Interplanetary Spacecraft Design using Solar Electric Propulsion

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Emphasis of the NASA electric propulsion technology program is now on the application of solar electric propulsion to scientific missions. Candidate planetary, cometary, and geosynchronous missions are being studied. The object of this paper is to describe a solar electric propulsion spacecraft design proposed as the means to accomplish a comet slow flyby, a comet rendezvous, and an out-of-the-ecliptic mission. The discussion includes design differences foreseen for these missions and indicates those areas where spacecraft design commonality is possible. Particular emphasis is placed on a solar electric propulsion module concept which permits an attractive degree of design inheritance from mission to mission.

I. Introduction

THE introduction and integration of Solar Electric Propulsion (SEP) technology into a planetary spacecraft as the primary propulsion unit poses significant design challenges to the system engineers. Investigations of these challenges have been conducted in many past studies and more recently, in the NASA-sponsored SEP advanced system technology (AST) activity during FY74-75 with participation from the Jet Propulsion Laboratory, Lewis Research Center, Marshall Space Flight Center, and industry. The SEP spacecraft system which received considerable design and integration attention at JPL during the AST program is discussed here. This system, referred to as the bimodule design, has a mission module and solar electric propulsion module as its two major hardware elements. An alternate system concept, the SEP stage, is defined and described in Ref. 1 and 2.

The mission applications for which the bimodule SEP spacecraft design was considered were: 1) a comet Encke slow flyby launched in 1979, 2) an out-of-the-ecliptic solar observation mission launched in 1979, and 3) a comet Encke rendezvous launched in 1981. Either of the first two missions would serve as a meaningful precursor to the Encke rendezvous and future SEP missions of increased complexity such as those discussed in Ref. 3. That is, either mission would adequately demonstrate the spacecraft engineering design practices, interfaces, and techniques required to support an SEP module used for trajectory-shaping purposes to satisfy the scientific objectives.

A. Encke Slow Flyby (ESF)

The Encke slow flyby mission was studied for launch in early 1979 with a comet arrival late in 1980. The mission objectives are to determine the existence and character of the comet nucleus, its volatiles, and its interactions with the solar plasma. The elliptical orbit required to attain a 4-km/s flyby of the comet at 1000 km is shown in Fig. 1, which illustrates two of the spacecraft design drivers: the large variation in heliocentric distance from 0.7 to 2.5 a.u., and the thrust vector pointing profile which requires thrust beam steering across wide segments of the celestial sphere over the mission duration.

B. Out-of-the-Ecliptic (OOE)

The OOE mission studied places an SEP spacecraft into a 1-a.u. circular orbit (see Fig. 2) using a 15-kW electric propulsion capability to raise the heliographic inclination of the orbit to approximately 65° deg over a four-year mission duration. This type of orbit permits science data to be accumulated on the structure of solar features, solar wind properties, and the configuration of the outer planetary field at solar latitudes never before measured. To accomplish the inclination change, the spacecraft thrusts at periods centered about the orbital nodes, as illustrated by the arrows on Fig. 2.

C. Encke Rendezvous (ER)

In the Encke Rendezvous mission, spacecraft rendezvous with the comet occurs 50 days prior to perihelion (Fig. 3). The vehicle "station keeps" with the nucleus permitting visual and thermal mapping of the nucleus, analysis of the gases and solids flowing from the nucleus, and study of cometary temporal processes. The ER mission is considered to be launched in March 1981 and has a flight time of 1070 days.

II. Spacecraft Requirements

The solar electric propelled spacecraft must satisfy two separate and distinct sets of system design requirements. The first is the standard set of mission-dependent considerations

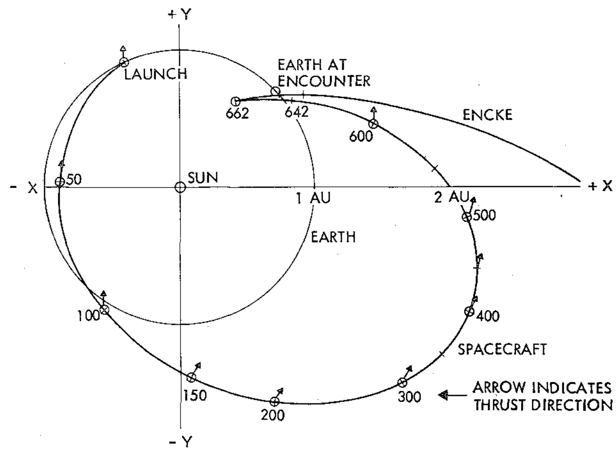

Fig. 1 Encke slow flyby trajectory, ecliptic plane projection.

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Index categories: Electric and Advanced Space Propulsion; Lunar and Interplanetary Spacecraft Systems, Unmanned; Spacecraft Propulsion Systems Integration.

