A Nuclear Electric Propulsion Vehicle for Planetary Exploration

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A design study currently is underway for a nuclear electric propulsion vehicle capable of performing detailed exploration of the outer plants. Evaluation of the design indicates that it also is applicable to orbit raising. Primary emphasis is on the power subsystem. Work on the design of the power system, along with both the mission rationale and preliminary spacecraft design, is summarized. A propulsion system at a 400-kWe power level with a specific weight goal of ≤25 kg/kW was selected for this study. Study results indicate that this goal can be realized, together with compatibility with the shuttle launch vehicle constraints.

Introduction

WHEREAS exploration of the inner planets of the solar system may be accomplished by the use of solar electric propulsion or solar sails, the detailed exploration of the outer planets (Jupiter to Pluto) cannot be accomplished by these sun-dependent spacecraft. Completion of the investigation of the solar system will require sun-independent self-propelled vehicles. The great distances involved and the desirability of accomplishing exploration of several planetary bodies in a given mission places great emphasis on long-life, high-reliability, automated vehicles. Long trip times, however, make these missions unattractive in terms of generating interest and the economic feasibility of maintaining mission operations over an extended period of time. A nuclear electric propulsion (NEP) vehicle with a lightweight propulsion system represents a highly attractive method of accomplishing these missions[1] within a reasonable flight time and with sufficient payload and propulsion capability to allow for highly automated and detailed investigation of several planetary bodies. Power levels of 300 kW to 1 MW appear to be the present requirement for onboard power, depending on the exact nature of the mission. The long life, high reliability, and power levels required for this application can be obtained by using thermionic conversion in conjunction with a nuclear reactor heat source for the power subsystem.[2] Space shuttle compatibility also will be required for these nuclear electric propulsion spacecraft. An out-of-core thermionic approach currently is being pursued. Both ion thrusters and magnetoplasmadynamic thrusters presently are being considered for propulsion of the spacecraft.

Planetary Missions

A typical mission scenario from Ref. 3 is shown in Table 1. The vehicle would be launched from a shuttle. It would consist of a NEP spacecraft of 9400 kg and a payload of 12,400 kg and would provide 400 kWe for the thrust subsystem, permitting a transit time to Jupiter of 1300 days. The payload could accommodate an 8000-kg orbiting laboratory and five 500-kg landers that could be used to probe surfaces and return samples to the spacecraft. These samples would be returned to Earth for a more detailed investigation. The spacecraft could orbit Jupiter and each of its moons in turn, performing explorations of these bodies sequentially.

The requirements that this type of mission will place on the propulsion system are listed in Table 2. These requirements, although somewhat soft, nevertheless provide the guideposts that can serve to direct the technology development required for the eventual implementation of advanced missions. The specific mass of the electric propulsion system should be less than 25 kg/kW to make the mission times attractive. Goals of 20 and 5 kg/kW therefore were set, respectively, for the constituent power and thrust subsystems.

The design of these systems must, by definition, be such as to increase reliability, minimize mass, simplify interfaces, and maintain flexibility for scientific information return. The effort is directed toward developing a conceptual spacecraft within the boundaries of this definition while meeting the requirements for outer planet exploration and remaining consistent with previous advanced propulsion comparison studies.[1]

Power Subsystem

The power subsystem consists of heat pipes, thermionic converters, nuclear reactor, reactor controls, coolant system, electric interconnections, and a nuclear radiation shield. The present effort has concentrated on a ~400-kW level power subsystem, which corresponds to the lower end of the effective jet power level listed in Table 2. This power level was chosen, since it appears to result in a spacecraft size that is compatible with the constraints imposed by the shuttle launch vehicle. Larger power levels may require in-orbit assembly techniques for the spacecraft or in-orbit propellant transfer.

A comparision has been made[4] of the power conversion methods suitable for this application. Thermionic converters were chosen on the basis of being able to provide the required long life with a high degree of reliability. This approach was believed to be suitable on the basis of no moving parts and applicability to a modular design without single-point failure being capable of disabling the system. Growth capabilities also are present for the thermionic converters, inasmuch as efficiency improvements will be realized with continuing work in this area.[5]

Past NEP studies have centered around a high-temperature configuration, with the thermionic converters located in the core of the reactor. This concept was studied extensively with feasibility established. The system was very complex, expensive, heavy, and lifetime limited. It would require extensive improvements in order to meet the present requirements. A different approach therefore is being investigated in order to establish concept feasibility. This approach removes the thermionic converters and related hard-

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ware from the reactor assembly, with heat transmitted by means of high-temperature heat pipes. This design is consistent with a more compact reactor design, and it is believed that this will result in lower costs due to the separation of reactor and thermionic converters.

