Thruster Subsystem Module for Solar Electric Propulsion

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Solar electric propulsion (SEP) is currently being studied for possible use in a number of near-Earth and planetary missions. Thruster systems for these missions could be integrated directly into a spacecraft or modularized into a thruster subsystem module (TSSM). A TSSM for electric propulsion missions would consist of a 30-cm ion thruster, thruster gimbal system, propellant storage and feed system, associated power processing unit (PPU), thermal control system, and complete supporting structure. The TSSM would be wholly self-contained and be essentially a plug-in or strap-on electric stage with simple mechanical, thermal, electrical, and propellant interfaces. The TSSM described is designed for a broad range of missions requiring from 2 to 10 TSSM's mounted in a 2 by x configuration. The thermal control system is designed to accommodate waste heat from the power processors based on realistic efficiencies when the TSSM is operating from 0.7 to 3.5 a.u. The modules are 0.61 m (2 ft) wide by 2.29 m (7.5 ft) long and have a dry weight including propellant tank of 54.4 kg (120 lb). The propellant tank will hold 145.1 kg (320 lb) of mercury.

Introduction

THE 30-cm ion thruster described by Sovey and King[1] has reached the state of development where it is ready for application as a primary propulsive unit. The power processors for these thrusters are developed to the point where preliminary packaging studies of the electronics have started. Various concepts for packaging the power processors have been investigated in support of recent solar electric propulsion (SEP) mission studies. The dual shear plate approach of housing the components, as proposed by Franzon, Fredrickson, and Ross,[2] was generally accepted as a baseline approach to packaging. The next level of packaging to be considered was the integration of the thruster subsystem into a spacecraft. Duxbury proposed a spacecraft design for the Encke slow flyby mission where the various components of the thrust subsystem were individually integrated into the spacecraft.[3] This paper presents a concept where the elements of the thruster subsystem are integrated to form a thruster subsystem module (TSSM). The TSSM would be a wholly self-contained strap-on electric stage with simple mechanical, thermal, power, and control interfaces with a spacecraft.

Present planning for SEP missions must accommodate a wide range of design requirements and options. The number of 30-cm ion thrusters required to fly the mission ranges from 2 to 10.[2-4] The quantity of propellant (mercury) required for these missions ranges from several hundred kg for earth orbit raising missions to over 1000 kg for some of the high-energy planetary missions. Satisfying this variety of requirements could lead to a family of relatively expensive custom-designed SEP spacecraft. However, a modular approach to the propulsion system for SEP spacecraft could lead to a standardized thrust system that would reduce costs and ease planning. From 2 to 10 thruster subsystem modules can be assembled into a thrust system and integrated with a spacecraft. Such thrust systems would be capable of performing all of the missions now under consideration.

A TSSM for these missions would consist of a 30-cm ion thruster, a thruster gimbal system, a propellant storage and feed system, a power processing unit (PPU), a modular thermal control system, and a modular support structure (Fig. 1). The TSSM's could be designed to be virtually identical with each other and thus completely interchangeable. The TSSM would have structural, power, telemetry, and command interfaces with spacecraft, but only structural, propellant, and thermal interfaces with other TSSM's.

Many benefits accrue from the use of TSSM's in an on-going series of SEP spacecraft. A qualification test program for the TSSM could be developed that would qualify the TSSM for the entire mission set. Since essentially identical TSSM's 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 thus be possible to use the flight spares of one mission as the flight units of the following mission to effect large cost savings. Reliability of the on-going missions would be greatly enhanced, since virtually identical hardware would be used.

