Application of MPD Thruster Systems to Interplanetary Missions

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Studies of space tests and applications of MPD thruster systems in the 1990s are described. They include the Advanced Space Experiment onboard the Space Flyer Unit (SFU) in low Earth orbit (LEO), the lunar polar orbiter mission in the early 1990s, and the asteroid rendezvous mission in the late 1990s. The SFU uses a 5 kw MPD thruster system (4×1.25 kW system) to assess the on-orbit performance and the potential impact on the LEO environment by thruster plasma injection. During the lunar mission, a 1.25 kW MPD thruster system lowers the orbiter from a high lunar polar orbit into a low orbit less than 100 kW in altitude. These early applications can be followed by an interplanetary asteroid rendezvous mission. Envisioning these missions, the MPD thruster system is required to have 10×10^6 pulse lifetime and to generate 40 mN/kW at a specific impulse of 2000 s.

Introduction

THE MPD thruster has several unique advantages in addition to its representative feature of a high specific impulse (up to several thousands of seconds). It is characterized by simple structural design, relatively low-voltage operation (ranging a few hundred volts), and feasibility of a wider selection of propellants consisting of atomic species abundant in the Earth's upper atmosphere. The MPD thruster is also free of the dc magnetic field that affects scientific magnetometers and that is also generated by the permanent magnets in some other electric propulsion systems. Because of these benefits, the MPD thruster is expected to be one of the most promising propulsion systems for future interplanetary spacecraft and cargo orbit transfer vehicles.

In the past decade, the MPD thrusters developed by the Institute of Space and Astronautical Science (ISAS) have had two successful space flights. In 1980, a 15 W MPD thruster system weighing 14 kg was carried by the MST-4 satellite and successfully changed the satellite spin rate.[1] In 1983, the space experiment with particle accelerator (SEPAC) package onboard Spacelab-1 used a 200 W MPD arcjet system weighing 63 kg to examine electron beam propagation and charge neutralization phenomena.[2] However, both experiments were not sufficiently powerful to demonstrate MPD thruster capability as a primary propulsion system driven by large solar arrays. Furthermore, the total experimental durations were too short to verify the thruster's lifetime. On the other hand, a 1×10^6 pulse endurance test using a 1 kW MPD thruster system was successfully accomplished at our laboratory in 1984.[3] Figure 1 shows the external view of the MPD thruster head used in the endurance test.

In spite of these pioneering efforts, MPD thruster applications to real space missions have been delayed. This situation owes much to the technological restrictions imposed on the available electrical power from the solar array and the system designers' conservatism against electric propulsion concepts. However, it is apparent that the high ΔV missions of interplanetary and other near-Earth targets will surely be developed in the 1990s, along with the fully matured technology of a large-scale solar array that can be the electrical power source for a solar electric propulsion system.

Here, we have studied three missions planned for the 1990s using MPD thruster systems. They include the advanced space experiments onboard the Space Flyer Unit, the lunar polar orbiter mission in the early 1990s, and the asteroid rendezvous mission in the late 1990s. These prospective missions have encouraged the development of an MPD thruster system most suitable for the near-term space missions.

Low-Altitude Exploration of the Moon by an MPD-Thruster-Propelled Polar Orbiter

Many planetary scientists are very interested in low-altitude observation of the moon from a polar orbiter that can survey the entire lunar surface. Such interest originates from the motivation to resolve the origin of the moon. Of course, the lunar mission will also be worthwhile from a practical point of view for the exploration of the polar district for future lunar-based construction.

Since a considerable amount of technological preparation (e.g., the development of launch vehicles and the establishment of orbit control techniques) is required in advance, the assessment of technological readiness at ISAS suggests at present that lunar mission will be realized by, hopefully, 1992. The mission objective is the low-altitude exploration of the moon from a lunar polar orbiter. The technological implications are 1) verification of launching capability of the upgraded vehicle, 2) verification of orbit control techniques, and 3) verification of MPD thruster capability as the primary propulsion system.

Figure 2 shows the mission scenario. In the first phase, the spacecraft is injected into a high lunar polar orbit of about 10,000 km altitude. In the second phase, where the incident solar radiation is nearly normal to the orbital plane and hence the solar array, the MPD thruster is operated to decelerate the spacecraft. This phase continues for about three months, during which the spacecraft orbit is spirally lowered to about 2000 km altitude. Remote observation of the lunar surface can be conducted for about three months in the third phase. In the fourth phase, the MPD thruster is operated again to bring the spacecraft down to an orbit lower than 100 km in altitude, where lunar magnetism, gravity, and other highly sensitive measurements can be performed.