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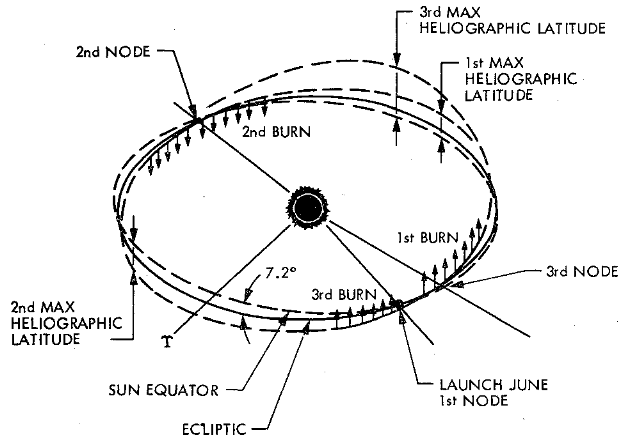

Fig. 2 **Out-of-the-ecliptic 1.0 a.u. circular orbit.**

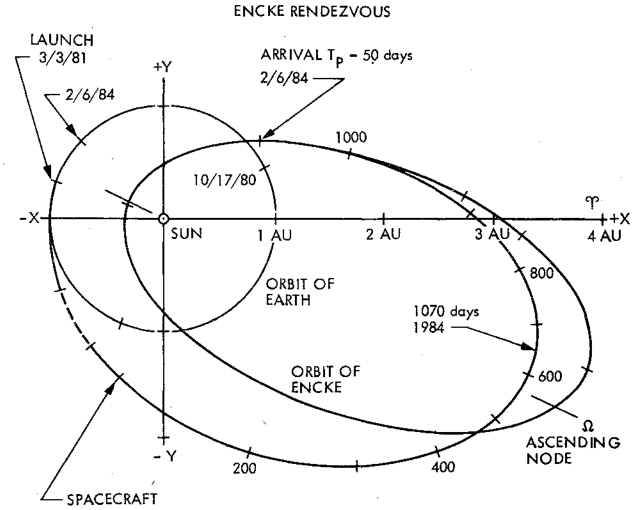

Fig. 3 **Encke rendezvous trajectory-ecliptic plane projection.**

related to the science payload, launch vehicle, and delivery path geometry and environment. The second is the set which arises solely from the introduction of SEP into the system. It represents a new or increased demand on the overall design in comparison with the conventional ballistic spacecraft.

Further elaboration of the mission-dependent system design requirements is given in Table 1. Detailed descriptions of the science payloads can be obtained from Refs. 4-6. Further discussion of the SEP-induced system design considerations is as follows:

Propulsion power: One dramatic departure from ballistic spacecraft is the high power level required to produce the ion thrust. Large deployable solar arrays are necessary to support the generation of the low thrust levels applied for long durations in the mission. For example, 15 kW of power into the propulsion subsystem produces 0.15 lb of thrust. A single solar wing in a two-wing 15-kW array can be 3 m in width and 30 m in length, assuming the 10 W/ft^2 solar cell technology. These large-celled areas increase the magnitude of the problem of selecting an overall vehicle configuration which satisfies science instrument viewing requirements, etc. In addition, flexure of the long, slender solar wings can have an impact on the attitude control subsystem design.

Temperature control: Attendant with handling high power levels is the problem of accommodating large heat losses due to the electrical inefficiencies of the power conversion and distribution hardware. For example, the ESF and OOE missions require approximately 15 kW of power into the thrust subsystem. At a power to thrust conversion efficiency of 63%, the spacecraft thermal control design must handle 5.5 kW of heat generated by the propulsion unit. The radiator area of each power processor required to reject the waste heat into black space to maintain component temperatures within acceptable limits approaches 1.0 m^2 for the comet missions. Depending upon the number of power processor units and the mission thermal environment, the physical size of the SEP module, considering just the temperature control factor, can exceed the dimensional constraints imposed by standard launch vehicle shrouds.

Thrust vector pointing: Thrust vector pointing encompasses three functions: 1) alignment of the resultant thrust vector from a cluster of thrusters through the spacecraft center-of-mass, 2) articulation of the ion thrusters to produce torques about the spacecraft primary axes as required to maintain three-axis attitude stabilization, and 3) orientation of the resultant thrust vector in inertial space to achieve the desired trajectory shaping. These factors will directly impact the spacecraft configuration and the attitude control subsystem interface and control requirements.

Command control: Due to considerations such as thrust vector control, throttling, and thrust parameter variation in flight, new command control requirements must be addressed. Means must be provided to articulate the ion thrusters to meet thrust steering requirements, to command thruster startup and shutdown sequences, and to react to SEP module anomalies. This control will require stored on-board computer programs and digital interfaces, which are all more complex than required for propulsion on previous planetary spacecraft.

Table 1 Mission-induced spacecraft design requirements

Requirement	ESF mission	OOE mission	ER mission
Navigation	1000 ± 500 km flyby altitude 4 km/s flyby velocity	65 deg heliographic inclination in 1.0 a.u. orbit	Comet rendezvous 50 days prior to periapsis
Mission lifetime	662 days	1554 days	1070 days to rendezvous
Thrust profile	Continuous to E-20 days	9 burns centered about the orbital nodes	Continuous to rendezvous and as needed for circumnavigation
Power	15-kW solar array	Dedicated 16-kW solar array	18-kW solar array
Data handling and telecommunications	Data rate: 60 kbps; no data storage; articulated HGA	Data rate: 2 kbps; no data storage; articulated MGA	Data rate: 60 kbps; tape recorder: 5 × 10-bit capacity; articulated HGA
Temperature control	0.7 to 1.5 a.u. heliocentric range; power processor louvers required	Constant 1.0 a.u. heliocentric range; power processor louvers not required	0.5 to 3.6 a.u. heliocentric range; power processor louvers required
Thrust equipment	6 thrusters; 6 power processors; 530-kg mercury	6 thrusters; 6 power processors; 1600-kg mercury	8 thrusters; 6 power processors; 585-kg mercury
Attitude and articulation control	Solar array articulated; gimbaled engines; gimbaled star tracker; nucleus detector; xenon cold gas; scan platform	Solar array body-fixed; gimbaled engines; gimbaled star tracker; no nucleus detector; N_2 cold gas; no scan platform	Solar array articulated; gimbaled engines; gimbaled star tracker; nucleus detector; xenon cold gas; scan platform
Science payload	9 instruments—62 kg, 70W	6 instruments—29 kg, 1018 W during coast	9 instruments—86 kg, 204 W

Electromagnetic interference (EMI): Present designs indicate solar array voltages will exceed 150 V and currents can exceed 50 A per wing. These current and voltage levels represent EMI design challenges that must be addressed to assure compatibility with the science and electronic subsystems on the spacecraft.