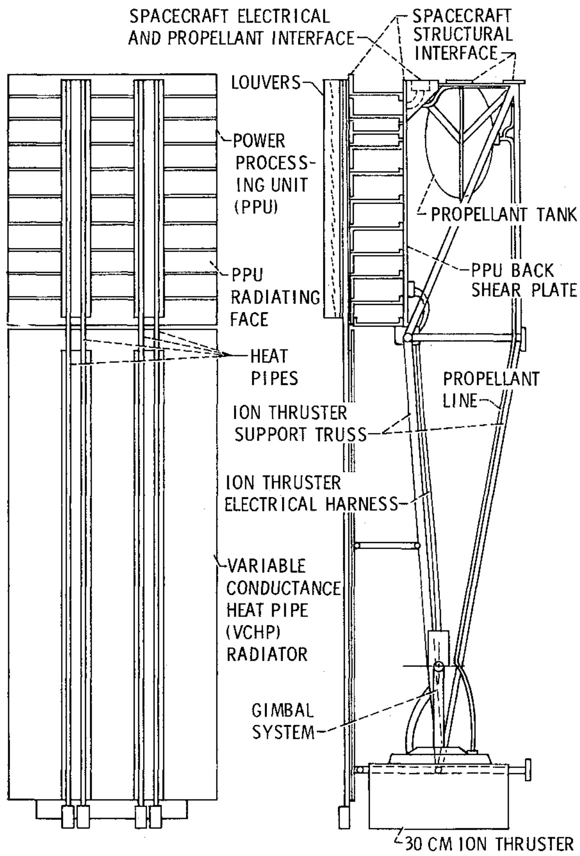

Fig. 1 Thruster subsystem module baseline design.

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However, all of these advantages would not accrue without some liability. It is not be possible to design a modular system that would be as weight optimized as a completely customized design. For example, weight penalities always occur where redundant structure (necessary for the testing of the individual modules) occurs and also the thermal control radiators for a 2 by x modular configuration can radiate only from one side (radiator weight would be halved if thermal radiation could occur on two sides). Another disadvantage is that the mandatory use of modules in later SEP spacecraft would not allow the design flexibility that might be gained by using just the individually qualified SEP components. However, the advantages of direct hardware cost savings associated with flight spare carryover, test program simplicity, reliability from previous flight experience, and prospective production cost savings far outweigh the disadvantages.

This report presents the TSSM general design requirements and approach. Integration and interfacing of TSSM's into 2 to 10 module assemblies is also discussed. A performance summary of the TSSM baseline concept is presented, including weight, dimensions, thermal range, gimbal angles, and total impulse.

TSSM Design Requirements

Launch Environment

The TSSM must be designed to meet the launch environmental requirements for all prospective missions. The maximum thrust and lateral quasistatic accelerations (launch acceleration plus launch low-frequency sine vibration) for the prospective launch vehicles are given in Table 1. The unamplified spacecraft sine and random vibration environments for the various prospective launch vehicles are given in Figs. 2 and 3. Figures 2 and 3 also depict a proposed vibration test specification for A TSSM.

Thermal

The TSSM must also be designed to meet the thermal requirements of a number of prospective electric propulsion missions. These requirements can be broken down into specific cases for each of these missions. These missions and their thermal environment requirements on which the thermal requirements are based can be found in Refs. 4-7. Further details on the thermal control system requirements can be found in Ref. 8.

Mechanical

Since the TSSM is to be used for a variety of missions it must be designed to meet the varied mechanical requirements of these missions. The chief mechanical requirements are those for the thrust vector orientation system or, more specifically, the thruster gimbal angles. The required maximum gimbal angles vary depending on the number of modules needed for the thrust system and the type of gimbal system chosen. The TSSM is designed to be assembled in clusters of from 2 to 10 units in a 2 by x configuration. The required gimbal angles for 10 modules are $\pm 31°$ and $\pm 8°$ to point the thrust vector of any one thruster directly through the spacecraft center of mass. The center of mass is assumed to be at the TSSM/spacecraft interface, as shown in Fig. 4. An additional gimbal angle capability of $\pm 5°$ is estimated to be required for spacecraft attitude control. However, the final gimbal angle requirements will be set by addition of 1) torque for attitude control, 2) torque for maneuvers (earth orbiting missions) and c.g. alignment. In conjunction with the thruster gimbal angle requirements, the thruster electrical harness and propellant feed lines are required to flex to allow full thruster gimbaling.