Figure 3 illustrates a conceptual view of the spacecraft. A winged solar array 1.0 m wide and 4.0 m in length generates

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1.77 kW of electrical power in total and supplies a maximum of 1.25 kW to the MPD thruster system installed on the frontal face of the spacecraft. The two MPD thruster heads normally consume 0.90 kW each and are operated alternately. Other spacecraft faces are occupied by the high-gain and omni antennas, the star and sun sensors, many other scientific instruments for lunar observation, and the attachment to the final stage of launch vehicle.

The weight summary of the spacecraft is given in Table 1. As shown, the MPD thruster system weighs about 51.5 kg, excluding 18 kg of hydrazine propellant whose tank is common to the reaction control system (RCS). The weight of each component in the MPD thruster system was estimated from the updated values verified in the recent system endurance test.[4] Other system weights except for the solar arrays were estimated from the existing values found in the conventional spacecraft launched by ISAS. As for the solar array, both 170 W/m^2 and 50 W/kg are our development goals for the 1990s time frame.[5]

Although the generated power from the solar arrays is supplied primarily to the MPD thruster system, the power is also usable for scientific instruments during nonthrusting times. The major payloads contain scientific instruments such as the penetrators with seismographs, an imaging camera, a thermal radiometer, x-ray and γ-ray spectrometers, a magnetometer, a radar altimeter, and a detector of interplanetary dust. The penetrators with a decelerating rocket motor will be launched from a proper altitude onto the lunar surface. In addition, the self-field induced by the pulsed MPD system has proved to have little influence on the key scientific instrument—the magnetometer—in our recent thruster system test.

The performance characteristics of the MPD thruster system for the lunar mission are summarized in Table 2. The thrust power ratio will be designed as 30 mN/kW at a specific impulse of 2000 s. The thruster system lifetime should exceed at least 15×10^6 pulses, which corresponds to continuous operation at 1 Hz for about 166 days. Most of these requirements have already been demonstrated in terms of thrust performance during laboratory tests, as shown in Fig. 4. In 1984, we completed an endurance test of a 1 kW MPD thruster discharge chamber in a vacuum tank and obtained data for electrode erosion rate.[3] The value of 0.6 μg/C was adequate to 10×10^6 pulse endurance. In 1985, a 1 kW MPD thruster whole system was verified to have 1×10^6 pulse endurance and the tested capacitors were found to have further longevity.[4] For the next step, a 1.25 kW MPD thruster system will be tested in 1987 to verify a 10×10^6 pulse endurance.

The lowest required velocity increment shown in Table 2 was given by a simple calculation of quasistatic orbit transfer. This ΔV might also be attainable using a chemical propulsion system; however, it cannot lower the orbit to less than 5000 km at the given mass ratio. The MPD thruster system provides a more versatile use of the spacecraft, such as cruising at a 20 km altitude above the lunar surface and then ascending to a higher orbit with little consumption of propellant. Such maneuvering facilitates the precise measurements made by the magnetometer and the relay of signals transmitted from the penetrators on the lunar surface.

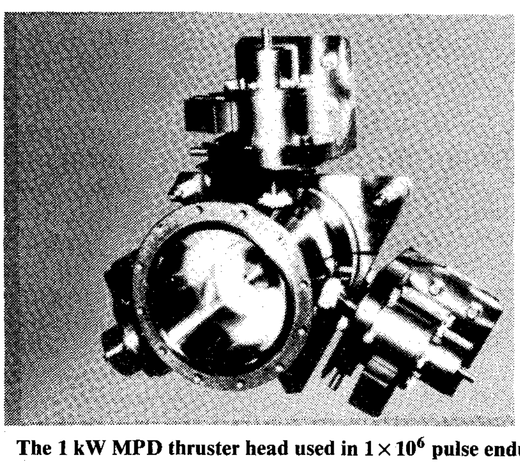

Fig. 1 The 1 kW MPD thruster head used in 1×10^6 pulse endurance test.

