NASA TECHNICAL MEMORANDUM

NASA TM X-73502

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MODULAR THRUST SUBSYSTEM APPROACHES TO SOLAR ELECTRIC PROPULSION MODULE DESIGN

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TECHNICAL PAPER to be presented at the
Twelfth International Electric Propulsion Conference sponsored by the
American Institute of Aeronautics and Astronautics
Key Biscayne, Florida, November 15-17, 1976

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Abstract

Three approaches are presented for packaging the elements of a 30 cm ion thruster subsystem into a modular thrust subsystem. The individual modules, when integrated into a conceptual solar electric propulsion module are applicable to a multimission set of interplanetary flights with the Space Shuttle/Interim Upper Stage as the launch vehicle. The emphasis is on the structural and thermal integration of the components into the modular thrust subsystems. Thermal control for the power processing units is either by direct radiation through louvers in combination with heat pipes or an all heat pipe system. The propellant storage and feed system and thruster gimbal system concepts are presented. The three approaches are compared on the basis of mass, cost, testing, interfaces, simplicity, reliability, and maintainability.

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Introduction

During the Advanced Systems Technology program conducted by NASA up to 1974, spacecraft system design studies were undertaken[1, 2] which in part focused on the integration of a solar electric propulsion (SEP) system into interplanetary spacecraft. The spacecraft design philosophy adopted by the Jet Propulsion Laboratory was to physically separate and house the SEP systems in a SEP module and the spacecraft engineering and support systems in a mission module.[3] The elements of the thrust subsystem (30 cm ion thrusters, thruster gimbal system, power processing units, and propellant storage and distribution) were individually integrated into a SEP module.

A continuing task at the Lewis Research Center has been to define the critical technology elements of the thrust subsystem interfaces and evaluate their solution. As part of this task, studies of various approaches to packaging the elements of the thrust subsystem and integrating them into a SEP module have been performed. In addition to defining the interface for the thrust subsystem, these studies have provided baseline information for systems and mission analyses.

The concept of packaging the elements of the thrust subsystem in modules and in turn packaging the modules into a SEP module was first presented by Sharp.[4] The use of a modular thrust subsystem in an on-going series of SEP spacecraft would accrue many benefits. A qualification test program for the thrust modules could be developed that would envelop a multimission set. Because essentially identical modules could be used for several on-going missions, only a flight acceptance test program would need to be performed on the modules to be used for each mission. It would therefore be possible to use the flight spares of one mission as the flight units of the following mission thus effecting a large cost savings. Reliability of the follow on missions would be greatly enhanced since virtually identical hardware would be used.

In this paper, a comparative description of three approaches to a modular thrust subsystem is given, including the Thrust Subsystem Module presented in Ref. 4. Also, some aspects of the integration of the approaches into a conceptual SEP module are defined. Descriptions of the thruster and the power processor are widely available in the literature. In this paper, subsystem descriptions are presented only for the thruster gimbal system and propellant storage and distribution system concepts. Finally, the benefits of the modular thrust subsystem concept are reviewed and the three approaches to the module are compared on the basis of mass, cost, testing, interfaces with the spacecraft, simplicity, maintainability, and reliability.

SEP Module Designs

Functional and Configuration Requirements

The SEP module has the primary functions of generating and distributing photovoltaic power to the power processing units, converting the power into a directed thrust, providing control torques with ion thrusters about the three principal axes of the spacecraft, and storing and distributing mercury propellant to the 30 cm ion thrusters.[3] For current missions under consideration, a total of six or eight thrusters and power processors is required. Figure 1(a) shows schematically a side-view of a six thruster SEP module using the JPL assembly approach. The thrusters are arranged in pairs to form a 2- by 3-matrix of thrusters when viewed from the bottom of the SEP module. One of the attractive features of the 2- by n-thruster configuration discussed in Ref. 3 was that the number of thrusters in the assembly may be increased or decreased without changing the basic design concept of the SEP module. However, to accommodate the change in the number of thrusters and power processors, changes are required in the lengths of the thrust assembly, power assembly, and SEPM adapter structure.

STAR category 20

Figure 1(b) shows the modular approach to the SEP module which integrates the thrust subsystem components in the vertical direction and retains the basic configuration concept of the JPL approach. The growth feature of the SEP module is enhanced by using the modular thrust subsystem approach, because to increase to an eight thruster configuration, only the design of the interface truss is affected and not the individual modular thrust subsystems.

Figure 2 shows the elements of the SEP module discussed herein. The solar array system has a beginning of life power requirement, depending upon the mission, of between 18 and 25 kW. The solar array is a foldout design extended via a deployable mast and is based on the current technology typified by the 25 kW SEP Solar Array System described in Ref. 5. The interface truss is a structural interface between the modular thrust subsystems, the solar array system, and the rest of the spacecraft. Power distribution components and a SEP module control interface unit are located within the interface truss. Because of their mission dependency, attitude control sensors and electronics and reaction control components normally considered a part of the SEP module are not considered in this paper.

Design Requirements

The design requirements for the modular thrust subsystems and the SEP module have been specified to envelop a planetary multimission set currently under consideration. These missions include Tempel II and Flora rendezvous, Saturn, Jupiter, and Mercury orbiters, and a 1 A. U. solar observatory. Table I lists the thermal environment which bounds the requirements for the mission set. It is assumed that the ion thrusters are qualified for these environments and that the thrusters can be thermally isolated from the remainder of the modular thrust subsystem. The worst case condition for the power processor thermal control system design is the 80° C solar array temperature encountered between 0.3 and 0.85 A. U. for the Mercury orbiter mission. At 1 A. U. the solar array temperature is 50° C with a solar array tilt angle of zero degrees. Between 1 and 0.85 A. U. the solar array temperature increases to 80° C and the array power increases. From 0.85 to 0.3 A. U. the solar array temperature is held constant at 80° C by gradually tilting the array to an angle of 80° at 0.3 A. U. Thus, the array output power between 0.85 to 0.3 A. U. is constant. Permitting the array temperature to increase to its design limit of approximately 140° C by using a different array tilt program (74° tilt at 0.3 A. U.) would result in an increased array output power level for heliocentric distances below 0.85 A. U. Table II lists the assumptions employed in the thermal control system design for the worst case condition of 0.3 A. U.

The launch load requirements of the Shuttle/Interim Upper Stage launch vehicle were used to determine the structural member sizes for the modular thrust subsystems and the SEP module. These launch load requirements were multiplied by an ultimate (1.4) and yield (1.1) factor of safety to generate the ultimate and limit design loads shown in Table III. For the purpose of minimizing amplifications of the Shuttle/IUS induced mechanical vibrations, a minimum allowable structural frequency of the spacecraft of 5 Hz was imposed.

Modular Thrust Subsystems

The three modular thrust subsystems which have been studied are compared in Fig. 3. The TSSM consists of one 30 cm thruster and gimbal system, a power processor, a propellant storage and distribution system and a modular thermal control system and support structure. The BIMOD consists of two thrusters and gimbal systems, two power processors, a common thermal control system, and a common structure. The BIMOD concept employs a single remote propellant storage system. The HYBRID module consists of two thrusters and power processors, a structure and thermal control system identical to the TSSM approach, but uses a remote propellant storage system similar to the BIMOD concept.

Details of each of the modular thrust subsystems are given in Fig. 4. The power processing units of each approach and the propellant tanks of the TSSM are located near the top of the module so that when integrated with the rest of the spacecraft the large masses are concentrated near the total spacecraft center of mass. The structure for all three approaches is a light-weight truss constructed of graphite reinforced plastic (GRP) tubes which are inserted into GRP end fittings. The thruster gimbal system concept is discussed in a later section of the paper. Each TSSM has an individual propellant storage tank while the BIMOD and HYBRID approaches utilize central propellant tanks. The details of the propellant storage and distribution systems for the three approaches are discussed in a later section.

The thermal control system for the TSSM and HYBRID concepts consists of a combination of louvers, a variable conductance heat pipe system (VCHPS), a radiator fin, and multilayer insulation. For normal spacecraft operation there is no solar incidence on the radiating face of the PPU. Heat that cannot be radiated to space directly through the louver system is conducted by heat pipes to an adjacent space facing radiator. As shown in Figs. 4(a) and (c) two heat pipes are attached to each VCHPS saddle, with the second heat pipe of each saddle being redundant. An exploded view of the power processor and thermal control system for the TSSM and HYBRID is shown in Fig. 5.

For the BIMOD approach, the two power processors are mated to a common heat pipe system and are interior to two remote single sided radiators. There are no louvers in this approach because the radiating flanges of the power processors are not exposed directly to space. An exploded view of the BIMOD thermal assembly is shown in Fig. 6. For the BIMOD assembly, the high heat dissipation flanges of the power processor are bolted directly to the heat pipe evaporator saddles. Figure 6 shows three heat pipes on each of the two heat pipe saddles. The A3 and A4 modules of the PPU contain the large thermal dissipation components of the power processor, with the highest heat dissipation components of PTU A on or near heat pipe saddle A of Fig. 6. Because of the inverted orientation of PPU B its high thermal dissipation components of the A3 and A4 modules are now located on or near heat pipe saddle B. With this orientation, the heat loads going into each of the two heat pipe saddles are equal when both PPUs are operating. Figure 6 illustrates that one heat pipe of saddle B and two heat pipes of saddle A are extended to the one heat pipe radiator. The remaining three heat pipes are capped on the near side of the PPU but extend to the radiator on the far side of the modules.

In the BIMOD configuration, the PPUs are unable to radiate any heat directly to space and, therefore, the total heat load must be dissipated by the heat pipe radiators. This requires that the heat pipe radiator area be larger than in the TSSM. As a design margin, the radiator lengths for all three approaches are 25 percent larger than the required length calculated when using the 72 percent radiator efficiency shown in Table II.

SEP Module Configurations

The SEP module configurations employing the TSSM, BIMOD, and HYBRID approaches are shown in Fig. 7. The configuration chosen for illustration contains eight thrusters. The major elements of the SEP module are the package of modular thrust subsystems, the interface truss between this package and the rest of the spacecraft, and the solar array which attaches to the interface truss. The distance between adjacent thruster centerlines is 0.61 meter. The central propellant tanks for the BIMOD and HYBRID are shown supported in the interface truss. Although not shown on these figures, several components are mounted within or on the interface truss. These include a raw power distribution assembly which distributes the raw solar array power to individual power processors, a preregulator assembly which converts the input array voltage of 200-400 volts to a reduced bus voltage for the spacecraft engineering systems, and a control interface unit which is used for control of the thrust subsystem.

