AN OVERVIEW OF ELECTRIC PROPULSION ACTIVITIES IN JAPAN

Michio Nishida* and Hirokazu Tahara†

*Department of Aeronautics and Astronautics, Kyushu University
   Fukuoka 812-8581, Japan

†Area of Aerospace Engineering, Division of Mechanical Science,
   Department of Systems and Human Science, Osaka University
   Toyonaka, Osaka 560-8531, Japan

Abstract

The paper describes the current status of electric propulsion technology development and related studies in Japan. Ion engine systems (IES) onboard the Engineering Test Satellite (ETS-VI) and the Communications and Broadcasting Engineering Test Satellite (COMETS), and a quasi-steady MPD thruster onboard the Space Flyer Unit (SFU) were successfully demonstrated. The National Space Development Agency of Japan (NASDA) has a plan to launch an improved IES onboard the ETS-VIII in 2002. 1.8 kW / 250 mN - class hydrazine DC arcjet thrusters are to be used on the Data Relay Test Satellite (DRTS). The Institute of Space and Astronautical Science (ISAS) is promoting the sample and return space mission from an asteroid, the MUSES-C, which is scheduled to be launched in 2002 and to bring back some specimens to the Earth in 2006 from an extra-terrestrial object. In the MUSES-C mission, a cathode-less microwave discharge ion engine system is ready as a primary propulsion in the interplanetary space. Electric propulsion studies at universities, research institutes and industries are also aiming at applications as well as fundamental physics.

FLIGHT PLANS AND RESULTS

National Space Development Agency of Japan (NASDA)

NASDA launched Communications and Broadcasting Engineering Test Satellite (COMETS), NASDA's second satellite equipped with an ion engine system (IES) for its north-south station keeping (NSSK), in 1998. However, because of the malfunction of the H-II Launch Vehicle, it did not reach the geostationary orbit. Therefore, IES was operated merely as an experimental equipment. There were no troubles in nine operations with total time of 10 hours 40 minutes. The followings were confirmed:

(1) IES operated properly and showed quite good similarity between the operation on orbit and the ground test,

(2) IES showed the characteristics of the Engineering Test Satellite VI (ETS-VI), and the design, manufacturing and test of IES is considered to be established,

(3) IES only needs to be proved for a long duration operation on orbit.

Engineering Test Satellite VIII (ETS-VIII) is another program equipped with an ion engine system. The design of engine system components mainly follow the precedent of ETS-VI and COMETS. Development focuses on simplifying of IES function to result in improvement of system reliability and reduction of cost. ETS-VIII is scheduled to be launched in 2002.

Institute of Space and Astronautical Science (ISAS)

Cathode-less Microwave Discharge Ion Thruster System for MUSES-C

The Institute of Space and Astronautical Science (ISAS) has been promoting the sample and return space mission from an asteroid, the “MUSES-C” which is scheduled to be launched in 2002 and to bring back some specimens to the Earth in 2006 from an extra-terrestrial object. The MUSES-C will demonstrate the space engineering technologies that are essential to the future scientific sample and return missions. They are 1) the ion engine system as a primary propulsion in the interplanetary space, 2) an autonomous navigation and guidance via optical measurements, 3) a sample collection mechanics and 4) direct reentry and sample recovery from an interplanetary field. The MUSES-C spacecraft shown in Figs.1 and 2 will be launched by the ISAS medium class launch vehicle M-V forward a Near Earth Asteroid 1989ML in 2002, and after four years, it will bring the specimens to the Earth in 2006. In the heliocentric trajectories to the asteroids and to the Earth, the ion engine system will propel the spacecraft. In general a round trip space mission requires too large fuel consumption of the conventional chemical thrusters to maneuver the spacecraft. Only the electric propulsion makes a spacecraft approach some of the asteroids and keeps a return way to the Earth. The spacecraft weight is about 500 kg including chemical and ion engine propellant of xenon gas. The relatively large solar cell panels will generate approximately 1.8 kW at the Earth distance for the elec-
electric propulsion. The z-axis (High Gain Antenna (HGA) aperture axis) is pointed to the Sun, and the ion engines are mounted perpendicular to the z-axis. The thrust direction is steered by rotating spacecraft attitude around the Sun direction. Four ion engines are mounted on the plate gimbaled, so that the thrust axis can track the spacecraft center of mass to drift due to fuel consumption. The round trip mission is designed using three of the ion engines, and an extra thruster is carried by the spacecraft purely for redundancy. A single unit of the ion engines generates a thrust force of 8.1 mN and 2,920 s as the nominal performance at 400 W input power including power consumption of the propellant management system and the thruster control unit etc. It can be throttled down to 60% of its full power consumption. A funnel like sampling horn is installed on the bottom panel. When the spacecraft touches down the asteroid, a projectile will be shot and shatter its surface. Generated fragments are guided to the canister through the horn under the ultra-low gravity field. The canister with the specimens of the asteroid is finally stowed in the reentry capsule, which reenters into the Earth atmosphere directly from the interplanetary trajectory.