Electric Propulsion Experiment on the SFU

The Space Flyer Unit (SFU) is a small space platform to be launched on the Space Shuttle and deployed from the Orbiter (STS). To be operated around 1992, the SFU is an appropriate test bed for a 5 kW (4×1.25 kW) MPD thruster system. The SFU accommodates several payload units in its octagonal structural body, as depicted in Fig. 5. Examples of the advanced space technology experiments performed onboard the SFU are two-dimensional array deployment, high-voltage solar array operation, microwave power transmission, and high-precision pointing system tests.[6] The weight of the fully equipped SFU will be as large as 3000 kg and will be launched by STS into low Earth orbit (LEO). The electric propulsion experiment

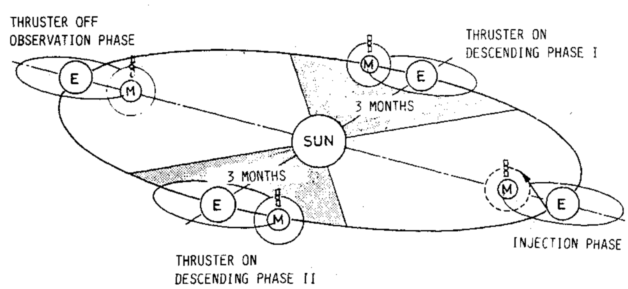

Fig. 2 Mission scenario in lunar exploration.

Table 1 Weight breakdown of MPD-propelled lunar polar orbiter

System	Subsystem, kg	Component, kg
1) Electrical power source	45.5	
Solar arrays		35.5
2) Communication	10.0	
3) Command and data management	7.5	
4) Orbit/attitude control	73.5[a]	
N_2H_4		36.5
5) Thermal control	7.5	
6) Structure	52.5	
7) Scientific instruments	102.0[b]	
8) MPD Thruster	51.5	
Charging control unit		8.5
Capacitor bank		30.0
Discharge head		6.0
Propellant supply		3.0
MPD controller		4.0
Total weight	350.0	

[a]Including momentum wheel, sensors, RCS, and MPD propellant. [b]Including penetrators with seismographs.

Table 2 Design and performance parameters of MPD thruster system for lunar exploration mission

Velocity increment (ΔV for 166 days)	1.02 km/s
Endurance	20×10^6
Specific impulse	2000 s
System input power	1.25 kW max[a]
Thrust/power ratio	30 mN/kW
MPD pulse width	600 μs
Repetition rate	1/1.5 Hz
Propellant	N_2H_4

[a]0.9 kW nonimal.

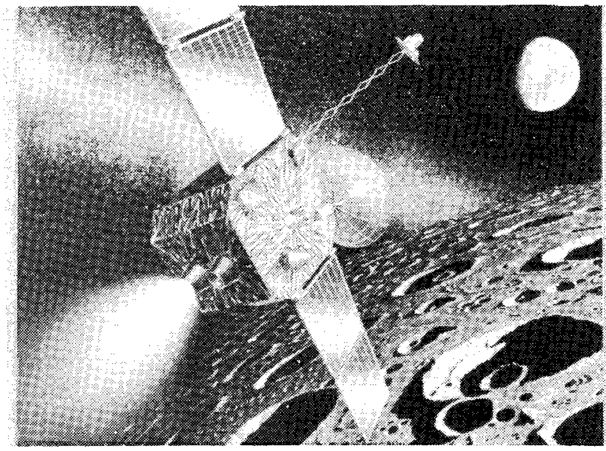

Fig. 3 Conceptual view of MPD-propelled lunar polar orbiter.

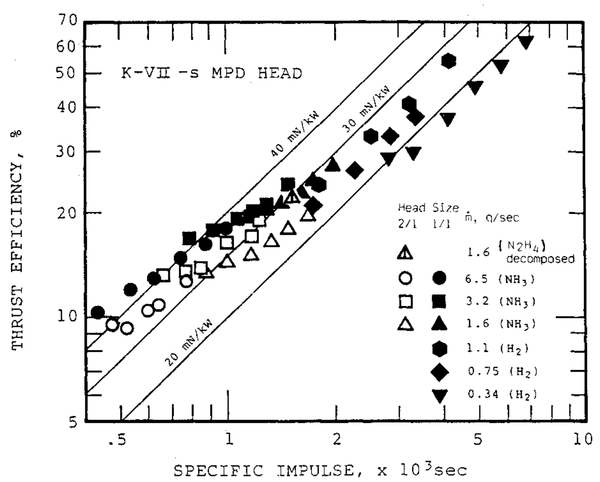

Fig. 4 Thrust performance of MPD thruster.