Figure 7 shows the attach points between the TSSM, BIMOD, or HYBRID and the interface truss, and also indicates a set of four launch adapter support points on the interface truss. Using the TSSM configuration, Figs. 8 and 9 show the end and side views of the stowed SEP module. The centerlines of the launch adapter tower are shown on these figures. The launch adapter, which bolts to the IUS-spacecraft interface, supports the entire spacecraft, with the exception of the solar arrays, at the support points on the interface truss near the center of mass of the assembled spacecraft. A spring loaded telescoping section of the solar array deployment booms has been provided to avoid transferring spacecraft launch loads to the stowed solar array structure. This design and support approach provides a savings in the mass of the structure of the SEP module and launch adapter.

The separation of the spacecraft from the IUS is accomplished by first firing the separation devices at the launch adapter tower support points shown in Fig. 8. The four launch adapter tower segments then pivot about their hinged bases in flower petal fashion to swing clear of the spacecraft. As the adapter tower swings clear, the solar array deployment booms immediately extend about 2 inches to their stops, thus leaving the spacecraft attached to the IUS only at the solar array blanket containment box (Fig. 8). When the separation devices at the solar array/IUS interface are fired, the spacecraft can be safely separated from the IUS using small ejection springs at the solar array/IUS interface.

When the spacecraft is a safe distance from the IUS, the solar array deployment boom lock (Fig. 8) is released, allowing the solar array booms to swing 90° and latch in the final deployed configuration. The fold out solar arrays can now be deployed in their normal manner. Because the telescoping section solar array booms have extended to solid stops, they provide increased stiffness, thereby increasing the total solar array system natural frequency.

A mass study for the modular thrust subsystem approach to a SEP module was recently presented in Ref. 6. The mass comparison given in Table IV for a SEP module using six thrusters and an 18 kW solar array shows that there is a negligible mass difference between the TSSM, BIMOD, and HYBRID approaches. This same conclusion holds when comparing the three approaches to a SEP module comprising eight thrusters with a 25 kW solar array or eight thrusters with an 18 kW solar array.

Subsystem Descriptions

Thruster Gimbal System

The functional requirements of the thruster gimbal system for the modular thrust subsystems are to direct the individual thrust vectors in two axes such that three axis attitude control and spacecraft reorientation control is provided by the ion thrusters, and to provide a mounting base and interface between the thruster and thrust subsystem truss. The travel angle and slew rate requirements for the two axis gimbal system are currently under review for the SEP planetary mission opportunities being considered. Figure 10 defines the gimbal angles α and β which are rotations about axes nominally parallel to two of the spacecraft principal axes. The travel angle in each axis is the sum of the angle to direct the thrust vector through the spacecraft center of mass and the additional angle required for thrust vector control. By arbitrarily adding a 5° angle for thrust vector control to the approximate angles for pointing the outboard thruster of an eight thruster SEP module through the center of mass, the gimbal angle requirements assumed are a total angle in the α direction of 70° and a total angle in the β direction of 30°.

Figure 11 shows the conceptual gimbal system interfacing with the 30 cm thruster. The two linear actuators and a cross pin hinge or gimbal pivot provide the thruster gimbal directions in two mutually orthogonal axes. These components are mounted on a thruster mounting bracket which is attached to the mounting pads on the sides of the thruster and to standoffs at two of the four ground screen mounts on the back of the thruster. The two jackscrew type actuators are driven by a stepper-motor-gearhead assembly. The actuators have a universal joint at both ends for compliance. A guide pin that is attached to the thruster mounting frame rides in the slot of a support bracket that is mounted to the lower truss of the module. One of the advantages of this system is that the arrangement of the actuators, cross pin hinge and guide pin provides stiffness in all directions thus eliminating the need for pin puller restraint during launch. The static and dynamic launch loads are carried in the x direction by the two actuators and the cross pin hinge, in the y direction by the thrust washers in the cross pin hinge, and in the z direction by the cross pin hinge and the support bracket. The angle indicator system consists of two resolvers that are attached to the cross pins of the hinge and provide direct readout of the α and β gimbal angles. The flexible propellant feed line is a coiled spring tube.

A good thermal design is provided by the linear actuator gimbal system because the actuators are placed behind the thrusters. A thermal barrier could be placed between the thrusters and the lower truss of each module and thereby the gimbal actuators would be located within the controlled thermal environment.

The gimbal system concept is a derivative of the 8 cm ion thruster gimbal system designed, fabricated, and tested at the Lewis Research Center.

Propellant Storage and Distribution System

The functional requirements of the propellant storage and feed system are to store the required mercury propellant for the lifetime of the mission, to isolate the propellant from the thruster during launch so that the dynamic environment does not have a detrimental effect on the operation of the thruster, and to supply the propellant to the thruster within a pressure range that satisfies the requirements of the thruster vaporizers.

As mentioned previously, the TSSM approach contains a complete propellant storage and distribution system within each module, as compared to the BIMOD and HYBRID approaches which employ central propellant tanks in the interface truss section of the SEP module. The schematic of the propellant storage and distribution system concept for a TSSM is shown in Fig. 12. The feed lines are 0.16 centimeter in diameter. Field joints are placed in the system to make the assembly/disassembly of the system more convenient. The latching valve serves the purpose of isolating the mercury propellant from the thruster vaporizer during the launch environment. The mercury propellant and nitrogen gas fill valves of the system are brought to the outside wall of the TSSM to provide for convenient filling of the system after assembly of the module into the spacecraft. A system of external valves and lines will be necessary to fill and drain the system. This hardware is envisioned as ground support equipment which will interface with the exposed propellant and gas fill values of the TSSM system.

Figure 13 shows the schematic for the system concept applicable to the BIMOD or HYBRID thrust subsystems. For this approach, a central propellant tank in the interface truss stores the propellant for all of the modules. Although only one tank is shown in the schematic, trade studies may show the desirability of two tanks for some missions. Propellant lines from the tank feed a propellant manifold which is also contained in the interface truss. The distribution system interface between the interface truss and the modular thrust subsystems are the field joints between the manifold and the two propellant feed lines for each module. For this approach, the fill valves for the single tank are brought to the exterior wall of the interface truss.

Consideration must be given to the redundancy of both systems. If a thruster in a TSSM fails, and the propellant loading does not contain some contingency, there would be insufficient propellant in the remaining TSSMs to complete the mission. The possibility of increasing the system reliability by placement of an interconnecting valve network between tanks is being studied.

A storage tank using a nitrogen gas expulsion technique to supply propellant to the thruster has been selected for all approaches. The mercury propellant volume of the storage tanks is determined by the nominal propellant loading for the mission, contingency for launch window and thruster utilization, and the propellant utilization of the tanks, lines, and components of the system. The gas volume is determined from the operating characteristics of the thruster vaporizers. The maximum operating pressure is that which will cause the mercury to intrude the porous plug of the thruster vaporizer and result in the vaporizer not functioning. Based on past testing, the intrusion pressure is about 52 N/cm^2.[7] Adding a safety factor for uncertainties such as the pore size of the plug, weld cracks, and accelerations the maximum operating pressure selected is 31 N/cm^2. The minimum operating pressure is determined from the partial pressure of the mercury at an operating temperature of the vaporizer. If the pressure is too low, the liquid/vapor interface could move away from the vaporizer and vaporizer control would be lost. The minimum operating pressure selected is 10 N/cm^2. These pressure limits correspond to a blowdown range of 3:1.

The storage tank design selected is shown in Fig. 14(a). The tank design is a derivative of the approach employed for the SERT II spacecraft.[8] An elastomeric bladder separates the mercury propellant volume from the pressurized nitrogen gas volume. The tank contains an internal liner which supports the bladder during the launch environment and thus minimizing slosh effects. The liner holes permit the pressurized gas to pass through the liner and move the bladder. A storage tank of the same outside dimensions can be employed for some missions which do not require the full sphere of mercury propellant. However, the shape of the bladder support liner is modified as shown in Fig. 14(b), so that only the volume of required mercury is supported by the liner. This concept minimizes slosh effects during the launch environment.

The utilization of the propellant in the storage and feed systems described is approximately 98 percent. Some of the mercury is trapped by the ribs of the bladder which are designed to prevent the bladder from blocking the exit orifice of the tank. For the six missions under consideration, four of the missions can be accomplished with a propellant loading of 732 kg and the remaining two can be accomplished with a propellant loading of 1500 kg. Table V lists the required tank inner sphere diameter for the two propellant capacities when using individual tanks in the TSSM or one central tank in the BIMOD or HYBRID approach.

Design/Development Comparison of Modular Approaches

The TSSM, BIMOD, and HYBRID approaches to the SEP module have been compared using a number of criteria. The BIMOD approach has a slight advantage over the other two approaches because it

(1) Has a slightly lower recurring cost.

(2) Requires a less complex life test in a thermal vacuum environment.

(3) Is marginally easier to maintain.

(4) Results in a less severe vibration and acoustic response of the power processors.

(5) Does not require the strict mechanical tolerances attendant with louver interfaces.

(6) Offers greater configuration flexibility.

As shown in Table IV, the differences in the mass among the three approaches is within the accuracy of the mass study. Comparing the reliability of the approaches on a subjective basis indicates that they are very similar.

Cost

The relative hardware costs of the three modular concepts for the modular approach to the SEP module are given in Table VI. It was estimated that the BIMOD would be about 10 percent less expensive than either the HYBRID or two TSSMs. Most of this difference is caused by differences in the thermal control systems. The BIMOD does not use louvers and employs 25 percent fewer heat pipe systems when compared to two TSSMs or a HYBRID.

Thermal Testing

There are some major testing differences between the TSSM, BIMOD, and HYBRID approaches. Two categories of thermal testing are envisioned as part of the development of any of the modular thrust subsystems. The first type of test would be a thermal vacuum life test of the power processors in the thermal configuration of the module, and the second type would be the flight qualification test program of the individual modular thrust subsystems. Consideration should not only be given to the test complexity inherent in each modular approach, but also to the facility operational costs.

For the thermal vacuum life testing, auxiliary cooling loops can be used in the heat pipe saddles for all three approaches. Rather than using auxiliary cooling loops, the flight qualification test program might also test the heat pipe systems in the gravity field. The techniques required for testing a similar system for the Communications Technology Satellite have been demonstrated at Lewis Research Center.[9, 10]

One of the design requirements of the modular thrust subsystem which may greatly simplify the thermal analysis and the qualification level testing is whether the radiating surface of the module is subjected to a solar incidence angle during any particular mission. These situations would need to be individually analyzed for both the TSSM and HYBRID approaches. This is because the thermal absorptance of the louver set varies depending on both the incidence (or cone) angle and the azimuth (or clock) angle of the sun with respect to louvers. For the BIMOD approach, the solar absorptance of the heat pipe radiator is dependent only on the incidence angle and not the azimuth angle, thus greatly simplifying the thermal analysis.