Table 1 Maximum thrust and lateral quasistatic launch vehicle accelerations

Launch vehicle	Thrust acceleration (g)	Lateral acceleration (applied simultaneously with thrust) (g)
Delta (2 stage)	+2.9/-1.0	2.0
	12.0	0.65
Atlas Centaur	7.2	1.0
Titan Centaur	5.6	1.3
Shuttle	+1.0	2.8
	-0.8	2.2
	-3.3	0.75
TSSM flight requirements	12.0	2.8
TSSM qualification requirements	18.0	4.2

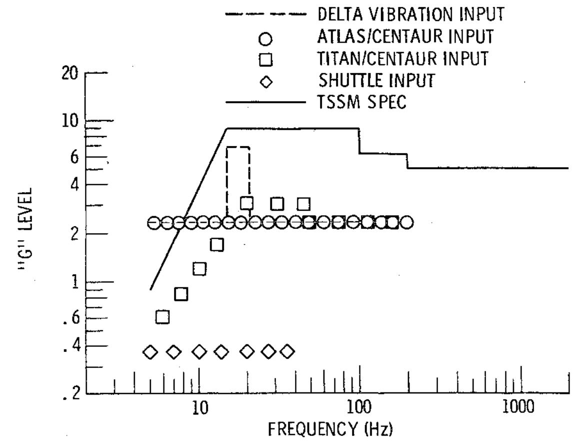

Fig. 2 Sine vibration qualification environment of module.

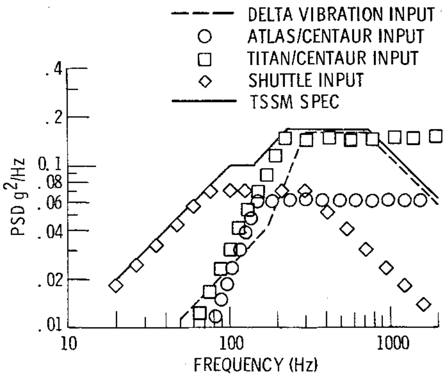

Fig. 3 Random vibration qualification environment of module.

Electrical

The mission electrical requirements are chiefly those of the PPU for the 30-cm ion thruster. The requirements principally concern the PPU operating electrical characteristics, command interface, and conducted and radiated electromagnetic interference. A detailed summary of the PPU electrical requirements can be found in Ref. 1 and 9.

The TSSM must, in addition to the PPU electrical interface, have a gimbal drive and gimbal telemetry electrical interface with the spacecraft.

General

The TSSM must, in addition to the specific requirements listed heretofore, satisfy the general requirements of: a) The overall weight of the propulsive section of a spacecraft using TSSM's must be within competitive range of that of a custom designed spacecraft. b). The TSSM must be easily assembled and disassembled from a SEP spacecraft. c) The TSSM must be easily serviced and repaired.

TSSM General Design Approach

The TSSM is designed as a strap-on electric stage. It consists of one 30-cm ion thruster, a two-degree-of-freedom gimbal system, a propellant storage and feed system for the entire TSSM which can be integrated into a multiple TSSM system with minimum change, and a support structure coupling all the subsystems into the TSSM. The TSSM has simple plug-in electrical and propellant interfaces and is essentially a unitized modular assmbly.

A baseline TSSM design (Fig. 1) was selected based in part on the results of the PPU packaging study[8] and on the additional need for some redundancy in the PPU thermal control system. The baseline design maximizes the separation between the thruster and the PPU and between the thruster and the propellant stowage tank to minimize the thruster gimbal angles required to point the thrust vector through the spacecraft center of mass (Fig. 4). However, there is also a need to keep the overall spacecraft height short to maintain the highest possible structural frequencies and to allow for maximum payload height. The TSSM height was minimized by positioning the thruster behind the variable conductance heat pipe system radiator. Figure 5 shows an artist's isometric of the TSSM and how several TSSM's would be integrated into a typical SEP spacecraft.

Thruster and Thruster Gimbal System

The thruster utilized in the TSSM is the 900 series thruster described in Ref. 1. It is an electron-bombardment mercury ion thruster of 30-cm nominal anode (and beam) diameter. The thruster gimbal system uses linear actuators because of their light weight and inherent simplicity. The gimbal system is capable of tilting the thruster $\pm 40°$ parallel to the length of the coupled TSSM's and $\pm 20°$ perpendicular to the PPU radiators. The 40° tilt is 9° further than the angle necessary to point through the assumed worst case spacecraft center of mass when 10 TSSM's are assembled as a spacecraft thrust system (see Fig. 4). The 20° tilt capability perpendicular to the PPU radiators is 12° greater than the angle necessary to point the thruster through the spacecraft center of mass. This excess gimbal angle capability exceeds that estimated for the attitude control function for most missions.