Figure 1 shows the out-of-core reactor concept. The heat pipe is the "heart" of this design, and therefore one of the heat-pipe modules is depicted separately. The heat pipe has nuclear fuel bonded to the evaporator section, with space for the reactor reflector at the center of the pipe and thermionic converters located along the condenser section. The design utilizes 90 heat pipes and a total of 540 converters. The thermionic converter power level required is approximately 500 kWe to deliver a nominal 400 kWe to the thrust subsystem and spacecraft payload. The thermionic converters are arranged such that each unit is connected with both series and parallel leads. This matrix connection network permits operation with failed units, thereby increasing system reliability. Accessibility of individual converters also allows variation in the selection of the current and voltage output of the system to meet power conditioning or other requirements. Electrical insulation between heat pipes, converters, and coolant passages permits this flexibility. The converters are cooled by several coolant loops, eliminating the single-point failure mode that was present in the previous in-core system. An overall power balance for the propulsion system is shown in Fig. 2. Some details on the critical portions of the power subsystem are described below. Additional information may be found in Refs. 2, 6, and 7.

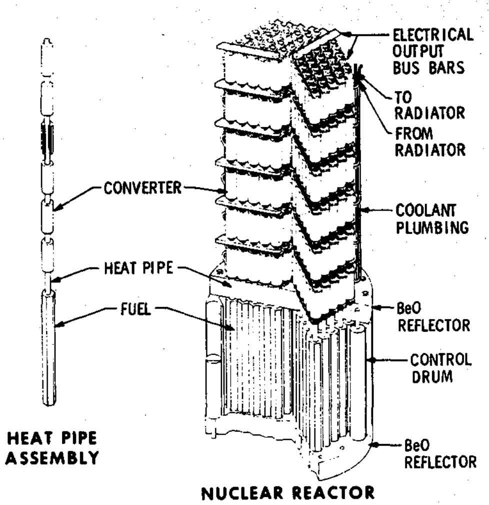


Fig. 1 Out-of-core thermionic reactor concept.

Reactor

The fast spectrum reactor is fueled with Mo-40% UO_2, with some variation of UO_2 content for power-flattening purposes. The reactor system includes 90 heat pipes, each with the fuel cermet bonded to one end. The array is shaped hexagonally, as shown in Fig. 3. The core is surrounded by a thermal radiation heat shield, a thin thermal neutron absorber, and a cylindrical BeO reflector. The neutron absorber reduces power peaking produced at the periphery of the core by reflecting low-energy neutrons. The reactor has BeO end reflectors of the same mean thickness as the radial reflector, resulting in a right cylindrical reactor.

Reactor control is achieved with control drums positioned in the reflector as shown in Fig. 3. The 18 drums containing BeO and B_4C produce a reactivity change, ΔK swing of more than 0.1, which is more than adequate for control of the system. The rotating drum configuration is preferable to other schemes, since this approach allows the diameter of the reactor to be minimal. It also assures a safe reactivity level if flooding should occur as a result of mission abort. The hazards of nearby large neutron-reflecting masses also are eliminated.

The fuel selection of MO-UO_2 was based on good chemical compatibility and a good thermal expansion match to the Mo heat pipes, to which it is bonded easily. Thermal conductivity and thermal shock characteristics of the fuel are better than other candidates such as UC-10 ZrC. Irradiation data for the

Table 1 Planetary extended mission sketch for Jupiter and Saturn and their satellites, using a 400-kWe spacecraft launched by shuttle to low Earth orbit

Jupiter		Saturn	
(1300 days)	21,800 kg in 26.0/0.0 orbit	(1900 days)	21,500 kg in 20.2/0.0 orbit
(25 days)	Orbit Callisto and drop 500-kg lander; sample return	(28 days)	Orbit Titan and drop 500-kg lander; sample return
(102 days)	Orbit Ganymede and drop 500-kg lander; sample return	(73 days)	Orbit Rhea and drop 500-kg lander; sample return
(73 days)	Orbit Europa and drop 500-kg lander; sample return	(36 days)	Orbit Dione and drop 500-kg lander; sample return
(68 days)	Orbit Io and drop 500-kg lander; sample return	(15 days)	Orbit Tethys and drop 500-kg lander; sample return
(14 days)	Drop a 500-kg Jupiter atmosphere probe	(109 days)	Minor arc traverse of Saturn's rings and drop 500-kg Saturn atmosphere probe
(~1500 days)	Return surface samples to Earth orbit	(~2100 days)	Return surface samples to Earth orbit

Table 2 Advanced planetary mission requirements and constraints for a NEP spacecraft

Effective Jet Power Level	270 kWj to 840 kWj
Lifetime	70,000 Hrs or Greater Total, including 30,000 full power hours available for thrust
Specific Impulse	9,000 Seconds
Specific Weight of Electric Propulsion System	25 kg/kw or less
Shield	
Neutron dose to electronics	10^{12} NVT
Gamma dose to electronics	10^{6} RAD
Propulsion System Reliability	HIGH (redundant system with single point failures eliminated)
Payload	Up to 13,000 kg
Size (undeployed)	Max. of 18 m x 4.5 m dia[a]
Weight	Max. of 29,500 kg[b]

[a] Compatible with shuttle payload bay.
[b] Subject to c.g. constraints and pallet/caging.