The department of electric propulsion at ISAS is developing the cathode-less microwave discharge ion thruster system for the MUSES-C in which plasmas of the ion source and the neutralizer are generated by the method of the electron cyclotron resonance (ECR) discharge. The schematic diagram of the cathode-less microwave discharge ion thruster system is illustrated in Fig.3 and the photograph in Fig.4. The ion source and neutralizer are fed by a single microwave generator and a single gas flow controller through passive dividers. The microwave ion thruster is applicable to the MUSES-C mission because of long life in the space operation and the easy maintenance in the ground operation due to the elimination of the hollow cathodes, which frequently reduce their functions due to the atmospheric exposure. The MUSES-C program demands the lifetime over 16,000 hours of the microwave discharge ion thruster. In order to verify the long life, the engineering model (EM) is devoted to the endurance test, and the prototype model (PM) is under preparation for it. The EM
endurance test was planned in order to achieve the following objectives:

(1) to access the endurance of the carbon-carbon composite grid system,
(2) to identify the failure modes of the ion thruster system,
(3) to obtain the preliminary data to determine the designs of the PM.

The EM endurance test was initiated in the vacuum chamber of 2 m in diameter and 5 m long in February 1997. The operation and data acquisition of the EM thruster and the vacuum facility are fully automated by a workstation, which is connected with the internet, so that the real time operational information is disclosed in the address "http://www.ep.isas.ac.jp/open". The performance test, thermal cycle test, frequency tuning test and inspection etc. were voluntarily interrupted several times. At the beginning of May 1999 the total accumulated operational time achieved 16,000 hours without fatal problems and is still progressing. In the middle of 1999, the EM endurance test will accomplish the operational time of 18,000 hours. What is of great concern is the grid mass loss owing to the erosion process. This is one of the most critical parameters that constrains the thruster life. The MUSES-C ion thrusters adopt the carbon-carbon composite grids instead of the conventional molybdenum metal grids, aiming at the long-term operation. In the inspection periods, the weight of each grid was measured and the slight erosion of the acceleration grid was found, which was not serious to satisfy the MUSES-C requirements. The screen grid never caused weight loss because it has no potential gap to the plasma in the ion source.

ION THRUSTERS

National Aerospace Laboratory (NAL)

Carbon-carbon composite grids were fabricated for a 14-cm diameter xenon ion thruster, and the endurance test of the grids is underway. A two-grid system was adopted consisting of the screen and accelerator grids. The screen grid is 0.5 mm thick and has an open area fraction of 68%. The accelerator grid is 0.8 mm thick and has an open area fraction of 27%. The thruster is being operated at a beam voltage of 1000 V, accelerator grid voltage of 200 V and discharge chamber propellant utilization efficiency of 90%. The discharge current is adjusted to maintain a constant beam current of 478 mA. The accelerator grid current is as small as 2.2 mA. By the end of March 1999, the total operation time of 2,773 hours was achieved with no problem, and the test is going to be continued hopefully till failure.

NAL / NASDA / Toshiba Corporation

Research of 35-cm diameter ring-cusp xenon ion thrusters was conducted to develop basic technology for primary propulsion systems of orbit transfer vehicles. Two thrusters (BBM1-MK1-1 and 2) as the first breadboard model were fabricated and tested to design another model that was expected to reach the following targets. Performance targets were set to a thrust of 150-180 mN, a specific impulse of 3,500 s and an ion production cost less than 140 W/A at a propellant utilization efficiency of 90%. The other targets were set to lifetime more than 30,000 hours, startup time less than 10 minutes, magnet temperature less than 200 °C, an accelerator grid mass change less than 3 mg/h and a characteristic frequency more than 100 Hz for an acceleration system.

The BBM1-MK1-1 thruster was operated intermittently for 1,000 hours with no trouble at the thrust of 150 mN and the specific impulse of 3,518 s, and performed with the ion production cost of 104 W/A at the propellant utilization efficiency of 90%. The startup time was 13 minutes on the average, and the accelerator grid mass change was 6.12 mg/h at the accelerator grid voltage of 300 V. It was difficult to continuously operate the thruster at the thrust of 180 mN.

The BBM1-MK1-2 thruster was designed mainly to achieve stable thruster operation even at the thrust of 180 mN and to acquire the vibration characteristics of the acceleration system. This thruster is discriminated from the previous one by the thickness of screen and accelerator grids, the accelerator grid voltage and the spring constants. Figure 5 shows a photograph of BBM1-MK1-2. It has the external dimensions of 45 cm in diameter and 29 cm in length, and 12.5 kg in weight.

The discharge chamber is the same as that for the BBM1-MK1-1 thruster and has a 1-mm-thick iron anode consisting of a seamless sidewall and an endwall. The sidewall is 17.4 cm long, and its cross section is a regular 24-sided polygon which has the inner osculating circle of 37 cm in diameter. The endwall has the curvature of a 2m radius and a dish depth of 7.1 mm. Four magnet rings of Sm2-Co17 are attached on their inside surfaces, i.e., two on the sidewall and two on the endwall. Their inside surfaces are coated with molybdenum to capture sputtered particles from the acceleration system, and their outside surfaces are coated with white alumina to facilitate infrared radiation. A ground screen consists of a cylindrical shell with an open area fraction of 63% and a donut plate coated with titania on both sides to facilitate thermal radiation. Hollow cathodes are designed to have maximum permissible emission currents of 20 A for the main and 4 A for the neutralizer.

