after capsule separation. The subsystem design is based on the use of a LM descent engine modified to operate in either a high or low thrust mode.

Figure 7 shows the propulsion module, with some details of tank installations and line routings. A schematic is shown in Figure 11. Propulsive capability requirements are tabulated in Section 2.5. Additional data on the propellants, pressurizing medium, and their associated tankage are given in Section 4.

6.1.1 Performance

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

- Three interplanetary trajectory corrections
- One Mars orbit insertion
- One planetary vehicle orbit trim
- One flight spacecraft orbit trim

The subsystem will provide propellant settling or propellant feed capability required by the zero gravity condition.

Table 2. Flight Spacecraft Characteristics

Item	Function	Description
Propulsion	Provides velocity increments for interplanetary trajectory corrections, Mars orbit insertion, and orbit trim before and after capsule separation	• Modified LM descent engine with ablative nozzle, helium pressurization system • Aerozene 50 and N_2O_4 propellants with mixture ratio adjusted to use equal diameter spherical tanks for both propellants, two tanks for each • Start tank in each system for zero g start capability
Guidance and Control	Three-axis attitude control of planetary vehicle using sun-Canopus, antenna and PSP orientations.	• Gyro assembly, accelerometer, Canopus sensors, sun sensors, earth detector, limb and terminator crossing detector • Gaseous nitrogen pressure vessels with redundant valves and regulators, and thrusters • Medium- and high-gain gimbal drive mechanisms • Engine gimbal electromechanical actuators • PSP gimbal drives
Power	Distributes power as required, and when sun-oriented, derives power from solar array.	• Body-mounted solar array, 165 ft^2; single crystal solar voltaic cells • Power control unit • Nickel-cadmium batteries • Redundant 4kHz inverters • Redundant 400 Hz inverters

Table 2. Flight Spacecraft Characteristics (Continued)

Item	Function	Description
S-Band Radio	Implements a two-way S-band radio link between spacecraft and DSN stations.	• S-band receiver • Redundant receivers, switches, selectors • High-gain antenna: gimballed rigid parabolic dish • Medium-gain antenna: gimballed rigid elliptical paraboloid • Cup turnstile low-gain antenna
Telemetry	Accepts data from flight capsule, spacecraft subsystems, and transducers and processes for input to the radio transmitter	• Redundant pulse code modulation encoders
Data Storage	Records digital data from science subsystem and telemetry subsystem during maneuver phase.	• Tape recorders
Command	Demodulates and decodes command signals from the communications subsystem, and provides outputs to the computer and sequencing subsystem, science equipment, other spacecraft components, and the flight capsule.	• Dual decoders and dual command detectors; approximately 170 direct discrete commands plus 21 serial commands
Computing and Sequencing	Provides time-referenced commands to flight spacecraft subsystems and flight capsule for switching and sequencing operations. Ground command override capability on all functions.	Primary and secondary units which include: • Command input register • Timing and mode logic • Function generator

Table 2. Flight Spacecraft Characteristics (Continued)

Item	Function	Description
		• Redundant accelerometer counters • Telemetry data registers • Command event registers
Structure and Mechanical	Equipment module and propulsion module structures and planetary vehicle adapter providing the required strength and rigidity, and support and alignment for equipment and appendages. Also provides support for flight capsule.	• Semimonocoque structures including exterior sandwich panels • Provide meteoroid shielding to all equipment • Shear resistant beams for propellant tankage support
Electrical Distribution	Distributes power to all subsystems.	• All spacecraft wiring harnesses, junction boxes, umbilical connectors, and in-flight jumper
Planetary Vehicle Adapter	Supports planetary vehicle with respect to the shroud.	• Tubular truss structure, supporting the planetary vehicle at eight points and interfaces with Voyager shroud at four points
Pyrotechnics	Actuation and disconnection functions, including launch vehicle-planetary vehicle separation release systems for antennas and science experiment, and propulsion valve operations.	• Electro-explosive devices, with double-dual safing circuitry and redundant arming circuits • All fragments and gas produced by the devices retained by the element • Control assembly
Thermal Control	Provides adequate thermal environment for spacecraft.	• Surface coatings and multilayer insulation • Bimetallically actuated louvers • Thermostatically controlled heaters • Ablative engine nozzle

