Modular Thrust Subsystem Approaches to Solar Electric Propulsion Module Design

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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.

Introduction

DURING the Advanced Systems Technology program conducted 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 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.

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 ongoing series of SEP spacecraft would accrue many benefits. A qualification test program for the thrust modules could be developed which would envelop a multimission set. Because essentially identical modules could be used for several ongoing missions, only a flight-acceptance test program would need to be performed on the modules to be used for each mission. It therefore would 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 enhanced greatly, 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 (TSSM) 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.

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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 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 to be 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 1 lists the thermal environment that 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 0 deg. 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 deg at 0.3 A.U. Thus, the array output power from 0.85 to 0.3 A.U. is constant. Permitting the array temperature to increase to its design limit of approximately

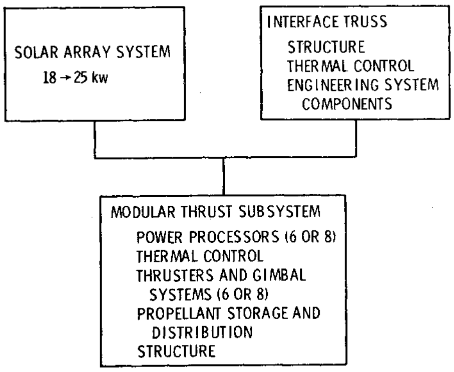

Fig 1 Definition of SEP module.

Table 1 PPU thermal design requirements

	Mercury orbiter	1 A. U.	Jupiter orbiter
Distance from sun, A. U.	0.3	1	5
Solar array temperature, °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 2 PPU thermal control system assumptions

Solar array temperature,[a] °C	80
Radiator temperature, °C	50
PPU baseplate temperature, °C	51
Solar array emittance	0.80
Louver emittance - fully open (where applicable)	0.65
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

[a] Assumes a tilt angle of 80 deg at 0.3 A.U.

140°C by using a different array tilt program (74-deg tilt at 0.3 A.U.) would result in an increased array output power level for heliocentric distances below 0.85 A.U. Table 2 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 (IUS) 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 3. 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 that have been studied are compared in Fig. 2. 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.

Table 3 Launch vehicle design load factors based on shuttle/IUS

	Flight event	Axis	Load factors (g's) Ultimate		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

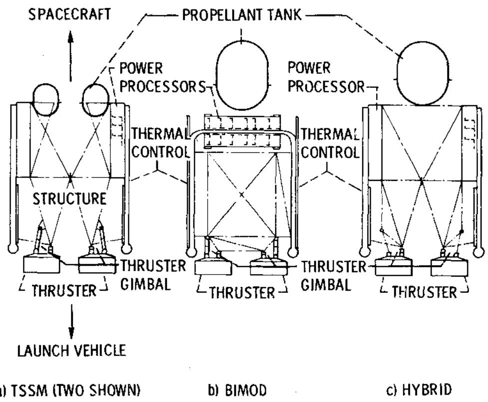

Fig. 2 Comparison of modular thrust subsystems.

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. 3. The power-processing units (PPU's) 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 lightweight 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, whereas 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. 3a and 3c, two heat pipes are attached to each VCHPS saddle, with the second heat pipe of

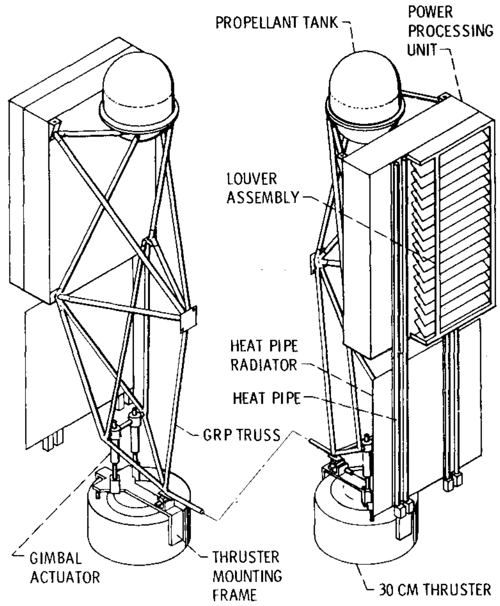

Fig. 3a Isometric view of TSSM.