Propellant Storage and Feed System

Each thruster of each TSSM has an individual propellant storage tank. These tanks are located as near to the main spacecraft as possible to move the overall spacecraft center of mass away from the thrusters and thus minimize the thruster gimbal angles. The storage thanks each have lines feeding the individual thrusters. The storage tanks are also cross coupled to other TSSM's by a common manifold. If the storage tanks

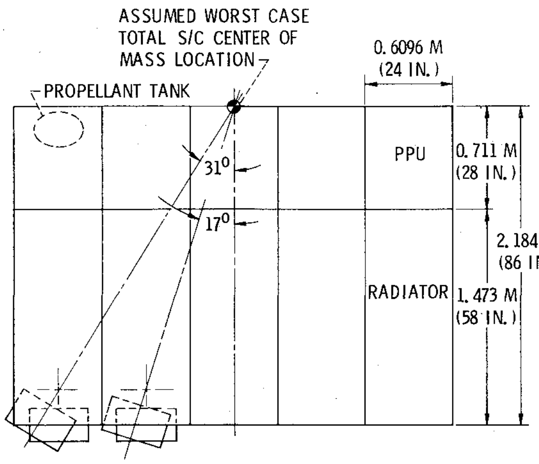

Fig. 4 Ten-module configuration.

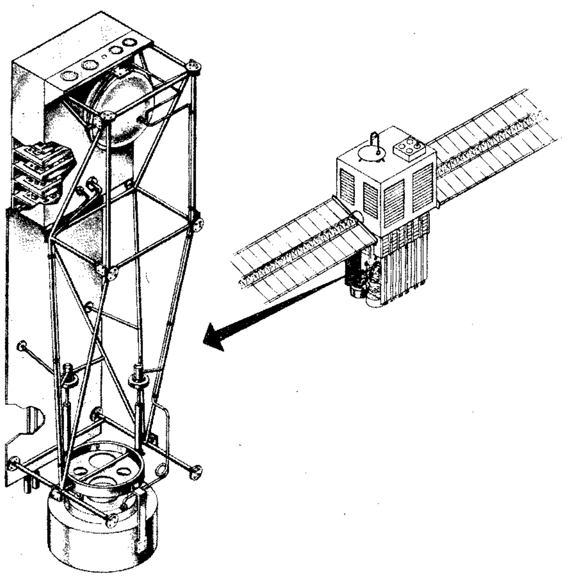

Fig. 5 Thruster subsystem module with typical solar electric propulsion spacecraft.

are not fully loaded, the spacecraft center of mass could be kept on the spacecraft centerline or shifted radially in any direction from the centerline by varying the amount of propellant in each tank. These tank propellant levels could be controlled either by controlling the tank pressurization gas temperature or by electromagnetically pumping the liquid metal propellant through a propellant valve-manifold system to other propellant tanks.

Power Processing Unit

The PPU for the 30-cm ion thrusters is discussed in detail in Refs. 10 and 11. The PPU packaging concept used in the TSSM has been discussed in detail.[8] This packaging concept compresses the PPU into a relatively small package.

This PPU packaging concept is ideally suited for the TSSM since the PPU and propellant tank center of mass can be moved as far as possible from the thrusters, thus minimizing the required thruster gimbal angles.

Thermal Control System

The TSSM baseline thermal control system uses direct thermal radiation of waste PPU heat through thermal control louvers combined with heat conduction of additional PPU waste heat by means of a variable conductance heat pipe system to an adjacent auxiliary radiator.[8] This sytem was chosen because it meets all the requirements of the cases of Table 2.

The thermal control system is sized by the thermal requirements of case Ia of Table 2, the 0.7 a.u. case. At 0.7 a.u. the PPU must radiate 387 W toward a relatively hot solar array (140°C) and thus a large radiator area is required. The heat that cannot be radiated directly from the PPU through the now open louvers on the PPU radiating face is conducted to an adjacent thermally isolated radiator by the heat pipes. The heat pipe radiator thermal isolation is necessitated by the need to limit PPU heat loss when either far from the sun (Table 2, case 1b) or when the spacecraft is eclipsed (Table 2, case 2b). When the TSSM is far from the sun, (Table 2, case 1b), the solar array is cold (-112°C) and the PPU dissipation low (12 W heater power only with an inoperative PPU). The inert gas of the heat pipe system expands and prevents heat conduction through the heat pipes. The louvers located on the PPU close and thus the PPU heater power requirements are kept low.

