ELECTRIC PROPULSION RESEARCH AND DEVELOPMENT IN JAPAN

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1. Introduction

The present paper gives an overview on electric propulsion activities and its future plans in Japan. Japan covers a variety of electric propulsion activities, ion thrusters, MPD thrusters and arcjet thrusters, from fundamental studies to flight tests. These activities have been performed by government research institutes such as NASDA (National Space Development Agency of Japan), NAL (National Aerospace Laboratory), ISAS (Institute of Space and Astronautical Science), University of Tokyo, Osaka University, Kyushu University, and etc. Some manufacturers, such as IHI (Ishikawajima-Harima Heavy Industries Co. Ltd.), MELCO (Mitsubishi Electric Corporation) and Toshiba Corporation, join and support these activities.

As for ion thrusters, research and development works on the 20 mN class xenon ion engine have been conducted for the flight test onboard the Engineering Test Satellite (ETS) VI which will be launched in 1993. Magnetic cusp type thrusters were also studied both theoretically and experimentally.

In R & D activities on MPD thrusters, a 1 kW class quasi-steady MPD thruster system has been developed for Electric Propulsion Experiment (EPEX) onboard Space Flyer Unit (SFU) which will be launched in 1994. Fundamental efforts were focused on improvement of thrust performance, acceleration mechanism and discharge phenomena of MPD arcjet.

On arcjet thrusters, a low power arcjet system is under development for stationkeeping of geosynchronous satellites and reaction control system for future platforms and spin-stabilized satellites. In addition, modeling and computation on arcjet flowfields were also conducted.

2. Plasma Propulsion

2.1 Institute of Space and Astronautical Science (ISAS)

2.1.1 Space Test of MPD Thruster

EPEX (Electric Propulsion Experiment) onboard SFU-1

EPEX is a space test on MPD propulsion system onboard the Space Flyer Unit - Mission One (SFU-1) scheduled in 1994 (Fig.1). Japanese MPD arcjet has already been flown with MS-T4 satellite in 1980 and with Spacelab-1 in 1983. However the EPEX is virtually the first test for a practical use of MPD arcjet as a thruster system with satisfactory electrical power and more tactical space-storable propellant.[1] The MPD propulsion in this country has the most concetrated interests on the repetitively pulsed quasi-steady thruster system with hydrazine propellant. This concept arises from the commonality use of conventional rocket propellant sharing with the electric propulsions and the complete throttlability of bus electrical power consumption. Since the SFU-1 provides a lot of opportunities for all the onboard experiments, the EPEX system is incorporated into a PLU (Payload Unit) box together with the other type experiments.

The EPEX objectives are:
1) To checkout the MPD thruster system in space,
2) To verify propulsion function,
3) To dump hydrazine into space.

The primary objective is to verify the survivalability of the MPD thruster system against launch and space environments. The secondary objective is planned so that the MPD thruster system is driven by 430 W averaged input power in a repetitively pulsed mode. The generated thrust will be evaluated on the ground test in advance and will be confirmed as the external disturbance against the SFU-1 attitude control by momentum wheels. Although the third objective sounds strange for the unmanned space platform, the SFU-1 will be retrieved after 6 months experimental period by a Space Shuttle, which is a manned system. Since all the heaters for experiments onboard SFU-1 are cut off before retrieval, hydrazine has the potential cause of leakage from tubings or fittings failed by volumetric inflation by melting during the reentry phase. Since this leads to hazardous circumstances inside the Shuttle cargo bay, the residual hydrazine of EPEX was decided to dump it into space before retrieval.