III. Spacecraft System Design

An SEP spacecraft concept that has received considerable attention at JPL over the past several years is the bimodule design.[7] For this concept, the spacecraft system consists of two major separable hardware modules–the mission module and the SEP module (Figs. 4 and 5).

A. Mission Module

The mission module is the communications and control center of the SEP vehicle; it also serves as the mounting base for the scientific instruments and contains science support elements, such as a scanning platform. Specific subsystems assigned to the mission module are radio, antenna, modulator/demodulator, flight data, data storage, attitude control, pyrotechnics, and power conditioning and distribution for these units. These subsystems are mostly derivatives of existing Mariner and Viking hardware designs. Where possible, advantage can also be taken of NASA standard hardware, such as the standard tape recorders currently under development.

The mission module electronic subsystems are contained within a 10-bay electronics bus. The electronic compartment size is compatible with Viking Orbiter and Mariner Jupiter-Saturn spacecraft sybsystems, allowing inheritance of the previous packaging designs, which reduces design effort and cost. A short synopsis of the electronic subsystems housed within the mission module is as follows:

1) Telecommunications (radio frequency, modulation/demodulation, and antennas): S-band uplink and downlink using 10/30-W transmitter. Downlink requires high-gain antenna for the comet missions and a medium-gain antenna for the OOE mission. Low-gain omni-antenna is used to support a 4 b/s command uplink.

2) Power: Converts raw power from SEP module to 2.4-kHz and 400-Hz ac power, 30-V and 56-V dc power; 30-A-h NiCa battery for launch and solar occulation phases.

3) Data handling and control: Flight data subsystem: Collects and formats all data; sequences science instruments; 8192 16-bit word programmable memory; spacecraft master clock. Computer command: decodes commands, sequences mission module and SEP module subsystems, and monitors SEP module performance; 8192 18-bit word programmable memory; data storage capacity is 5×10^8 bits for reduced data rate playback.

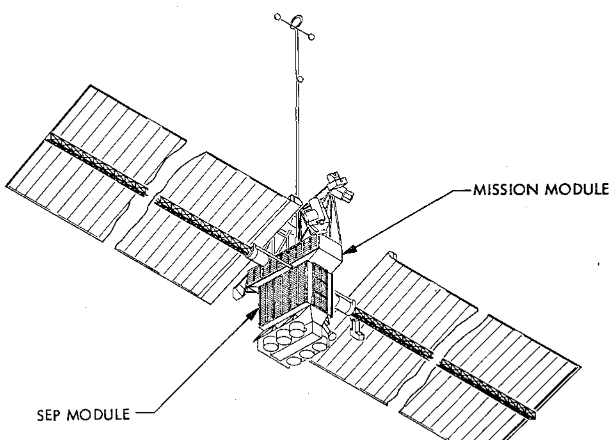

Fig. 4 Encke slow flyby spacecraft.

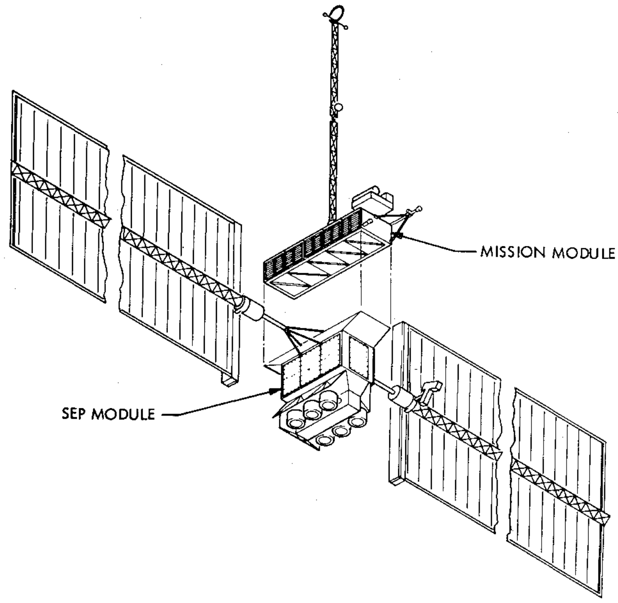

Fig. 5 Out-of-ecliptic SEP Spacecraft.

4) Attitude and articulation control: Maintains three-axis stabilized spacecraft using sun and star sensors. Control torques from SEP module during thrust operations and from cold gas system during coast. 8192 18-bit word programmable memory. Articulates thrusters, antennas, solar arrays and science instruments.

