REVIEW OF ELECTRIC PROPULSION ACTIVITIES IN JAPAN

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Abstract

This paper reviews the present status of research and development activities on electric propulsion in Japan. Four 25 mN xenon ion thrusters were successfully operated on ETS-VI. The thrusters had been planed to be operational for its north-south stationkeeping, but were operated as an in-orbit experiment because of the failure in putting into geostationary orbit. The same types of the thrusters are to be onboard COMETS, which will be launched in 1997. Research and development have been continued for DC ion thrusters with 150 mN thrust, RF ion thrusters, microwave discharge ion thrusters, and Hall thrusters. Studies have been started in other fields such as feasibility of C_{60} for ion thruster propellant, carbon-carbon grids for ion thrusters, and mission analysis for electric propulsion. An MPD arcjet thruster was successfully operated on SFU. It generated over 40,000 pulsed firings, and the thrust impulse obtained in the flight showed very agreement with the ground test results. Experimental and computational studies have been continued for DC arcjets, quasi-steady MPD arcjets, and their non-propulsion applications. Two electro-thermal hydrazine thrusters were successfully operated in a flight experiment on ETS-VI.

Introduction

Japan covers a variety of research and development activities in the field of electric propulsion, on ion thrusters, DC arcjets, MPD thrusters, and other types of thrusters, from fundamental studies to in-flight experiments. These activities have been performed by several governmental research institutes, universities, and industries. Research and development of ion propulsion systems have been performed by the National Space Development Agency of Japan, the National Aerospace Laboratory, University of Tokyo, Tokyo Metropolitan Institute of Technology, Mitsubishi Electric Corporation, and Toshiba Corporation. DC arcjets and MPD thrusters have been studied by the Institute of Space and Astronautical Science, Osaka University, University of Tokyo, Kyushu University, Nagoya University, some other universities, and Ishikawajima-Harima Heavy Industries. Research on other types of thrusters are also being conducted by some of these organizations.

As for ion propulsion, four 25 mN xenon ion thrusters were successfully operated on ETS-VI. The thrusters had been planed to be operational for its north-south stationkeeping, but, in fact, were operated as an in-orbit experiment because of the failure in putting into geostationary orbit. The same types of the thrusters are to be onboard COMETS, which will be launched in 1997. In research and development, efforts have been continued for DC ion thrusters with 150 mN thrust, RF ion thrusters, microwave discharge ion thrusters, and Hall thrusters. Studies have been started in other fields such as feasibility of C_{60} for ion thruster propellant, carbon-carbon grids for ion thrusters, and mission analysis for electric propulsion.

On plasma propulsion, an MPD arcjet thruster was successfully operated on SFU. It generated over 40,000 pulsed firings, and the thrust impulse obtained in the flight showed very agreement with the ground test results. Experimental and computational studies have been continued for DC arcjets, quasi-steady MPD arcjets, and their non-propulsion applications. Two electro-thermal hydrazine thrusters were successfully operated in a flight experiment on ETS-VI.

This paper reviews these activities performed by each of the organizations introduced above. Details are described in the reference papers [1-27].

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Ion Propulsion

National Space Development Agency of Japan (NASDA)

NASDA has developed a 25 mN ion thruster for the NSSK of NASDA's Engineering Test Satellite-VI (ETS-VI). This ion thruster (Fig. 1) is an electron bombardment type thruster. BBM, DM, EM and PM thrusters were manufactured and tested before FM thrusters were manufactured. The performance test results including endurance test results of preceding four models were reflected to the design of FM thrusters.

Nine thrusters (a BBM, two DM, four EM and two PM) experienced total operation time of 72,000 hours with over 7,000 hours for each thruster. The long total operation time insures high reliability of the thruster. On the other hand, minimum operation time of each thruster insures minimum life time. The operation time of 6,500 hours was required for ETS-VI NSSK.

The thruster and a propellant storage/supply system including a propellant flow controller were developed by Mitsubishi Electric Corporation (MELCO). Power supplies and a sequence controller were developed by Toshiba Corporation (Toshiba). The interface matching test between thrusters and power supplies was carried out by MELCO. Two EMs of the power supply were used in the thruster endurance test for verification of long time interface matching with thrusters.

FM thrusters were installed on ETS-VI for the first time at Tanegashima Space Center three months before launch. Figure 2 shows ETS-VI just after FM thrusters were installed. N_2 gas purge had been conducted for FM thrusters until just before launch. It is not desirable that the hollow cathode in ion thruster is exposed in the moist atmosphere. The exposure time in the atmosphere is restricted to 1,000 hours for ETS-VI ion thrusters.

The ion thrusters and the power supplies were checked out a month after launch. The check out was carried out in satellite's low spin mode, since the satellite was not put into the geosynchronous orbit. The high voltage break down due to the outgoing gas from grids occurred frequently for a while after high voltage (1 kV) was supplied. After several hours operation, however, it reduced rapidly with the reduction of outgoing gas. The imbalance of two neutralizer emission current at two thrusters' simultaneous operation was adjusted by the change of propellant mass flow rate for two neutralizers.

Since ETS-VI was not put into the geosynchronous orbit and the NSSK maneuver came to be unnecessary, the

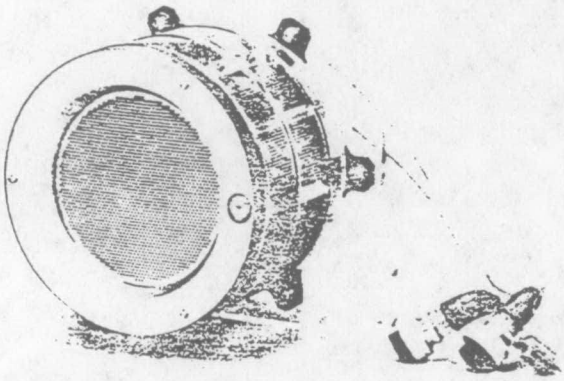


Fig. 1 Photograph of ETS-VI ion thruster.

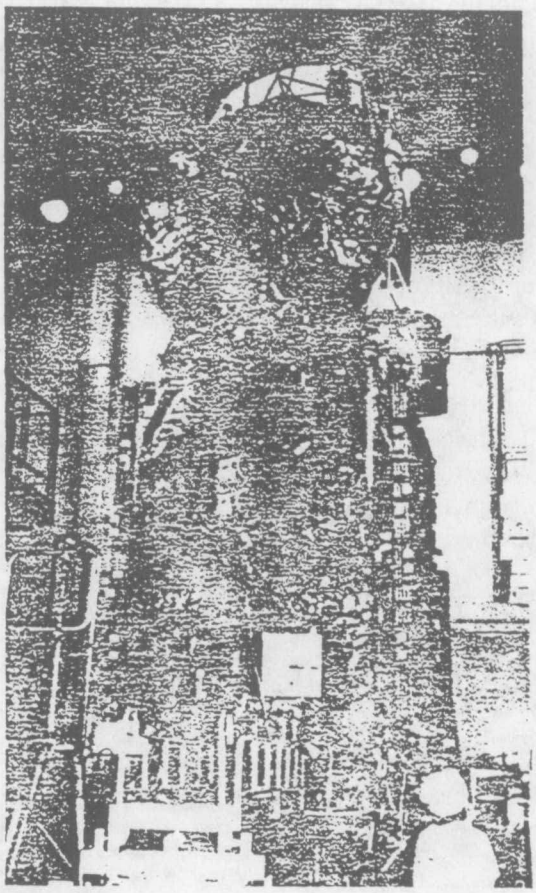


Fig. 2 ETS-VI with FM thrusters installed.