The acceleration system uses a three-grid system of 35 cm in diameter. The screen grid is made of a 0.6-mm-thick molybdenum plate with an open area fraction of 70.4% and the convex curvature of 1.6 m. The accelerator grid is made of a 0.8-mm-thick molybdenum plate with an open area fraction of 63%.
fraction of 25.9% and the convex curvature of 1.5 m. The decelerator grid is made of a 0.6-mm-thick molybdenum plate with an open area fraction of 50.0% and the convex curvature of 1.5 m. Each grid has 18,241 holes for ion extraction arranged in the hexagonal arrays. The holes and the outer shapes of the grids are formed on the 0.2-mm-thick molybdenum plates using photochemical etching. Then, the etched plates are piled and press-welded to obtain the required grid thickness and dished using press formning with carbon dies. Each grid support is of flexible type using 12 piano wire springs. The spring constants are 5.1 kN/m for the screen grid springs and 10.8 kN/m for the accelerator and decelerator grid springs in the nominal operation condition and 353 kN/m for all grid springs in the vibration test. The fundamental natural mode of vibration is 114 Hz for the acceleration system.

The BBM1-MK1-2 thruster was operated stably at the thrusts of 150 and 180 mN, the specific impulse of 3,518 s, the total electric powers of 3.35 and 4.00 kW, and the total xenon flow rates of 3.25 and 3.88 A equivalent, and the foregoing designs were proved effective. The ion production costs of 123 and 121 W/A are obtained under the following conditions: the propellant utilization efficiency of 90%, the discharge voltages of 35.8 and 31.3 V, the accelerator grid voltage of 200V, the accelerator grid currents of 14 and 19 mA, and the decelerator grid currents of 6 and 8 mA. The highest magnet temperatures are 203 °C and 207 °C, and exceed the target limit of 200 °C. The vacuum chamber pressures corrected for Xe are 0.35 and 0.42 mPa. The startup time, the accelerator grid mass change at the accelerator grid voltage of 200 V and the fundamental natural mode of vibration will be measured in the planned tests.

**Mitsubishi Electric Corporation (MELCO)**

The Ion Engine Subsystem (IES) on NASA’s COMETS was operated successfully early in March 1999.6 This IES has the same design as that of ETS-VI.

Following to the success in ETS-VI and COMETS, a 1kW/20mN class improved IES, whose design is based on the ETS-VI/COMETS IES, is undergoing at MELCO under the contract with NASA. This is to be used for NSK on ETS-VIII.7,8 The main objectives of the IES improvement are

(1) Longer life capability that can cover NSK of a 2 to 3 ton class, over a 10-year-life GEO satellite,
(2) Simple subsystem construction,
(3) Easier operation,
(4) Lower cost.

The life test for the development model has already started in late 1998 at MELCO. Engineering model components of the improved IES are in process of manufacturing and tests in the year of 1999 to 2000 by MELCO.

**ISAS**

Development of 30mN class Microwave Ion Engine

For near future space missions, ISAS is develop-
ing a 30mN class ion thruster that is a scale-up version of the 7mN class microwave ion engine for the MUSES-C’s main engine. For the 4.2 GHz microwave power of 100 W used for the plasma generation of both an ion source and an neutralizer, the ion beam current of 600 mA and a maximum thrust of 30 mN are expected. This engine is targeting at ISAS’s future science missions by a small spacecraft weighing less than 500 kg, hence the engine subsystem weighs only 40 kg requiring 1 kW electrical power for both the microwave power generator and the DC power supplying unit. A laboratory model with a 20-cm-diam. discharge chamber shown in Fig.6 was designed, and a preliminary test showed easy ignition and enough plasma generation of the ion source. The engine’s beam extraction system is also in development.

![Fig.6 A 20-cm-diam. microwave ion engine’s discharge chamber.](image)

<table>
<thead>
<tr>
<th>Table 2 Typical 30-cm-diam. cusp ion thruster performance</th>
</tr>
</thead>
<tbody>
<tr>
<td>Forward Propellant Flow Rate</td>
</tr>
<tr>
<td>Reverse Propellant Flow Rate</td>
</tr>
<tr>
<td>Main Cathode Flow Rate</td>
</tr>
<tr>
<td>Discharge Chamber Configuration</td>
</tr>
<tr>
<td>Beam Voltage</td>
</tr>
<tr>
<td>Beam Current</td>
</tr>
<tr>
<td>Accelerator Grid Voltage</td>
</tr>
<tr>
<td>Accelerator Grid Current</td>
</tr>
<tr>
<td>Discharge Voltage</td>
</tr>
<tr>
<td>Discharge Current</td>
</tr>
<tr>
<td>Thrust</td>
</tr>
<tr>
<td>Specific Impulse</td>
</tr>
<tr>
<td>Ion Production Cost</td>
</tr>
<tr>
<td>Propellant Utilization Efficiency</td>
</tr>
<tr>
<td>Thruster Efficiency</td>
</tr>
<tr>
<td>Total Power Consumption</td>
</tr>
<tr>
<td>Thrust to Power Ratio</td>
</tr>
</tbody>
</table>

Tokyo Metropolitan Institute of Technology
30cm Diameter Cusp Ion Thruster

A study of a 30-cm diameter cusp ion thruster has started with the support of MELCO. In this study, the effects of the propellant distribution and the magnetic field configuration on the performance have been evaluated experimentally. Thruster operations were conducted for various conditions of the propellant feed and the magnetic field to obtain the performance characteristics. The plasma diagnosis was also performed by using electrostatic probes in some cases. Table 2 shows the summary of the performance obtained in this experiment.