B. Solar Electric Propulsion Module

The SEP module contains the primary propulsion unit and power source (solar arrays) of the SEP spacecraft. During the thrust phases of the mission, vehicle attitude stabilization torques are generated by the SEP module ion thrusters. The SEP module design takes advantage of capabilities existing in the mission module for such support functions as data handling and transmission, attitude control electronics, and command computer rather than duplicating these functions at increased mass, volume, and cost. Unlike the mission module, many of the hardware elements in the SEP module represent new technology currently under development by NASA. A building-block architecture has been selected for the SEP module, which allows the base design to grow or shrink in incremental hardware steps to match the varying mission-to-mission propulsion performance requirements. The intent of this particular design approach is to minimize SEP module redesign costs for follow-on applications beyond the initial mission.

The primary functions of the SEP module are: 1) to convert solar electric power into a directed mercury ion beam, which imparts a net thrust force upon the spacecraft in a controlled and acceptable manner relative to the required mission thrust profile. 2) To provide torques about each of the three principal axes of the spacecraft during thrust periods to maintain stable spacecraft attitude control. 3) To store and distribute mercury propellant to the 30-cm mercury electron-bombardment ion thrusters. 4) To generate the primary photovoltaic power necessary to operate the SEP module and mission module. 5) To distribute primary photovoltaic power directly to the power processors. 6) To distribute approximately 500 W of raw dc power to the mission module.

The SEP module is subdivided into three assemblies: the solar array assembly, the power assembly, and the thrust assembly. Together they provide the functions just listed. Figure 6 illustrates the functional interfaces between the SEP module and mission module. The SEP module is physically attached to the mission module via the structural element designated the SEP module adapter.

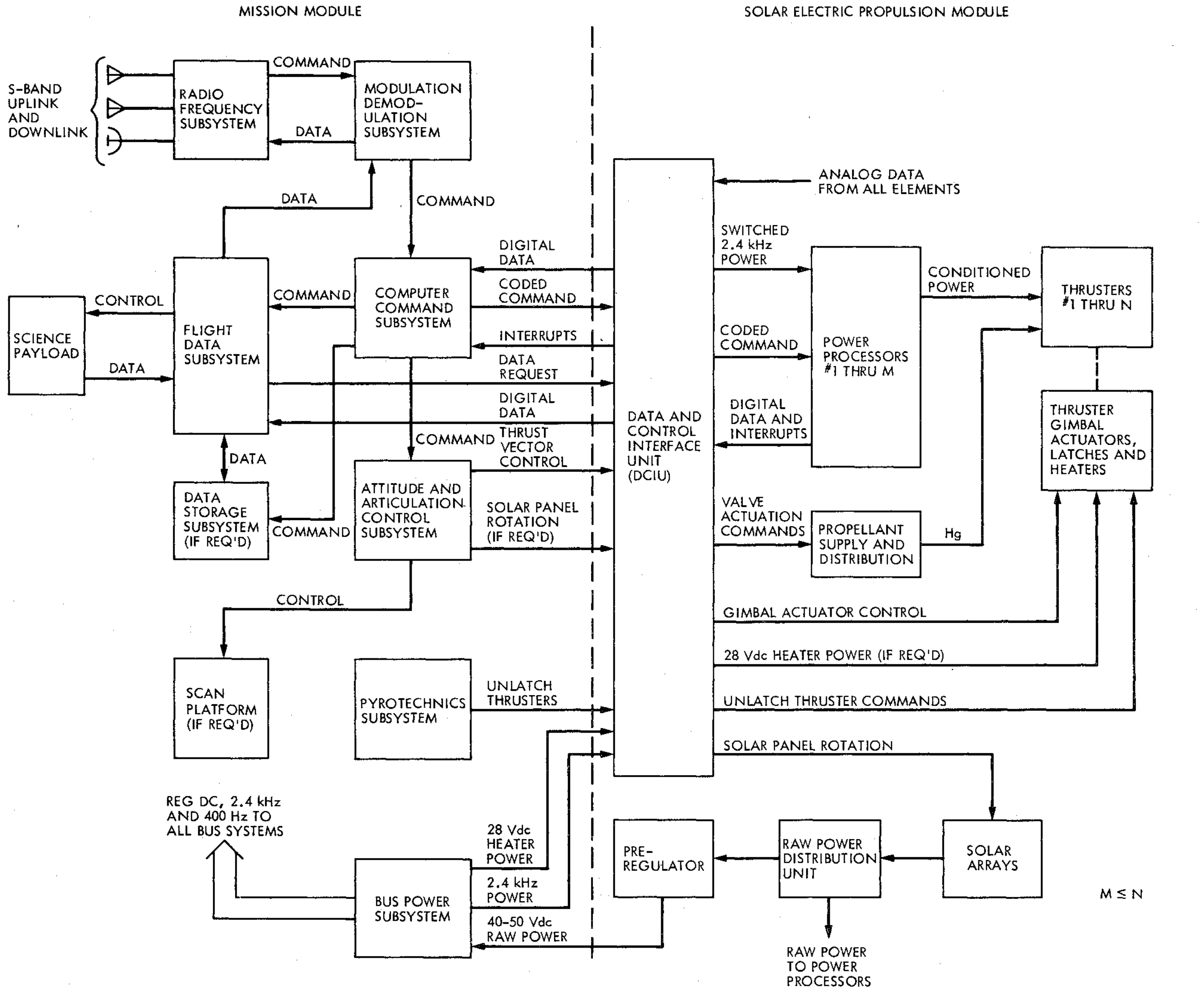

Fig. 6 SEP module functional block diagram.

C. Solar Array Assembly

The solar assembly provides the raw dc power required for operation of the SEP spacecraft. The power is produced from flat-fold, deployable solar arrays containing photovoltaic cells. The cells are mounted on a blanket, which is stored in a cannister at launch and deployed by means of a boom after launch.