Mechanical Interfaces

The TSSM and HYBRID each have louver interfaces. The surfaces at these interfaces must be kept very flat in order to avoid binding the louvers. A tighter surface flatness requirement would tend to make PPUs used in the HYBRID or TSSM costlier than for the BIMOD.

Vibration Response

The HYBRID or TSSM and the BIMOD employ similar graphite truss construction and the ion thrusters are supported in a similar manner for all three approaches. Similarities also exist regarding the use of multilayer insulation for thermal control. It would be safe to assume that the vibration responses of these similar areas of the modular concepts would be about equal. The primary design differences between the HYBRID or TSSM and the BIMOD are in how the PPUs are integrated into the structure. The PPU's for the HYBRID or TSSM are mounted singly to a heat pipe system as shown in Figs. 4(a) and (c). For the BIMOD the PPU's are mounted back to back to a common heat pipe system (Fig. 4(b)). Thus, for the BIMOD, the two PPU baseplates and the common heat pipe system combine to form a very stiff internally reinforced structure. Since the PPU structure for the BIMOD is stiffer and since the PPU's in the BIMOD are relatively isolated from direct acoustic excitation as compared to the TSSM or HYBRID, the PPU's in the BIMOD should have a less severe vibration and acoustic response to the launch environment.

Maintainability

Exploded views of both the TSSM and the BIMOD are shown in Figs. 5 and 6, respectively. A typical maintenance operation might be the replacement of one of the circuit modules of the PPU. To do this on the TSSM it would be necessary to remove the louvers, the propellant tank and its support structure, the thruster support structure and the PPU frame. The circuit module and its back plate could then be electrically disconnected and unbolted from the heat pipe saddles and louver supports structure and replaced. This procedure would be reversed for reassembly. The procedure for the HYBRID would be similar with the exception of propellant tank removal since a central propellant tank is used.

For the BIMOD, the procedure for replacing a PPU circuit module in the PPU mounted nearest to the thrusters would be different than for the PPU mounted furthest from the thrusters. For the PPU mounted nearest to the thrusters, the thruster support structure and PPU outer frame would need to be removed before the circuit module and its back plate could be electrically disconnected and unbolted. Only the PPU outer frame would need to be removed from the PPU furthest from the thrusters in order to service a PPU module.

Thus, for one PPU only, the BIMOD should be simpler to service.

Configuration Flexibility

The mechanical integration of the modular thrust subsystems with the interface truss of a SEP module has been discussed herein. One of the remaining unknowns is the required interface between the truss and the payload. On the basis of total spacecraft configuration, the central propellant tank approaches of the BIMOD and HYBRID appear to offer more flexibility in arriving at a well balanced spacecraft. After defining the payload mass and volume, the overall dimension of the interface truss may not satisfy the requirements of the user. An alternate approach to a standard interface truss is to standardize on central propellant tank sizes and the remaining components within the interface truss. The truss structure could then be made mission dependent, if necessary, and still retain the basic design features of serving to integrate the modular thrust subsystems into the SEP module and to provide support points for a launch adapter tower.

Concluding Remarks and Recommendations

The three approaches to the modular thrust subsystems described herein represent viable options to the design and development of a SEP module. Each approach has satisfied a set of requirements for a number of representative planetary missions. The number of thrusters required on a given mission may be changed from six to eight without affecting the design of the individual modules. Only the interface truss between the modular thrust subsystems and the payload need be changed to accommodate the growth of the SEP module. And, the concept of a standard modular thrust subsystem is consistent with the concept of a standard SEP module for a number of missions and the attendant objective of minimum development cost.

The design study has concentrated on defining the mechanical/thermal interfaces of the power processor and thruster. The successful flight performance of the heat pipe system on CTS has again demonstrated that this technology is ready for application to a SEP module. With the use of a heat pipe/louvers or an all heat pipe thermal system, the mass penalty for the multimission thermal design is small, because only the radiator length, heat pipe length, and the size of a lightweight structure are affected. For this reason and because of the basic structural design approach, it is believed that a comparison would indicate that the modular approach is competitive in terms of mass with the assembly or individual integration approach. Continued interface definition is required for the elements of the propellant storage and distribution system and the thruster gimbal system. Finally, the BIMOD is the recommended choice for further design and development as the modular thrust subsystem.

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TABLE I. - PPU THERMAL DESIGN REQUIREMENTS

	Mercury orbiter	1 A. U.	Jupiter orbiter
Distance from sun, A. U.	0.3	1	5
Solar array temperature, ^{o}C	80	50	-140
PPU thermal dissipation, W (measured at full power)	277	277	324
PPU efficiency, percent	91.1	91.1	89.7

TABLE II. - PPU THERMAL CONTROL SYSTEM ASSUMPTIONS

Solar array temperature,* ^{o}C	80
Radiator temperature, ^{o}C	50
PPU baseplate temperature, ^{o}C	51
Solar array emittance	0.80
Louver emittance - fully open (where applicable)	0.05
Radiator emittance	0.88
Radiator efficiency, percent	0.72
Radiator view factor to space	0.83
Radiator view factor to solar array	0.17
Louver view factor to space (where applicable)	0.83
Louver view factor to solar array (where applicable)	0.17

*Assumes a tilt angle of 80^{o} at 0.3 A.U.

TABLE III. - LAUNCH VEHICLE DESIGN LOAD FACTORS BASED ON SHUTTLE/IUS

	Flight event	Axis	Load factors (g's) Ultimate		Load factors (g's) Limit
Shuttle load factors	Maximum accel.	X (longitudinal)	+4.7	Conservatively combined	+3.7
	Liftoff	Y	±1.4		±1.1
	Landing	Z	-4.0		-3.2
	Emergency landing	X (longitudinal)	-9.0	Separate	
		Y	±1.5		
		Z	+2.0		
			-4.5		
IUS load factors	Earth orbital	X (longitudinal)	7.0	Combine	5.5
		Y	±4.2		±3.3
		Z	±4.2		±3.3

TABLE IV. - MASS COMPARISON FOR A SIX THRUSTER, 18 kW SEP MODULE

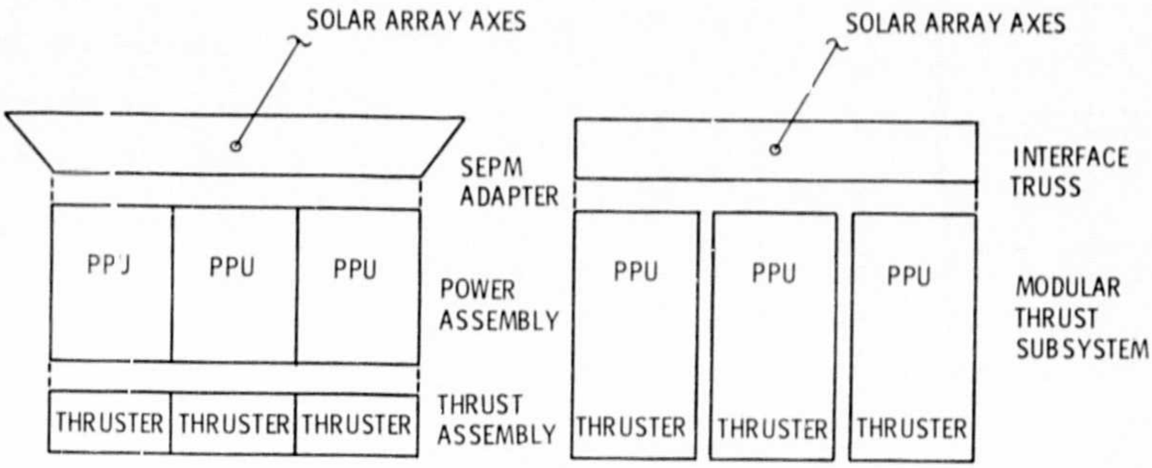
	TSSM		BIMOD		HYBRID	
Power processor	SIX TSSM	176.87 kg	THREE BIMOD	176.87 kg	THREE HYBRID	176.87 kg
PPU thermal control		53.90		58.54		53.90
Thruster		48.96		48.96		48.96
Thruster gimbal system		17.70		17.70		17.70
PPU to thruster harness		3.24		5.22		3.24
Structure		14.79		15.33		14.79
Propellant storage and distribution system		26.99	Interface truss	20.01	Interface truss	20.01
Propellant tank and line residuals		15.27		15.04		15.04
Tank support truss	Interface truss	———		2.12		2.12
Truss tubes, end fittings, module attach points and hardware, solar array drive brackets		14.74		15.81		15.81
Truss insulation		2.45		2.56		2.56
Component shelf stiffeners, controller shelf, and attach brackets		1.90		1.90		1.90
Component thermal control		2.55		2.43		2.43
Raw power distribution		9.8		9.80		9.80
Preregulator		5.2		5.20		5.20
TSS controller		4.5		4.50		4.50
Power harness		1.2		1.20		1.20
Array drive and electronics	Solar array	9.1	Solar array	9.1	Solar array	9.1
Array deployment boom		12.16		12.16		12.16
Array to truss harness		4.86		4.86		4.86
Array mast canister, structure, and mechanisms		105.56		106.56		106.56
Array blanket leaders		2.90		2.90		2.90
Array blanket, mast elements, and harness		198.36		198.36		198.36
Total SEP module		734.00 kg		737.43 kg		730.27 kg

TABLE V. - PROPELLANT TANK SIZE REQUIREMENTS

Propellant mass, kg	Applicable missions	Inner tank sphere diameter, cm	
		TSSM	BIMOD, HYBRID
732	Tempel II rend. Flora rend. Saturn orbiter Jupiter orbiter	25.7 (6 module)	46.7
1500	Mercury orbiter Out-of-ecliptic	29.7 (8 module)	59.4

TABLE VI. - SEP MODULE SYSTEM RELATIVE RECURRING HARDWARE COST COMPARISON

	Module type	
System	TSSM × 2 HYBRID	BIMOD
Thruster	0.141	0.141
Gimbal system	.018	.018
Propellant system	.019	.009
PPU	.615	.615
Louver systems	.039	0
Heat pipe systems	.105	.079
PPU frame structure	.021	.014
Truss structure	.042	.025
Total	1.00	0.901



(a) JPL assembly approach.

(b) LeRC modular approach.

Figure 1. - Comparison of component integration approaches to a SEP module.