An alternative thermal control system for the TSSM would eliminate the louvers and use only the heat pipe system with its

Table 2 TSSM Missions and mission thermal environmental requirements

Mission	Case	PPU thermal dissi-pation (W)	PPU heater power (W)	Distance from sun (a.u.)	Solar array temper-ature[e] (°C)	PPU radiating face tem-perature (°C)	VCHPS radiator temperature (°C)
Comet[a] rendezvous	Ia	387	0	0.7	140	[d]60	+50
	Ib	0	12	3.5	-112	[d]-40	-135
Earth[b] orbital 14,824km (8000n.mi.) altitude or greater	IIa	387 c	0	1.0	50	[d]60	50 (15-deg periodic solar incidence angle to radiation faces)
	IIb	0	0	1.0	-220	[d]-40	
	IIb						-200 (72-min eclipse)

[a]Ref. 1. [b]Refs. 2,4. [c]Based on an 87% efficient PPU. [d]Based on 85°C PPU solid-state-device junction temperatures. [e]Solar array is assumed to be 69 in. away from PPU radiating surfaces. Assumed view factor from PPU to solar array is 0.17 (Ref. 9).

radiator. The main advantage of this system would be that it would further limit the heat losses associated with case Ib when the TSSM is far from the sun. Less heat is radiated because the PPU heat loss through the closed louvers is eliminated. The only heat losses that occur are through conduction of heat through only the metal of the heat pipes to the heat pipe radiator. If low conductivity metal (stainless steel) is used for the heat pipes, this loss can be minimized (see Ref. 8 for a weight-tradeoff study of PPU thermal control by an all-louver system, louvers with heat pipes and an all-heat pipe system).

Structure

The baseline TSSM structure consists of the PPU structure coupled with a graphite reinforced-plastic support truss. Figure 1 shows that the heaviest part of the TSSM, the propellant storage tank, is located very close to the TSSM-spacecraft structural interface, which is also in the same plane as the spacecraft-launch vehicle interface. Thus, the tank needs only to be supported by stabilization struts, since the tank lateral shear forces and thrust acceleration forces can be transferred directly into the spacecraft structure and then directly into the launch vehicle adaptor truss.

The PPU is also located close to the TSSM-spacecraft structural interface. However, the PPU structure also acts as a part of the overall TSSM structure. TSSM truss compressive and tensile forces are transferred through the PPU to the spacecraft by the PPU side members and the PPU back shear plate. Lateral truss forces and lateral PPU acceleration forces (parallel to the PPU radiating face) are transferred to the spacecraft by shear through the PPU back shear plate (see Ref. 8 for a more complete PPU structural description).

The ion thruster, thruster gimbal system, thruster electrical harness and propellant feedlines, and the heat pipe radiator are all supported by the graphite support truss. The truss is designed to transfer these loads to the spacecraft independent of the number of TSSM's coupled together. Graphite was chosen because of its low coefficient of expansion, light weight, and high stiffness to weight ratio.

The heat pipe radiator was not incorporated as part of the overall spacecraft structure because of its large temperature excursions (250°C), and resulting large expansions and contractions [0.864 cm (0.34 in.)] total for the 1.473 m (58 in.) long radiator shown in Figs. 1, 4, and 5). Also, the heat pipe radiator must be thermally isolated from the spacecraft to avoid large heat losses during spacecraft eclipse (Table 2, case 2b) or PPU shutdown (case 1b). Thermal insolation is virtually impossible to achieve through a good structural interface.

TSSM Baseline Weight and Performance Summary

TSSM Weight Summary

Table 3 summarizes the TSSM component weights. The PPU system weights are identical with those listed in Ref. 8 for the baseline PPU with thermal control by louvers in combination with heat pipes. The weights listed in Table 3 and in Ref. 8 for the louvers, louver support structure, evaporator heat pipes and saddles, and heat pipe gas reservoirs are also identical. However, the weights listed in Table 3 for the condenser heat pipes and saddles and the heat pipe radiator are heavier than those given in Ref. 8 since the heat pipe radiator was lengthened to meet the worst case thermal conditions.