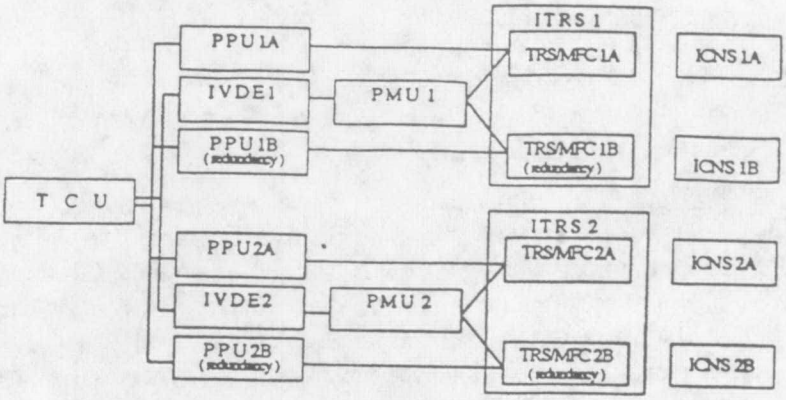
beam thrusting of ion thrusters was carried out as one of ETS-VI in-orbit experiments. The beam thrusting time is about 50 hours for two primary thrusters and about 30 hours for two redundant thrusters by now. The sequence controller permits single thruster operation and also dual thruster operation. The single operation mode was used to measure the thrust of an ion thruster in the way of measurement of satellite wheels' momentum. The thrust efficiency and the specific impulse were both satisfactory. The same ion thrusters are to be aboard on NASDA's Communications and Broadcasting Engineering Test Satellite (COMETS) for NSSK. At present, the acceptance test is being conducted. COMETS will be launched in 1997.

NASDA has started research of large ion thrusters with the National Aerospace Laboratory. The thrust is nominal 150 mN. MELCO and Toshiba manufactured the research models. This ion thruster is expected to be aboard on EOTV experimental vehicle whose concept is under study now. Basically, the EOTV experimental vehicle is one ton class vehicle and put into the low earth orbit by a launch vehicle. Then, the orbit altitude will be raised by the thrust of ion thruster beyond geostationary orbit.

Mitsubishi Electric Corporation (MELCO)

ETS-VI IES

MELCO has developed Ion Engine Subsystem (IES) for Engineering Test Satellite VI (ETS-VI) under the contract with NASDA [1, 2]. Figure 3 shows the subsystem block diagram. IES was successfully operated on ETS-VI, and on-orbit performance data were obtained [3]. At the beginning of October in 1994, the performance of IES was successfully verified by initial on-orbit check out. After the initial check out, about 20 times of operation were conducted and total operation time (sum of four thrusters) is about 160 hours (55 hours for the longest operated thruster) by the end of June in 1995.



Note : TCU, IVDE and PMU have a full redundancy in one unit

TCU = Thruster Controle Unit
PPU = Powe Proccessing Unit
IVDE = Ion Engine Valve Drive Electronics
PMU = Propellant Managing Unit
TRS = Thruster
MFC = Mass Flow Controler
ITRS = Ion Cluster
ICNS = Ion Engine Contamination Shield

Fig. 3 IES Subsystem block diagram for ETS-VI.

Operating parameters of ion thruster 1A on orbit are shown in Table 1, with the corresponding

Table 1 Operating parameters of Thruster 1A.

TRS-1A	23.3mN-6%		23.3mN		23.3mN+6%	
	Ground Test	On Orbit	Ground Test	On Orbit	Ground Test	On Orbit
BEAM Voltage	1050.V	1045.V	1050.V	1045.V	1050.V	1041.V
BEAM Current	444.mA	439.mA	472.mA	468.mA	500.mA	505.mA
ACCEL. GRID Voltage	492.V	(523.V)	496.V	(523.V)	525.V	(520.V)
ACCEL. GRID Current	1.8mA	1.6mA	1.9mA	1.7mA	2.1mA	1.9mA
DISCHARGE Voltage	35.8V	36.7V	37.3V	36.4V	36.1V	37.6V
DISCHARGE Current	3.55A	3.55A	3.54A	3.53A	3.90A	3.87A
MHC KEEPER Voltage	3.5V	3.3V	3.9V	3.6V	3.2V	2.8V
MHC KEEPER Current	0.5A	0.5A	0.5A	0.5A	0.5A	0.5A
NHC KEEPER Voltage	17.3V	12.9V	17.4V	12.9V	16.8V	12.6V
NHC KEEPER Current	0.5A	0.51A	0.5A	0.51A	0.5A	0.51A
Propellant (Xe) Flow : CHANBER	7.30sccm	7.31sccm	7.80sccm	7.83sccm	8.15sccm	8.19sccm
Propellant (Xe) Flow : NHC	0.42sccm	0.58sccm	0.42sccm	0.58sccm	0.42sccm	0.58sccm
THRUST	22.0mN	21.7mN	23.4mN	23.2mN	24.8mN	24.9mN
SPECIFIC IMPULSE*	2,984sec	2,886sec	2,983sec	2,886sec	3,030sec	2,978sec

NOTE *: Isp on orbit is lower than that of ground test because NHC Xenon flow rate is increased to balance the neutralizing current

data obtained in the ground test. By comparing these data, including data of other thrusters, there are little differences between on-orbit and ground data except for the neutralizer keeper voltage. The difference of neutralizer keeper voltage is caused by the amount of neutralizing current and also by the increase of neutralizer mass flow rate.

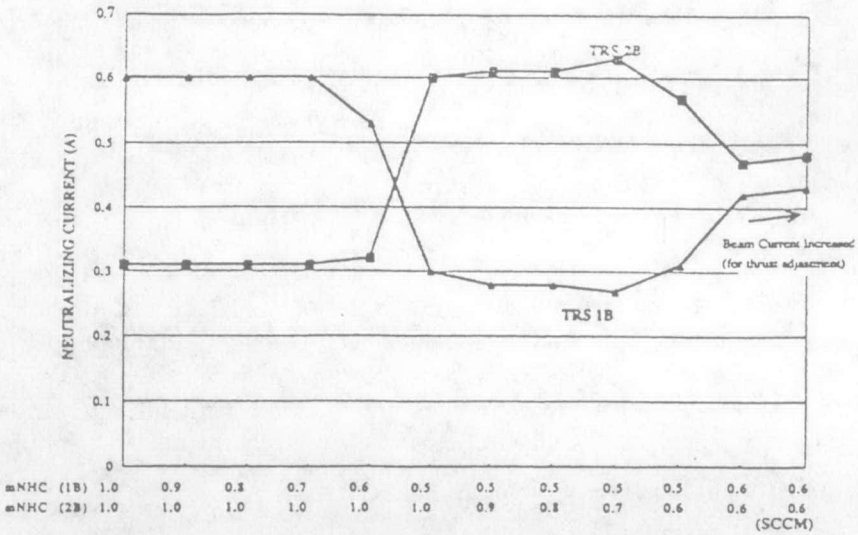


Fig. 4 Neutralizing current balancing procedure.