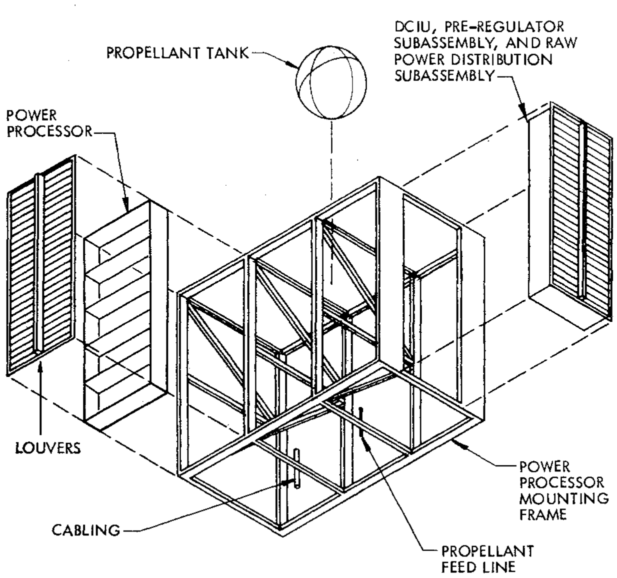

Fig. 7 Power assembly.

For the missions studies here, the solar array size varies from 15 to 18 kW of raw dc power at Earth. The associated solar cell area varies from approximately 1500 to 1800 ft^2, assuming the 10-W/ft^2 cell technology.

D. Power Assembly

The power assembly (Fig. 7) contains the ion engine power processors and houses the propellant tank and support electronics for the SEP module. The power assembly receives raw dc power from the solar arrays. The raw power distribution unit routes this power to the power processors and to the mission module. For the OOE mission, the power for the mission module can be obtained directly from a dedicated solar array section, which produces voltages compatible with Mariner power equipment (40-50 V dc). In this option, the mission module power subsystem is electrically isolated from the SEP module. For the comet missions, the mission module power must first be processed at the power processor voltage level (e.g., 200-400 v dc) by a preregulator unit and converted to the desired 40-50 V dc level.

E. Power Processor

The power processor accepts raw power from the solar arrays through the raw power distribution unit and conditions this power for the ion thrusters. The 12 different voltage-current supplies required to operate the thruster are provided by the power processor. For a thruster operating at full thrust (2A beam current for 30-cm mercury ion bombardment thruster), the power processor input is 2960 W, assuming a conversion efficiency of 91%. In addition, the processors have several control loops, which can be preset by the spacecraft command computer for stable operation.

The exterior face of the power processor operates as a radiator of internally generated heat when operating near the sun. Temperature-sensitive louvers are added to the outer face of the power processor to retain this waste heat when operating at or near aphelion in the comet missions. The constant heliocentric distance characteristic of the OOE mission relaxes the thermal control design problem, permitting removal of the louvers.

The power processor mass will vary depending on redundancy employed and the radiating area required. The projected power processor mass for the comet missions is 30.0 kg including louvers; the estimate for the OOE mission is 26.0 kg assuming removal of louvers and reduction in radiator area consistent with the fixed 1.0 a.u. heliocentric distance.

F. Propellant Storage and Distribution Assembly (PSDA)

The mercury propellant, ejected as an ion stream by the thrusters, is stored in a stainless steel tank that contains a bladder and pressurant to expel the mercury. The mercury is routed to the thrusters by means of a series of propellant feedlines and latching valves. The propellant load required is under 600kg for the comet missions and 1600 kg for the OOE mission. Detailed PSDA definition and component design is provided in Ref. 12.

G. Data and Control Interface Unit

The data and control interface unit (DCIU) interfaces command and telemetry data between the mission module and SEP module. The DCIU accepts digital commands from the mission module computer command and flight data subsystems. These commands enable the mission module to control the SEP module from preprogrammed algorithms that can be updated in flight by ground command.

The DCIU provides the analog-to-digital conversion of SEP module analog telemetry signals, such as propellant tank pressure. In addition, the DCIU contains logic for switching 2.4-kHz power to the power processors. Interrupts from the power processors are monitored within the DCIU and an alert is generated to the mission module when a problem is detected.

H. Thrust Assembly

The thrust assembly (Fig. 8) attaches to the bottom of the power assembly and contains the thrusters. The structural element of the thrust assembly supporting the thrusters, gimbal actuators, valves, cabling, and launch latches is termed the thruster mounting frame. The frame can easily tolerate growth in the number of thrusters without significantly altering its basic design.[8] Figure 8 illustrates a 2×3 array of thrusters. In the building-block concept, the array can be increased or decreased to obtain a $2\times N$ configuration while maintaining the same design concept. In the same fashion, the number of power processors in the power assembly can be changed by widening or narrowing the power processor mounting frame.

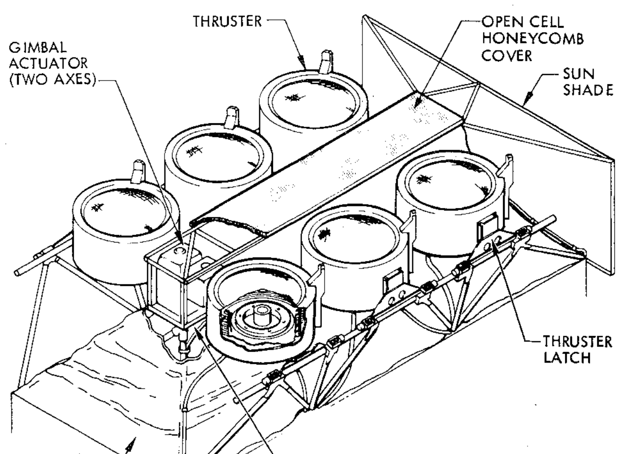

Fig. 8 Thrust assembly.