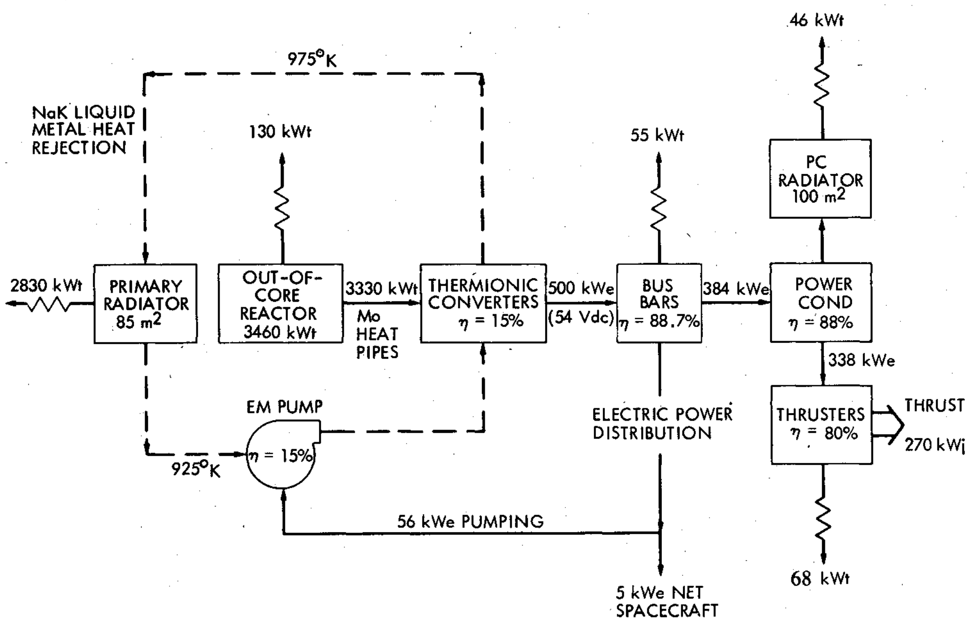


Fig. 2 Power balance for 400-kWe nuclear electric propulsion system.

fuel indicate that less than 1% volume swelling will occur as a result of the 30,000 hr at full power required for thrust during the longest one-way missions. This low amount of swelling is accommodated readily by the reactor design.

Thermionic Converters

The converters that were chosen for this application are second-generation units that are expected to be available within several years. The geometric layout and requirements for a typical converter are shown in Fig. 4. The converter components can have a small range for variation in thickness without impairing performance. Electrode thickness is determined by the allowable I^2R drops along their length. The unit diameter is determined effectively by heat-pipe diameter. Thus, since total system electrical power output is fixed, the required heat-pipe area can be determined for a given conversion efficiency. This area can be obtained either through a large number of small-diameter heat pipes or a smaller number of large-diameter heat pipes. Minimum size and hence minimum mass are obtained with the minimum number of large heat pipes. This is limited by radial heat fluxes in the heat pipes and the temperature differentials in the nuclear fuel.

The total length of the thermionic system is determined by the area needed to produce the power within the converters. The length of each converter is limited by I^2R losses within each converter as a result of conduction of the generated power along each electrode to the electrical leads at each end of the converter.

This sizing approach to the power system results in a reactor that is somewhat larger than is required from criticality constraints. This has resulted in the long lifetime

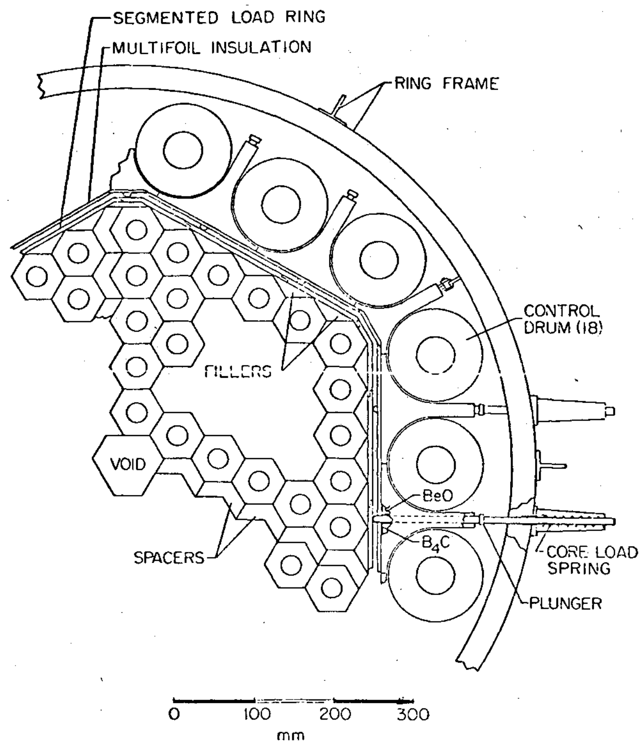


Fig. 3 Heat-pipe out-of-core thermionic reactor design with hexagonal reflector.