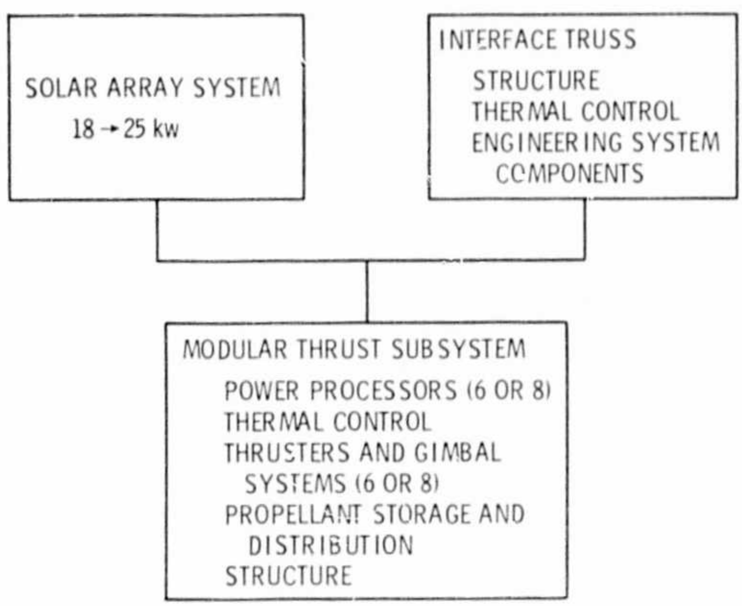


Figure 2. - Definition of SEP module.

E-8800

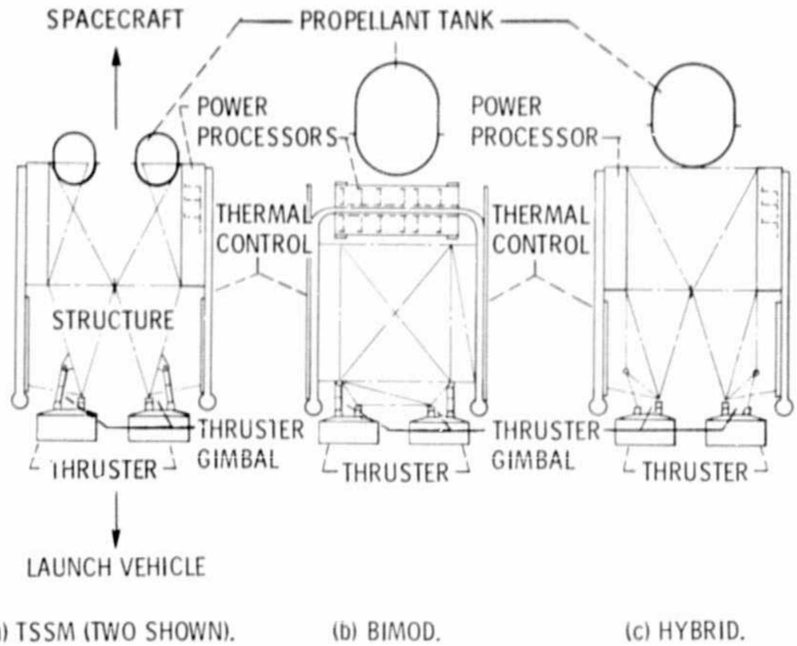


Figure 3. - Comparison of modular thrust subsystems.

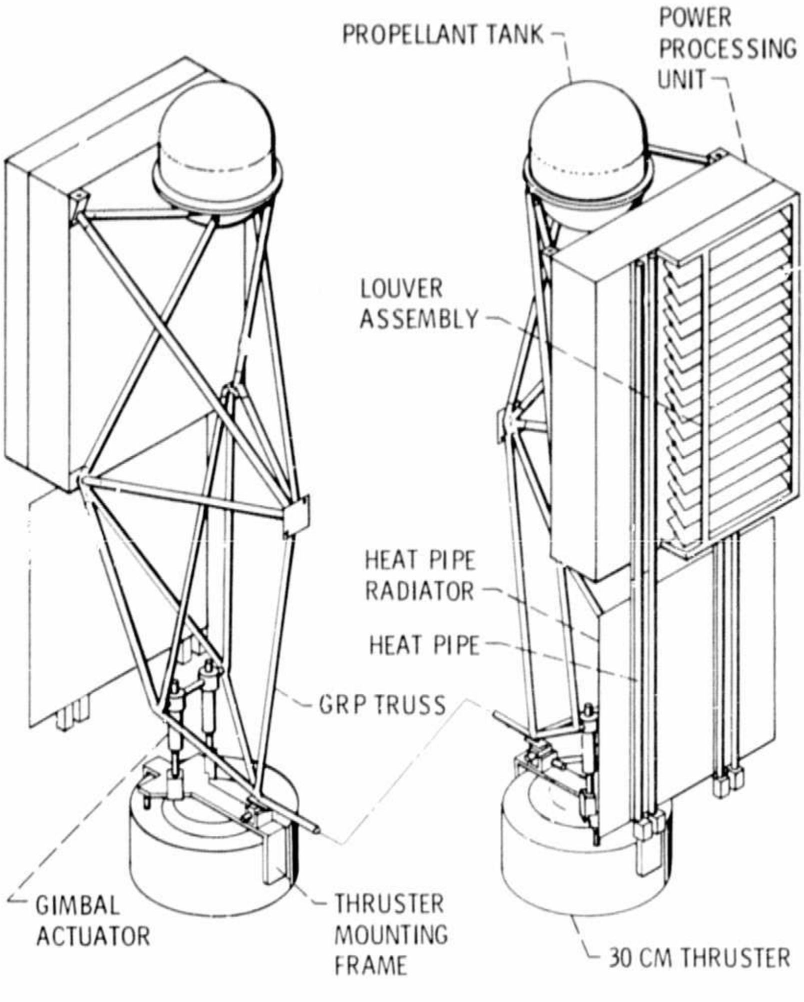


Figure 4(a). - Isometric view of TSSM.

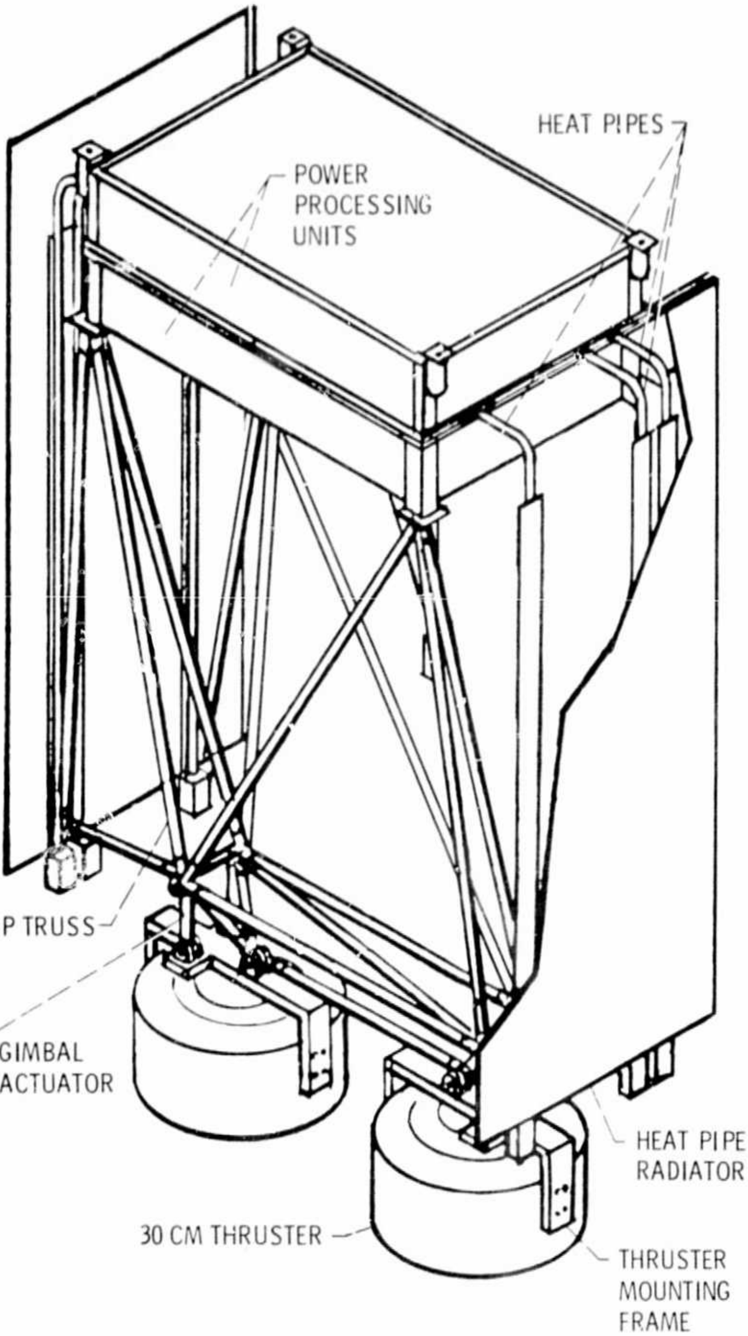


Figure 4(b) Isometric view of BIMOD.

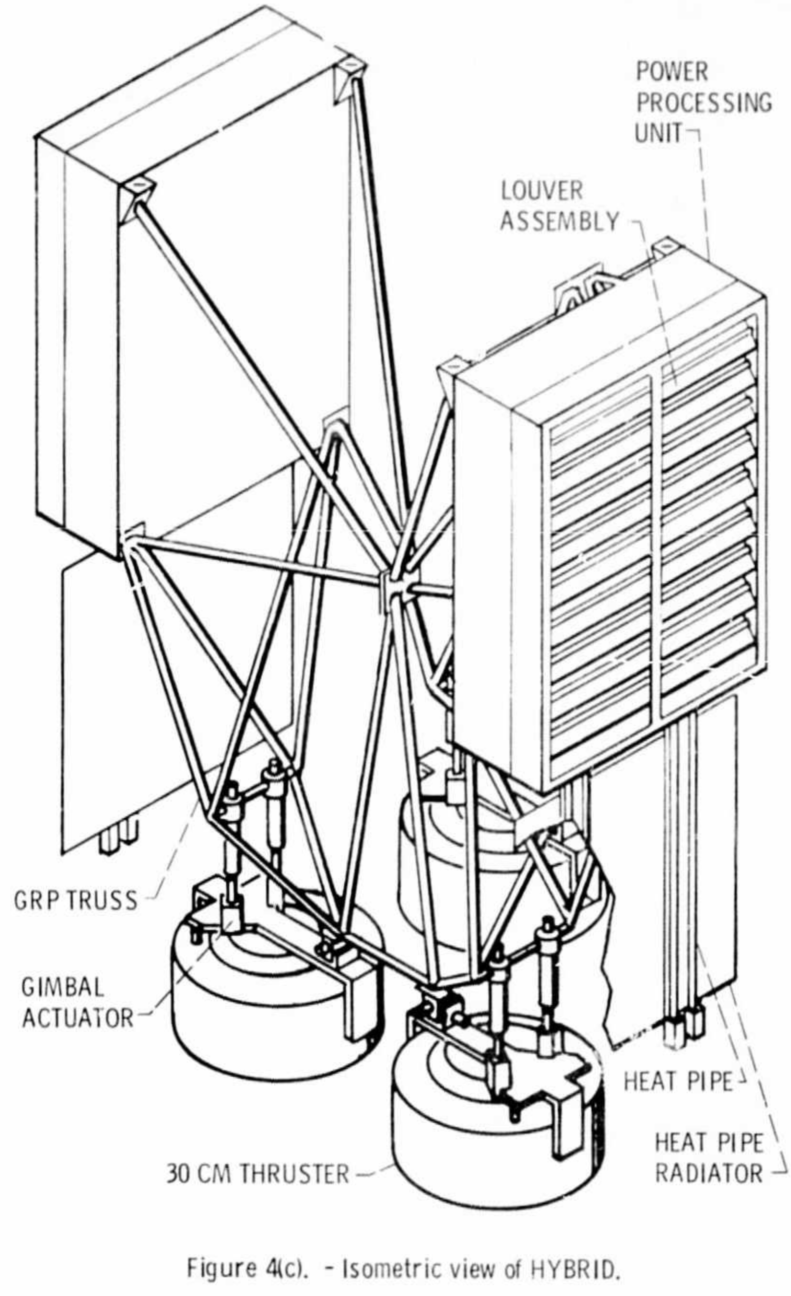


Figure 4(c). - Isometric view of HYBRID.