The light weight of the support truss reflects the use of the graphite-reinforced-plastic for the truss members in combination with magnesium end fittings. The PPU to thruster electrical harness weight is derived from the known 30-cm thruster electrical power requirements.

Table 3 Thruster subsystem module weight

	Subtotals (kg)(lb)	Totals (kg) (lb)
1) PPU electrical components	13.88 (30.61)	
2) PPU harness	0.68 (1.50)	
3) PPU connectors	1.13 (2.50)	
4) PPU hardware	1.34 (2.96)	
5) PPU cross beams	2.37 (5.23)	
6) PPU back shear plate	0.42 (0.94)	
7) Total PPU		19.84 (43.74)
8) Louvers and louver support structure	2.49 (5.50)	
9) Evaporator heat pipes and saddles	0.84 (1.85)	
10) Heat pipe gas reservoirs	0.80 (1.76)	
11) Condenser heat pipes and saddles (58 in. long)	2.52 (5.56)	
12) Heat pipe radiator	2.06 (4.54)	
13) PPU thermal control total		8.71 (19.21)
14) Thruster support truss struts	1.24 (2.74)	
15) Thruster support truss fittings	0.51 (1.12)	
16) Thruster support truss hardware	0.10 (0.22)	
17) Thruster support truss total		1.85 (4.08)
18) PPU to thruster electrical harness		0.61 (1.35)
19) Thruster		9.07 (20.00)
20) Thruster gimbal system		4.58 (10.10)
21) Tank and bladder	9.07 (20.00)	
22) Feed system weight	0.76 (1.68)	
23) Feed system total		9.83 (21.68)
24) TSSM total weight		54.49(120.16)

Table 4 TSSM baseline performance summary

1) TSSM weight: dry weight = 54.49 kg (120 lb)
   wet weight = 199.5 kg (440 lb)
2) Overall size: 0.61 m (24 in.) wide × 0.61 m (24 in.) deep × 2.29 m (90 in.) long
3) Thermal range: 0.7 to 3.5 a.u.
4) Thruster gimbal angles: ±40-deg parallel to PPU radiator and ±20-deg perpendicular to PPU radiator
5) Thruster specific impulse: 3000 sec
6) Total impulse: 4,270,290 Newton-sec (960,000 lb-sec)

The thruster weight is for the 900 series 30-cm ion thruster (Ref. 1). The propellant storage tank and propellant feed system weights are for a 145. 1-kg (320-lb) capacity ellipsoidal mercury propellant tank. The propellant storage tank design is similar to that flown on the SERT II mission. The total dry TSSM weight is, then, 54.49 kg (120 lb) for a system capable of carrying 145.12 kg (320 lb) of propellant.

Performance Summary

The baseline TSSM performance summary is given in Table 4. The TSSM 199.5 kg (440 lb) wet weight is for a propellant load of 145.12 kg (320 lb). If more propellant is needed, larger tanks can be used. The propellant tanks can then be off-loaded for the less energetic missions. The 2.29 m (90 in.) overall length includes the thruster protrusion beyond the end of the VCHPS radiator.

The 3.5 a.u. thermal capability of the TSSM can be extended simply by the addition of more PPU heater power. However, the 0.7 a.u. limitation is governed by the permissible thruster temperature (thrusters have been tested up to a 0.63 a.u. thermal environment[12,13]) the permissible solar array temperature when tilted (currently 140°C) and the overall spacecraft thermal design. Thus, the 0.7 a.u. limit is not really a TSSM limit.

Conclusions

A thruster subsystem module for solar electric propulsion missions can be designed and built based on existing technology. The TSSM would have simple electrical, mechanical, thermal, and propellant interfaces and be essentially a strap-on electric stage. The module would be capable of meeting all of the environmental and operational requirements of the SEP mission set. The TSSM could be environmentally qualified for the entire mission set, so that the spares of one mission would become the flight units of the next mission, thus effecting large cost savings.

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