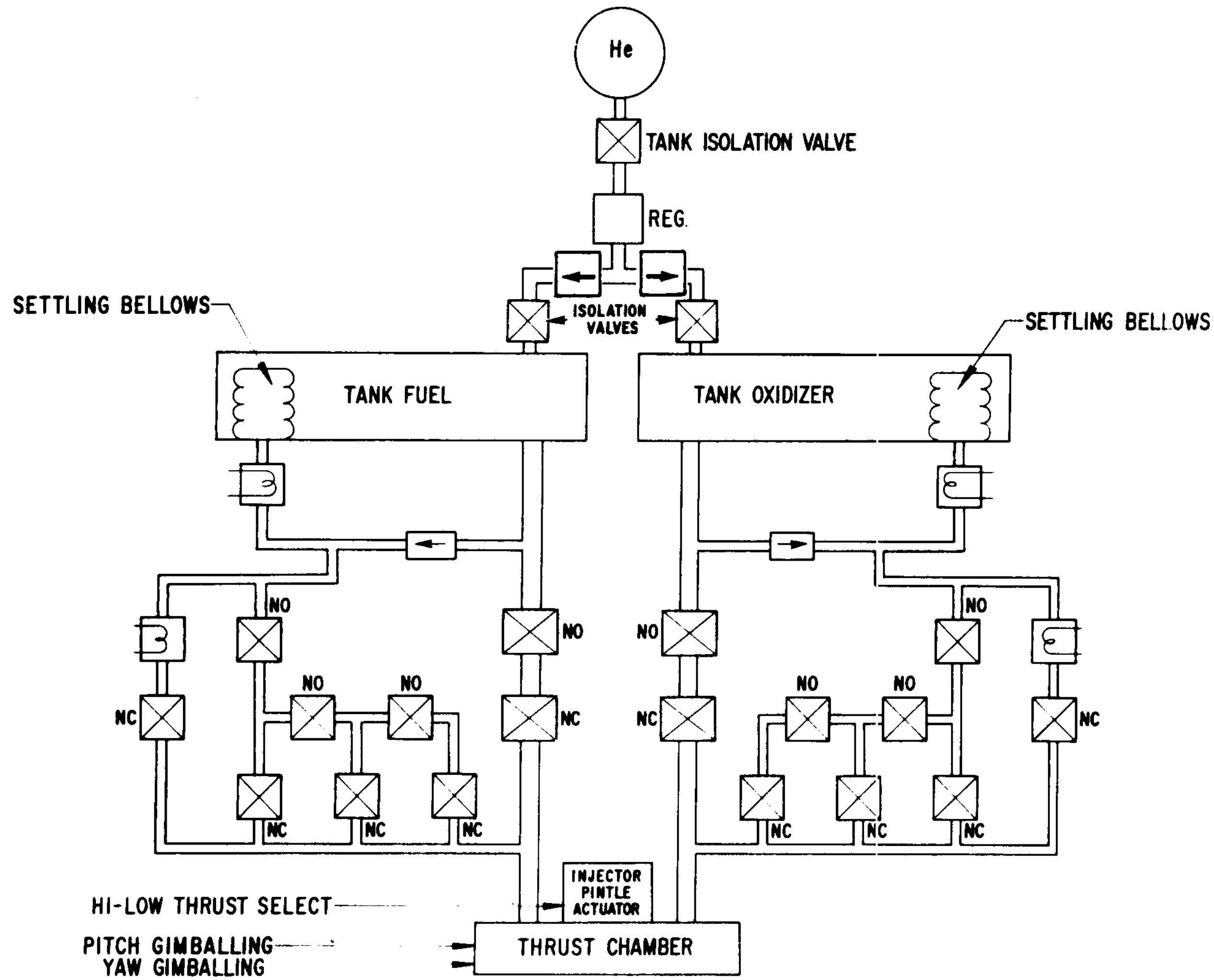


Figure 11. Propulsion Subsystem Schematic

It was determined that the 1973 mission is the worst case as far as propellant requirements are concerned. Therefore the following velocity profile has been used per the mission specification:

Maneuver	Velocity Requirement (m/sec)	(f/sec)	I_{sp} (secs)	Propellant Fraction (W_p/W_o)	Mode of Operation
Midcourse correction	210	689	385	0.072	Blow-down
Orbit insertion	1,760	5,780	305	0.446	Pressure-regulated
Orbit Trim*	150	493	285	0.052	Blow-down

The integrated propellant fraction is 0.513.

6.1.2 Configuration

The propulsion subsystem is wholly contained within the propulsion module. The engine is a LM descent engine with an ablative nozzle, modified to operate at a thrust level of 1050 or 7750 pounds.

Aerozene 50 and N_2O_4 are the propellants, with the mixture ratio adjusted such that the propellant tanks for the fuel and oxidizer are identical. Sufficient ullage volume is provided to attain 95 psia nominal at the completion of the midcourse corrections for the transit phase. Additional data on the propellant tankage is in Section 4.

Engine start tanks, lines, and valves have been based upon the Task B design, with the spherical propellant tanks sized to the maximum allowable planetary vehicle weight, i.e., 22,000 pounds less the adapter, with appropriate allowance for ullage.

The pressurization system consists of the ullage helium, and pressure-regulated helium tanks with associated valves and plumbing. Items that have been resized for this study are the helium gas and tanks. Design criteria are listed in Section 4.

* Capability with capsule not separated

The regulated external pressurization system has been sized to maintain a constant 235 psi in the propellant tanks during the insertion maneuver with account taken of the residual pressure in the tanks at the end of the midcourse maneuvers.

Weights of residual propellant have been based upon the results of earlier propulsion studies. The resulting equation is:

$$\text{Residuals} = 382 + 0.0032\ W_{pu}$$

where W_{pu} = the usable propellant weight

6.2 GUIDANCE AND CONTROL

The guidance and control subsystem controls the orientation of the Voyager spacecraft at all times after separation from the launch vehicle, including acquisition of celestial references and use of inertial devices for short term reference when required. It provides instrumentation to control velocity adjustments, implements antenna pointing control, and generates signals associated with terminator and limb crossings during Mars orbital operations.

6.2.1 Performance

6.2.1.1 Celestial References and Cruise Orientation

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

6.2.1.2 Roll Maneuvers

The guidance and control subsystem will allow the spacecraft to roll about the sun-stabilized roll axis at a constant rate to permit magnetometer and star track calibration if required. It will provide a backup during such roll maneuvers so as to preclude spacecraft spin-up and depletion of the reaction gas supply. The subsystem will also allow an incremental roll turn of ± 2 degrees upon command. In the event that the sun or Canopus is occulted, the subsystem will automatically maintain the spacecraft's attitude at the beginning of the occultation to an accuracy of 2 degrees (3σ) in each axis for a period of 3 hours.