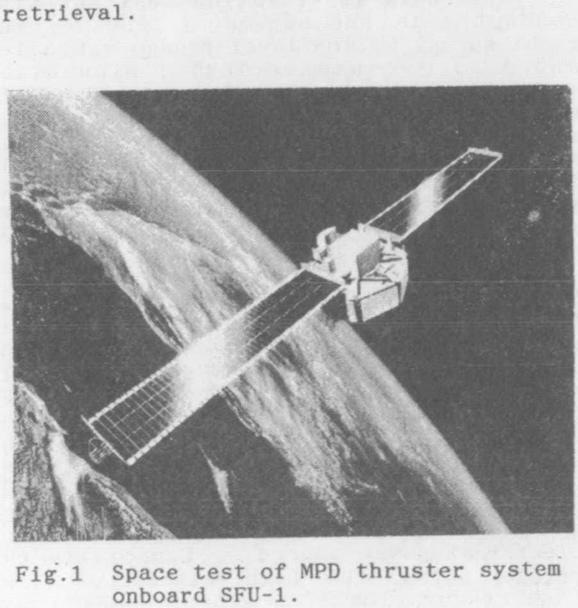

Fig.1 Space test of MPD thruster system onboard SFU-1.

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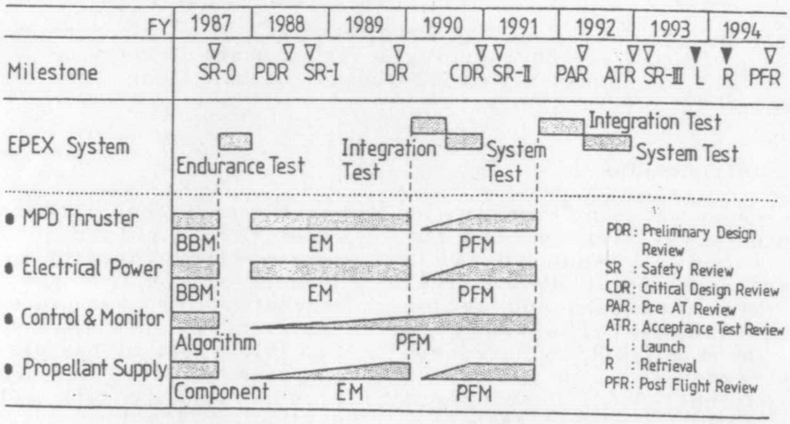
Table 1 EPEX development schedule.



EPEX Development Status

From 1990 to 1991 the EPEX system reached an EM (Engineering Model) phase and successfully finished the CDR (Critical Design Review) after the environmental tests of thermal vacuum test, vibration test, and mass property test (Table 1) . Major characteristics are summarized in Table 2 .

Thermal Vacuum Test / Vibration Test A thermal vacuum test was conducted in the summer of 1990 using thermal/structural dummy components installed inside the PLU (EM) . In this test the worst cold case (-30 °C), the worst hot case (+40 °C) and periodic transient case corresponding to the sunshine/sunshade intervals were simulated to evaluate the thermal analysis model adequacy. As the consequence, heat pipes and thermal louvers on the PLU worked properly (Fig.2) and hydrazine components were verified to be kept higher than its freezing temperature except for a part of the hydrazine dump nozzle exposed into space. Some of the electrical devices on the side panel exceeded their low temperature limit and the reduction of radiation panel was required.

A structural vibration test was also conducted in the autumn of 1990 . The modal survey by low level random vibration and AT (Acceptance Test) sinusoidal vibration level for X-, Y-, Z- axes of PLU were applied in order to confirm the resonance frequency and to evaluate the maximum g-level during the flight. All the components will be dedicated to QT (Qualification Test) level vibration test on the basis of these data before FM (Flight Model) integration test.

EMI Measurement Test Plasma generated EMI (Electromagnetic Interference) was evaluated for the MPD thruster firing inside a glass chamber employing the MIL-STD 462 method (Fig.3) . The RE-02 and RE-01 which are the MIL-STD 461C reference limits were investigated including the electrical power supply system. It was found that the EMI entirely came from the generated plasma plume but there was no evidence of EMI generation from its electrical power source. Especially the trigger pulse with high voltage of 1 kV (short-duration of 5 microseconds) was

Table 2 EPEX system characteristics.