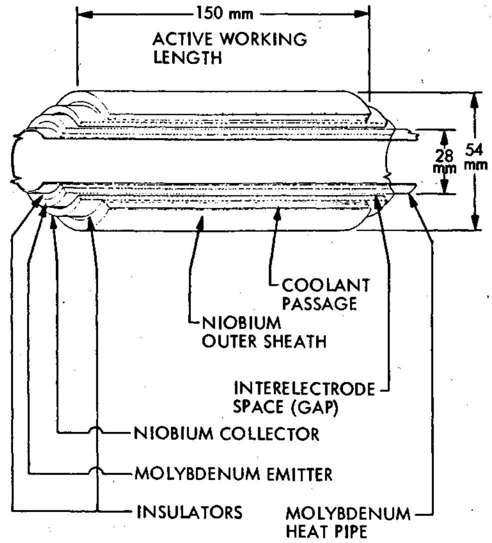
and low fuel swelling projected for the reactor. This design is based on thermoinic power levels of 6 W/cm^2 at 1650 K, with a conversion efficiency of 15%. These parameters represent near-term technology, which have been observed in research investigations.

Connection of the heat pipes with their converters into a system is based on minimum power losses in the electrical circuit and avoiding single-failure points in the system. Hence, the converters are connected into a matrix array of 90 in series and 6 in parallel, 540 in all, each producing approximately 1 kW of power at 0.6 V. The 90 converters theoretically will produce 54 V total; however, some of the voltage is lost as an I^2R drop in electrical leads, resulting in an input to the power conditioners of roughly 45 V, or $\pm$22.5 V from ground.

As can be seen from the design approach, there are many variables that can be modified to tailor the conversion system to a particular application. The modular approach allows increasing or decreasing system size to fit a wide range of power requirements. These changes can be made by varying either the number of heat-pipe/converter modules or the geometry of the modules without sacrificing reliability that is based on the matrix connections of the system.

Heat Pipes

The 1650 K thermionic converter operating temperature permits the use of lighter, less brittle, and lower-cost materials than tungsten. The wide variety of heat-pipe designs considered for the system therefore were based on molybdenum-lithium. Molybdenum is compatible with the nuclear fuel as well as the thermionic converter. Lithium has been demonstrated with molybdenum heat pipes to exhibit long life coupled with axial heat-transfer rates of 15 kW/cm^2 of vapor area. Radial heat fluxes of hundreds of watts per square centimer also have been demonstrated. The major variable is wick design to achieve reliability and good performance, coupled with ease of manufacture. Multiple pumping channels are required. The approach of a grooved wall covered with a wick offers reliability but represents significant manufacturing problems.



CONVECTOR PARAMETERS	
EFFICIENCY η (%)	15
EMITTER TEMPERATURE T_E (°K)	1650
COLLECTOR TEMPERATURE T_C (°K)	950
POWER DENSITY (W/cm)	6
CURRENT DENSITY (A/cm^2)	10
TOTAL POWER/CONVERTER (W)	1000
INTERELECTRODE SPACE (GAP) (mm)	.25
EMITTER THICKNESS (mm)	2.75
COLLECTOR THICKNESS (mm)	2.75

Fig. 4 Typical converter cross section.

The design under consideration uses six arteries formed from a molybdenum screen. This approach offers reliability of multiple pumping channels and minimum diameter, since part of the wall thickness no longer is needed for grooves. Ease of manufacture is based on smooth wall tubing with forming of the screen easily accomplished, resulting in good performance not only at the operating temperature but also during startup and less than full-power operation (spacecraft coast periods). A combination of cleaning and gettering during operation is being considered to assure the lifetime and performance that is needed.[8] The axial heat-transfer design point used was 10 kW/cm^2 (two-thirds of demonstrated capability). This corresponds to less than 100 W/cm^2 radial heat flux, which is required to assure acceptable performance throughout the spacecraft lifetime.

Insulator

The out-of-core concept requires a relatively low-temperature insulator on the cool or collector side of the converter. Demonstrated technology is available for this use. The high-temperature insulator between the thermionic emitter and heat pipe requires advances in the state-of-the-art. It is required to conduct heat adequately and not allow electrical shorting of the system. The task has been made more difficult by the requirement of matching the thermal coefficient of expansion of the molybdenum heat pipe. Single oxides have high thermal expansion coefficients and therefore are not usable. There are some mixed oxide materials that might be applicable, but little data are available.

A candidate material presently undergoing evaluation is sialon. Sialon is the acronym for a ceramic formed from silicon, aluminum, oxygen, and nitrogen. This material has reasonably good electrical resistivity coupled with a good thermal expansion match to molybdenum. Several phases can be present in this material. The Y-phase, $SiAl_4O_2N_4$, sialon

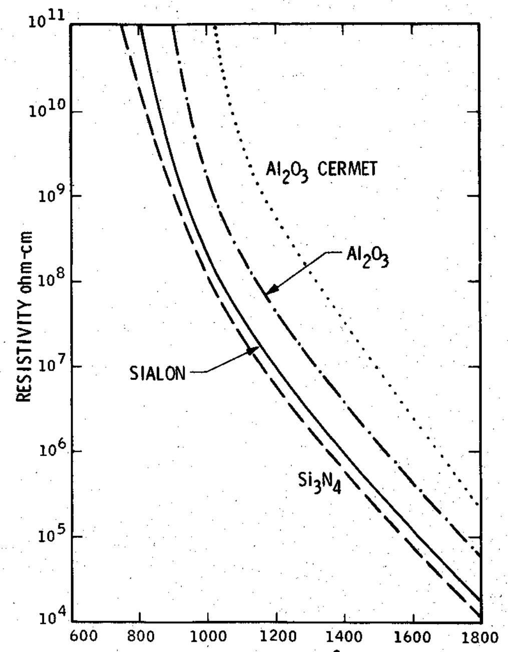


Fig. 5 Electrical resistivity of ceramics and cermets.