On-orbit thrust measurement was conducted using the satellite attitude control data. It resulted that both 1A and 2A thrusters were verified to generate about 24 mN of thrust with 3,000 sec of specific impulse at nominal 23.3 mN operating point. This thrust is 95 to 98 % of ideal thrust. These results show good agreement with two ground test results; direct thrust measurement by pendulum method and estimation from ion beam characteristics measurement.

Unbalance of neutralizing current was observed on orbit when two thrusters were simultaneously operated, as was observed in the ground test. However, it was revealed that we can balance them by changing the mass flow rates to neutralizers as predicted from ground test data [2, 4]. Figure 4 shows the balancing process between simultaneously operated thrusters (TRS 1B and 2B) at beam current of 0.45~0.48 A. At first, there was unbalance of 0.3 A at 1.0 SCCM neutralizer mass flow rates (mNHC) for both thrusters. So we changed mNHC and finally we were able to balance them at 0.6 SCCM mNHC for both thrusters. Once they were balanced, no fluctuation was observed. Acceptance tests of IES for COMETS are now under way.

Thruster Design

BBM of 150 mN class ion thruster, cusp-type electron bombardment thruster, has been designed and fabricated under the contracts with NASDA and NAL. Trial manufacture of carbon-carbon composite grids, for thruster life elongation, has been completed and the performance test of these grids is to be conducted.

Toshiba Corporation

Toshiba developed four power supplies and one sequence controller for the ion engine subsystem of ETS-VI. Two power supplies are for primary thrusters, and the other two are for redundant thrusters. The ion engine subsystem was checked out a month after launch of ETS-VI. Beam thrusting of ion thrusters was then carried out as one of ETS-VI in-orbit experiments. Successfully, each power supply could feed each thruster with electric powers including high voltage of 1 kV for beam power, and the sequence controller could control the power supplies and propellant storage/supply system.

On the COMETS ion engine system, the acceptance test of each component including power supply, sequence controller and propellant storage/supply system, has completed. At present, acceptance test of the ion engine subsystem is being conducted by Toshiba.

National Aerospace Laboratory(NAL)

Research on a xenon ion thruster for auxiliary propulsion has been continued. The thruster has an ion-beam exhausting diameter of 14 cm, and exhausts an ion beam of 480 mA to produce a thrust of 25 mN at a specific impulse of 3,500 s. In this thruster, ion-beam divergence angles were measured for three kinds of accelerator grids with different grid hole diameters and compensation [5]. Test results (Fig. 5) revealed that the ion-beam divergence angle (95 % half angle) decreases with decreasing the accelerator

grid voltages. The smallest divergence angle was 23.1°. Appropriate design parameters were obtained for the accelerator grids.

Fabrication and testing were conducted of carbon-carbon composite grids for a 14 cm ion thruster [6]. Among several drilling methods examined, mechanical drilling with tungsten-carbide bits gave the best result in terms of hole finish and drilling cost. Then, ion optics of two-grid system was fabricated from carbon-carbon panels. The screen grid achieved an open area fraction of about 67 %. The grids were installed to the 14 cm diameter thruster. Performance tests revealed that the carbon-carbon grids gave thruster performance comparable with the molybdenum grids (Fig.6). The ion beam was less divergent for the carbon-carbon grids than for the molybdenum grids. However, arcing between the grids occurred more frequently and at lower total voltages for the carbon-carbon grids. Moreover, the carbon-carbon grids suffered matrix erosion due to the arcing. These suggested that carbon-carbon grids require further improvements to mitigate arcing problems, and to have higher strength, particularly, of screen grids.

National Aerospace Laboratory/Toshiba Corporation

Research has been continued on 30 cm diameter ring-cusp xenon ion thrusters for primary propulsion. The thrusters were designed to produce a thrust of 150 mN and a specific impulse of 3,500 s at an exhaust ion beam energy of 1 keV. The second laboratory model LM2-M2 (Fig. 7) is the same as the LM2-M1 thruster except for a curvature of an accelerator grid [7]. The radius of curvature for the accelerator grid has been reduced to 1.3 m, resulting in an increased dish depth of 9.76 mm. Performance tests showed an ion production cost of 160 W/A and a discharge voltage of 30 V at a discharge chamber propellant utilization efficiency of 90 %. The ion production cost was improved by 6 W/A compared with the LM2-M1 thruster.

Tokyo Metropolitan Institute of Technology

<u>Evaluation of Fullerene (C_{60}) Application to Ion Thruster</u> [8]

The feasibility of C_{60} application to ion thruster has been studied. The re-solidification of the sublimated C_{60} on the Pyrex discharge chamber of RF ion thruster was already presented [9]. Based on the above and the results obtained at NASA-JPL [10] and MIT [11], C_{60} application to electron bombardment thruster was examined. Several considerations, such as discharge chamber body heater, were taken into account to prevent the re-solidification, fragmentation and multi-ionization.

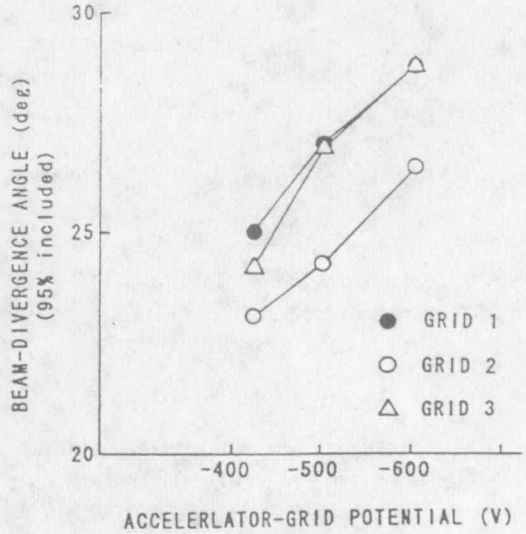


Fig. 5 Ion-beam divergence angles for NAL 25 mN ion thruster.

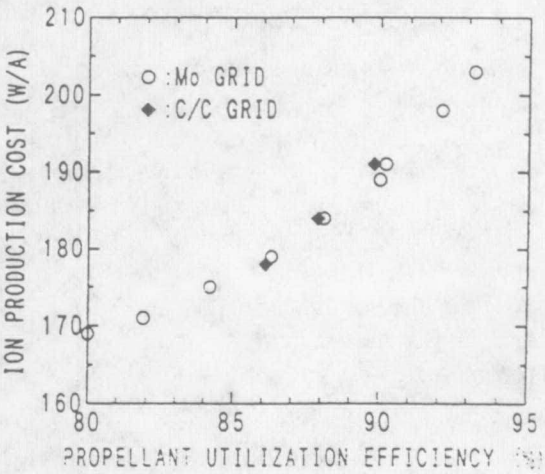


Fig. 6 Comparison of thruster performance for carbon-carbon and molybdenum grids.