Evaluation of C₆₀ Application to Ion Thruster

Based on the previous work, C₆₀ application to the EB ion thruster with a hollow-cathode, a plasma confinement magnetic coil (similar to Kaufmanns type), where several considerations in its design were taken into accounts to prevent the C₆₀ re-solidification, fragmentation and mult ionization, was evaluated. Using this new designed thruster, preliminary experiments using Xe or a mixture of C₆₀ and Xe, were performed. The obtained results were (1) inspection of ExB-probe showed the detection of C₆₀ and Xe ion beam, (2) ion beam current was suppressed by space charge limitation, (3) produced plasma was dominated by C⁺, C₆₀ and Xe⁺ ions, judged from Langmuir probe diagnosis, and (4) negative ion production occurred in the downstream region.

However, the thruster performance did not reach the predicted one. For the performance improvement, appropriate C₆₀ flow rate, beam extraction voltage for C₆₀ and design modification may be necessary.

Miscellaneous

Moreover, the studies on amorphous-carbon grid application to ion optics, neutral propellant initial distribution simulation using DSMC method, ion beam extraction simulation by PIC for the detailed evaluation of plasma sheath condition, Teflon PPT evaluation for a small satellite, and electromagnetic interference between electric propulsion and spacecraft/communications have been carrying on.

University of Tokyo

An efficient code that can estimate the lifetime of an ion thruster due to charge-exchange was developed. With this tool, it is possible to gain knowledge about the ion thruster lifetime, and it enables the design of ion thrusters in shorter periods and at a lower cost. In the case of a three dimensional grid system, the grid erosion is concentrated on the inside wall region of the hole, and an axisymmetric approach can be adopted as well (Fig.7). For the 2 grid system, the erosion pattern is three dimensional, and this requires an analysis using the three dimensional code. However, using the information obtained from the two dimensional code and in-
Introducing the emission surface model reduces the computational cost.

The analytical and computational results indicate that the thruster lifetime imposed by charge exchange erosion can be modeled in a very simple form. The dependence of lifetime on ion beam density is very strong both in the 3 grid and 2 grid systems. Especially for the 3 grid system, the lifetime is in inverse proportion to the 4th power of beam current density (with a fixed propellant utilization efficiency).

![Fig. 7 Ion extraction features with change in grid shape (MUSES-C grid system).](image)

![Fig. 8 Schematic block diagram of PMU of the MUSES-C ion engine system.](image)

Kyushu Institute of Technology

A new type of ion thruster grids made of carbon material is being developed by the group which consists of Kyushu Institute of Technology, Tokyo Metropolitan Institute of Technology, Fukuoka Prefecture Industrial Research Center, Kasyu Kogyo, and Astec Irie. Laser mining of carbon plate is tried to produce an ion thruster grid which is easy to fabricate and strong against sputtering. Samples are now attached to an ion thruster and now tested in ground experiment.

Mitsubishi Heavy Industries, Ltd. (MHI)

MHI is developing the propellant management unit (PMU) of the MUSES-C spacecraft ion engine system, as shown in Fig. 8, in collaboration with ISAS and NEC. A pulse width modulation regulator system is applied for the PMU.

ARCJET AND MPD THRUSTERS

ISAS

Flowfield Observation and Development of Low Power DC Arcjet

Both the physical research and technological development have been underway for the stabilization and life-enhancement of a DC arcjet for its practical application. The stabilization problem has been recognized as the operational modes (high voltage / low voltage mode). The quick transition to the high voltage mode or the maintenance of the stable mode is the main point. For that purpose, a flow visualization arcjet (VAJ) which makes it possible to observe the flowfield in the discharge region was used. Figure 9 shows the observation region of the VAJ in operation with an arc column in the nozzle. The observation was carried out by a spectroscopic technique, and there obtained inside distributions of electron temperature (Fig.9), electron density and other plasma parameters. The absorption spectroscopy of the VAJ inner flow was also conducted. The absorption profile (+15 deg, λr: 811.75 nm) was obtained, whereby atomic temperature and flow velocity could be determined. In parallel, a simple model of an arcjet flowfield was constructed and found compatible with experimental results.

Osaka University

Low Power Arcjet Thruster Development

The laboratory-model radiation-cooled arcjet thruster RAT-VII shown in Fig.10 was developed to widely examine stable operation range, aiming at lowering power level for near-future use. The arcjet thruster was operated in the wide electric power range of 200 - 1000 W with a mixture of N₂ + 2H₂ at mass flow rates of 15 - 40 mg/s. The influences of electrode gap, propellant swirl flow injection and constrictor diameter on operational stability and thrust performance were investigated to achieve high-performance stable operations around 500 W. Furthermore, the temperatures of the thruster body surface were measured to examine the thermal characteristics. The characteristics of heat transfer inside the body were analyzed computationally using the measured data.

As a result, the arcjet with a constrictor diameter of 0.4 - 0.5 mm, an electrode gap of 0.1 mm and with swirl flow injection stably operated below 500 W, and a high performance of thrust efficiency.
Fig. 9 Discharge region of a low power DC arcjet (top) and the temperature distribution (bottom; vertical/horizontal unit: mm).

42% and specific impulse 458 sec were achieved at 500 W. Furthermore, the calculated thermal characteristics of the thruster body showed that a 30%-downsize body was preferable to achieve a higher thrust performance below 500 W by decreasing the heat loss from the body surface and to achieve a more stable operation with a higher-temperature body by shortening a transient time from start to a steady state.