Mass,	44.8 kg (including N_2H_4)
Size,	PLU box (1.1x1.5x1.0m)
Power,	430 W,max
Propellant,	Hydrazine (0.5 kg)
Current,	6 kA (Peak Value)
Voltage,	130 V (Peak Value)
Pulse Width,	0.15 msec
Rep. Rate,	1.8 Hz,max
Thrust/Power,	30 mN/kW
Isp,	600 sec (Peak Value)
Bus Commands,	9
Data Rate,	640 bps,max

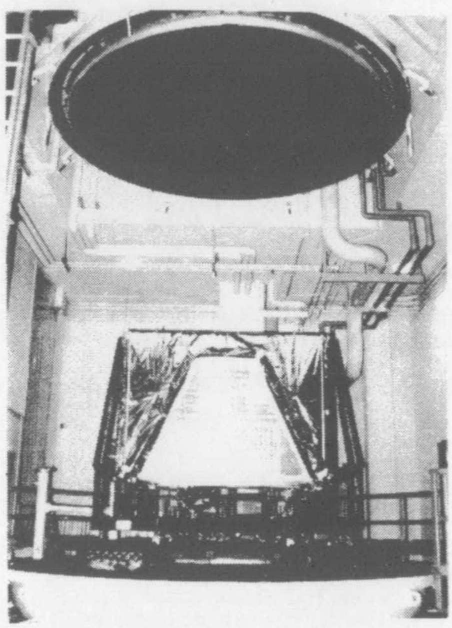


Fig.2 Thermal vacuum test set-up for PLU box (Trapezoidal box has 3 thermal louvers).

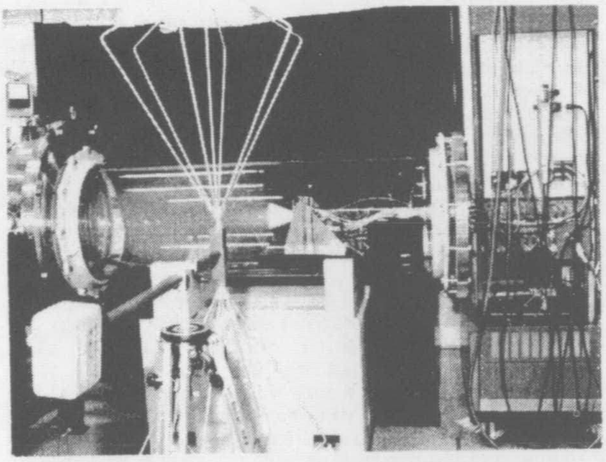


Fig.3 EMI test set-up for RE-01 and RE-02.

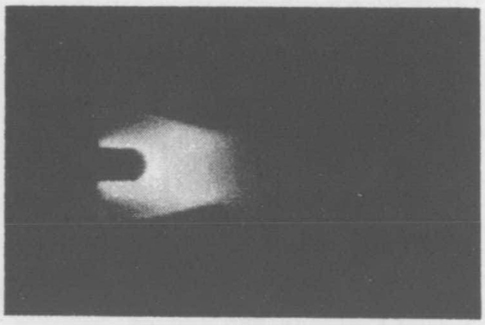


Fig.4 2-dimentional MPD arcjet firing.

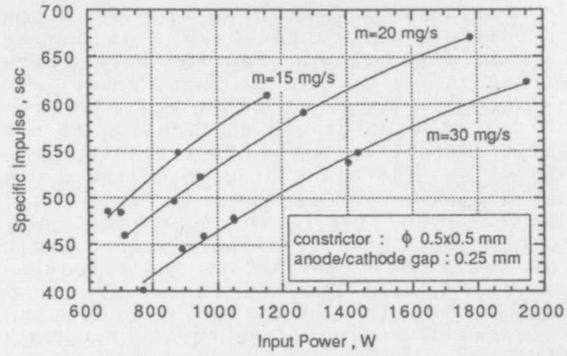


Fig.5 Thrust performance of DC arcjet SAGAMI-I.

identified to be the major source of EMI. The magnitude showed almost isotropic distribution surrounding the plasma plume. In the frequency domain higher than 1 GHz which is usually the communication band revealed no noticeable EMI spectrum for both narrow and broadband and in the low frequency domain of 100 kHz or lower, there found to be severe over specification by 40 to 60 dBμ V/m/GHz. As this over specification gradually decreased with the frequency towards 1 GHz, most of the bus components have far higher susceptibility and the assessment board concluded that there is no problem anticipated.