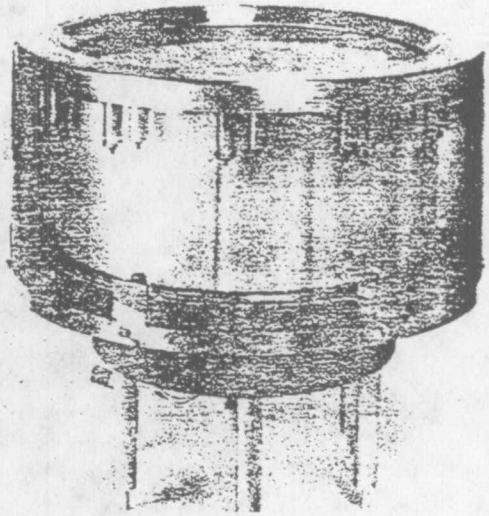


Fig. 7 NAL/Toshiba 150 mN thruster LM2-M2.

Flow rate of 0.5 SCCM was obtained and the body heater was effective against the re-solidification and the fragmentation was not observed. As shown in Fig. 8, both positive and negative charge generations were detected by the target and these charges seem C_{60} ions by FTIR. In the preliminary ion extraction, the ion beam with subtle luminosity was observed. The obtained performance using C_{60} is inferior to that using xenon, and the improvement of the mass sublimation and the plasma confinement are necessary for the next step.

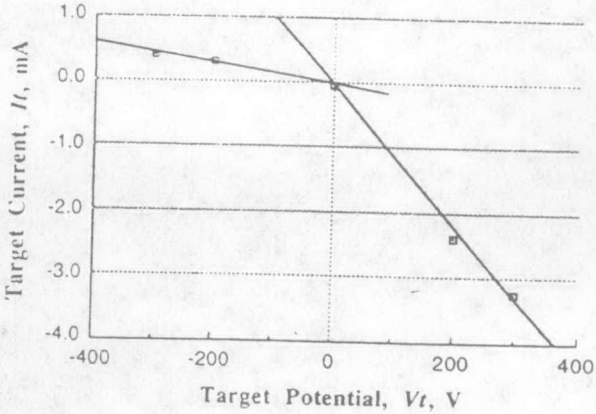


Fig. 8 Target current versus target voltage.

RF Ion Thruster Study [12]

The effects of the propellant feed method on the performance have been evaluated. These results show that the radial propellant gas feed from the axis to the outside in the discharge chamber is effective for the performance improvement. The best obtained performance shows that its thrust density and thrust power ratio is comparable or more with the practical thruster.

Institute of Space and Astronautical Science (ISAS)

Microwave Discharge Ion Thruster

A working group in ISAS plans the sample return mission from the asteroid named Nereus. A spacecraft of 360 kg (Fig. 9) launched by the Japanese M-V rocket on January 2002 will rendezvous Nereus, and bring rocks or sand to Earth by a reentry capsule. This spacecraft will be propelled to Nereus and to Earth by the microwave discharge ion thruster. The microwave discharge ion thruster is adaptive to this mission due to the features as follows: (1) A power supply system is drastically simplified due to the plasma generations in an ion source and a neutralizer fed by a single microwave generator (Fig. 10). (2) The electron-cyclotron-resonance microwave plasma generation is released from the degradation of the discharge electrodes and the thermionic electrodes, and the erosion of screen grid. (3) Plasmas in an ion source and a neutralizer are ignited quickly by microwave injection without any preheat sequence.

ISAS will start the development program of the ion thruster system with a long-life, light weight and low electrical power, soon. The performance of 300 eV of the ion production cost and 85 % of the propellant utilization efficiency has been achieved. Previous to the development program the 300-hour

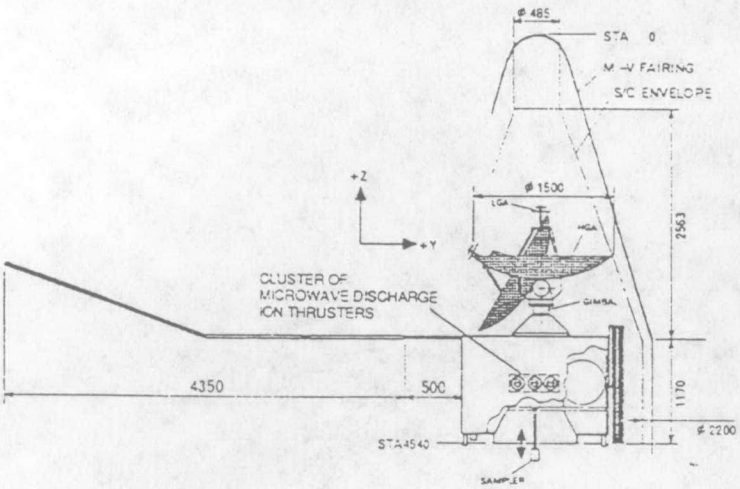


Fig. 9 Spacecraft plan for asteroid sample return mission.

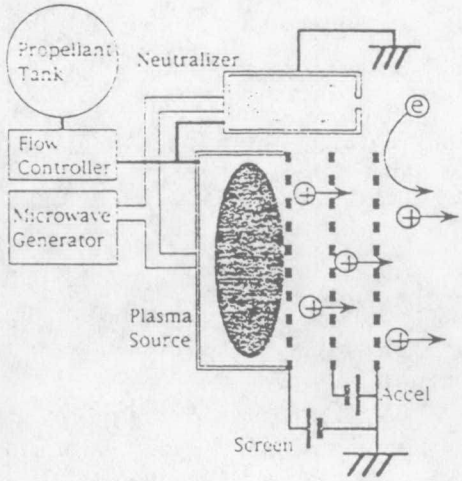


Fig. 10 Schematic diagram of ISAS microwave discharge ion thruster.

automatic operation was successfully demonstrated using a carbon-carbon composite grid system. Neither a degradation of the performance nor any marks just like a discharge flake in the discharge chamber was observed after the test. The screen grid showed slight weight loss, which is acceptable to the long operation over 10,000 hours. A durability of the accel grid against the charge exchange sputtering was not identified due to a contamination of the facility effect.

Nagoya University

Hall Thruster

Experimental works have been continued on a Hall thruster [13]. The optimum channel length for Hall thruster operation was investigated using a variable channel thruster (Fig. 11). The channel length at which maximum efficiency is marked is found to depend on operating conditions and propellant gases, but not on applied-field strength.

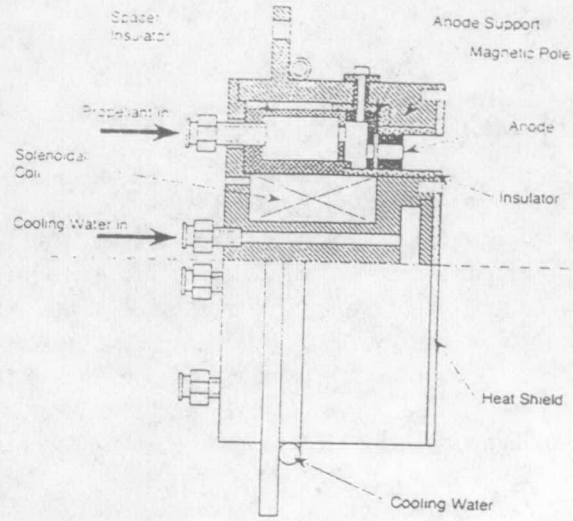


Fig. 11 Variable channel Hall thruster.