Medium Power Arcjet Research

Spectroscopic and electrostatic probe measurements were carried out to understand the plasma features inside and outside a 10kW class DC arcjet. Ammonia or a mixture of nitrogen and hydrogen was used as a working gas. The NH₃ and N₂+3H₂ plasmas in the expansion nozzle and in the plume were in thermodynamical nonequilibrium state although the state at the constrictor throat was nearly in thermal equilibrium. As a result, the H-atom excitation temperature and the N₂ rotational excitation temperature decreased from 11,000 K at the constrictor to 4,000 K or to 2,000 K at the nozzle exit for 0.2 g/s, although the NH rotational temperature did not show an axial decrease even inside the nozzle. On the other hand, several temperatures were almost kept within some ranges in the plume under an ambient pressure 130 Pa except for the NH rotational temperature in NH₃.

MPD Arcjet Research

A new research program for understanding of quasiside MPD arcjet acceleration mechanism was started. That was based on a unique concept that two electromagnetic forces, blowing and pumping forces, should be independently examined with two different simple discharge chambers.

Plasma diagnostic measurement and flowfield analysis were conducted with coaxial MPD channels to understand the blowing acceleration process. Plasma velocity at the exit of the channel was measured by a Doppler shift with a Fabry-Pérot interferometer. For the MC-II MPD channel with longer electrodes and a smaller electrode gap than those for the conventional MPD thrusters, exhaust plasma velocities were 5,500 to 7,500 m/s, which were lower than the velocity predicted from the theoretical electromagnetic thrust. In addition, the numerical analysis of an axisymmetric MPD channel flow was also conducted. The calculated results roughly agreed with the measured ones, although an intensive current concentration calculated was not observed in the experiment. It is inferred from the result that there exist two acceleration zones near the inlet and outlet of the channel, although a drastic acceleration near the upstream end of the channel as predicted by the analytical result is not expected. Furthermore, the plasma diagnostic measurement in the PF-II MPD arcjet with enhanced pumping force designed for near-cathode observation was conducted to examine the pumping acceleration mechanism. The ratio of the pumping force to the overall thrust reached 85% of the theoretical electromagnetic thrust near the critical current operations.
Kyushu University
Numerical Analysis of a Thermochemical Nonequilibrium Flow in an Arcjet Thruster

Efforts have been made to numerically reproduce the flow field inside a low power DC arcjet thruster with nitrogen as propellant. Governing equations are formulated by assuming thermally and/or chemically nonequilibrium flow and solved using a finite volume method. A typical result is shown in Fig. 11. Work is presently underway to investigate the effect of thermal non-equilibrium on the thruster performance, and to develop more efficient algorithms that enable the numerical analysis to be a practical tool for design process.

Nagoya University
Numerical codes for an axisymmetric MPD thruster was developed, where the sonic conditions at the inlet was determined from a one-dimensional flow analysis. In order to investigate the mechanism of onset phenomenon in the axisymmetric flow, the calculations were performed by applying the code to the straight-configuration electrodes (Fig. 12). On the other hand, a plume diagnosis of an arcjet is underway using a diode laser absorption technique.

Tohoku University
A high-power (up to 2MW) and quasi-steady (more than 1ms) MPD arcjet is being investigated to measure the plasma blow-off speed with externally applied magnetic field in the HITOP device (0.8 m in diameter, 3.2 m in length) of Tohoku University. A plasma flow field in the magnetic nozzle was measured by several diagnostics in order to clarify an acceleration mechanism of the MPD thruster and to carry out basic plasma physics experiments on MHD phenomena in the supersonic flow.

Simple Estimation of an Arcjet Thruster Performance Characteristics

In the design of arcjet thrusters, it is needed to calculate the flowfield inside the thruster for specified thruster geometry and to predict the performance characteristics of it. This is done by the numerical calculations using the axisymmetric Navier-Stokes equations, which needs very long computation time to obtain the steady state solutions. Therefore, in certain cases, numerical calculations are inconvenient as practical design tools. Nothing would be better than quasi one-dimensional calculation, if that could give satisfactory results to the thruster design. On the view of this point, quasi one-dimensional calculations which are available as a simple design tool have been tried and the results have been compared with experiments.

Low Power Arcjet Thruster Experiments

A small power (< 1kW) arcjet thruster which can satisfy the conditions for the installation on small satellites and also provide experimental data necessary to verify the numerical results of arcjet performance has been developed. The systematic tests of operational characteristics and thruster performance, current-voltage characteristics and the thrust measurements are underway.

Fig. 11 Heavy particle temperature distribution for a nitrogen arcjet thruster.

Fig. 12 Onset discharge current $J_{onset}$ and onset magnetic force number at the cathode base $S_{onset}$ against mass flow rate.

Fig. 12 Onset discharge current $J_{onset}$ and onset magnetic force number at the cathode base $S_{onset}$ against mass flow rate.
particle energy analyzer. Typical plasma parameters measured at 2 m downstream of the MPD source are as follows: plasma density: $5 \times 10^{12} - 3 \times 10^{14}$ cm$^{-3}$, half-maximum diameter: 10 - 40 cm, ion temperature: low energy component 5 - 15 eV, high energy component 50 - 100 eV, electron temperature: 5 - 10 eV, ion Mach number: 1 - 4, Alfvén Mach number: 0.1 - 3, axial flow energy: 20 - 50 eV and azimuthal rotation energy: 3 - 5 eV, depending on the experimental conditions such as the field magnetic configuration, the arc discharge current and the gas flow rate.