6.2.1.3 Antenna Pointing

The high-gain and medium-gain antennas will be positioned relative to the spacecraft so that they point towards the earth as required. The antenna axes will be pointed to an accuracy of 1 degree (3σ) relative to the desired direction in the celestial reference system.

6.2.1.4 Thrust Vector Control

The guidance and control subsystem will provide thrust vector control during engine firings. Actuators will rotate the thrust direction relative to the spacecraft by means of the engine gimbals.

6.2.1.5 Velocity Increment Control

The subsystem will provide a pulse rebalance accelerometer output signal to the computer and sequencer subsystem so as to allow the engine to be shut down when the desired velocity increment has been achieved. The proportional error is to be no more than 0.1 percent.

6.2.1.6 Fine and Coarse Attitude Control

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

6.2.1.7 Slosh Stability

To eliminate an intrinsic unstable powered flight slosh mode due to phase relationship and coincidence of frequencies between the rigid body and the spherical tank propellant slosh mode dynamics, it is necessary that the spacecraft center of mass be forward of the propellant tank center. Baffles and screens within the propellant tanks can provide the required slosh stability if the spacecraft center of mass is adversely located; however, for the purposes of this study no damping aids were to be required. Section 7 reviews the slosh stability margins for each of the three configurations for the worst case of early capsule jettison; with the capsule present the margins are increased in each case.

6.2.2 Configuration

The guidance and control subsystem consists of a gyro assembly, accelerometer, Canopus sensors, sun sensors, earth detector, link and terminator crossing detection, attitude control pneumatic system consisting of gaseous nitrogen pressure vessels with redundant valves and regulators with thrusters, medium- and high-gain antenna gimbal drive mechanisms, and engine gimbal electromechanical actuators. All this equipment is mounted in the equipment module.

6.3 POWER SUBSYSTEM

The power subsystem distributes power as required throughout the flight spacecraft. When sun-oriented, the spacecraft derives power from photovoltaic solar cells arranged on panels. When not sun-oriented, such as during maneuvers or during planetary orbit when the sun is eclipsed by Mars, power is derived from batteries. The batteries also provide power for reacquisition in the event of a noncatastrophic loss of attitude control.

6.3.1 Performance

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

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

The battery energy requirements in the following table consist of average spacecraft loads plus short duration loads for propulsion valves and engine gimbal actuators:

Battery Energy Requirements

Mission Phase	Battery Output Energy Required (w-hrs)
Prelaunch, launch, injection, and acquisition	944
Trajectory correction	1103
Orbit insertion	1283
Planetary vehicle orbital trim	1084
Capsule separation	1112
Flight spacecraft orbital trim	1082
Eclipses in Mars orbit (150 cycles)	1270

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

6.3.2 Configuration

The power subsystem consists of the solar array, power control unit, nickel-cadmium batteries, redundant 400 and 4 kHz inverters. The solar array is body-mounted to the equipment module, and contains 165 ft^2 of single crystal solar voltaic cells with 6-mil cover glass.

6.4 S-BAND RADIO

The S-band radio subsystem implements a two-way radio link between the spacecraft and the DSN stations to provide telemetry to earth; command capability to the flight spacecraft; and doppler, ranging, and angular tracking. It consists of antennas, receivers, transmitters, and related microwave equipment plus redundancy control and selection circuitry. The basic frequencies are 2100 to 2120 MH_z receive and 2290 to 2300 MH_z transmit.

6.4.1 Performance

Uplink communications with each planetary vehicle will be accommodated from interplanetary trajectory injection through six months after Mars orbit insertion by a low-gain nonsteerable antenna utilizing a 210-foot ground antenna and 100-kw ground transmitter. Hemispheric coverage relative to the spacecraft zero cone angle axis (aft direction) is provided by the low-gain antenna.

Ranging will be available simultaneously with telemetry during cruise but not concurrently with high data rate in Mars orbit. Ranging is not required through the low-gain antenna.

Downlink telemetry is required from each planetary vehicle within the view capability of the DSN, from prelaunch to interplanetary trajectory injection. After interplanetary trajectory injection the spacecraft allows simultaneous doppler tracking, telemetry, and command of both planetary vehicles by the DSN.

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

6.4.2 Configuration

The S-band radio subsystem includes three antenna (low-, medium-, and high-gain) which are shared for uplink and downlink communications. Each antenna is connected through a diplexer to a phase-locked receiver which synchronously detects the uplink ranging data and command subcarriers and provides a submultiple of the received frequency to the downlink transmitters for retransmission to permit a two-way doppler measurement at the DSN stations.

The detected command subcarriers are connected through receiver selection logic to redundant command detectors. Selection logic is controlled by the command sequencer and a receiver output signal indicating that carrier phase lock has been achieved (signal present). The command detectors synchronize to the pseudo-noise code and coherently extract the command bits. The detected command bits are then transferred to the command decoders for subsequent processing.

The downlink incorporates two 50-watt transmitters for normal communications plus a 1-watt unit for launch operations and backup support. Switching is provided so that any of the three transmitters can be connected through diplexers to any of the three antennas.

A modulation combiner combines the ranging data from the receiver outputs with the biphase-modulated telemetry subcarriers from the telemetry data handling subsystem. The resultant composite signal is applied to the phase modulator in the selected exciter. Switching is provided to inhibit ranging or telemetry transmission if required by interference or power budget constraints.

6.5 TELEMETRY

The telemetry subsystem accepts data from the flight capsule and other flight spacecraft subsystems and transducers and processes this data for input to the radio transmitter as a serial pulse-code-modulated digital signal.