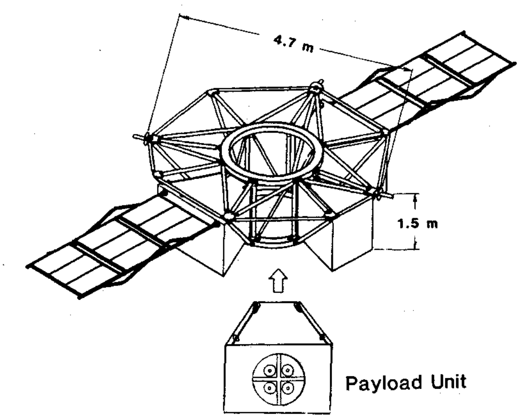

Fig. 5 Electric propulsion experiment onboard Space Flyer Unit.

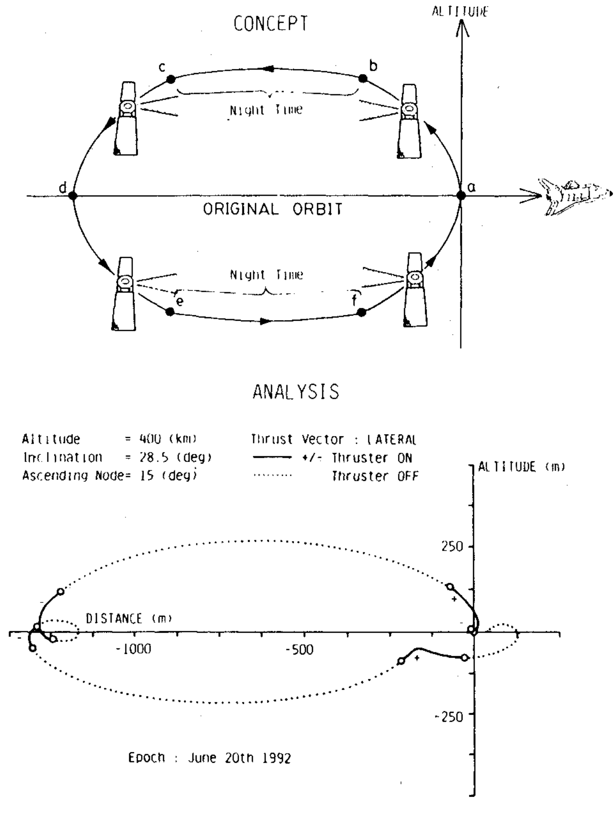

Fig. 6 Concept and analysis of loop orbit operation in EPEX.

(EPEX) is one of the SFU missions and is partly coupled with the high-voltage solar array experiment to be described later. The objectives of EPEX are, in order of priority:

1) To assess the on-orbit performance of the MPD thruster as an SFU primary propulsion system.

2) To measure plasma particles, waves, and luminosity caused by the MPD thruster plume.

3) To verify the altitude maintenance capabilities of the MPD thruster as an SFU rebooster.

4) To assess the life cycle of the MPD thruster system.

At present, the fourth objective is not taken into account in the test of the MPD thruster system at 5 kW. A prolonged stay of six months in space will permit such endurance testing at reduced power. In meeting the above objectives, the thruster operational modes are defined in the following paragraphs.

Loop Operation

As illustrated in Fig. 6, the orbit of an MPD-thruster-propelled SFU is deformed with reference to the co-orbiting STS, which is referred to as the drag-free object. The acceleration by the MPD thruster from point a to point b raises the SFU above and behind STS. This is followed by an inertial flight between points b and c, which corresponds to night coverage. The deceleration by the MPD thruster from point c to point d lowers the SFU until the next night coverage. The symmetrical procedure closes the SFU orbit like a racetrack loop. It takes about 6 h for this operation. Although flight simulations by computer analysis in which the aerodynamic drag is taken into account result in only an approximate loop orbit, the loop has dimensions of 1250 m in longitude and 500 m in altitude, as shown in Fig. 6. This experiment will verify the MPD thruster capability as a primary propulsion system.