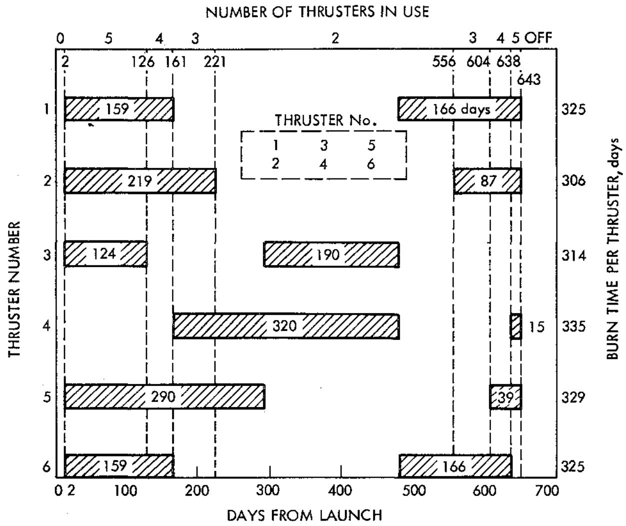

Fig. 9 Thruster sequencing for ESF mission.

I. Thrusters

The number of thrusters required in the SEP module is dependent on thruster lifetime and total velocity increment required. The thruster lifetime constraint factors into determining the number of thrusters required to meet the total impulse (A-hr) demanded. Figure 9 illustrates a thruster usage profile for an ESF mission that uses a thruster sequencing plan to meet the thruster lifetime constraints while accomplishing the required mission velocity increment. A similar profile has been developed for the ER mission. The OOE mission requirement, on the other hand, is met by operating five thrusters continuously for their rated lifetime.

To avoid a single failure of a power processor or thruster from jeopardizing the mission, at least one spare thruster and one spare power processor are included in each of the mission designs.

J. Gimbal Actuator

Each gimbal actuator permits motion of its thruster in two orthogonal directions. With two degrees of freedom on each thruster, it is possible to steer them to provide the torques necessary for maintaining 3-axis stabilization of the SEP spacecraft.[9] As discussed earlier, the orientation of each thruster is controlled by signals from the articulation and attitude control subsystem located in the mission module. This subsystem is a derivative of the unit being designed for the Mariner Jupiter/Saturn spacecraft.

IV. SEP Module Design Requirements and Performance

The information presented in this section defines key design parameters of the SEP module for each of the missions of interest. The mission design assumptions related to this information are listed in Table 2. The Titan C/Centaur D1-T launch vehicle was a common assumption for the mission set, as was the 30-cm thruster performence documented in Ref. 7.

A. Encke Slow Flyby

The relationship between final spacecraft mass, expelled mercury propellant, and installed solar array power (plotted in Figs. 10 and 11) was obtained from Ref. 10. The dashed lines correspond to the design point selected for the SEP module. In other words, the SEP module design requirements based on a 1260-kg final spacecraft mass are: 1) 15 kW solar array—installed power, 2) 530 kg of usable propellant, and 3) provisions to run a maximum of five thrusters simultaneously at 1.5-A beam current whenever adequate power is available

Table 2 Mission assumptions related to SEP performance calculations

Parameter	ESF	OOE	ER
Launch data	1/14/79	6/6/79	3/3/81
Launch azimith	Constrained	Unconstrained	Constrained
Flight time (days)	662	1554	1070
Thrust subsystem lifetime (khr)	10	20	15
Thrust vector steering	Unconstrained	Perpendicular to orbital plane	Unconstrained
Thrust profile	Minimum of two thrusters operating at aphelion; terminal coast period of 20 days	9 thrust periods centered about nodes; coast periods centered about antinodes	Minimum of two thrusters operating at perihelion
Target	Flyby comet at 4 km/s	Maximize final heliographic inclination	Rendezvous with comet 50 days before perihelion

to the thrust subsystem. A further listing of SEP module requirements is given in Table 3.

B. Out-of-the-Ecliptic

The relationship beween net spacecraft mass and final heliographic inclination for various power levels into the thrust subsystem is plotted in Fig. 12.[11] Net mass is defined as the final spacecraft mass less dry SEP module mass. It equates directly with the maximum allowable mission module mass. The design point for the OEE SEP module allows delivery of a 370-kg mission module, including 29-kg science payload, to a heliographic latitude of 63 deg in about 4.25 years. The characteristics of the SEP module consistent with this design point are 1) 16-kW solar array-installed power, of which 15 Kw is available to the thrust subsystem, 2) 1600 kg of usable propellant, and 3) provisions to operate five thrusters simultaneously at a 2.0-A beam current during all thrust phases of the mission. A listing of SEP module requirements for the OOE mission is included in Table 3.