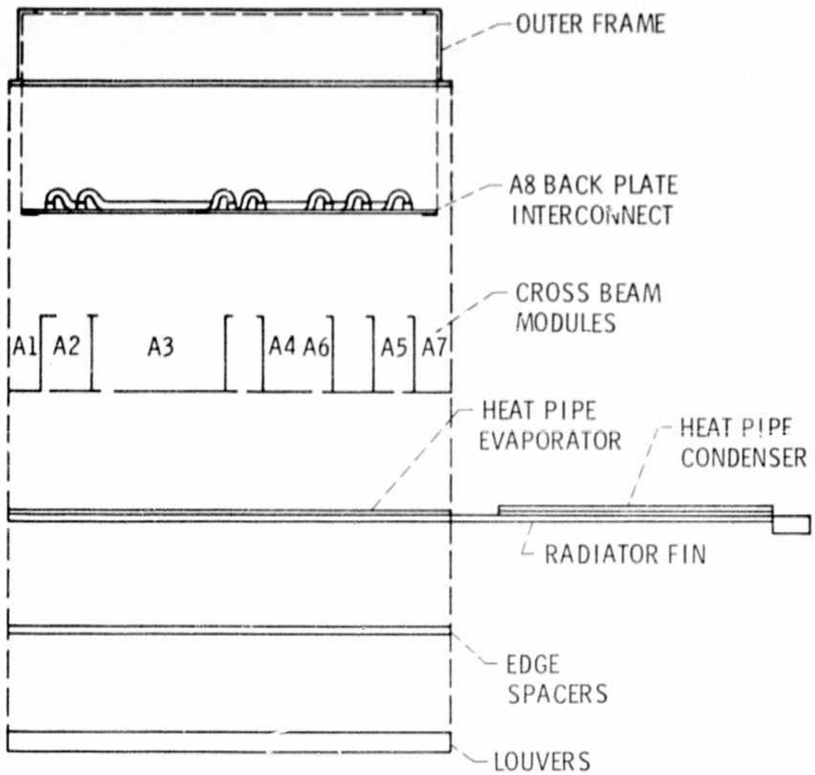
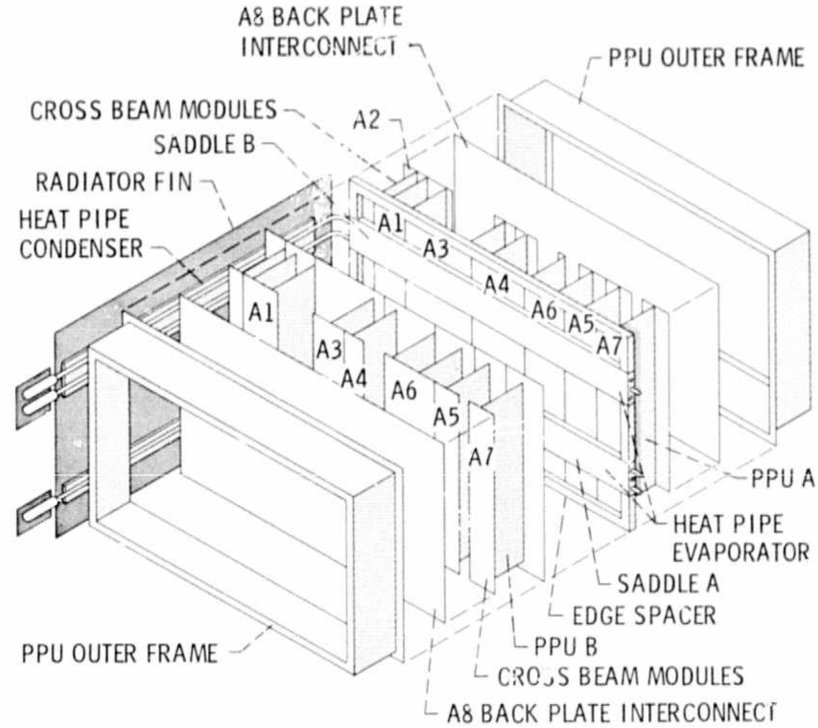


Figure 5. - Top view of exploded power processing unit - TSSM assembly.



NEAR SIDE RADIATOR FIN
AND HEAT PIPE CONDENSER
NOT SHOWN

Figure 6. - BIMOD assembly.

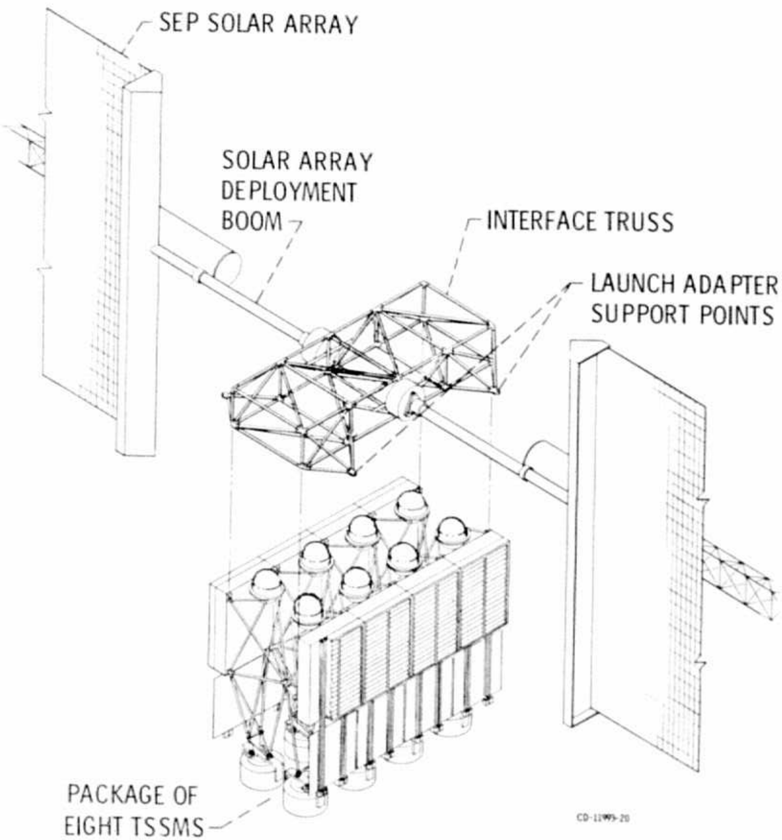


Figure 7(a). - Conceptual propulsion module using TSSMS.

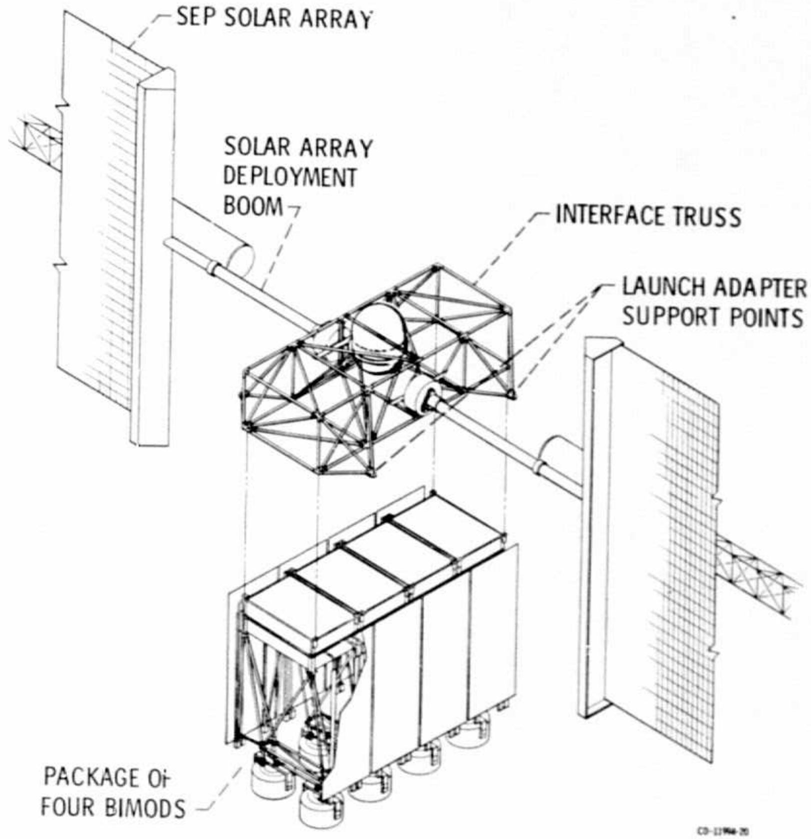


Figure 7(b). - Conceptual propulsion module using BIMODS.