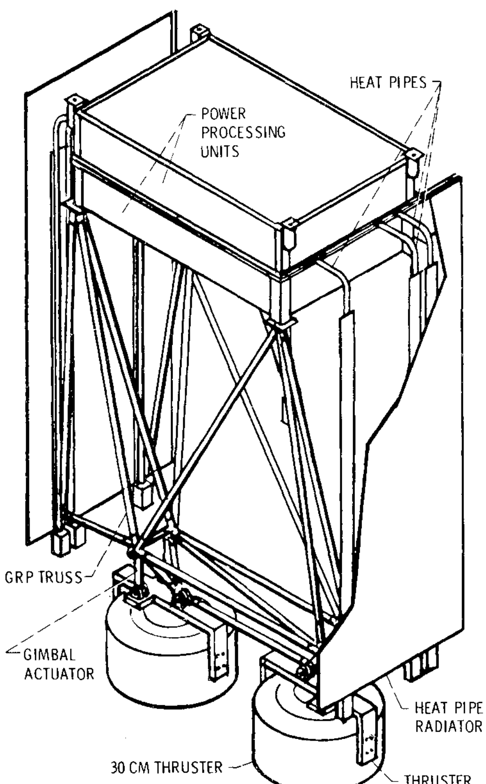

Fig. 3b Isometric view of BIMOD.

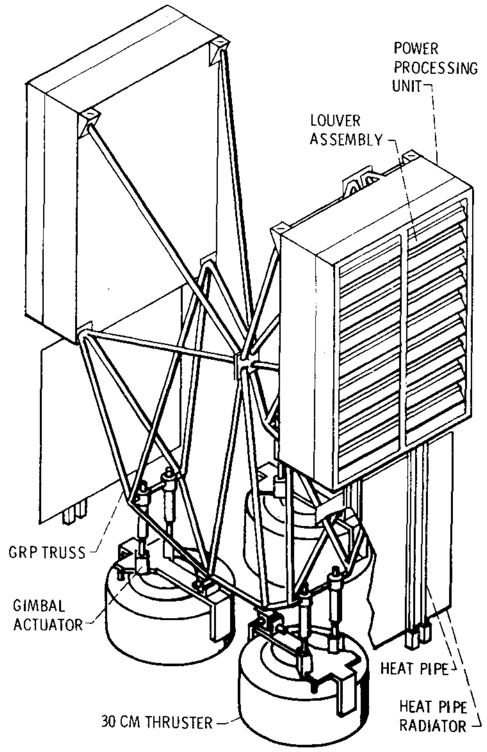

Fig. 3c Isometric view of HYBRID.

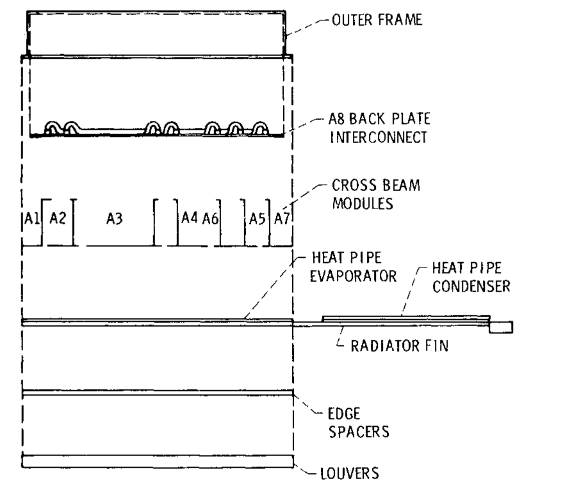

Fig. 4 Top view of exploded power-processing unit: TSSM assembly.

each saddle being redundant. An exploded view of the power processor and thermal control system for the TSSM and HYBRID is shown in Fig. 4.