Plasma Disturbance Measurement

The plasma diagnostic measurements are also conducted in EPEX with the assistance of the co-orbiting STS as shown in Fig. 7. In case 1, the MPD thruster onboard the SFU injects plasma parallel to the geomagnetic field line toward the STS; in case 2, the plasma is injected in the direction orthogonal to the geomagnetic field lines. The plasma particles, excited waves, and airglow are measured by the STS, while keeping a distance of about 20 km from the SFU. The acquired data will also be useful for the environmental assessment of LEO with potential contamination caused by a great deal of plasma exhaust. Moreover, the electromagnetic compatibility will be retested, although the MPD arcjet in SEPAC proved to be harmless to the command and telemetry system of Spacelab-1.

Altitude Maintenance Experiment by MPD Thruster

Figure 8 plots the required power for the SFU rebooster at different altitudes in 1985 and in 1992. Since the solar activity reaches a maximum around 1992, the exospheric aerodynamic drag increases drastically; however, the result indicates that only 1 kW of the solar array power will keep the SFU orbit at a constant altitude of 400 km. If the bus power supply is adequate for this requirement even at night, the SFU reboosting experiment using the MPD thruster can be performed throughout 24 h. Such an experiment will demonstrate another possible application of the MPD thruster to orbit maintenance, as well as its long-term durability.

MPD Thruster System in EPEX

A systems block diagram is presented in Fig. 9. In the EPEX, a 5 kW MPD thruster system is comprised of four 1.25 kW systems that are to be used in the lunar exploration mission. The electrical power system has the alternative of either a low- or a high-voltage line. The former comprises a conventional charging control unit (CCU) connected to a 55 V array. The latter is connected directly to a series-parallel switching array consisting of six or eight modules that generate appropriate voltages up to 330 V so as to allow the maximum charging efficiency.[7] This solar array is referred to as the high-voltage solar array (HVSA). The HVSA is one of the dedicated devices for the SFU experiments and is capable of supplying a maximum power of 8 kW to the EPEX. Most of the generated electrical power is stored in the capacitor bank (CB) [consisting of a pulse-forming network (PFN), trigger driver (TRG), and drivers for the fast-acting valves (FAV)] and is then supplied to the MPD head system. The hydrazine propellant is supplied into the MPD head system by FAVs.

The MPD thruster system is integrated in one module and packed into a single payload unit. The module weight is 234 kg. This value is roughly four times as large as that used in the lunar mission with the increased weight of the structural and thermal control system. One or two modules will be installed on the SFU. The number of modules and their arrangement depends upon the limit of interference between the thruster plume and solar arrays and also upon the simplicity of the thrust vector control.

In this experiment, it is stringently required that the MPD thruster system verify the thrust power ratio of 30 mN/kW at a specific impulse of 2000 s and guarantee a thermal efficiency of 80%. These technological readiness demonstrations will be required for the interplanetary mission of asteroid rendezvous beyond the mid-1990s.

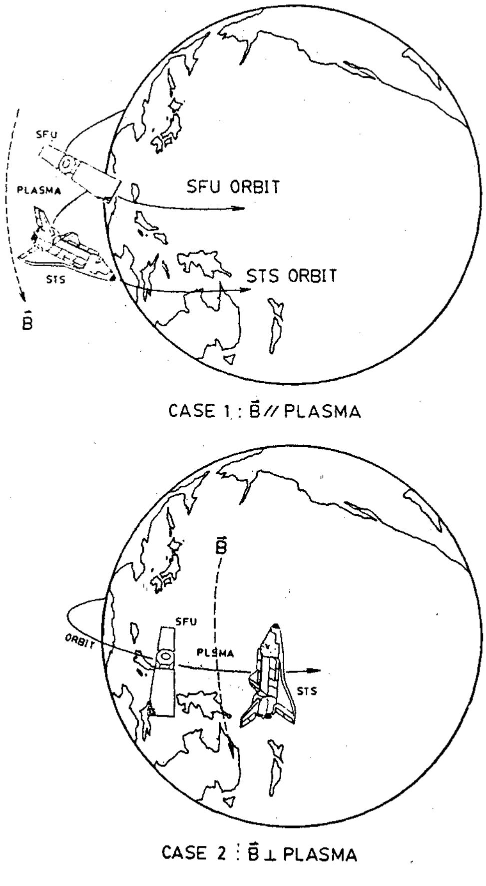

Fig. 7 Plasma disturbance measurements in EPEX.