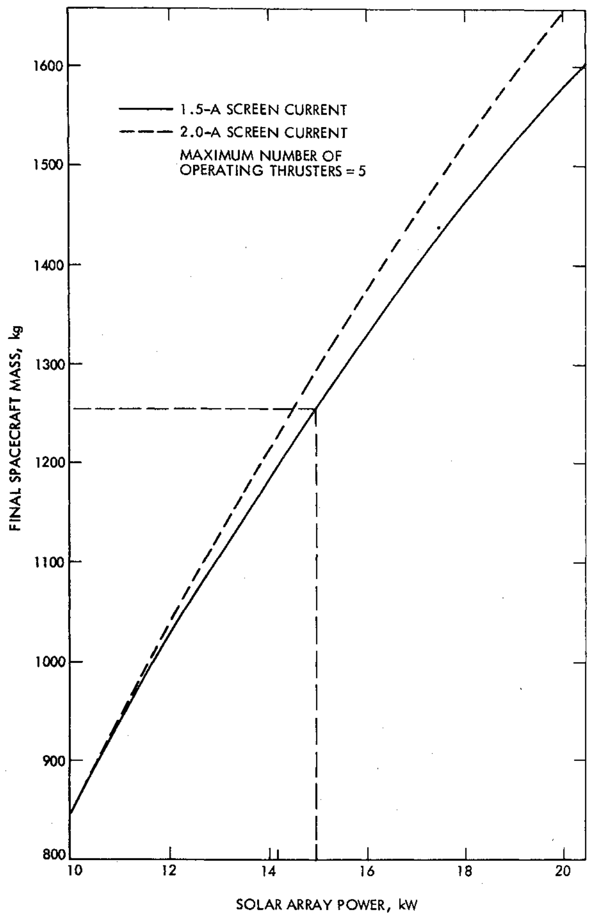

Fig. 10 Encke slow flyby final spacecraft mass.

C. Encke Rendezvous

The relationship between final spacecraft mass at rendezvous and expelled propellant vs power into the thrust subsystem is plotted in Fig. 13. The SEP module design requirements for the ER mission include 1) 18-kW solar array-installed power, 2) usable propellant of 585 kg, and 3) provisions to run five thrusters simultaneously at 2.0-A beam current whenever adequate power is available to the thrust subsystem. This design is capable of delivering a final

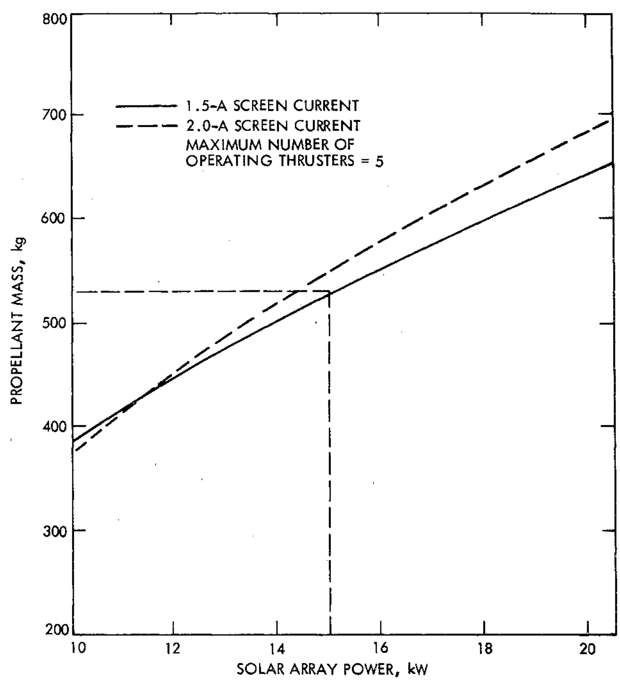

Fig. 11 Encke slow flyby propellant mass.

Table 3 SEP module design requirements

SEP module	ESF	OOE	ER
Solar array assembly			
No. of wings	2	2	2
Installed power (kW)	15	16	18
Rotatable	Yes	No	Yes
Thrust assembly			
Array geometry	2×3	2×3	2×4
Min operating	2	0	2
Max operating	5	5	5
Spare thrusters	1	1	3
Power assembly			
Array geometry	2×3	2×3	2×3
Spare PWR processors	1	1	1
Switch matrix	No	No	Yes
Louvers	Yes	No	Yes
Propulsion tank capacity (kg)	530	1600	585

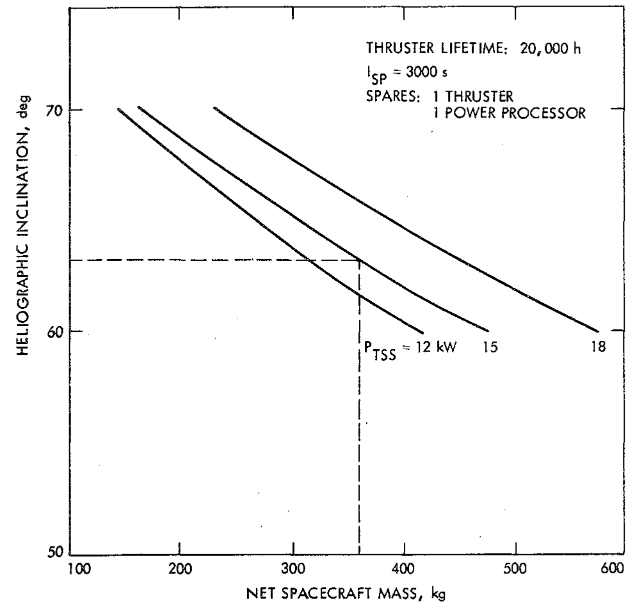

Fig. 12 OOE mission, circular 1.0 a.u. orbit.