Hydrazine Dump Test Almost all the hydrazine propellant will be consumed by MPD thruster firing during the flight test, however, the residual hydrazine will amount to 50 g in normal design and to 500 g maximum in case of unintentional failure of the system. For both cases the residual hydrazine must be dumped into space before a Space Shuttle retrieval. As the first step we conducted a hydrazine tank evacuation keeping its temperature by electrical heaters up to 30°C . This test was successfully finished and the residual hydrazine vapor pressure was found to be less than 5 ppm within 10 hrs. The most difficult problem is the hydrazine boiling and freezing just outside the dump port which is exposed to vacuum space. The former leads to contamination of SFU-1 by splashing and the latter leads to improper evacuation of hydrazine tank. Hence, the precise temperature control of the tank and flow passage and the hydrodynamic design verification for non-freezing non-splashing dump nozzle are being attempted.

2.1.2 MPD Arcjet Research

A 2-dimensional MPD arcjet has been used for the basic understandings of the flowfields pattern with argon and hydrogen propellant (Fig.4) . Electromagnetic acceleration and velocity profile inside the discharge chamber were examined employing conventional doppler shift measurement techniques. The acceleration patterns were identified to correspond with the Lorentz force distribution for argon, and a distinct compressing acceleration pattern was found along the centerline for hydrogen discharge. In order to get a more informative 2-dimensional image of velocity patterns, a LIF (Laser Induced Fluoresence) mothod using a pulsed dye laser sheet is examined.[2]

On the other hand, a quasi-one dimensional analysis concluded that the magnetoacoustic choking occurs at the location of magnetoacoustic Mach number being unity, where both the discharge current and the discharge voltage are controlled.

2.1.3 DC Arcjet Research

A low power DC arcjet SAGAMI-I recorded an I_{sp} of 600 sec with the input power less than 2 kW for a N_2+2H_2 flow rate of 30 mg/s (Fig.5) . This model employs a 0.5 mm diam. constrictor and regenerative cooling by propellant gas. The ignition stability is investigated by low current regime pursuing a low erosion and ignition reliability of DC arcjet. Three ignition phases of Unstable, Transition, and Steady mode were identified for lower current operation than 6 amperes. The minimum discharge sustaining current was also found out as function of operating pressure. At less than this current the discharge was spontaneously distinguished after several seconds or a few tens of seconds.[3]

A laboratory model of SAGAMI-II was fabricated aiming at a few seconds pulsed operation of DC arcjet. The SAGAMI-II is a miniature model designed after SAGAMI-I but is extremely reduced both in size and weight. Its dimensions are 1 inch × 1 inch × 1 inch and the weight is only 130 g. The initial test was successfully conducted using N_2+2H_2 propellant showing quick start and quick cut-off. This type of pulsed DC arcjet will be able to serve as a high I_{sp} RCS (Reaction Control System) for future spacecrafts, platforms and spin-stabilized satellites.

2.2 Osaka University

2.2.1 Quasi-Steady MPD Arcjet

Continuous Operational Tests The repetitively operational tests have been undertaken using the developed MPD arcjet system.[4] The present main objectives are to evaluate possibility of higher power operation (more than 10 kW) on the base of data obtained from the repetitively firing tests at average input power levels between 0.7 and 3 kW , and to solve technological problems in application to orbital transfer, interplanetary transportation and etc. The block diagram of the present MPD arcjet system is shown in Fig.6 . The system consists of the MPD head, power supply unit, propellant supply unit including two fast acting valves and operational control unit. The MPD arcjet system has been tested in the following conditions: the used propellants are Ar, H_2 and N_2+2H_2 mixture; discharge current 6-10 kA; repetively operational period 0.5-1.0 sec; duration time for single shot 1.1 msec; specific impulse 600-2,000 sec . The MPD arcjet system was successfully operated in 0.7-3 kW of average input power. The possibility for higher power operation depends on how to reduce the cathode tip temperature.