6.5.1 Performance

The following six data rates will be provided:

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

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

- 27 channels sampled at the frame rate
- 70 channels sampled at 1/20 of the frame rate

- 126 channels sampled at 1/200 of the frame rate
- 209 channels sampled at 1/400 of the frame rate

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

6.5.2 Configuration

The telemetry subsystem consists of two redundant PCM encoders. Each encoder has the following major elements:

- Programmer
- Frequency generator
- Analog multiplexer
- Analog-to-digital converter
- Pseudo-noise generator
- Digital multiplexer
- Capsule data buffer
- Modulator-mixer

6.6 DATA STORAGE

The data storage subsystem records digital data from the flight spacecraft science subsystem and engineering and cruise science data from the flight spacecraft telemetry subsystem. The capability of recording the high rate capsule data from the relay link receiver is within the capsule system and not part of the flight spacecraft data storage subsystem.

The subsystem contains six tape recorders, each assigned a separate function. Each recorder is independent of the others, with separate power, control, and signal lines. The data storage subsystem will have a total science storage capacity of 2.225×10^8 bits, as follows:

- Television data recorder: two recorders, each with a capacity of 10^8 bits
- Spectrometer data recorder: one recorder, with a capacity of 10^7 bits
- IR scanner data recorder: one recorder, with a capacity of 10^7 bits
- Fields and particles recorder: one recorder, with a capacity of 10^6 bits
- Maneuver recorder: one recorder, with a capacity of 1.5×10^6 bits

6.7 COMMAND

The command subsystem demodulates command signals from the output of the S-band radio subsystem, decodes the resulting digital commands, and provides outputs to the computer and sequencer subsystem, the science subsystem, the flight capsule, and other spacecraft components as required by the addresses in the command data. Dual decoders and command detectors are included.

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

- Direct discrete commands: approximately 170
- Quantitative commands (serial outputs)
- 7-bit serial words to each of two science decoders
- 20-bit serial words to each of six registers in the computer and sequencer
- 12-bit serial words to three antenna pointing registers in the guidance and control
- 32-bit serial words to two computer and sequencer command input registers, two science sequencer command input registers, and six flight capsule registers

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

Synchronization acquisition time will be minimized, and the command subsystem will process command words at the rate of 1 bit/sec $\pm$10 percent, after detector lock has been acquired.

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

6.8 COMPUTING AND SEQUENCING

The computing and sequencing subsystem provides time-referenced commands to other flight spacecraft subsystems and the flight capsule to achieve required switching and sequencing for maneuvers, science sequencing, antenna alignment confirmation, cruise operations timing, and Canopus acquisition calibration. There is ground command override capability on all the functions of the subsystem.

6.8.1 Performance

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

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

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

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

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

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

6.8.2 Configuration

The computing and sequencing subsystem consists of primary and backup units containing the following major elements:

- Command input register
- Timing and mode logic
- Function generator
- Redundant accelerometer counters
- Telemetry data registers
- Command event registers

6.9 STRUCTURAL AND MECHANICAL

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

The structure provides for a propulsion module and an equipment module which can be separately assembled, integrated and tested.

6.10 PYROTECHNIC

The pyrotechnic subsystem includes the electro-explosive devices and their associated firing circuitry to provide various actuation and disconnection functions for the flight spacecraft. The subsystem is utilized as follows:

- Launch vehicle-planetary vehicle separation (commanded from launch vehicle)
- Antenna release (3)
- PSP release and uncaging
- Propulsion valve openings (N.C. valves)
- Propulsion valve closings (N.O. valves)
- Jettisoning of science covers (if required)

6.10.1 Performance

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

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

6.10.2 Configuration

The pyrotechnic subsystem includes a control assembly and the electro-explosive devices, which have double-dual safing circuitry and redundant arming circuits. The pyrotechnic control assembly includes the safe-arm circuit which controls application of power to the subsystem, the power conversion circuitry which rectifies the AC input to provide the proper DC voltage for the energy storage circuits, and solid state firing circuits which provide initiating current to individual explosive devices on command. The attach-release devices are mechanical assemblies which utilize the explosive pressure impulse as the source of motive power.

6.11 TEMPERATURE CONTROL SUBSYSTEM

The temperature control subsystem provides an adequate thermal environment for the entire flight spacecraft during all phases of the Voyager mission. The following design features are associated with spacecraft temperature control:

- Surface finishes to attain desired thermal properties particularly of external equipment
- Appropriate location of electronic components
- Structural design to achieve various degrees of thermal coupling, generally close coupling within the main compartment and poor coupling between the main compartment and solar array and capsule, and between the external equipment and the solar array substrate structure.
- Multilayer aluminized Mylar insulation on the equipment module and propulsion module to provide an acceptable environment for the electronic equipment, propellant lines and tanks, and to minimize thermal gradients within the compartment envelope
- Bimetallic-actuated louvers to regulate electronic equipment temperatures
- Thermostatically controlled heaters
- Ablative engine nozzle design

Temperature control of the main compartment is achieved by insulation of the external surfaces of the equipment module, the propulsion module, and the capsule adapter. Bimetallic-actuated louvers are used to provide temperature control for the electronic equipment. The electronic equipment and the propulsion tanks and lines are radiatively coupled for thermal control purposes by the use of thermal coatings.

The solar array on the preferred configuration will be uninsulated. Appropriate thermal properties on the front and back side of the array will limit the maximum temperature to acceptable values. The mass of the substrate and the array combined with the thermal properties of the front and back is adequate to maintain the solar array above its allowable lower temperature limit during a solar eclipse while in Mars orbit.