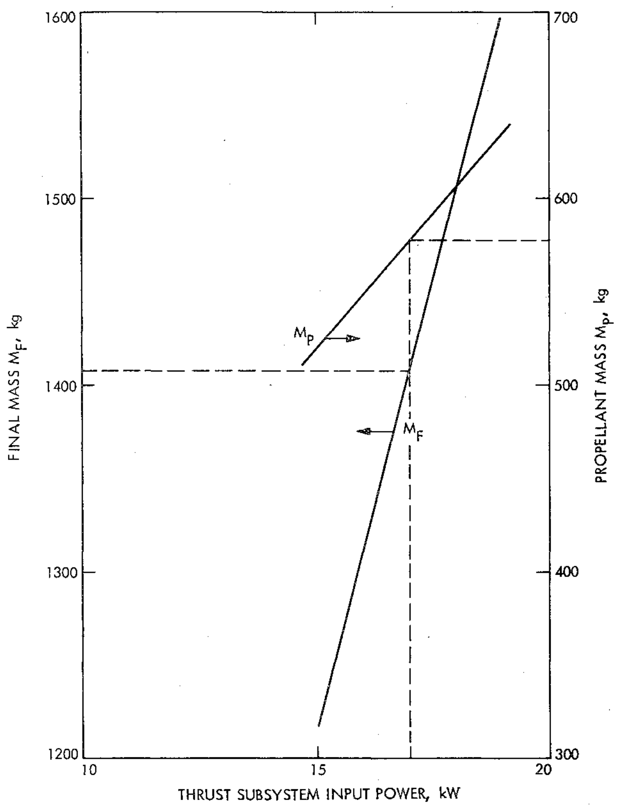

Fig. 13 Encke rendezvous final spacecraft mass and propellant mass.

spacecraft mass of 1410 kg to the desired rendezvous point. A further breakdown of SEP module requirements for the ER mission is included in Table 3.

V. Mass Summary

Table 4 lists representative spacecraft masses by subsystem for the three spacecraft designs discussed. The final spacecraft masses for the three missions are compatible with the performance assumptions and calculations described in Sec. IV.

Table 4 Spacecraft mass summary

Component	ESF	OOE	ER
Mission module			
Science instruments	62	29	86
Engineering subsystems	350	327	350
SEP module			
Solar array assembly	254	268	300
Power assembly	430	337	441
Thrust assembly	91	10	10
Spacecraft final mass	1197	1062	1311
Propellant	530	1600	585
Launch vehicle adapter	50	50	50
Total launch mass (kg)	1777	2712	1946

VI. Conclusions

The bimodule SEP spacecraft system design will accomplish the missions investigated. An attractive level of mission module and SEP module design commonality appears possible from mission to mission. The solar electric propulsion module building-block approach suggested allows expansion or reduction in propulsive capability without requiring a new propulsion system design effort for each mission. In addition, the mission module is comprised entirely of subsystems developed on previous planetary programs and will contribute heavily in keeping SEP spacecraft costs within acceptable levels.

References

[1]"Concept Definition and Systems Analysis Study for a SEP Stage," Boeing Aerospace, Seattle Wash., Final Rept. on NAS8-30921, D180, 18553, Jan. 1975.

[2]Concept Definition and System Analysis Study for a SEP Stage," Rockwell International, Anaheim, Calif., Final Rept. on NAS 8-30920, SD 74-SA-0176, Feb. 1975.

[3]Atkins, K., "U.S. Solar-Electric Propulsion Planetary Mission Candidates: Out-of-the Ecliptic, Small Bodies, and Orbiter of Mercury and Satury," III EEPC Paper 74-243, Oct. 1974.

[4]Newbury, R.L., "Science Rationale and Instrument Package for a Slow Flyby of Comet Encke," Jet Propulsion Lab., Pasadena, Calif., Internal Doc. 760-90, June 1, 1973.

[5]Duxbury, J.H., "A Circular 1.0 a.u. Out-of-the-Ecliptic Mission Using Solar Electric Propulation,"III EEPC Paper 74-242, Oct. 1974.

[6]Newburn, R.L., "A Rendezvous with Comet Encke Using Solar Electric Propulation," Jet Propulsion Lab., Pasadena, Calif., Internal Doc. 701-201, to be published.

[7]Gerpheide, J. and Duxbury, J., "Solar Electric Propulsion Encke Slow Flyby 1979 Mission and Spacecraft Description," Jet Propulsion Lab., Pasadena, Calif., Internal Doc. 701-200, June 28, 1974.

[8]Ross, R.G., "Thruster Array Approaches for a Solar Electric Propulsion Encke Flyby Mission," AIAA Paper 73-1115, Lake Tahoe, Nev., 1973.

[9]Marsh, E.L., "Multiple Gimbaling of Thrusters for Thrust Vector Control and Thrust vector Reorienting of Solar Electric Spacecraft," AIAA Paper 73-1116, Lake Tahoe, Nev., 1973.

[10]Sauer, C.G., "Trajectory Design of a Solar Electric Propulsion Slow Flyby Mission," AIAA Paper 75-421, New Orleans, La., 1975.

[11]Yen, C.L., "An Optimal Transfer to Out-of-the-Ecliptic Using a Solar Electric Spacecraft," presented at the AIAA 11th Electric Propulsion Conference, New Orleans, La., 1975.

[12]Womack, J.R. and Smith, R.H., "Development of a Mercury Propellant Storage and Distribution Assembly," AIAA Paper 75-408, New Orleans, La., 1975.