Hokkaido University

They have made experiment of a multichannel two-dimensional MPD thruster at ISAS and compared experimental data of the temperature in the thruster with numerical values since the beginning of 1990s. There has still been a discrepancy between them. Then, they introduced a new physical model into a conventional one in order to overcome the discrepancy, i.e., dependency of both the electrical and thermal conductivities on the temperature profile in the thruster. They describe temperature effects on electric and thermal conductivities of MPD models, and numerically solve the new model of a two-dimensional MPD thruster under several practical assumptions, using the TVD MacCormack scheme for fluid models and Successive Over Relaxation method for a steady state electromagnetic model. In the analysis, they apply boundary conditions that a total current flows from an anode to a cathode based on the stream function. For Ar propellant, they numerically analyze three kinds of transport models, i.e., (a) electrical and thermal conductivities to be constant, (b) only the electrical conductivity depending on the temperature, and (c) both the electrical and thermal conductivities depending on the temperature in the thruster. The temperature profile obtained shows that the maximum temperatures for (a) and (b) are higher than experimental data, but numerical results for (c) are in good agreement with experimental data. Then, it is found that (c) affects the current profile and the temperature distribution in the thruster.

Kyushu Institute of Technology

A solid propellant arcjet type of thruster (1 N class), which is capable of electrically controlling and sustaining its combustion, has been developed and a comparative examination of the thruster performance with different chemically reactive propellants has been conducted. The tests demonstrate that this type of motor can be controlled stably using non self-combustible AN-based composite propellants at a discharge power level around 1 kW. Currently the use of solid propellants as substitute for teflon in pulsed plasma thrusters are also being examined.

Tokai University

An experimental study for a performance evaluation of an arcjet operation at very low power levels ranging from 5 to 30 W was conducted with nitrogen as propellant. The propulsive performance is evaluated for two different nozzles, a conventional nozzle consisting of an assembly of tungsten nozzle parts and a modified nozzle consisting of an assembly of an insulator and a tungsten anode. In the modified nozzle, ceramic material or an insulator was used as a part of a constrictor to allow an arc column to penetrate further downstream of the constrictor or to maintain the high-voltage mode discharges, and to reduce the electrode losses. Stable operations with specific impulse of about 100 to 280 s at input power levels ranging from 5 to 30 W with the constrictor diameter of 0.3 or 0.5 mm, as shown in Fig.13, were confirmed at efficiency of approximately 47%, except in a singular case of a glow discharge or a glow jet, in which little improvement of propulsive performance was observed with an increase in electrical input power. The constrictor diameter was found to have a significant effect on the thrust performance of the device. For the partially insulated nozzle, the specific impulse and the thrust efficiency were significantly higher than those for the conventional nozzle.

Fig.13 Specific impulse vs input power characteristics (propellant mass flow rate: 5 mg/s, W,: tungsten nozzle, I,: insulator inserted nozzle, C: convergent nozzle, D: divergent nozzle).

Ishikawajima-Harima Heavy Industries Co., Ltd. (IHI)

IHI has been developing electric propulsion technologies for space application since 1978. Their flight experiences of the electric propulsion system are as follows:

(1) Two MPD thruster modules, including power processors of 200 J/pulse and propellant supply systems of ammonia, were developed for
being launched onboard the MS-T4 test satellite (TANSEI VI) of ISAS, 1980.
(2) A 1 kW-class MPD thruster system, including a thruster module and a propellant supply system of hydrazine, was developed for being launched onboard the SFU, 1995.
(3) An EHT (Electrothermal Hydrazine Thruster) system, including a thruster module and power processor, was developed for a mission apparatus launched onboard the ETS-VI (KIKU VI) of NASA, 1994.

Recently IHI has been developing a 1 kW DC arcjet system. A 100-hours firing test was conducted successfully using the engineering model of thruster, gas generator, power processor and domestic hydrazine. A flight type system of the 1 kW DC arcjet has been designed and manufactured by reflecting the test result.

MHI
MHI has been developing a 300W class hydrazine arcjet propulsion system, considering application to near future space programs, in collaboration with ISAS/NASA. The arcjet thruster was run with N2+2H2, which was substituted for hydrazine decomposed product in actual state. Main configurations and characteristics of the arcjet thruster are as follows:
- Nozzle throat diameter: 0.3 mm,
- Cathode gap: 0 mm,
- Thrust: 59 - 74 mN (nominal 64 mN),
- Input power: 370 W (at nominal thrust).

The on-off test of 1600 cycles, with a repetitive condition of 15sec firing and 15sec radiation cooling, was conducted. Specific impulse levels of 430 to 450 s were kept stably during the endurance test, as shown in Fig.14. The test results show that the durability and stability of the thruster in the long-term operation was demonstrated.

MELCO
Following to the success of the ion engine systems of ETS-VI and COMETS, applications of DC arcjet thruster systems are underway at MELCO under the contract with NASA.
A 1.8kW/250mN class hydrazine DC arcjet is to be used for NSSK on Data Relay Test Satellite (DRTS). Manufacturing of flight model components is almost completed by PRIMEX Aerospace Company, USA, and they will be installed on the spacecraft soon by MELCO.