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

6.12 ELECTRICAL DISTRIBUTION

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

6.13 PLANETARY VEHICLE ADAPTER

The planetary vehicle adapter comprises all spacecraft-supplied structure, cabling, and hardware located between a planetary vehicle in-flight separation joint and the associated points of attachment to the launch vehicle nose fairing. These are illustrated in the configuration drawings (Figures 5, 8, and 10).

6.14 SCIENCE SUBSYSTEM

The flight spacecraft will be designed to accommodate and provide support for a science subsystem that is capable of accomplishing a specified science mission. Detailed information and requirements data pertinent to integration of the science subsystem into the spacecraft are beyond the scope of this report. However, a reference science subsystem has been defined within the framework of the present preliminary design effort. The science subsystem is made up of the following:

- Science experiment equipment
- Data automation equipment
- Science command decoding equipment
- Science power switching electronics
- Planetary scan platform (PSP) and control
- Science deployment
- Fixed science packages

7. MASS PROPERTIES

Mass property estimates have been prepared for the three configurations under investigation. Estimates of weight and longitudinal center-of-mass location have been prepared for each, with estimates of inertia computed for the preferred configuration.

Weight estimates for the three configurations are summarized in Table 3; Tables 4 and 5 give detailed breakdowns. Comparing these three estimates it can be seen that the two semimonocoque configurations have satisfactory weight margins compared to the specification requirement of 22,000 pounds, while the truss configuration is marginally satisfactory.

In Table 6 are the moments of inertia and the longitudinal center-of-mass location for the preferred configuration. Values are shown for ignition and burnout, both with and without the flight capsule.

Figures 12, 13, and 14 are plots of spacecraft stability versus percentage of tank volume utilized for the three configurations. Only the semimonocoque configuration fails in this respect.

Figure 15 illustrates the coordinate system used in the mass property derivations.

In order to maintain a stable slosh mode during the mission, the spacecraft center-of-mass must at all times be forward of the propellant tank center (for spherical tanks). Additionally, a minimum gimbal moment arm of at least 32 inches is required for thrust vector control. These conditions can best be obtained by locating the equipment as far forward and the propellant tanks as far aft as possible. This has been one of the governing requirements for the configuration. Figures 12, 13, and 14 have been prepared to show the degree to which this has been achieved for the three configurations considered. The data are based upon a worst case condition, i.e., assuming an emergency capsule separation prior to midcourse corrections.

In the derivation of the flight spacecraft bus weights, the following subsystem weights were used from the Task B study:

Table 3. Planetary Vehicle Summary Weight Estimates

Item	Weight (lb)		
Configuration	AD 7-122	AD 7-100A	AD 7-125
Flight Capsule	5,000.0	5,000.0	5,000.0
Flight Spacecraft Science Subsystem	400.0	400.0	400.0
Flight Spacecraft Capsule Bus Support Equipment	50.0	50.0	50.0
Flight Spacecraft Equipment Module	1,980.3	2,433.0	2,821.7
Structure	502.6	768.8	1,157.0
Thermal control	132.2	188.7	233.6
Pyrotechnics	37.0	37.0	37.0
Power	364.1	452.9	373.2
Electrical distribution	228.9	228.9	228.9
Guidance and control	268.5	268.5	268.5
Communications	125.5	125.5	125.5
Telemetry and command	90.5	90.5	90.5
Computing and sequencing	36.0	36.0	36.0
Balance weights	15.0	15.0	15.0
Contingency	180.0	221.2	256.5
Flight Spacecraft Propulsion Module	13,453.4	13,142.1	13,426.8
Structure	512.0	258.1	385.4
Thermal control	29.4	29.4	29.4
Engine and valves	426.5	426.5	426.5
Propellant feed assembly	363.1	362.8	365.6
Pressurization system	414.4	413.9	418.9
Contingency	174.5	149.1	162.6
Residuals (propellant and helium)	462.5	462.3	463.4
Usable propellant	11,071.0	11,040.0	11,175.0
Planetary Vehicle Gross Weight	20,883.7	21,025.1	21,698.5
Planetary Vehicle Adapter	403.0	463.3	199.9
Planetary Vehicle Weight Margin	713.3	511.6	101.6
Planetary Vehicle Plus Adapter Gross Weight	22,000.0	22,000.0	22,000.0

Table 4. Flight Spacecraft Equipment Module Weight Estimates Breakdown

Item	Weight (lb)		
Configuration	AD 7-122	AD 7-100A	AD 7-125
Equipment Module Structure	502.6	768.8	1,157.0
Capsule support	18.8	126.5	-
Equipment panels	100.0	100.0	100.0
Hinges	2.2	2.2	2.2
Latches	4.8	4.8	4.8
Mounting rails	78.0	78.0	78.0
Structure equipment support	28.7	-	-
Meteoroid protection panels	208.8	198.5	748.0
Corner members	8.6	-	-
Attachments and miscellaneous	44.0	44.0	44.0
Miscellaneous supports	8.7	8.7	8.7
Solar array support linkage	-	30.0	-
Aft equipment module	-	176.1	-
Truss members	-	-	157.2
Solar array supports	-	-	14.1
Thermal Control	132.2	188.7	233.6
Insulation	106.8	163.3	208.2
Louvers	17.1	17.1	17.1
Heaters and thermostats	2.0	2.0	2.0
Attachments and miscellaneous	6.3	6.3	6.3
Pyrotechnics	37.0	37.0	37.0
Release and deployment system	7.7	7.7	7.7
Electrical connectors	2.2	2.2	2.2
Explosive valve pyrotechnic (18)	0.6	0.6	0.6
Pyrotechnic control assembly (1)	25.0	25.0	25.0
Attachments and miscellaneous	1.5	1.5	1.5
Power Supply	364.1	452.9	373.2
Solar array	132.0	220.8	141.1
Battery (3)	138.0	138.0	138.0
Inverters	20.6	20.6	20.6