Axial-Field MPD Arcjet Research A quasi-steady MPD arcjet with applied magnetic fields has been studied to clarify the influence of axial magnetic fields on the thruster performance and discharge feature.[5] Pulsed axial magnetic fields were applied by a few-turn coil, which was connected with a pulse forming network independent of the main discharge circuit. There existed the particular condition at which the discharge voltage with the axial magnetic field was smaller than that only with the self field, though an increase in axial field intensity raised the voltage at most of operational conditions. The thrust characteristics for H_2 showed that the thrust increased with axial field intensity at every discharge current and for mixture of N_2+2H_2 that there was the optimum axial field intensity, with which the maximum thrust was achieved. The discharges for most of operational conditions occurred more upstream with an increase in axial field intensity. However, it was expected that the discharge for H_2 at a low current of 5 kA occurred more upstream up to a transitional field intensity of 0.2 Tesla and more downstream beyond 0.2 Tesla.

2.2.2 DC Arcjet

Low-Power DC Arcjet Thruster The research program has been focused on the development of a 1 kW class hydrazine arcjet propulsion system for application to the north-south stationkeeping for geosynchronous communication satellites, in cooperation with Ishikawajima-Harima Heavy Industries Co. Ltd. A cross sectional schematic of the arcjet thruster in 0.6-mm constrictor diameter is shown in Fig.7.[6,7] The arcjet has operated at power levels between 0.5 and 1.5 kW using hydrazine decomposition products as the propellant.

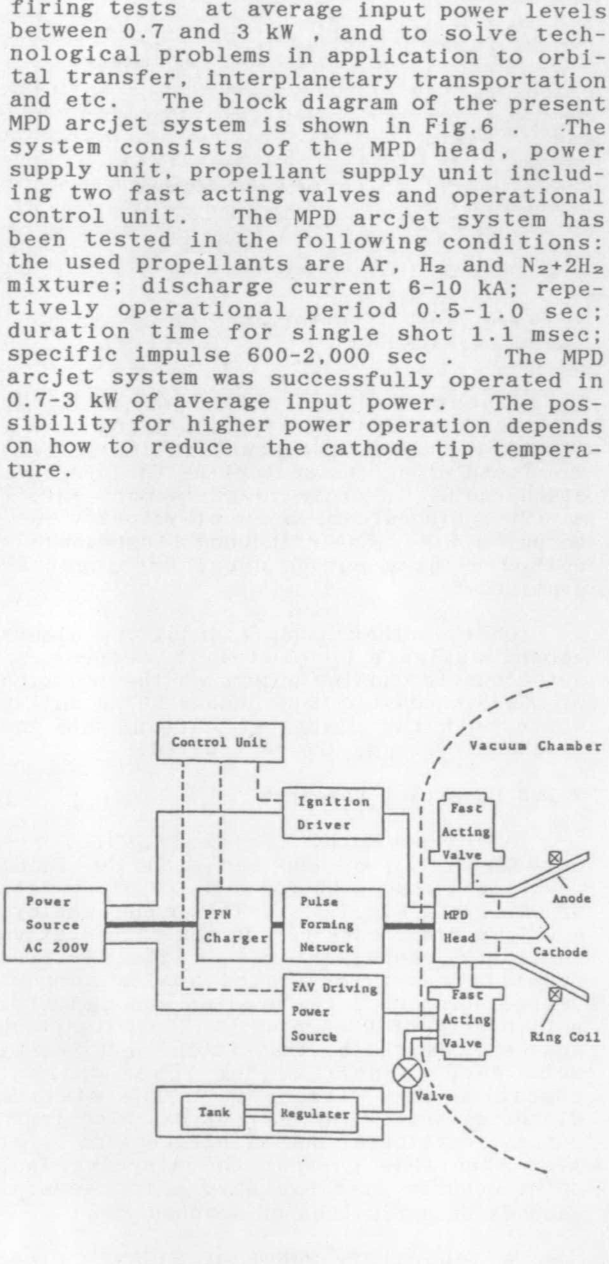


Fig.6 Block diagram of MPD arcjet system.