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. 5. For the BIMOD assembly, the high heat dissipation flanges of the power processor are bolted directly to the heat-pipe evaporator saddles. Figure 5 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 PPU A on or near heat-pipe saddle A of Fig. 5. Because of the inverted orientation of PPU B, its high thermal dissipation components of the A3 and A4 modules now are 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 PPU's are operating. Figure 5 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 PPU's 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% larger than the required length calculated when using the 72% radiator efficiency shown in Table 2.

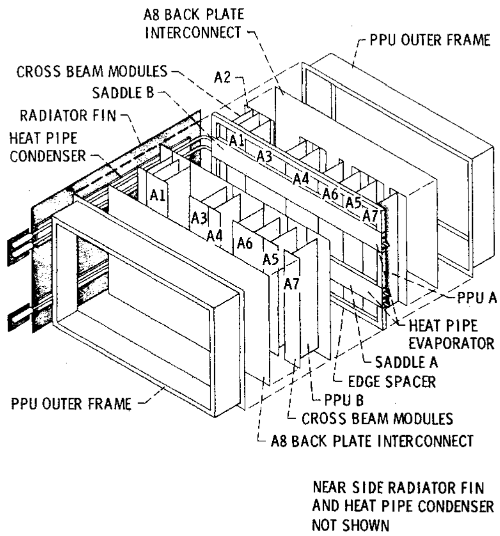

Fig. 5 BIMOD assembly.

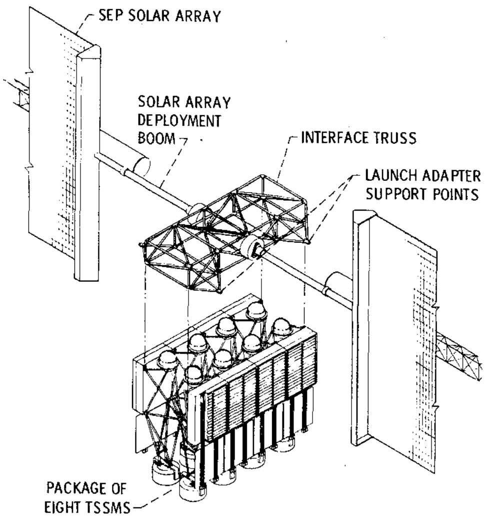

Fig. 6a Conceptual propulsion module using TSSM's.

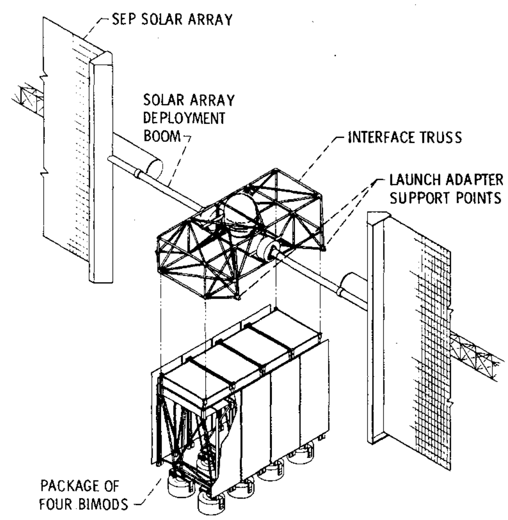

Fig. 6b Conceptual propulsion module using BIMOD's.

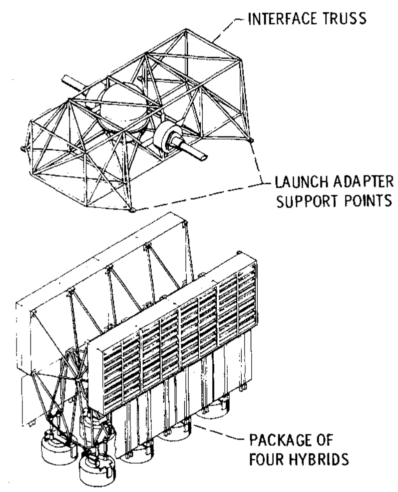

Fig. 6c Conceptual propulsion module using HYBRID's.