University of Tokyo

Hall Thruster

In a Hall thruster, the assumption of quasi-neutrality cannot be used in the neighborhood of the anode and of the wall surface and in the downstream region of the acceleration channel. It is because the charge neutrality can be violated in low-density plasma with strong magnetic field. Therefore, we have developed the particle simulation method to examine plasma phenomena in such regions. In our code, the charged particle simulation is based on a Particle-in-Cell (PIC) method coupled with Monte Carlo techniques which presume the electrostatic approximation [14]. Using this code, we observed the periodic plasma fluctuation in the azimuthal direction. Two curves in Fig. 12 show the azimuthal distributions of the space potential at different times. With the resulting azimuthal electric field, electrons come to more easily move across the magnetic field lines. Figure 13 is a typical calculated result with a calculated azimuthal electric field and qualitatively agree with the measured one. The results suggest that this particle simulation method is useful to examine plasma phenomena.

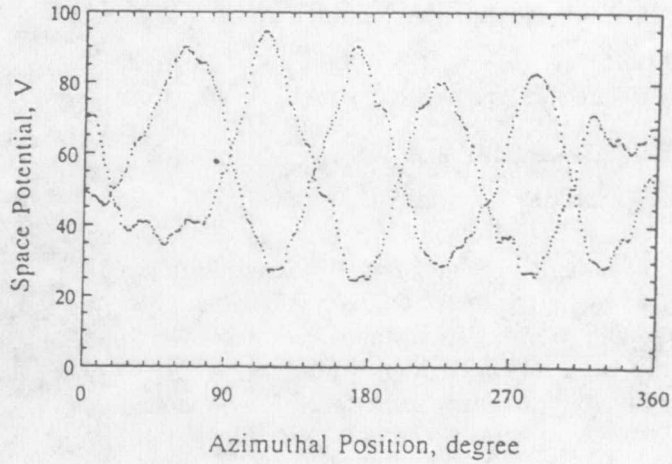
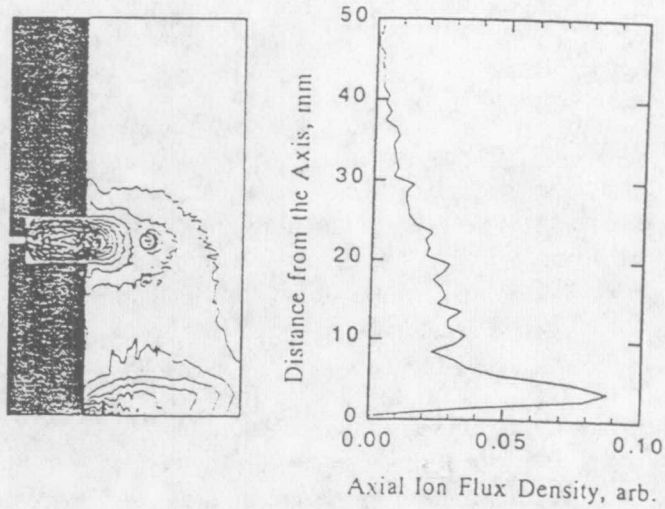


Fig. 12 Azimuthal distributions of space potential.



(a) Ion density. (b) Axial ion flux density.

Fig. 13 Typical calculated distributions.

Mission Analysis [15]

Although mission analysis is important for electric propulsion system design, optimizing trajectories of low-thrust missions is so difficult and time-consuming that parametric study

on mission analysis using electric propulsion system is rather difficult.

Table 2 Comparison of computation time.

Optimization method	DCNLP	SCGRA	Total
DCNLP(16 time divisions) + SCGRA(256 time divisions)	1	1	2
DCNLP(16 time divisions) + SCGRA(512 time divisions)	1	3	4
DCNLP(64 time divisions) only	80	–	80

To overcome this problem, we have developed an optimum trajectory calculation code that has good convergence to initial estimates and takes much lower computation cost than the codes developed so far. To realize this, two mathematical methods for obtaining optimum solutions are employed and combined in the present work. One method is DCNLP (Direct Collocation with Nonlinear Programming), which has robust convergence to initial estimates, however, computation cost becomes too high for precise final solutions. The other one is SCGRA (Sequential Conjugate-Gradient Restoration Algorithm), which is superior in computation cost, but it is inferior in convergence unless initial estimates satisfy constraints. In this study, the DCNLP with small number of time steps is used to get initial approximate values and then the SCGRA is employed to obtain precise final solutions. Computing times of the combined method for a typical Earth-asteroid rendezvous case are summarized in Table 2. From the result, it is found that the present method is very effective and efficient to compute optimum trajectories.

Plasma Propulsion

Institute of Space and Astronautical Science (ISAS)

EPEX onboard SFU

The EPEX (Electric Propulsion Experiment, Fig. 14) is an MPD (Magnetoplasmadynamic) arcjet thruster experiment installed on the SFU (Space Flyer Unit, Fig. 15), which was launched on March 18, 1995. In this experiment the EPEX thruster system has been successfully operated to generate over 40,000 pulsed firings of plasma on-orbit using hydrazine propellant (Table 3). The thrust was confirmed as the attitude disturbances to this 4 ton spacecraft by the on-board NGC system. Obtained impulse of about 3 mNs per pulse showed very good agreement with the ground test result.

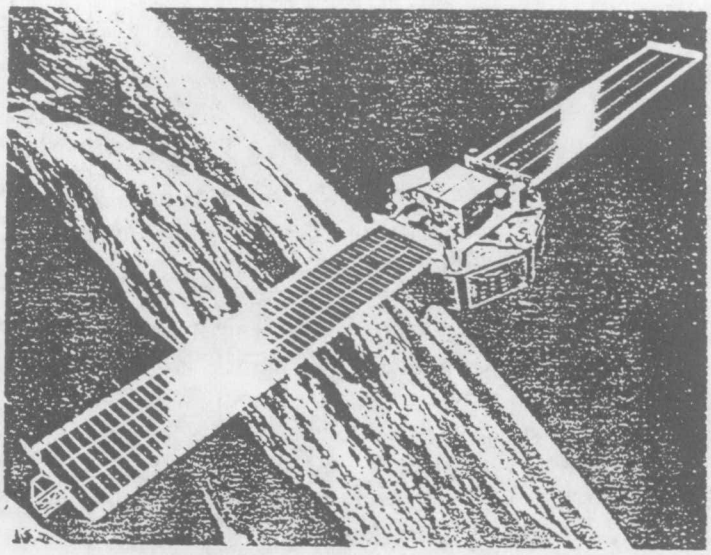


Fig. 14 Space Flyer Unit (Japanese Free-Flying Platform).