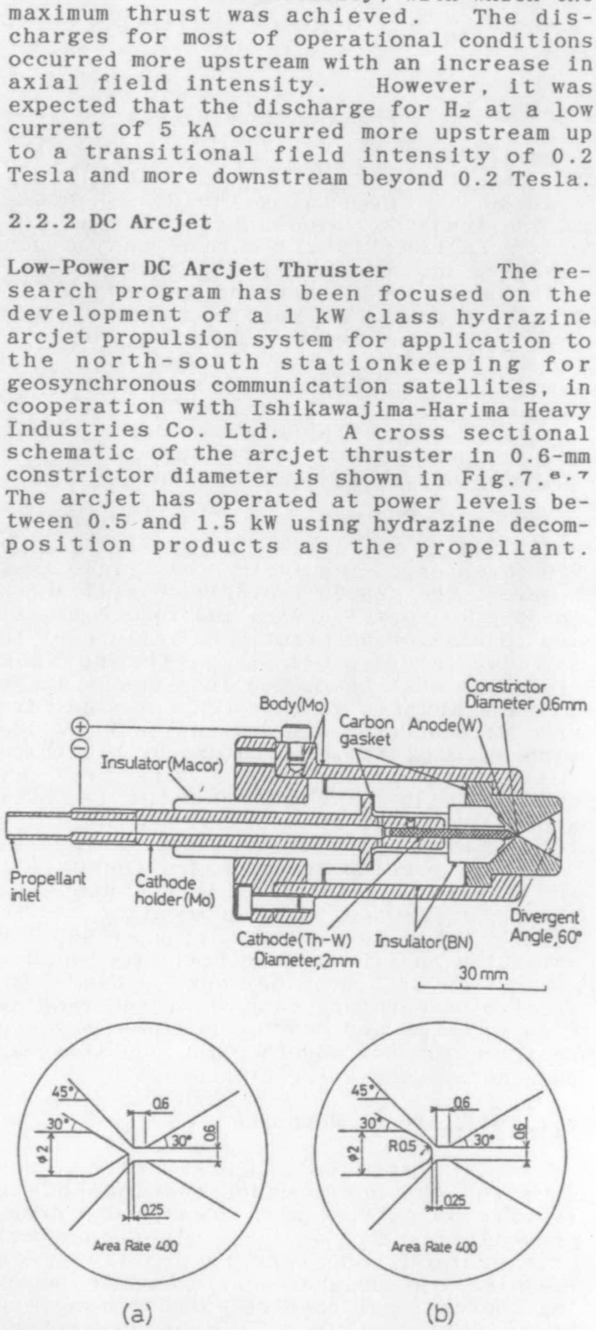


Fig.7 Cross-sectional schematic of DC arcjet thruster (RAT-V) :

Stable, reliable long-term operations under these conditions at specific impulse levels in the 400-550 second range have been demonstrated in the laboratory. A 50 kHz PWM (pulse width-modulated) power supply (20 A, 150 V dc) incorporating high-voltage pulsed starting circuits (2 kV, pulse width of 20 μ sec) has tested. Endurance tests for 50 hours and on-off cyclic tests of 1000 times have been completed. A greater amounts of erosion in the constrictor region occurred at the low voltage mode and the start-up. To shorten the unstable operational time of the low voltage mode, the following were undertaken; (1) the arcjet was operated in higher plenum chamber, that is, higher mass flow rate and higher input power. For example, the operational conditions were at 1.2 kW, 11 A and 30 mg/sec . (2) The arc-chamber pressure increased by automatically opening valves of flow meter once after the low-voltage limit was violated. In addition a two-pulse starting technique was adopted to achieve a smooth transition to steady-state operation.