SEP Module Configurations

The SEP module configurations employing the TSSM, BIMOD, and HYBRID approaches are shown in Fig. 6. 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 that attaches to the interface truss. The distance between adjacent thruster centerlines is 0.61 m. The central propellant tanks for the BIMOD and HYBRID are shown supported in the interface truss.

Figure 6 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. 7 and 8 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.

A mass study for the modular thrust subsystem approach to a SEP module was presented recently in Ref. 6. The mass comparison given in Table 4 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 9 defines the gimbal angles α and β, which are rotations about axes nominally parallel to two of the spacecraft principal axes. By arbitrarily adding a 5-deg 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 deg and a total angle in the β direction of 30 deg.

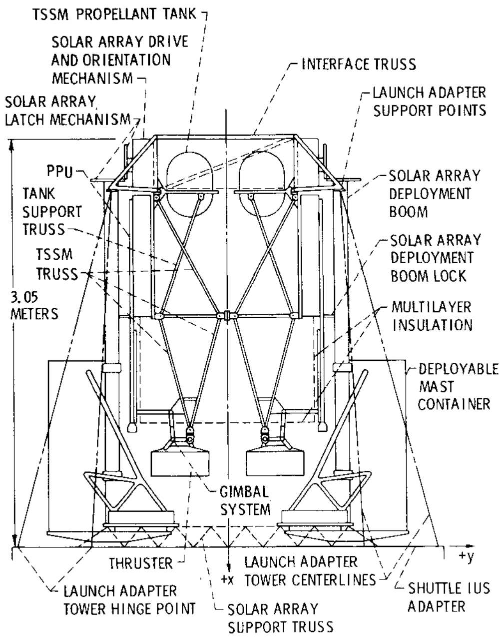

Fig. 7 End view of TSSM based SEP module.

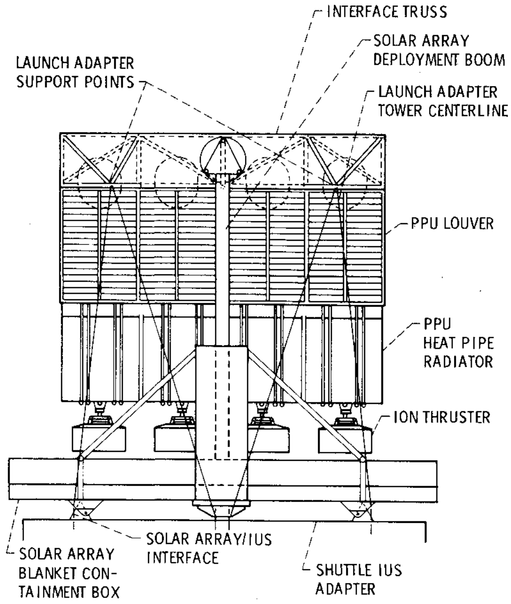

Fig. 8 Side view of TSSM based SEP module.

Figure 9 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

Table 4 Mass comparison for a six-thruster, 18-kW SEP module

	TSSM		BIMOD		HYBRID	
Power processor		176.87 kg		176.87 kg		176.87 kg
PPU thermal control		53.90		58.54		53.90
Thruster		48.96	THREE	48.96	THREE	48.96
Thruster gimbal system	SIX	17.70	BIMOD	17.70	HYBRID	17.70
PPU to thruster harness	TSSM	3.24		5.22		3.24
Structure		14.79		15.33		14.79
Propellant storage and distribution system		26.99		20.01		20.01
Propellant tank and line residuals		15.27		15.04		15.04
Tank support truss		------		2.12		2.12
Truss tubes, end fittings, module attach points and hardware, solar array drive brackets		14.74	Interface	15.81	Interface	15.81
Truss insulation		2.45	truss	2.86	truss	2.86
Component shelf stiffeners, controller shelf, and attach brackets	Interface truss	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		9.1		9.1		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	Solar array	106.56	Solar array	106.56	Solar array	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