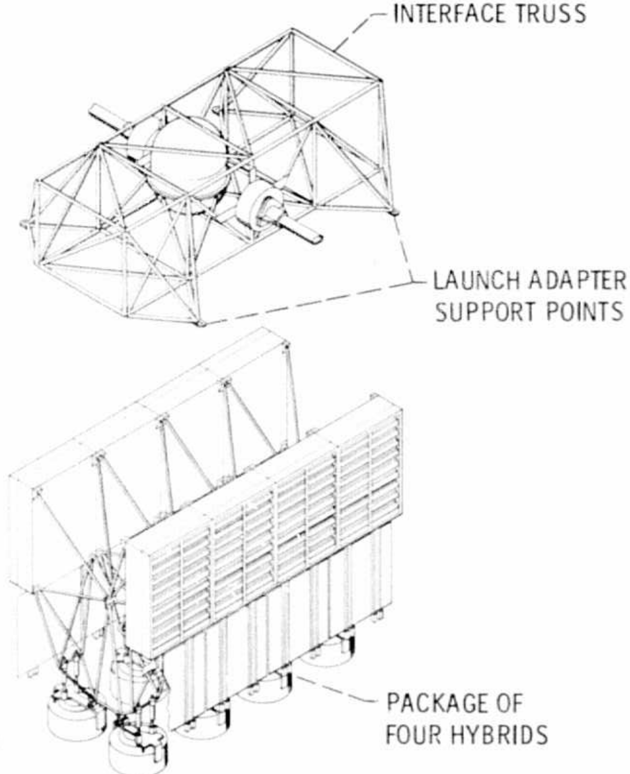


Figure 7(c). - Conceptual propulsion module using HYBRIDS.

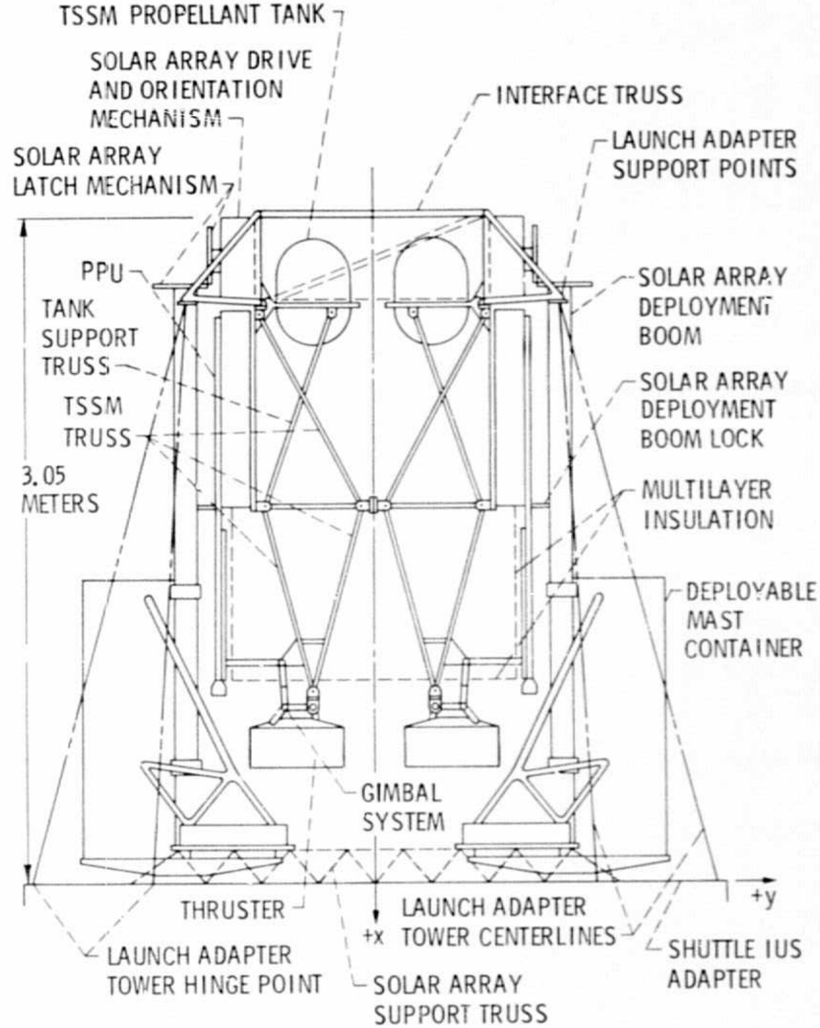


Figure 8. - End view of TSSM based SEP module.

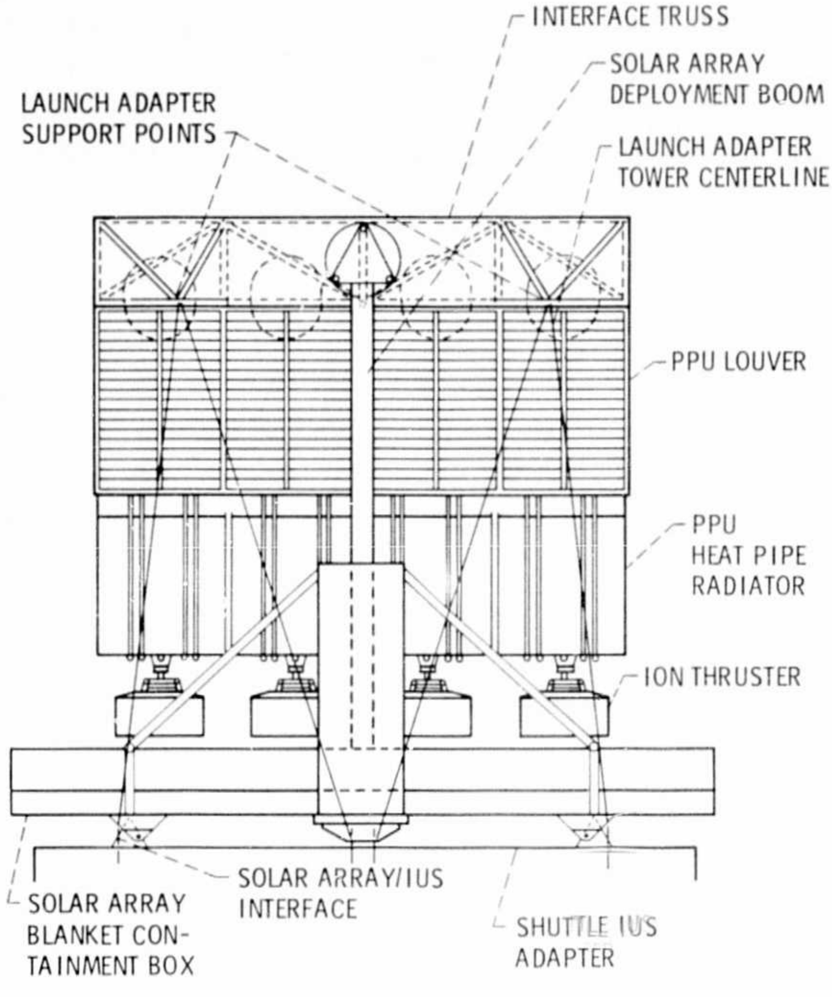


Figure 9. - Side view of TSSM based SEP module.

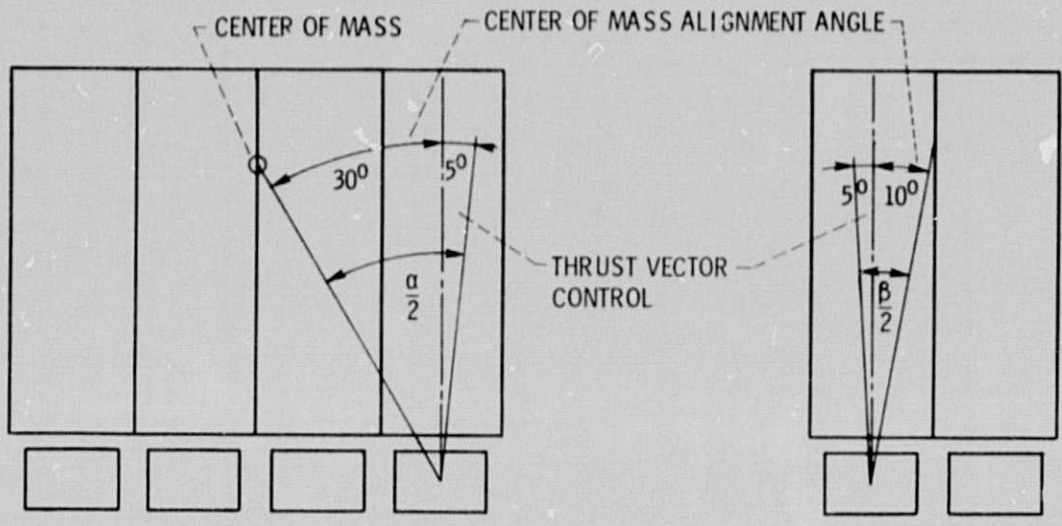


Figure 10. - Gimbal angle requirements.

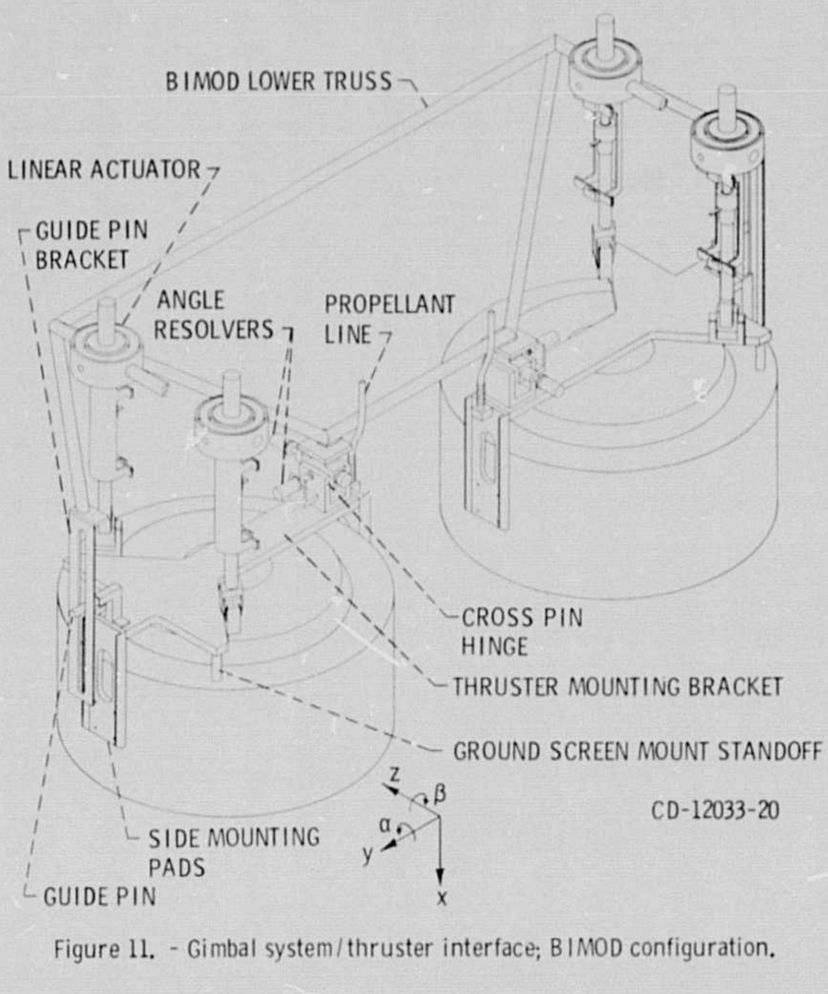


Figure 11. - Gimbal system/thruster interface; BIMOD configuration.