Medium-Power DC Arcjet Research A medium-power (10 kW class) dc arcjet has been studied to understand the correlation between the operational characteristics and arc features in the discharge chamber.[8] The arejet was operated at discharge currents of 80-150 A with He; Ar, H_2, N_2 and mixture of N_2+2H_2 . The arc-heated flow-field was also analyzed numerically. The operational characteristics for molecular gases were found to be sensitive to constrictor diameter though for atomic gases to be insensitive. This is because their arc attachment characteristics on the anode are different. The computational flow-field analyses showed that the arc radii for Ar gradually increased downstream in the constrector and that the arcs attached to the constrictor wall though the arcs for N_2 passed through the constrictor. Consequently, it was found that the dependence of gas species on arc attachment characteristics was due to whether there was dissociation process. In addition, from emission spectroscopy, the excitation temperatures for Ar were smaller than those for N_2 in the constrictor, resulting in temperatures on the center line of 11,000-13,000 K for N_2 and of 6,000-7,500 K for Ar as shown in Fig.8 . This feature is chiefly because thermal pinch for N_2 is more effective than that for Ar.

2.2.3 Microwave-Discharge Electrothermal Thruster

Electrothermal thrusters using microwave discharge have been investigated in Osaka University since 1986.[9] The recent study is focused on stabilization of microwave discharge plasmas under high pressures and on understanding of electromagnetic field patterns in a resonant cavity with plasmas. An experimental microwave electrothermal thruster consists of a cylindrical resonant cavity and a quartz glass tube on its axis. Microwaves of maximum 1 kW (2.45 GHz) were introduced into the cavity, in which TM_{011} electromagnetic-field mode was excited. It was found that bluff bodies were effective to make plasma balls stable. Electromagnetic - field analyses in the cavity also showed that a modified TM_{011} mode was excited for variations of plasma density and collision frequency resulting in long cavity lengths and low quality factoies as shown in Fig.9 .

In addition, material degradation simulation due to atomic oxygen flow has been conducted using the microwave-heated plasmajet. The microwave plasmajet is useful as a high-velocity, clean atomic-oxygen flow source. Exposures of several materials to atomic oxygen flow are going on.

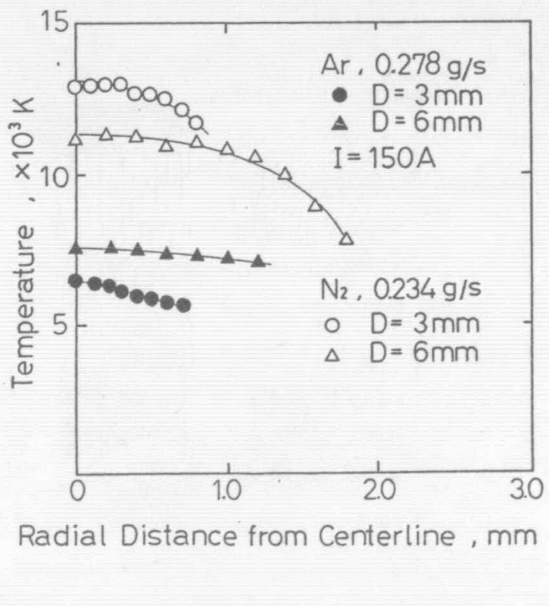


Fig.8 Radial variation of excitation temperature in constrictor.

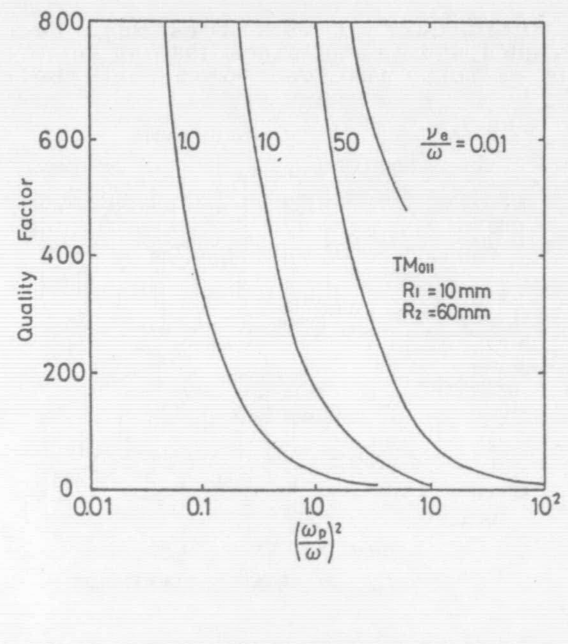


Fig.9 Quality factor vs plasma density characteristics.