This flight system was designed after a full-scale KW-class breadboard model which already finished 25 days endurance test on the ground in 1988 [16]. The MPD arcjet was already tested in space in 1981 onboard MST-4 satellite using ammonia propellant and in 1983 onboard Spacelab-1 (SEPAC: Space Experiment of Particle Accelerator) using argon propellant, but those were not authentic propulsion system with the sufficient electrical input power of multi-hundreds watts. In the EPEX the maximum system input power was 430 W from the bus solar array or battery with the bus voltage ranging from 34 V to 50 V and the propellant was hydrazine. Due to

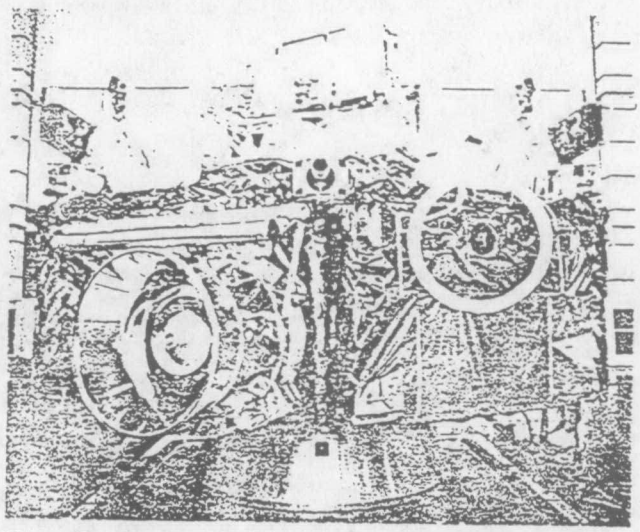


Fig. 15 MPD arcjet thruster installed as the EPEX.

strong requirement of mass saving of SFU, the EPEX system was reduced from the breadboard model in stages of pulse forming network and comprises only one-staged LC-ladder network of which total capacitance amounts to 2.5 mF. This capacitor bank produces 6 kA with 0.15 ms pulse width arc discharge and drives the igniter circuit together with the two set of FAV (Fast Acting Valves) for supplying hydrazine decomposed gas to the thruster. The system works as a repetitively pulsed MPD arcjet at the rates of 0.5 to 1.8 Hz maximum.

Since the safety consideration of SFU, namely the safe retrieval by a space-shuttle "Endevour" (STS-72), has the highest priority in the project, the residual hydrazine dump into space after the experiment is mandatory to the EPEX. On July 21, 1995 the EPEX successfully opened the hydrazine dump valve after about 3 days assigned experimental period to dispose the residual little liquid hydrazine with pressurant nitrogen gas.

Two-Dimensional MPD Arcjet

In order to acquire the design guideline of MPD arcjet, a multichannel two-dimensional MPD arcjet (2DMPD, Fig. 16) in quasi-steady operation was employed, whose two dimensionality enables the visualization of the two-dimensional flowfield and can also reveal the correlation between the internal flowfield and the thrust performance [17]. Thrust performance and internal plasma flowfield of a 1 MW class 2DMPD arcjet were measured in order to clarify their dependence on its cross-sectional geometry of electrodes [18]. According to investigations among six different electrode's configurations, the thrust performance of the hydrogen MPD heavily relied on its chamber cross-sectional geometry as shown in Fig. 17, especially in realistic Isp range around 1,000 s to 3,000 s. Regardless of the anode geometry, cathode geometry was the main factor which decided the performance, and the convergent-divergent anode with short cathode showed the best performance.

Table 3 EPEX result.

- Thrust of 3 mNsec / shot and peak Isp = 800 - 1,000 sec confirmed.
- Over 40,000 shots firings successfully achieved.
- Misfiring was less than 0.3 %

Experiment Date	Number of Firings	Remarks
March 18	-	Launched with hydrazine propellant of 130 cc
March 28	-	Systems checkout
May 29-June 2	2,641	Thrust performance confirmed by NGC data
June 24-26	25,184	0.5 - 1.8 Hz, Daytime
July 17-20	15,570	0.5 - 1.8 Hz, Nightime
July 21	-	Residual hydrazine dump

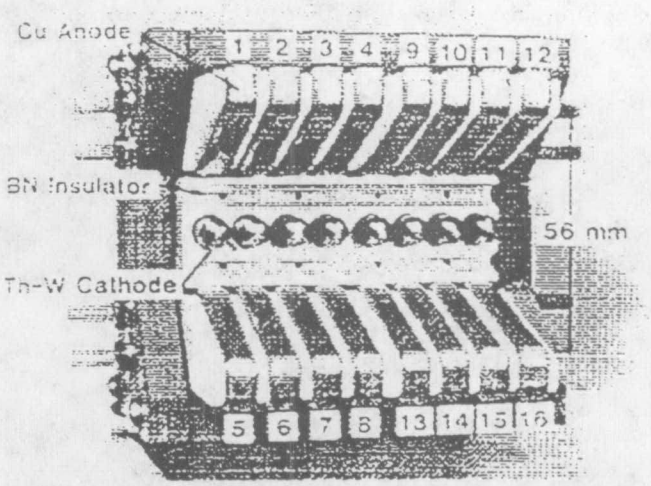


Fig. 16 Two-dimensional MPD arcjet.

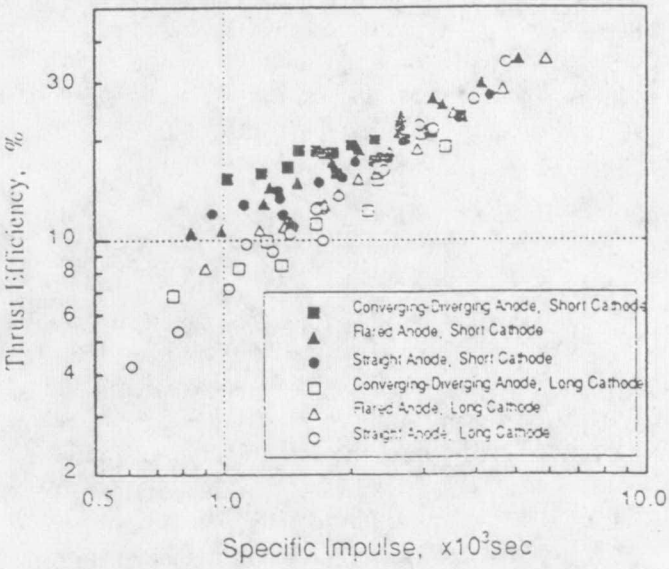


Fig. 17 Thrust performance of a two-dimensional MPD arcjet (Hydrogen propellant).

Low Power DC Arcjet

In ISAS, a 300 W class DC arcjet using hydrazine simulated gas as a propellant is newly developed for NSSK of 1 ton class geostationary satellites. For the purpose of reducing the weight and improving the thrust performance, the thruster is designed very small as the diameter of 16 mm and the weight about 200 g (Fig. 18) [19]. In the preliminary tests for the development of this new arcjet, some problems were perceived for designing and operation of it. Most of the problems were related to the insulator made by boron nitride which was exposed to discharge chamber. Due to the porous and fragile nature of boron nitride, the insulator composed by this material caused leakage of the propellant and damage of itself accompanied by reconstruction of the thruster.