Table 4. Flight Spacecraft Equipment Module Weight Estimates Breakdown (Continued)

Item	Weight (lb)		
Configuration	AD 7-122	AD 7-100A	AD 7-125
Battery regulator (3)	42.0	42.0	42.0
Power control unit (1)	8.0	8.0	8.0
Shunt element assembly (2)	16.0	16.0	16.0
Power distribution box (1)	7.5	7.5	7.5
Integration	228.9	228.9	228.9
Cabling and connectors (4)	190.0	190.0	190.0
Junction box (1)	20.0	20.0	20.0
Umbilical	8.0	8.0	8.0
Cabling channels	10.9	10.9	10.9
Guidance and Control	268.5	268.5	268.5
Gyro reference assembly (1)	10.0	10.0	10.0
Accelerometer (1)	1.0	1.0	1.0
Guidance and control electronics	13.0	13.0	13.0
Canopus sensor (2)	12.0	12.0	12.0
Fine sun sensor (1)	0.2	0.2	0.2
Coarse sun sensor (4)	0.8	0.8	0.8
Earth detector (1)	0.3	0.3	0.3
Solenoid valve (16)	19.5	19.5	19.5
Pressure vessel (2)	60.0	60.0	60.0
Nitrogen gas	49.0	49.0	49.0
Regulator (4)	3.5	3.5	3.5
Thrusters (4)	4.0	4.0	4.0
Lines (2)	4.0	4.0	4.0
High-gain drive assembly	32.0	32.0	32.0
Medium-gain drive assembly	17.0	17.0	17.0
TVC actuator (2)	36.0	36.0	36.0
Limb and terminator crossing detector	1.2	1.2	1.2
Antenna drive electronics	5.0	5.0	5.0

Table 4. Flight Spacecraft Equipment Module Weight Estimates Breakdown (Continued)

Item	Weight (lb)		
Communications	125.5	125.5	125.5
Modulator exciter (2)	6.0	6.0	6.0
4-port hybrid ring and power monitor (1)	0.6	0.6	0.6
1-watt transmitter and power monitor (1)	3.5	3.5	3.5
Power amplifier power supply and RF monitor (2)	15.6	15.6	15.6
Transmitter selector (1)	1.0	1.0	1.0
S-band receiver (3)	15.0	15.0	15.0
Receiver selector (1)	1.0	1.0	1.0
Circulator switch (4)	7.3	7.3	7.3
Diplexer (3)	3.9	3.9	3.9
Low-gain antenna (1)	3.0	3.0	3.0
Medium-gain antenna (1)	13.4	13.4	13.4
High-gain antenna (1)	55.2	55.2	55.2
Telemetry and Command	90.5	90.5	90.5
Tape recorders (6)	72.0	72.0	72.0
PCM encoder (2)	8.0	8.0	8.0
Decoder (2)	5.3	5.3	5.3
Command detector (2)	5.2	5.2	5.2
Computing and Sequencing	36.0	36.0	36.0
Balance Weights	15.0	15.0	15.0
Contingency (10 percent)	180.0	221.0	256.5
Gross Equipment Module Weight	1,980.3	2,431.1	2,821.7

Table 5. Flight Spacecraft Propulsion Module Weight Estimates Breakdown

Item	Weight (lb)		
Configuration	AD 7-122	AD 7-100A	AD 7-125
Propulsion Module Structure	512.0	258.1	385.4
Lower ring	11.6	-	-
Meteoroid protection panels	247.3	-	134.3
Reaction control supports	11.7	11.7	11.7
Attachments and miscellaneous	14.3	14.3	14.3
Base structure	25.0	25.6	-
Internal structure	79.0	90.7	-
Corner members	7.3	-	-
Tank supports	84.0	84.0	69.8
Engine supports	31.8	31.8	31.8
Trusses	-	-	123.5
Temperature Control	29.4	29.4	29.4
Insulation (Refrasil)	24.3	24.3	24.3
Heaters and thermostats	2.0	2.0	2.0
Attachments and miscellaneous	3.1	3.1	3.1
Engines and Valves	426.5	426.5	426.5
Injector	29.3	29.3	29.3
Combustion chamber assembly	280.0	280.0	280.0
Injector pintle actuator	4.0	4.0	4.0
Propellant lines and ducts	13.9	13.9	13.9
Electrical harness	9.0	9.0	9.0
Instrumentation	2.7	2.7	2.7
Gimbal assembly	27.2	27.2	27.2
Hardware-engine integration	9.4	9.4	9.4
Fuel control module	15.5	15.5	15.5
Oxidizer control module	18.0	18.0	18.0
Solenoid valves (8)	14.0	14.0	14.0
Quad check valves (2)	1.0	1.0	1.0
Trim orifices (2)	0.5	0.5	0.5
Filter (2)	2.0	2.0	2.0

Table 5. Flight Spacecraft Propulsion Module Weight Estimates Breakdown (Continued)