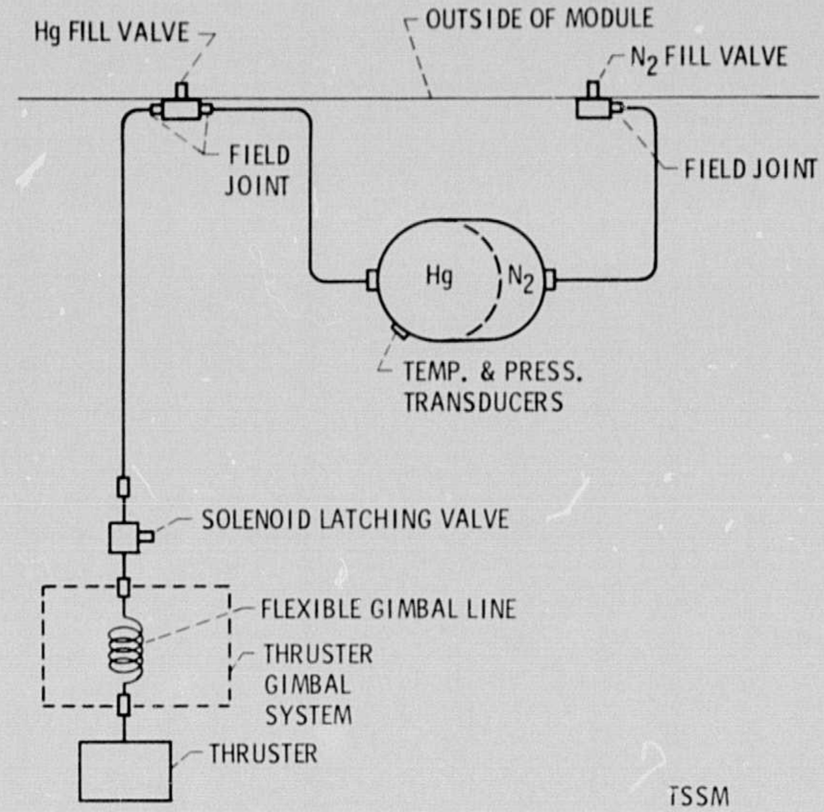


Figure 12. - TSSM Feed system schematic.

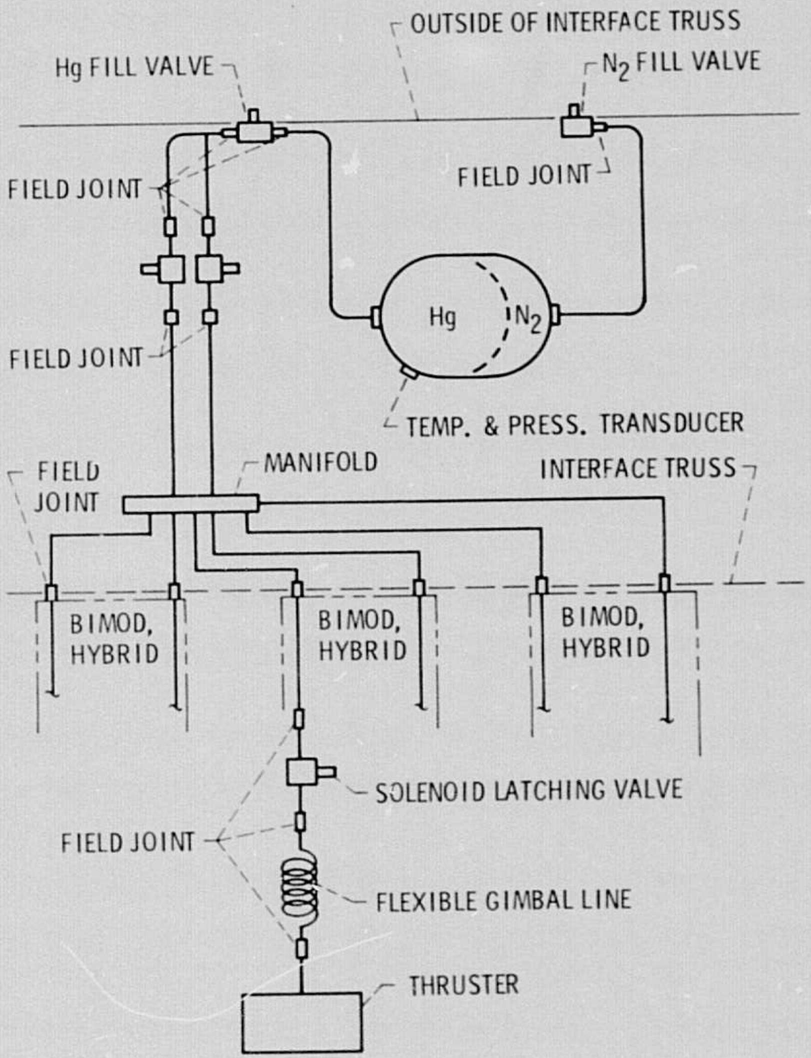
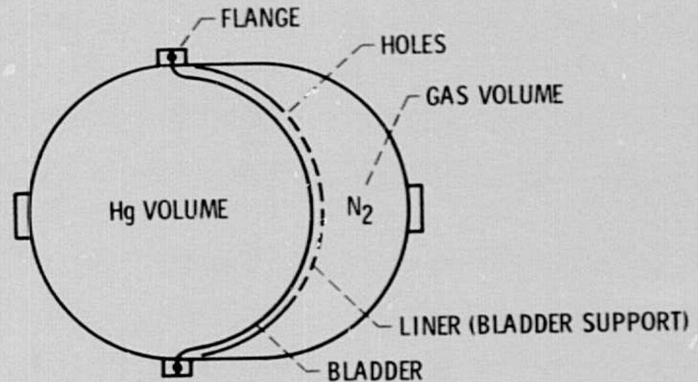
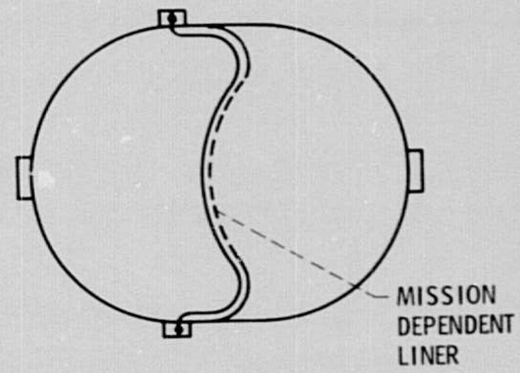


Figure 13. - BIMOD, HYBRID system schematic.



(a) Storage tank.



(b) Storage tank with mission dependent liner.

Figure 14. - Propellant storage tank concept.

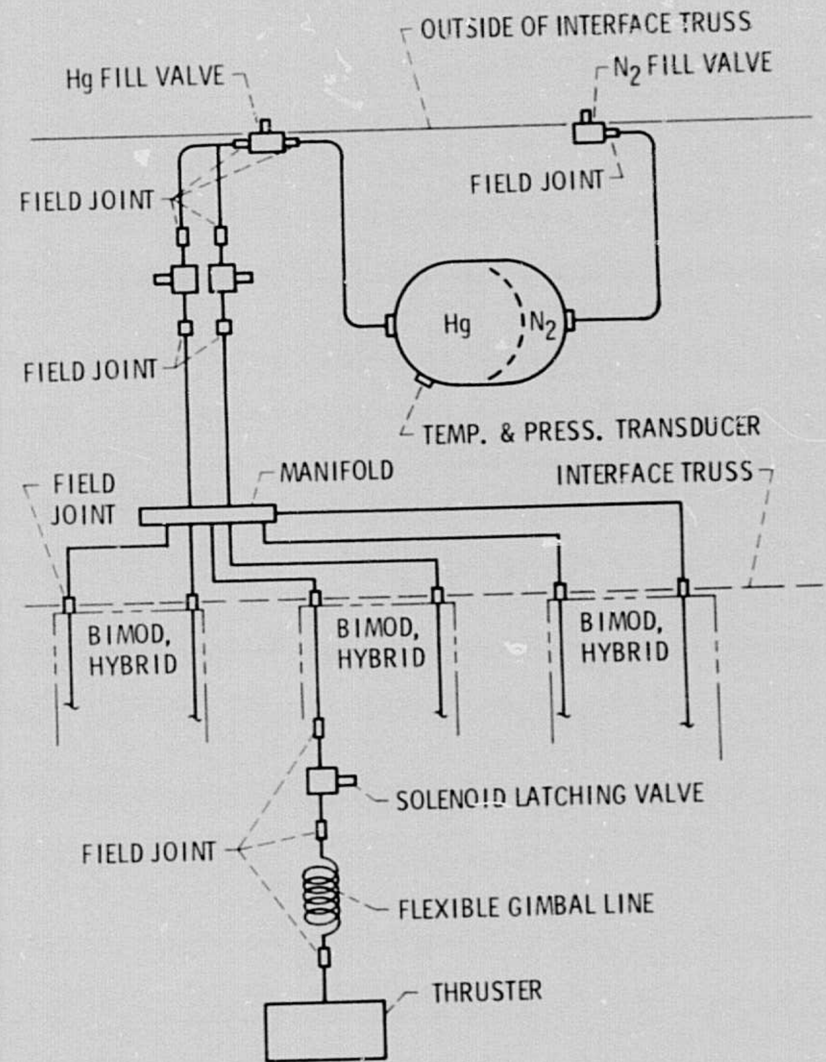
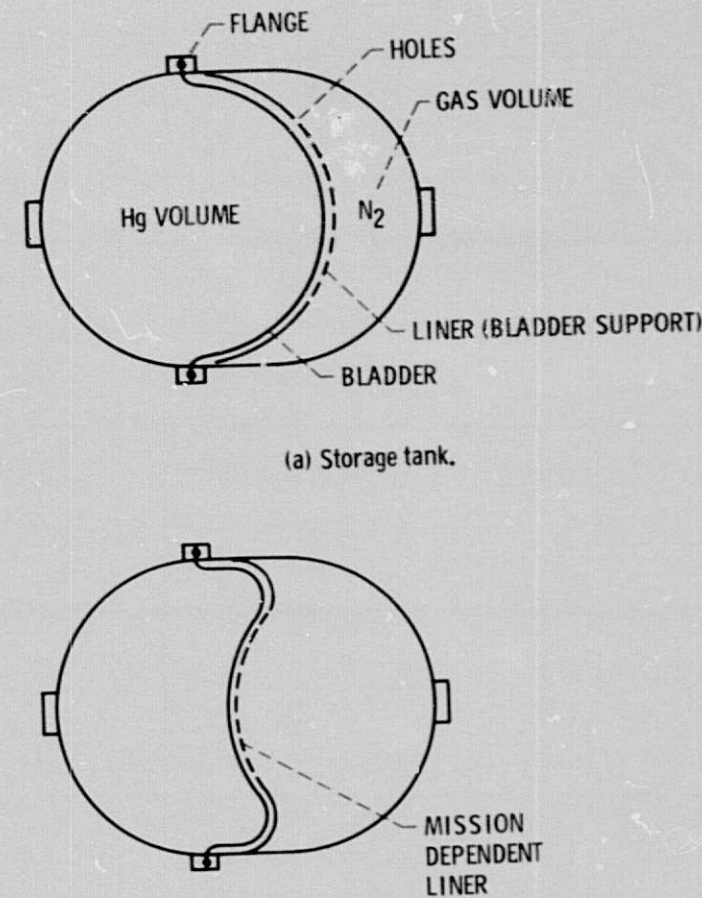


Figure 13. - BIMOD, HYBRID system schematic.