Astro Research Corporation
Astro Research is a design and consulting company, established in 1996. Their study of electric propulsion is focused on electric propulsion systems for small satellites and their missions. They are studying in co-operation with universities and organizations to develop new small satellite systems with electric propulsion. Also, they have developed a numerical simulation code for electric propulsion plumes based on the direct simulation Monte Carlo (DSMC) method.

HALL THRUSTERS

University of Tokyo
Oscillation phenomena in a Hall thruster were analytically studied. Discharge current oscillations at the frequency of 10 to 100 kHz are often referred to as the ionization oscillation, and the strong one can lead to discharge extinction of the thruster. A one dimensional discharge model was developed combining the conservation equations for ions and neutrals. Spatially sinusoidal fluctuation of the density along the channel was taken into account. Predicted oscillation frequencies well agreed with the observed ones (Fig.15).

![Fig.15 Predicted and observed discharge-current oscillation frequencies.](image)

Osaka University
An experimental facility was constructed to conduct the research and development of Hall effect
thrusters. The facility consists of a water-cooled vacuum tank that is 1.2 m in diameter and 2.25 m long, a compound turbo molecular pumping system with a total pumping speed of 10,000 l/s, several DC power supplies and a thrust measurement system. Endurance tests can be done and contamination due to thruster plumes under a clean and high vacuum environment is investigated. Preliminary experiments were made using two THT-series Hall thrusters as shown in Fig. 16 to obtain fundamental Hall thruster operational characteristics and to examine the effects of magnetic field shape and strength on Hall thruster performance.

![Cross section of the THT-II Hall thruster.](image)

The radial magnetic field strength in the acceleration channel of the THT-I thruster increased towards upstream although that for the THT-II thruster decreased as well as those of the well-known Russian Stationary Plasma Thrusters (SPT). The operational characteristics of discharge current, thrust and ion current vs discharge voltage agreed with those for the SPTs. The maximum thrust efficiency of 29% was achieved at the specific impulse of 1,500 sec for the THT-II thruster. This performance was lower than those for the SPTs because the Hall thruster design was not optimized. The operation for the THT-II thruster was more stable at high discharge voltages than that for the THT-I thruster. The magnetic field shape and strength were found to influence operation stability.

**LASER THERMAL THRUSTERS**

**University of Tokyo**

A laser thermal thruster powered by a CW CO₂ laser (Fig.17) was developed to examine basic characteristics of laser propulsion. It consists of a ZnSe condensing lens, ZnSe window, stainless steel chamber and tungsten throat. Thrust is measured by a load cell, and the heat loss to the chamber wall is evaluated by measuring the heat flux to the cooling water. Figure 18 shows energy transmission distribution of input laser power.

![Illustration of a 2kW-class laser thermal thruster.](image)

![Energy transmission distribution. Laser power: 700 W, propellant: argon.](image)

**LASER FUSION ROCKET**

**Kyushu University**

A magnetic thrust chamber concept in a laser fusion rocket is suitable for controlling the plasma flow, and it has an advantage in that thermalization with wall structures in a thrust chamber can be avoided. A three-dimensional hybrid particle-in-cell code has been developed to analyze the plasma behaviors in the magnetic thrust chamber and to estimate the thrust efficiency.33 It has been found from the simulations that the thrust efficiency is 65% for the configuration shown in Fig. 1. Figure 2 shows a contour plot of the plasma density in the $x-z$ plane at $y = 0$. It is also found that the thrust efficiency seemed to reach its maximum value around the cone angle $\theta_c = 50 \text{ deg.}$
Fig. 19 Schematic of the calculation model for a magnetic thrust chamber.

Fig. 20 Contour plot of the plasma density in the $z-z$ plane at $y=0$ ( $t=8$ msec ).

SPACECRAFT ENVIRONMENT

ISAS

Contamination Measurements of Electric Propulsion and Solid Rocket Motors

An experiment aboard the MUSES-C spacecraft to detect surface contamination by a xenon ion engine is studied. Solar cells were chosen as contamination sensors due to its simplicity and light weight. Sensor short circuit current is the contamination sensitive parameter. In NASA's second Space Electric Rocket Test-II (SERT-II) mission that was launched in 1970, two contamination sensors located in the field of view of the accelerator grid were rapidly covered with sputtered molybdenum. In the MUSES-C mission, carbon-carbon composite is chosen as an ion optics material, whose sputtering rate will be less than that of conventional molybdenum. In order to estimate the degradation of solar cells by deposition of sputtered materials, sensor samples are going to be coated with diamond like carbon (DLC), graphite like carbon (GLC), molybdenum or aluminum using ion plating, sputtering or chemical vapor deposition techniques. Although the embarkation of the sensors to the MUSES-C is now pending due to the lack of payload mass, the first flight opportunity is given for the LUNAR-A mission. Three contamination sensors are mounted near one of the two spin motors housed in B3PL (electronics bay atop the third stage M-34 motor) of M-V-2 rocket. Because the spin motor contains no alumina for fear of contamination of optical sensors and thermal control surfaces on the spacecraft, the contamination will be mainly caused by solid carbon. Three sensors observe high density plume, low density plume and no contamination as a reference. From the output difference of these sensors, the effect of solid rocket plume contamination will be able to investigate. The mechanical environment tests of the sensors and a memory recorder were successfully completed. The contaminants, which will contain carbon, have been sampled from an engineering model of the MUSES-C ion engine and solid rocket motors such as DOM-3 and DFM-1, 2 after the ground tests. A number of surface analysis methods will be applied to these thin contaminated films to investigate optical properties and film structures.