Item	Weight (lb)		
Configuration	AD 7-122	AD 7-100A	AD 7-125
Propellant Feed Assembly	363.1	362.8	365.6
Propellant tanks (4)	292.3	291.6	294.8
Lines and valves	48.2	48.6	48.2
Engine start tanks (2)	22.6	22.6	22.6
Pressurization System	414.4	413.9	418.9
Valves, regulator, etc.	33.2	33.2	33.2
Lines, fill and vent	13.2	13.2	13.2
Tank	368.0	367.5	372.5
Contingency	174.5	149.1	162.6
Residuals	462.5	462.3	463.4
Propellant (including start tanks)	417.4	417.3	417.8
Helium	45.1	45.0	45.6
Propulsion Module at Burnout	2,382.4	2,102.1	2,251.8
Usable Propellant	11,071.0	11,040.0	11,175.0
Propulsion Module at Ignition	13,453.4	13,142.1	13,426.8

Table 6. Moment of Inertia, AD 7-122

Condition	Weight (lb)	Longitudinal c.g. $\bar{Z}$ (in.)	Moments of Inertia (slug ft^2) I_x	I_y	I_z
Without Capsule					
Ignition	16,597	114.4	7,322	5,679	9,525
Burnout	5,526	128.3	4,791	3,163	5,218
With Capsule					
Ignition	21,597	141.4	23,094	21,448	14,037
Burnout	10,526	177.1	15,265	13,626	9,729

- Electrical distribution
- Guidance and control
- Communications
- Telemetry and command
- Computing and sequencing
- Balance weights

Weights for the flight capsule, flight spacecraft science subsystem, and flight spacecraft capsule bus support equipment were obtained from the Mission Specification Document. The bases for weights of the other subsystems are described in Section 6. A conservative contingency or growth provision factor of 10 percent of the dry weight of the equipment and propulsion modules has been added.