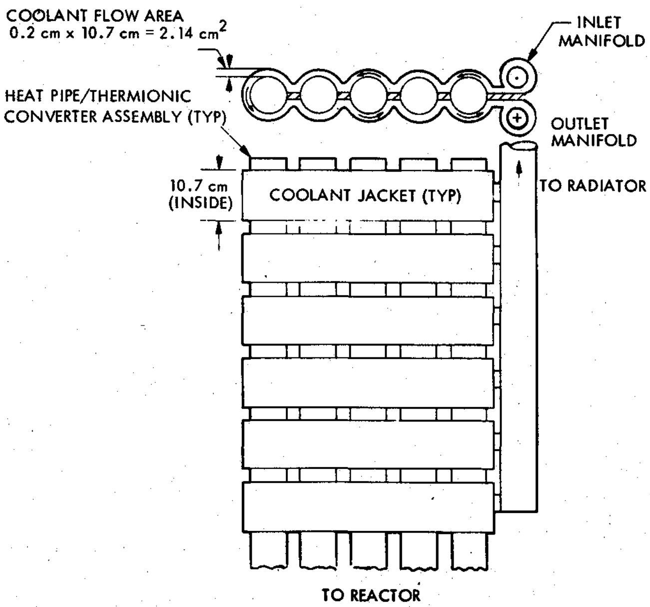


Fig. 6 One coolant loop arrangement.

has good vacuum stability, thermal expansion matching to molybdenum, and adequate electrical properties. This material presently is undergoing testing as a pure ceramic, with promising electrical resistivity results as shown in Fig. 5.

Previous work on oxide ceramics has shown that a cermet (ceramic with metal particles interspersed) can have higher electrical resistivity and improved high-temperature electrolysis behavior. Thus, the *Y*-phase ceramic is being prepared and tested as a molybdenum and a tungsten cermet.

Coolant System

The prime consideration in design of the coolant system for the thermionic converters is the elimination of single-point failures. In an out-of-core system, this can be accomplished readily through the use of multiple loops. A typical coolant loop through the thermionic system is shown in Fig. 6. This loop cools 30 converters, all of which are connected electrically in series. Thus, the loss of a loop will leave five of the six converters in parallel, with essentially no degradation of performance other than the power loss from the 30-converter string. Eighteen coolant loops are used in the system. Secondary design considerations are based on a tradeoff of fluid velocity and the ΔT across a series of five converters. Increased velocity reduces ΔT but increases pumping power requirements and erosion damage. The tentative design point uses a ΔT of 50 K and requires 56 kWe of pump power to move the fluid at ~ 10 fps within the coolant ducts surrounding the converters. The pumping power is based on an electromagnetic pump efficiency of 15%.

A heat-pipe radiator with beryllium armor for meteorite protection is envisioned for the system. The tentative design uses heat pipes that run the full length of the radiator, which is a maximum of 5.8 m. Each of the 18 loops is connected to each of the heat pipes for redundancy. This design approach keeps the coolant loops as close to the thermionics as possible, thereby minimizing coolant-system weight.

Radiator size is limited by the shuttle bay size. A 5.8-m-long $\times$ 4.5-m-diam radiator is compatible with 15% conversion efficiency of the thermionics and the 925 K mean operating temperature of the coolant. The maximum length radiator that can be accompanied by the shuttle in this spacecraft design is 9 m. Thus, power levels can be increased until this limit is reached. Further increases will require either improved converter efficiency or orbital assembly of a spacecraft.

Table 3 Electrical thruster comparison

	Ion Thruster	Magnetoplasma-dynamic Thruster
Size	30 cm dia	10 cm dia
Weight (each)	7 kg	100 kg
Exhaust Velocity (Average)	88 km/sec	49 km/sec
Efficiency	80%	50%
Power Level	16.2 kW Continuous	2.7 MW Instantaneous
Thrust Level	0.37 N	55.1 N
Propellant	Mercury	Argon

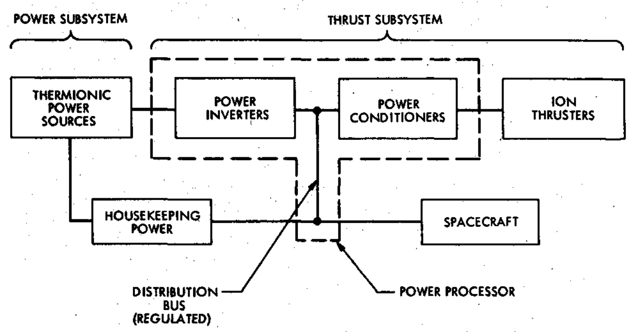


Fig. 7 Nuclear electric propulsion power and thrust system block diagram for ion thrusters.