Eros Rendezvous Mission

There are several thousands of asteroids of various kinds in the solar system. The asteroids belts are considered to be the birthplace of meteorites and to carry abundant information related to the origin and evolution of the solar system. Therefore, planetary scientists are eager to perform in-situ investigations of asteroids, ultimately including a sample return. Moreover, some scientists suggest that the asteroids will be the potential source of raw material for space industrialization.[8]

Several possible modes of asteroid exploration have been proposed. In order of mission requirements and difficulties, they are: multiflyby, rendezvous or orbiter, landing, and sample return. Although the target selection criteria are not yet fully established due to the diversity and number of targets in

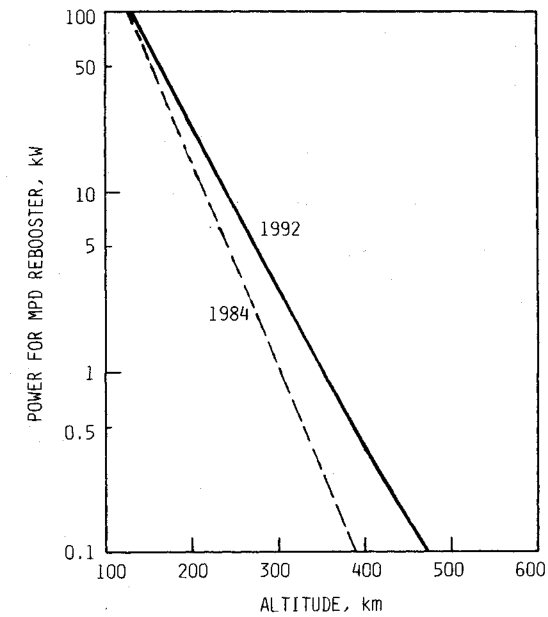

Fig. 8 Required power for altitude maintenance in EPEX.

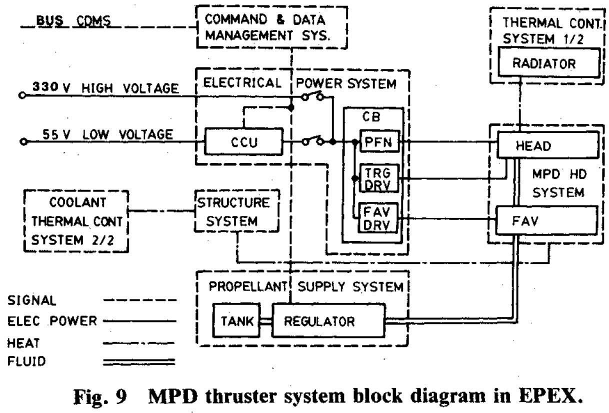

Fig. 9 MPD thruster system block diagram in EPEX.

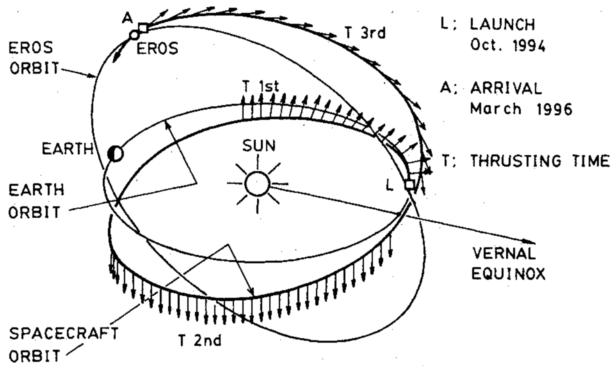

Fig. 10 Mission scenario of Eros rendezvous.

Table 3 Design and performance requirements of MPD thruster system for Eros rendezvous mission

Velocity increment (ΔV for 364 days)	8.20 km/s
Earth escape mass	400 kg
System input power (at BOL)	2.5 kW
Specific impulse	2000 s
Thrust power ratio	40 mN/kW
Endurance	30×10^6

space, they could be categorized into three groups: Earth-approaching asteroids, mainbelt asteroids, and asteroid families. Both the mission specification and the target selection are strongly dependent on the launching capability as well as the propulsion system and navigation technique in interplanetary space. The first trip to the asteroids should be a single rendezvous with one of the near-Earth targets.

Here, an Eros single-rendezvous mission using a 2.5 kW MPD thruster system (2×1.25 kW system) that will be ready in our scenario of the EPEX experiment onboard the SFU has been studied. The mission objective is the imaging and spectroscopic observation of the asteroids. The technological implications are 1) to verify low-thrust interplanetary navigation techniques and communication techniques and 2) to verify the MPD thruster system capability for interplanetary missions.