To solve these defects, new machinable ceramics composed mainly of aluminum nitride was employed. Several tests to evaluate thrust performance, thermal condition, and so on were conducted. "Thrust vector measurement" was carried out to evaluate the misaligned component of the thrust, because it was sometimes observed that the plume had not been directed to the normal direction around the power range of 300 W in the preliminary tests. Though no serious misaligned components were observed for the case in which the mass flow rate is higher than 20 mg/s, in the case of the mass flow rate of 10 mg/s misalignment of the thrust about 10 % of the axial component was detected. To understand the physics in the low power arcjet thrusters, a two-dimensional DC arcjet for visualization of inside of the constrictor and flowfield in the nozzle were newly designed (Fig. 19). Preliminary tests to acquire the characteristics of this thruster are to be conducted.

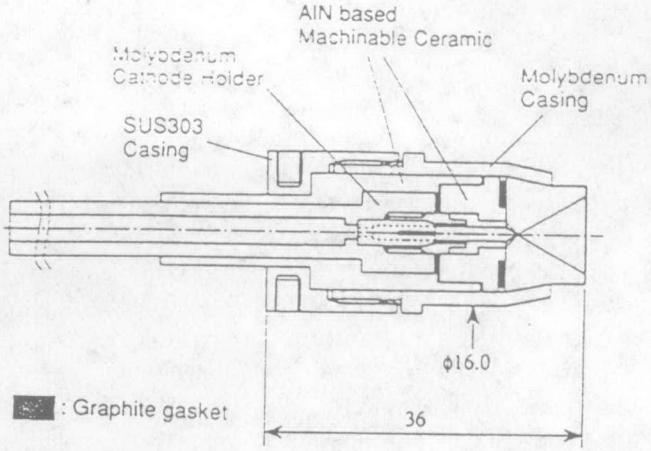


Fig. 18 Newly designed 300 W class low power DC arcjet.

Fig. 19 Two-dimensional DC arcjet.

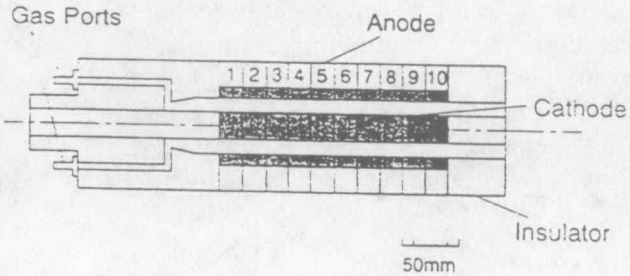


Fig. 20 Cross section of quasi-steady MPD channel MC-II.

Osaka University

Quasi-Steady MPD Arcjet Research

Plasma diagnostics and flowfield analysis in quasi-steady MPD channels (Fig. 20) almost realizing only electromagnetic blowing acceleration, i.e., one-dimensional flowfield, were made [20]. The discharge current concentrated near the downstream end of the discharge chamber, and particularly the current fractions for molecular gases were much larger than those for monatomic gases. However, the current concentration near the inlet, which was expected from the one-dimensional MHD flowfield analysis, was hardly observed. The axial variation of the velocity inferred from the electron number density measured showed that plasma was expected to be drastically accelerated only near the outlet, although the analyzed velocity profile had two acceleration zones near the inlet and the outlet. Also, the influence of applied magnetic fields on thruster performance and energy balance was studied using an

MPD arcjet periodical operation system.

Steady-State Arcjet Thruster Research & Development

The 1-kW-class radiation-cooled arcjet thruster RAT-VI (Fig. 21) was developed for north-south stationkeeping of geosynchronous satellites, in cooperation with Ishikawajima-Harima Heavy Industries Co. Ltd [21]. The arcjet was provided with a propellant passage on the inner wall of the anode. Higher thruster performances were achieved with the regenerative cooling of the anode than those with that of the cathode. Furthermore, to simulate propellant preheating due to exothermic reaction of decomposition of hydrazine in a gas generator in practical use, a mixture of hydrogen and nitrogen heated to 600℃ was introduced into the thruster. The practical use of hydrazine as arcjet propellant was expected to produce higher thruster performance and no problem on operational stability and durability.

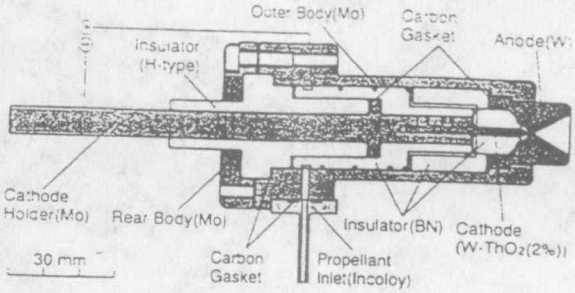


Fig. 21 Cross section of 1-kW-class radiation -cooled arcjet thruster RAT-VI.

Spectroscopic measurement was carried out to understand the flowfield in a 10-kW-class water-cooled arcjet [22]. In the expansion nozzle, as shown in Fig.22, the radial profiles of the physical properties for N_2 and N_2^+ showed that there existed a core flow with high vibrational and rotational temperatures and great electron number densities on the center axis even at the nozzle exit. Both temperatures on the arcjet axis at the nozzle exit increased linearly with the input power. The vibrational temperature ranged from 6,000 to 10,000 K in input power levels of 5-11 kW and the rotational one from 500 to 2,000 K. Arcjet flowfields were also numerically analyzed using a quasi-one dimensional core-flow model.

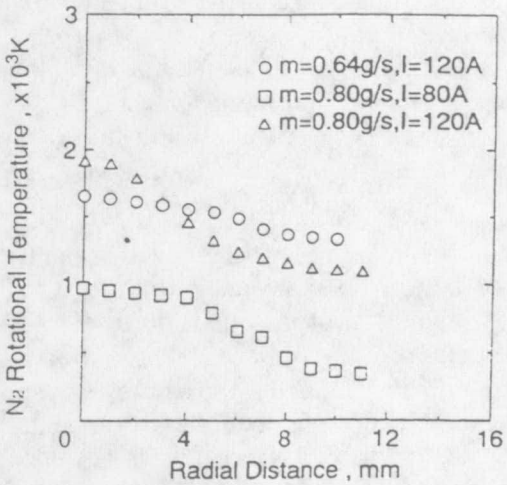
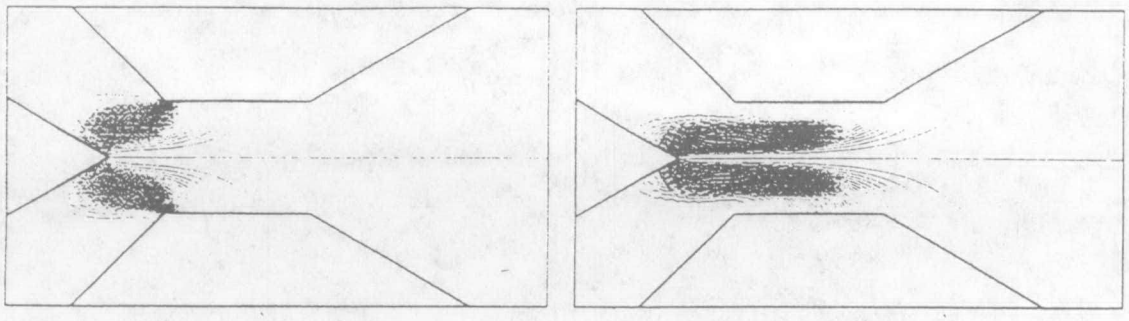


Fig. 22 Radial profiles of N_2 rotational temperature at expansion nozzle exit.