Thrust Subsystem

The thrust subsystem consists primarily of thrusters and power-processing equipment. Two options exist for the type of thruster that can be used. These include the electron

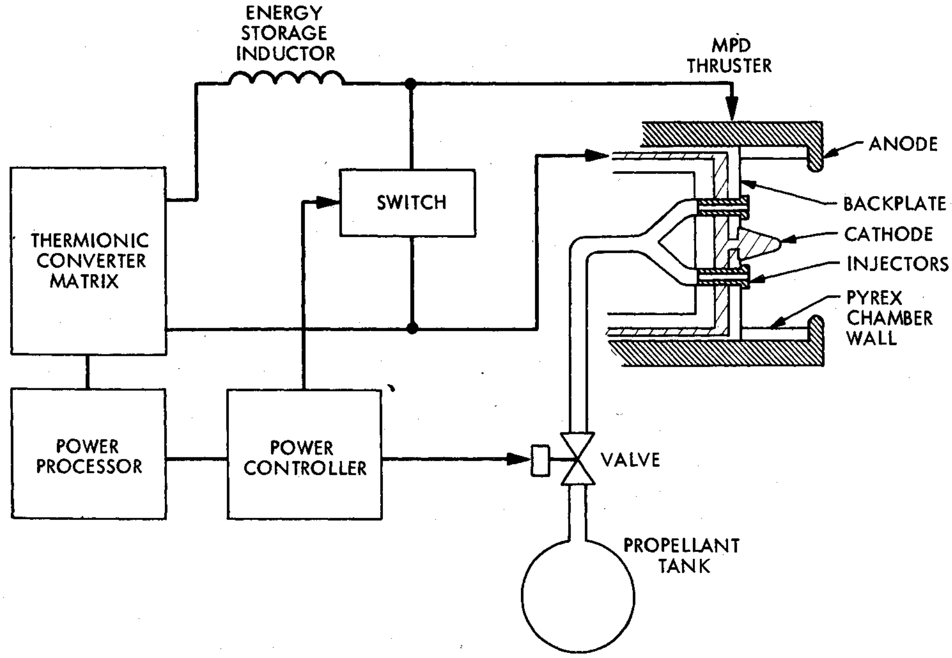


Fig. 8 MPD thruster system schematic.

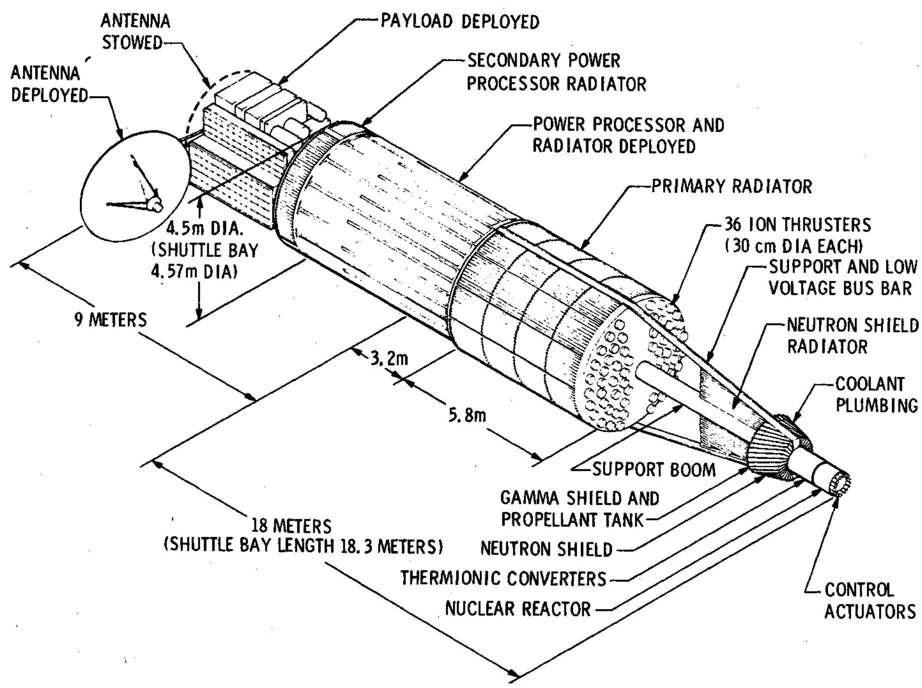


Fig. 9 NEP spacecraft utilizing ion thrusters: deployed configuration.

bombardment ion thruster, which presently exists in a highly developed state,[9] and the magnetoplasmadynamic (MPD) thruster,[10] which presently exists primarily as a laboratory device. The ion thruster presently is considered as the baseline for the thrust subsystem because of its well-defined performance and readiness for solar electric propulsion flight applications. The MPD thruster is of interest because of its high thrust density and overall simplicity of design and operation. The high thrust density also allows greater freedom of integration of the MPD thruster into a spacecraft design because the smaller thruster size allows greater freedom of placement of the thruster on the spacecraft. Previous limitations, which were believed to constrain the specific impulse available from the MPD thruster, appear to have been overcome, as indicated by recent research results.[10] Efficiencies somewhat comparable to those of the ion thruster will have to be obtained, however, for this thruster to be competitive for interplanetary applications. The MPD thruster presently appears to have future potential for reaching the higher levels of efficiency and specific impulse. Table 3 compares the two thrusters and the performance levels that presently are being used in Jet Propulsion Laboratory (JPL) studies.