The minor planet Eros is a relatively familiar asteroid because of its peculiar shape. Eros belongs to the Earth-approaching asteroids. A preliminary analysis in which the payload ratio is roughly optimized resulted in the operational modes of the MPD thruster as shown in Fig. 10. The spacecraft should be injected into an Earth escape orbit by a launch vehicle on Oct. 26, 1994. About 1.5 years after departure from the Earth following MPD thruster operation for 364 days, the spacecraft will arrive at Eros on March 8, 1996, just before perihelion. Once the rendezvous is achieved, a more than two-day stay within the distance of 10,000 km should be expected for observation with satisfactory resolution of the entire surface of the asteroid. Fine adjustment of the rendezvous orbit will be also controlled by the MPD thruster operation.

The design and performance requirements of the MPD thruster system are summarized in Table 3. The mission ΔV was not perfectly optimized, because launch windows yielding perihelion rendezvous near the Earth were primarily selected. Iterative calculation of the trajectory was made with thrust scheduling so that the required velocity increment was minimized. Assuming that the launch capability of our vehicle is increased to 400 kg for Earth escape, a 2.5 kW MPD thruster system will be suitable for the Eros rendezvous mission. A specific impulse of 2000 s is a reasonable compromise between the selection of mission time and payload ratio under the available power constraints. The payload weight of about 15 kg will have to contain at least an imaging camera and, if possible, some other spectroscopic instruments.

It should be noted here that this study might be too modest to take full advantage of the MPD thruster system, because the total spacecraft mass is substantially limited by the launch capability. Future upgrading in this aspect, as well as the development of highly efficient solar arrays and MPD thruster with long lifetime, will provide more extensive activities in space such as multitarget missions and deep space exploration.

Summary

Three missions with MPD thruster systems in the 1990s have been studied. In the electric propulsion experiment, a 5 kW MPD thruster system will be tested onboard the Space Flyer Unit for the purpose of verifying the technological readiness for the following lunar and asteroid rendezvous missions. The plasma interaction with a low Earth orbit environment will be also examined. The lunar polar orbiter will hopefully be the first space mission equipped with a 1.25 kW MPD thruster as the primary propulsion system, allowing the spacecraft closer access to the lunar surface, i.e., within a few tens of kilometers. The Eros rendezvous mission is only a preliminary exploration, but will be meaningful not only scientifically but also technologically as the first step to interplanetary missions with a 2.5 kW MPD thruster system employed as the main propulsion device. Besides the maturity of MPD thruster technology, a lightweight and highly efficient solar array is the requisite technology to be developed no later than the 1990s.

References

[1]Kuriki, K., Nakamaru, K., and Morimotor, S., "MPD Thruster Test on Engineering Satellite," *AIAA Progress in Astronautics and Aeronautics: Progress in Electric Propulsion and Its Application to Space Missions*, Vol. 179, edited by R.C. Fine, AIAA, New York, 1981, pp. 605–615.

[2]Kuriki, K., "The MPD Thruster Test on the Space Shuttle," *Journal of Spacecraft and Rockets*, Vol. 16, Sept.-Oct. 1979, pp. 326–332.

[3]Shimizu, Y., Kuriki, K., Morimoto, S., Suzuki, H., Uematsu, K., and Takiyama, T., "Endurance Test of MPD Arcjet Discharge Chamber," IEPC Paper 84-29, 1984.

[4]Toki, K., Shimizu, Y., and Kuriki, K., "Development of Repetitively Pulsed Quasi-Steady MPD Thruster," ISTS Paper 14, b-3-1, 1986

[5]Kuriki, K., "Advanced Technology Experiment onboard Space Platform," ISTS Paper 347, q-3-5, 1986.

[6]Kuriki, K., Nagatomo, M., and Ijichi, K., "Energetics Experiment on Space Station," IAF Paper 85-158, 1985.

[7]Ijichi, K. and Billerbeck, W.J., "Capacitor Bank Charging by Series-Parallel Switching of Solar Arrays," *Journal of Spacecraft and Rockets*, Vol. 21, March-April 1984, pp. 196-201.

[8]Gehrels, T. (ed.), *Asteroids*, University of Arizona Press, Tucson, 1979.