Non-Propulsion Applications

For applications of quasi-steady MPD arcjets to ceramic coatings, an MPD arcjet with a cathode covered with a ceramic material was developed [23]. The front velocities of ablated ceramic Al atoms inferred with a streak camera were much higher than velocities of 200-500 m/s for conventional plasma torches. This is effective for deposition of rigid films adhering strongly to substrate surfaces. The MPD plasma spraying showed that a dense uniform ceramic film with above 1,200 Vickers hardness was deposited.

To understand degradation of materials due to ion bombardment in a space plasma environment, polymers and glasses coated with MgF_2 were exposed to ion beams of oxygen and nitrogen with energy levels ranging from 600 eV to 5 keV [24]. For polymers, XPS analysis showed that the addition reaction of oxygen or nitrogen atoms or the disruption or separation of various structural compounds occurred depending on ion energy and dose. For the glass plate, the coating layer of MgF_2 was drastically sputtered. Also, the study of degradation, charging and discharge on material surfaces and solar cells in space plasma was initiated using an ECR oxygen plasma accelerator.



(a) Ar. $\dot{m} = 148.6$ [mg/s] (b) H_2. $\dot{m} = 14.9$ [mg/s]

Fig. 23 Current stream lines in (a) argon and (b) hydrogen plasma flows at I = 100 A.

University of Tokyo

Arcjet

For years, numerical analysis of the DC arcjet has been continued [25] in order to well-understand the fundamental physics and to develop a performance-prediction code applicable to a thruster design tool. A newly improved code called APAC (arcjet plasmo-dynamic analysis code) [26] takes accounts of finite rate chemistry, electron temperature disparity from the heavy species temperature, generalized Ohm's law, and an electrode-sheath model involving ionization and recombination. Equations of generalized multicomponent gas-dynamic system are constructed as a mathematical model, which is solved by fully implicit direct LU decomposition technique together with Yee's TVD.

One of new findings of APAC is that the temperature of electron becomes higher than that of the heavy particles with approaching the anode wall. Such electron temperature disparity determines the electron mobility near the anode surface wall as well as the electron momentum transfer collision frequencies, resulting in the discharge modes peculiar to the gas species, namely the low and high voltage modes in argon and hydrogen flows, respectively, as shown in Fig. 23. Quantitative comparison of the numerical results with the experimental ones has found that the thruster performance can be predicted by this plausible model with errors less than 16 % with respect to both the specific impulse and thrust efficiency.

Kyushu University

Arcjet Thruster

Computational studies of a flow in an arcjet thruster with molecular gas as propellant are being conducted. Nonequilibrium vibrational excitations, dissociations and ionizations are included in the analysis. The computation uses a two-temperature model (translational-rotational and vibrational-electron temperatures), and this analysis will be extended to a three-temperature (translational-rotational, vibrational and electron temperatures) analysis.

Numerical studies were made of nonequilibrium plume issuing from an arcjet thruster. The plume considered here is an arc-heated nonequilibrium plume issuing from an orifice into low-density stationary air as a freejet. Governing equations for calculations are axisymmetric Navier-Stokes equations coupled with species vibrational energy, electron energy and species mass conservation equations. These equations were numerically solved for various orifice temperatures and various orifice diameters.

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 simple design tool have been tried and the results have been compared with previous experiments. Discharge power is put into propellant gas as heat input and then total enthalpy of the propellant gas is increased. In this work, it is considered that the discharge power is employed not only for elevation of the propellant gas temperature but also for ionization of it.

Nagoya University

The interaction between plasma flows and applied fields is of our main interest both in electric propulsion and in plasma physics. Analytical researches have been conducted on MPD thruster flows [27] and on MHD shock reflections. Two-dimensional numerical codes could catch an oblique shock extending from the cathode tip to the downstream of the thruster (Fig. 24) and also reproduce various patterns of MHD-shock reflections on a conducting-wedge in applied fields. The shock-electrode or shock-nozzle configuration changes the flow patterns inside the thruster and determines its specific impulse. Experimental works have been continued on an applied-field MPD arcjet.

Fig. 24 Pressure contours in an MPD thruster.

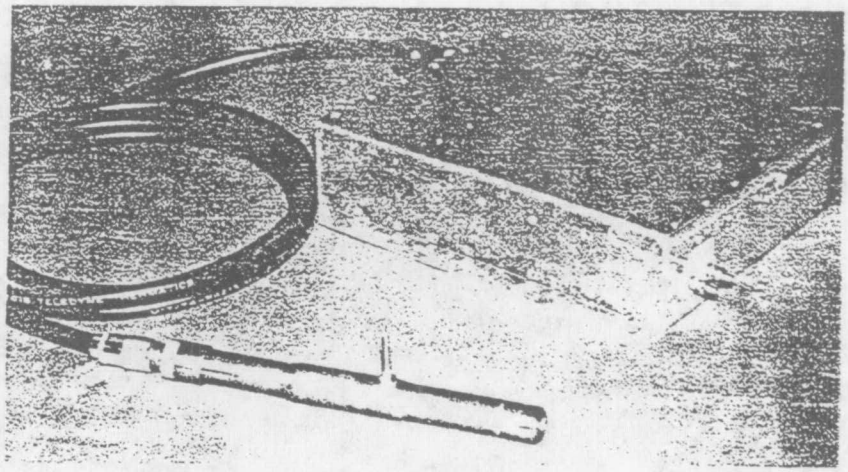


Fig. 25 1 kW-class DC arcjet thruster and power control unit.

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

IHI has been conducting research and development of electric propulsion systems since 1978. Its Research and development Institute has a role of basic study on MPD and DC arcjet thrusters. The Space Development Division has developed flight models of electric propulsion systems.

An MPD arcjet system, called EPEX (Electric Propulsion Experiment) was tested onboard SFU#1. The system used hydrazine as a propellant, and was operated successfully with more than 40,000 shots at maximum power of 430 W. IHI developed the MPD thruster module, propellant supply system and thermal control system under contract of ISAS. The thruster module includes an active thermal control device for continuous pulsed firing. The residual propellant was successfully dumped to space through the port of the propellant supply system after the EPEX system operation was completed.

Two electro-thermal hydrazine thruster systems were launched for experiment onboard ETS-VI in 1994. The system, including a thruster and a heater control unit, was developed as an advanced monopropellant gas jet system under contract of NASDA, and was successfully operated on orbit.

IHI is now focusing on developing DC arcjet system. Feasibility study has been almost completed. The system (Fig.25) has achieved specific impulse higher than 500 s and efficiency of power control unit higher than 90 %. The most important development items for DC arcjet system are propellant flow control technology and endurance performance enhancement. Flight model design of DC arc jet system with power range of 500 W to 1 kW starts now.

Acknowledgments

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