Interaction between Plasma Plume of Electric Propulsion and Spacecraft Communication

Electric propulsion exhausts a highly ionized and dense plasma that potentially can impact on a microwave signal such as communication, navigation or remote sensing of a satellite (Fig. 21). Since the interference with microwave causes fatal damage to the mission, it is necessary to reveal the interaction between the plume plasma and microwave.

The following computational and experimental investigations on the electromagnetic interaction between the EP plasma plume and microwave for spacecraft communication have been conducted:

1. The microwave attenuation and phase delay by an actual EP plume were clarified by both the experiment and numerical simulation of an ion engine and an arcjet. At the observation point, there was no interference between the X-band microwave and each EP plume. However, the S-band microwave was attenuated by the arcjet plume with dense plasma of the order of $10^{10}$ cm$^{-3}$ around the cutoff density.

2. This simulation code was applied to evaluate a large-scale interaction between the plasma of the order of km and the microwave. Computations using a simple plume model showed possibility that the plume pulsation might cause the phase delay of the microwave.

Osaka University

Ground facilities were developed for the simula-
Fig.21 Possible interaction between the exhaust plasma and telecommunication band.

The influence of high-energy charged-particles and atomic oxygen bombardment, and UV light irradiation on chemical structure of polymer films were investigated. The films were exposed to oxygen or nitrogen ion beams with 0.5 keV, to electron beams with 20 - 30 keV, to atomic oxygen with 2.5 eV, and to UV and visible light of 250 - 600 nm. The X-ray photoelectron spectroscopic (XPS) analysis showed that atomic oxygen bombardment and UV irradiation degraded most kinds of polymers. The effect of ion bombardment was also very sensitive to incident ion species, energy and dose.

In order to understand the interaction between a plasma flow and a solar array at a negatively biased voltage, a test plate which consists of an electrode side and a dielectric side, was exposed to an oxygen plasma flow. The collected ion current and spatial plasma potential distributions measured were varied with the biased voltage and attack angle of the plate to the plasma flow. Kapton films, located on the center of the negatively biased plate, were exposed to the oxygen plasma flow. The XPS analysis showed drastic changes of the chemical structure near their surfaces due to ion bombardment. The Kapton films negatively charged by exposure to high-energy electron beams were also exposed to the argon plasma flow in order to understand the relaxation of spacecraft charging by a plasma flow, i.e., construction of a transient ion sheath. Negative charging was rapidly relaxed, and the attack angle influenced the time history of the neutralization current.

Kyushu Institute of Technology

Spacecraft environmental interaction induced by the use of electric propulsion system is being studied. The use of electric propulsion in space brings an inherent concern about EMI (Electro-Magnetic Interference), because the electric propulsion system obtains its thrust by emitting an electric current into the space. Computer simulations using the Monte Carlo Particle-in-Cell method is carried out to study the effects of the plasma jet on communication wave and the electromagnetic noise radiated by the plasma jet. The plasma jet contains a large amount of unionized propellant gas. A fraction of the plasma is scattered back toward the spacecraft. The interaction between the backflow charged particles and the spacecraft surface, e.g., solar array, is also investigated by the computer simulation as well as laboratory experiments.

NON-PROPULSIVE APPLICATIONS

Osaka University

Ceramic Coatings Using MPD Arcjets

For applications of an MPD arcjet to ceramic coatings, two types of an MPD arcjet generator were developed. One was provided with a cathode covered by Mullite or Zirconia ceramics and the other with a titanium cathode. The former was operated with Ar for Mullite or Zirconia coating due to ablation of the cathode cover, and the latter with N₂ for titanium nitride spray coating due to reactive process between ablated titanium particles and nitrogen plasma. Uniform dense coatings 30 µm and 10 µm in thickness for Mullite and Zirconia, respectively, were sprayed with 110 shot operations, and 40 µm for titanium nitride with 50-shot operation. Mullite coatings with above 1000 Vickers hardness were deposited. TiN, Ti₅N and Ti mixed layers for titanium nitride spray coating were observed, and their contents depended on titanium cathode diameter. The hardness of the titanium nitride coatings increased with increasing cathode diameter although the thickness decreased from 40 µm to 3 µm.

Nitriding of Metals by Irradiation of Supersonic Hydrogen/Nitrogen-Mixture and Ammonia Plasmajets

A supersonic reactive plasma jet under a low-pressure environment was utilized for nitriding of metal surface. A supersonic pure N₂ plasma jet was proved to be reactive enough to construct a hard titanium nitride layer on the titanium surface by the few minutes plasma jet irradiation even at a processing chamber pressure of 30 Pa. Furthermore, nitriding of titanium plates and atmospheric thermal sprayed titanium coatings were carried out at 30 Pa using a supersonic H₂/N₂ mixture or ammonia plasmajet. It was seen that the nitriding of the surfaces was enhanced by hydrogen addition to the nitrogen plasma jet or by using a NH₃ plasma jet.
ACKNOWLEDGEMENTS

The authors are deeply indebted to the colleagues who provided the updated information on electric propulsion research/development activities of each organizations.

REFERENCES