(a) Storage tank.

(b) Storage tank with mission dependent liner.

Figure 14. - Propellant storage tank concept.

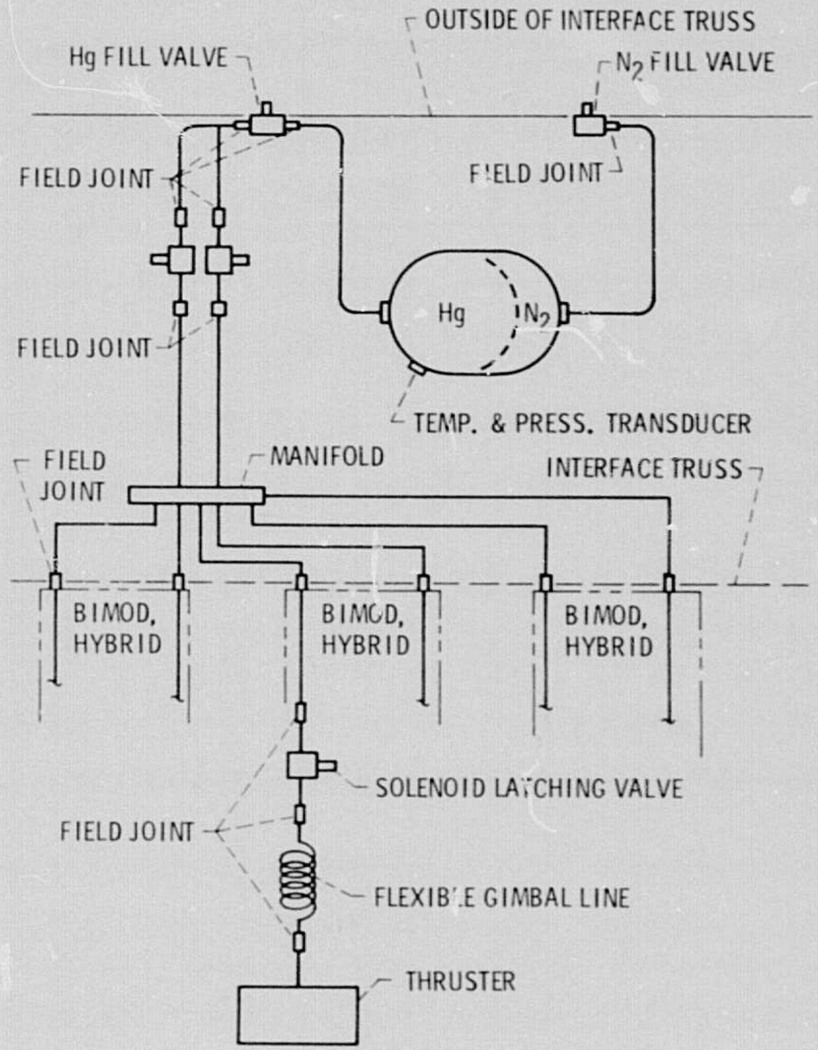
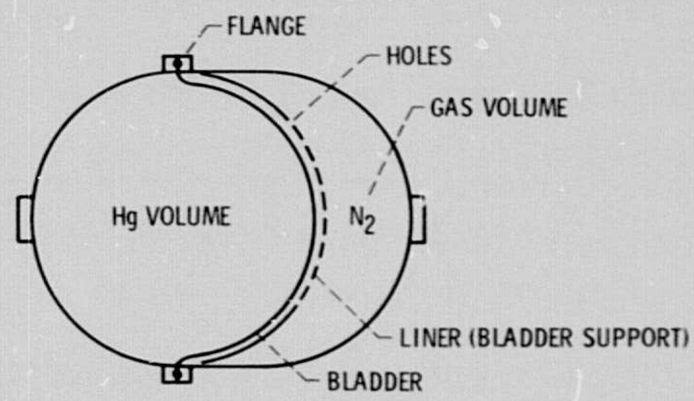
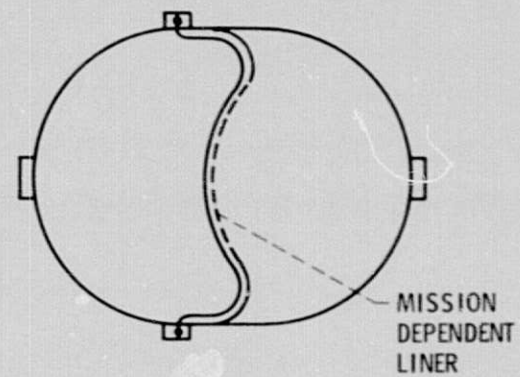


Figure 13. - BIMOD, HYBRID system schematic.



(a) Storage tank.



(b) Storage tank with mission dependent liner.

Figure 14. - Propellant storage tank concept.

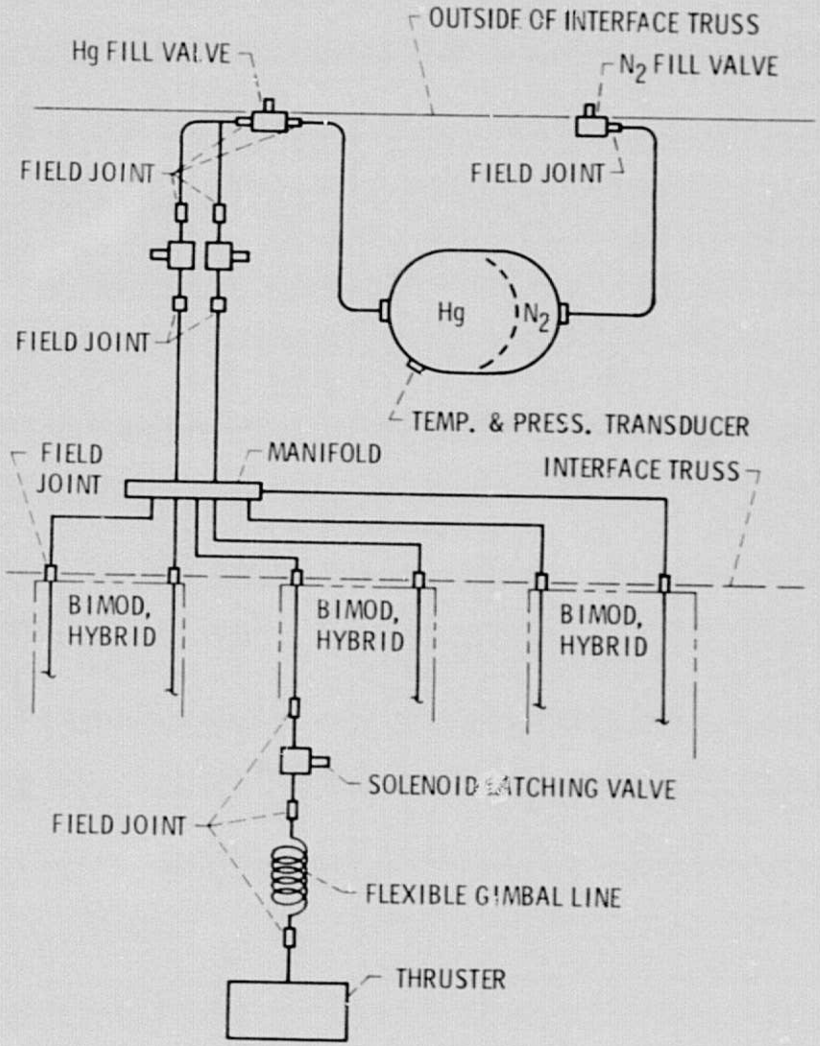
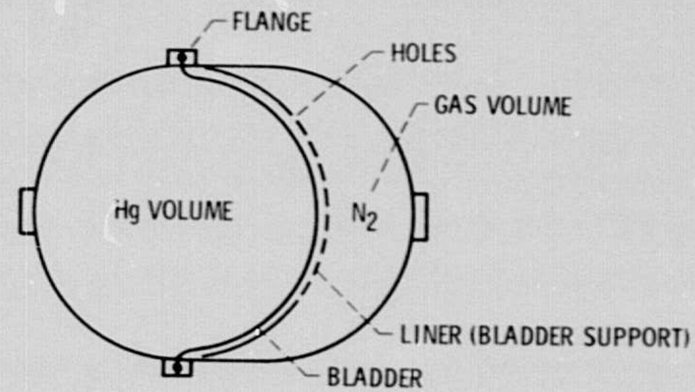
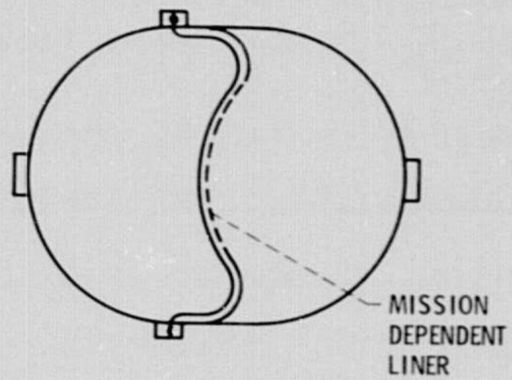


Figure 13. - BIMOD, HYBRID system schematic.



(a) Storage tank.



(b) Storage tank with mission dependent liner.

Figure 14. - Propellant storage tank concept.