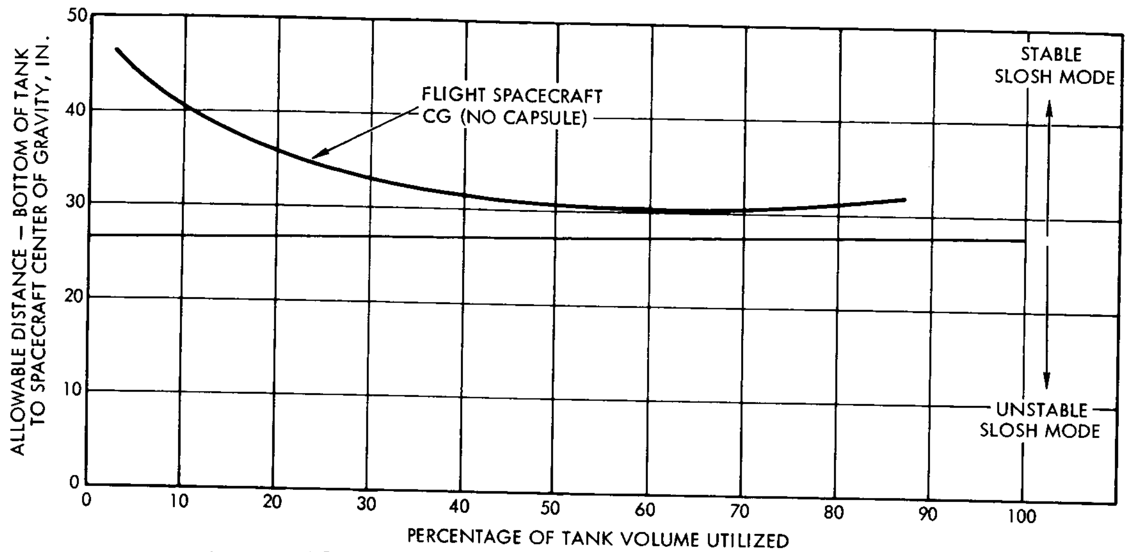


Figure 12. Allowable Center of Gravity Location for Preferred Configuration

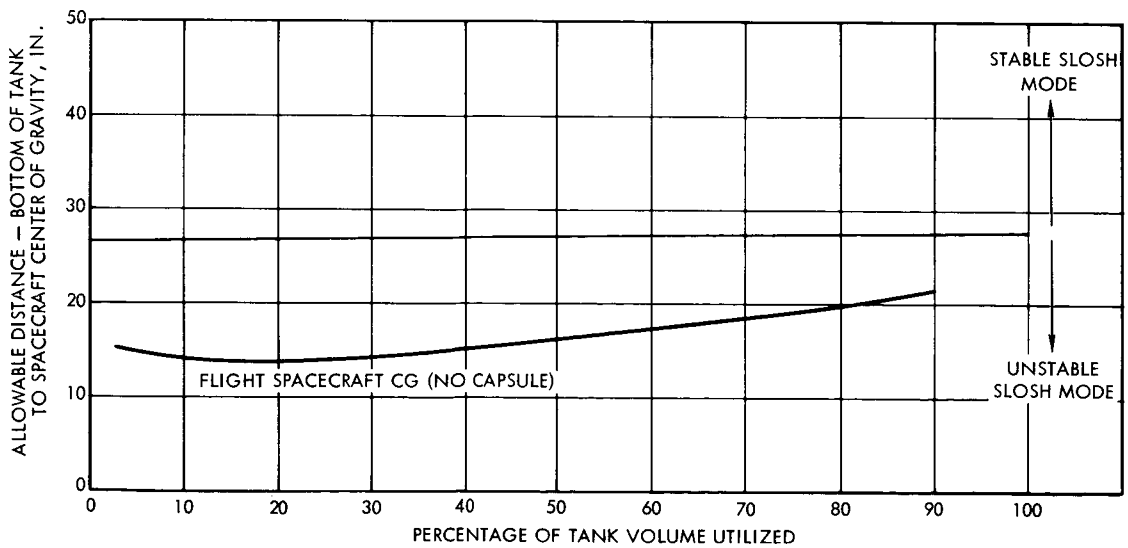


Figure 13. Allowable Center of Gravity Location for Semimonocoque Configuration

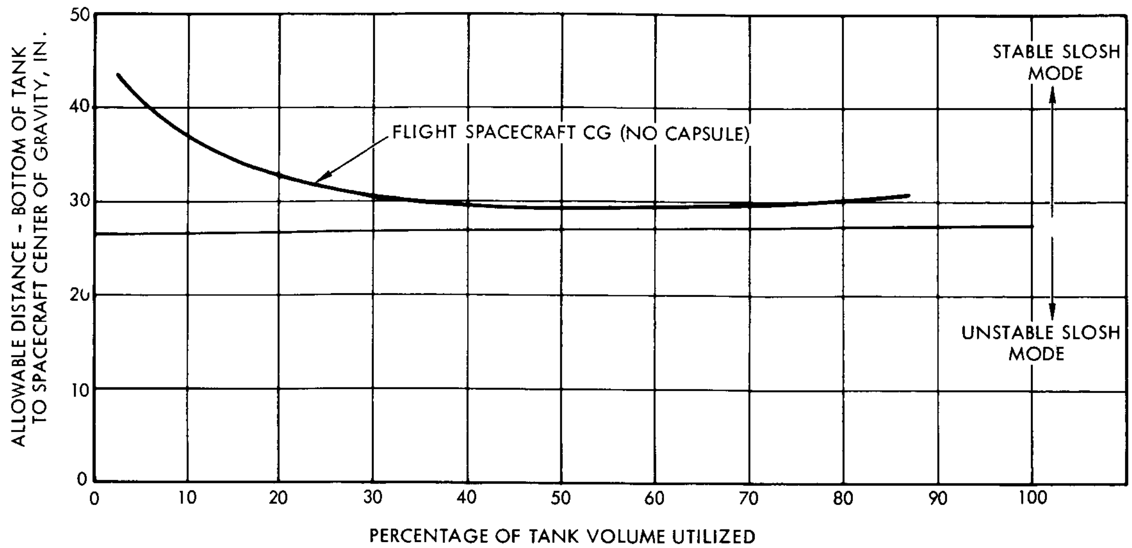


Figure 14. Allowable Center of Gravity Location for Truss Configuration

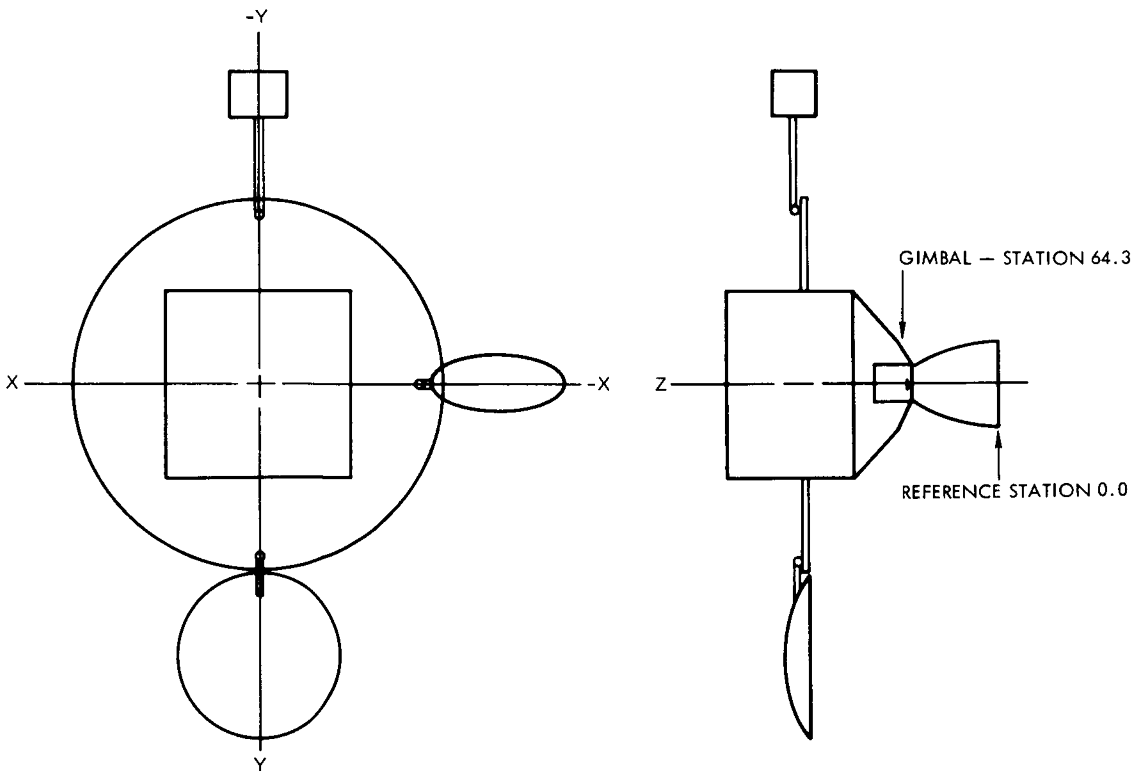


Figure 15. Axis Coordinate System

REFERENCES

1. "Performance and Design Requirements for the 1973 Voyager Mission, General Specification for," 1 January 1967, Jet Propulsion Laboratory.

2. "Voyager Environmental Prediction Document," draft dated 26 October 1966, Jet Propulsion Laboratory.