The power-processing equipment subsystem is shown in block diagram format for an ion thruster operating from a thermionic converter matrix (examined in Ref. 11) in Fig. 7. Inasmuch as the mass of the power processor is dependent on the source voltage, a thermionic converter array output voltage of 23 V or higher is preferred to maintain this power processor mass at an acceptable level. A study[12] to examine the feasibility of operating an MPD thruster from thermionic converters has indicated that there are no major problems as a result of operating from the converters. Results from this study are included in Table 4, which compares the expected weights of MPD thruster and ion thruster subsystems. Near-term and long-range predictions are used for the MPD thruster. The present 30-cm ion thruster was used in the ion thruster comparison. The ion thruster power processor presently has a specific mass α near 10 kg/kW. Reductions in the processor specific mass to 3 kg/kW are included in the comparison of Table 4.

A parametric study of MPD thruster applications to planetary missions has yet to be accomplished. There is some indication that higher exhaust velocity *Ve* is achievable at higher than 50% efficiency. System tradeoffs between power level, efficiency, and pulsed operation are as yet unknown. It is obvious from Table 4 that the primary criterion is to reduce propellant mass by increasing the exhaust velocity of the MPD thruster.

An inductive energy storage technique presently is being examined for the MPD thruster. Work on the feasibility of utilizing the inductive energy storage concept from the standpoint of the thruster presently is underway at Princeton University. The system that would be used for an MPD thruster is shown in Fig. 8. In this system, the thermionic generating unit operates continuously at a power level of ~0.4 MW, whereas the MPD thruster operates intermittently at higher voltages and power levels. Energy storage is provided by building up a large current in an inductor. Periodically, the charging current is interrupted, and the energy stored in the magnetic field of the inductor is utilized for a short-duration thrust pulse.

Spacecraft Configuration

A conceptual drawing of a NEP spacecraft that would employ ion thrusters is shown in Fig. 9. The shuttle under current development would be the primary launch vehicle for this spacecraft; therefore, the shuttle payload bay size and weight limitations were used to determine maximum dimensions and weight limitations. The major drivers were the power system radiator size, the thrust subsystem radiator size, and the shadow shield restrictions. The reactor and converters are located at the aft end of the spacecraft. A lithium hydride shield provides protection against neutrons, and the stored propellant provides gamma-ray protection for the spacecraft within a 23° cone angle. The radiation has been designed to limit the neutron flux at the power-processing radiator to 10^{12} NVT and the gamma dose to 10^6 rad. A neutron shield radiator may be required to maintain an acceptable vapor pressure of hydrogen over the lithium hydride. The mercury propellant and propellant tankage also are used as gamma shields. The primary radiator requires a 4.5 m diam × 5.8 m length and utilizes a large portion of the shuttle payload bay.

The thrust subsystem power processor radiator and spacecraft payload can be fitted within the shadow of the radiation shield and the shuttle bay by using telescoping sections that are deployed after the spacecraft is launched. The power processor radiator section can be stowed either over or inside the primary radiator. The spacecraft payload can be attached to the power-processing radiator or, if need be, telescoped from inside the radiator. In addition, it is necessary to deploy the high-gain antenna. The payload is located furthest from the reactor and enjoys an unrestricted view for the science instruments. The weight breakdown for this spacecraft is included in Table 4. Present projections indicate that it is possible to meet the goal of 25 kg/kW for the electric propulsion system. The design also will meet the shuttle size, weight, and c.g. constraints. Figure 10 shows the center of gravity of the spacecraft and the limitations imposed by the shuttle. With the heavy reactor shield and mercury propellant at the tail of the shuttle, the c.g. envelope should present no problems.

The ion thrusters are located midship and are canted out 9° from the centerline to minimize plume impingement. On the spacecraft, Kapton coatings are provided to limit further sputtering from areas exposed to the ion beam. If this problem is still unacceptable as far as effects of the Kapton on thruster operation, then a deployment of the thrusters away from the spacecraft also will have to be considered.

MPD thrusters with high-temperature, self-radiating anodes also can be used for propulsion of the spacecraft. In this case, the power processor and radiator in Fig. 9 would be deleted and an energy storage inductor placed behind the neutron shield. The MPD thruster would operate in the quasisteady-state mode at instantaneous power levels of approximately 2 MWe. As an alternative to a directly cooled anode, excess area could be added to the primary radiator and NaK coolant used to cool the anodes of the MPD thrusters.