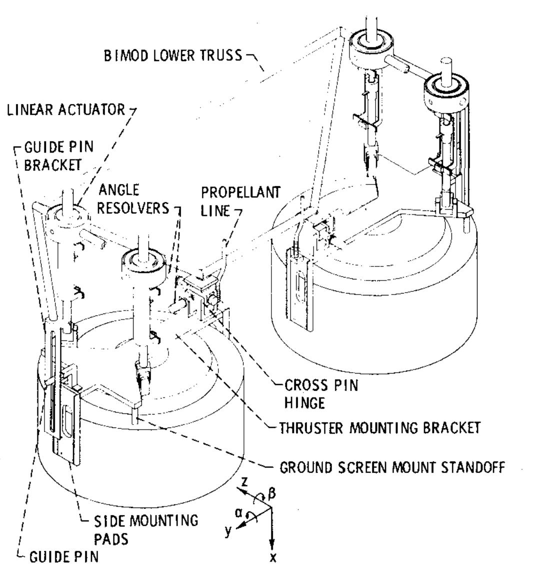

Fig. 9 Gimbal system/thruster interface, BIMOD configuration.

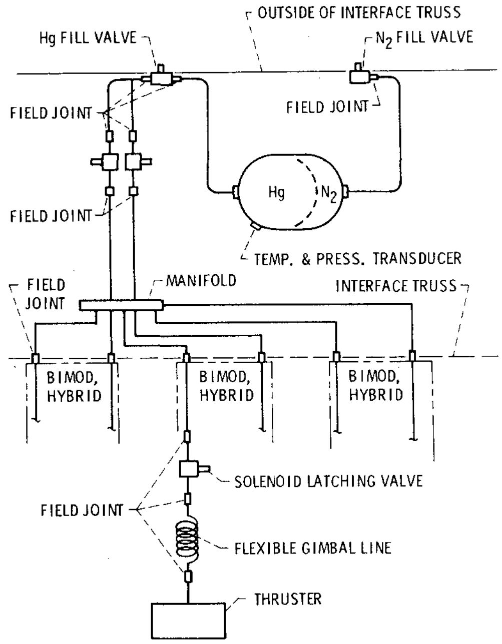

Fig. 10 BIMOD, HYBRID system schematic.

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.

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.

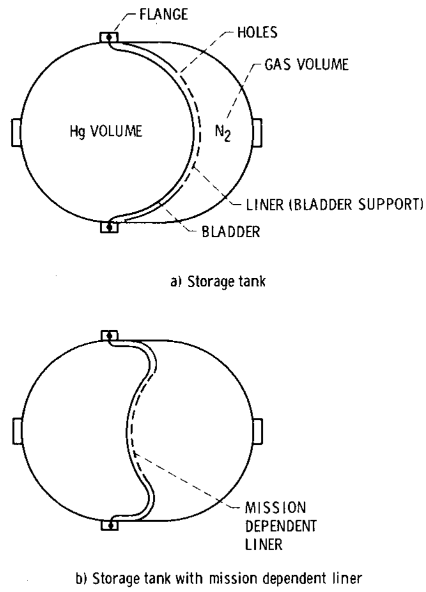

Fig. 11 Propellant storage tank concept.

Figure 10 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 also is contained in the interface truss. The distribution system interface between the interface truss and the modular thrust subsystems is 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.

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. 11a. 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, 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 that do not require the full sphere of mercury propellant. However, the shape of the bladder support liner is modified,

Table 5 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

as shown in Fig. 11b, 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%. 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 5 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, and 6) offers greater configuration flexibility. As shown in Table 4, 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.

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 the Communications Technology Satellite (CTS)[9,10] again has 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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