Table 4 Ion vs MPD thruster system mass breakdown

Subsystem	System With Ion Thrusters Mass, kg	System With MPD Thrusters Mass, kg
Power Subsystem		
Reactor Core (40 = 1)	932	932
Reactor Reflectors and Control	773	773
LiH Shield	2,030	2,030
Heat Pipes	364	364
Thermionic Converters	805	805
Low Voltage Conductors - Thermionic Converters to PWR Processor	1,104	1,104
Coolant Plumbing INL Accumulations	536	536
Coolant (Nak-78)	569	569
EM Pumps	120	120
Primary Radiator	793	793
Support Structure	400	400
Total	8,426	8,426
α(400 kWe)	21.07	21.07
Thrust Subsystem		
Power Processor	1,200	950
Thruster	247	200
Structure	100	100
Switchgear	20	50
Radiator		140
Total	1,567	1,440
α(400 kWe)	3.92	3.60
Propellant	8,960	17,280
Tankage	450	800
Spacecraft		
Payload	10,142	1,599

Orbit-Raising Application

An attractive supplemental use of a nuclear electric spacecraft may be as an integral part of the near-Earth orbit

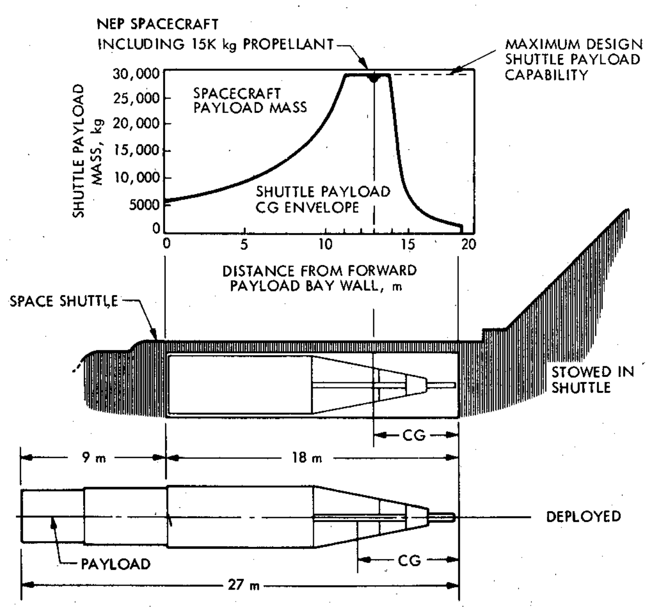


Fig. 10 400-kWe NEP spacecraft stowed in the space shuttle.

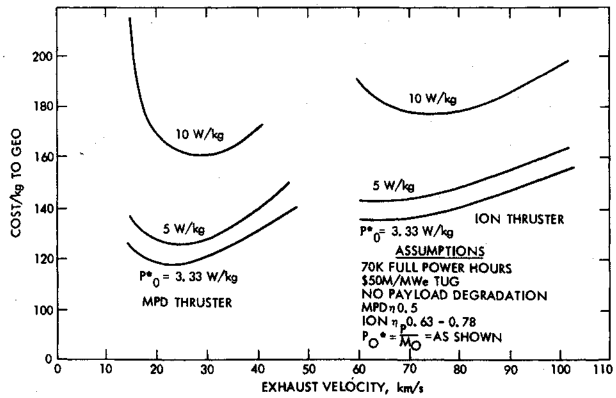


Fig. 11 Transport cost LEO-GEO round trip with reusable tug.

transportation system. Early study activities[13] in this area have identified electric propulsion as a potentially cost-effective mode of materiel transport. A nuclear electric spacecraft developed for planetary exploration also could be brought into service as a nuclear tug to transport equipment from shuttle orbit to geosynchronous orbit.

The NEP spacecraft could be modified to be used for transfer of cargo from shuttle orbit to geosynchronous orbit. The use of NEP for this application currently is under study.[14] Cargo would be transferred in containerized pallets, which would replace the payload shown in Fig. 9. The container would be brought to low Earth orbit in a shuttle bay and left at a logistics depot. The NEP vehicle would pick up the container at the depot and transfer it to a second depot in geosynchronous orbit near the site of its intended use. Cargo transfer vehicles of this type will be required to supply materials for manned geosynchronous satellites, space manufacturing, and also for construction materials if a space station were to be assembled at this altitude. Economics will determine the most optimum approach. The cost analysis from Ref. 13 is shown for this nuclear tug application in Fig. 11. The cost of this tug is lower than the cost for a chemical tug providing the same service and lowest when MPD thrusters are used. The low cost of the MPD thruster option is due to the smaller transfer time obtained with the lower specific impulse operation. In this case, a clear optimum is indicated at an exhaust velocity of about 20 km/sec, a region where MPD thruster operation can be obtained easily.

Conclusions

A nuclear electric propulsion vehicle with a lightweight propulsion system represents a highly attractive method of accomplishing the detailed exploration of the outer planets of the solar system. This system at power levels of 300 kW to 1 MW can accomplish these missions within a reasonable flight time and with sufficient payload and propulsion capability to allow for highly automated and detailed investigation of several planetary bodies. The long life, high reliability, and power levels required can be obtained by using thermionic conversion in conjunction with a nuclear reactor heat source. The system is compatible with the space shuttle. Both ion thrusters and magnetoplasmadynamic thrusters can be used for propulsion of the spacecraft. The spacecraft with MPD thrusters is a potentially cost-effective method of cargo transfer from shuttle orbit to geosynchronous orbit for missions such as manned satellites and space manufacturing.

Acknowledgment

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