2.3 University of Tokyo

2.3.1 DC arcjet

In DC arcjets, heat loss flux to electrode surface is so large to have unfavorable effects on their thruster performance. Therefore, it is important to understand the transport mechanism of heat from arc plasmas to the electrode surfaces, for the design and performance improvement of thrusters. From this point of view, experimental and analytical studies of the heat loss problem in DC arcjets have been started at University of Tokyo. A 5 kW-class arcjet, as shown in Fig.10, has axially segmented anodes, each of which is electrically and thermally isolated to one another and is separately water-cooled and connected to a separate power supply. Hence, one can measure not only heat loss and current distributions, but also operate the thruster with any current distribution to obtain higher thruster performance. The arc plasma in the electrode region is optically examined by spectroscopic measurement when quartz glass sheets are used as the insulators between the segmented anodes. The anode geometry is changeable by the selection of anode segments. The experiments conducted so far shows that there exists a close relation between the heat loss and current distribution. When discharge current is distributed in the downstream electrode region, the total heat flux loss to the anode is relatively small, resulting in higher thruster performance.

In the analytical work, a two-dimensional numerical plasma flow model in which the Navier-Stokes equation is coupled with the Maxwell's equations has been developed to help the understanding of heat loss mechanism. After the above equations are dispersed by the control volume method, the plasma flow is calculated by the time marching method while the electromagnetic field is calculated by the implicit method.

2.3.2 Hall-Current Thruster

Hall-current thrusters have been designed and tested since 1987 at University of Tokyo in order to improve their thruster performance and to investigate the plasma acceleration process.[10] The recent thruster model is shown in Fig.11 . This thruster offers much higher thrust density than conventional ion thrusters. Thrust density of 2.5 mN/cm^2 was obtained at a specific impulse of 1,400 sec in the experiment and this value is two orders of magnitude higher than that of conventional ones.

The maximum thrust efficiency could be raised from 16 % to 32 % by changing the geometric design; shortening the acceleration channel length together with arranging the magnetic field lines to be perpendicular to the axis. Plasm characteristics such as space potential, plasma density and electron temperature were measured by scanning Langmuir probes in radial and axial directions. Assuming that ions are accelerated only by electrostatic fields, one can estimate ion flux toward the channel exit and walls. In Fig.12, the domains in which the ions are extracted downstream of the exit as ion beams are illustrated. The ion loss fraction to the walls f_L was reduced from 0.7 to 0.4 by the .

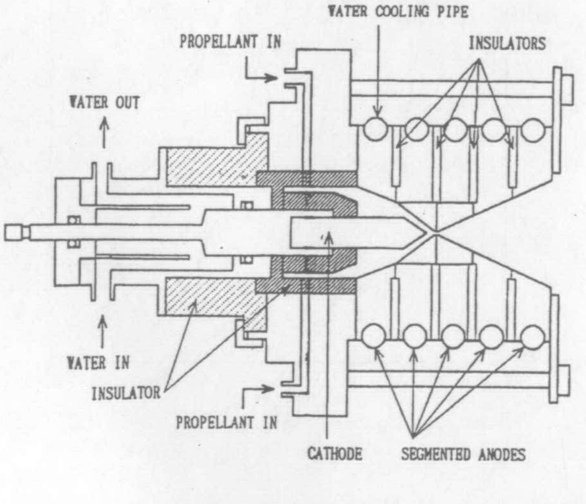


Fig.10 5 kW arcjet with segmented anodes.

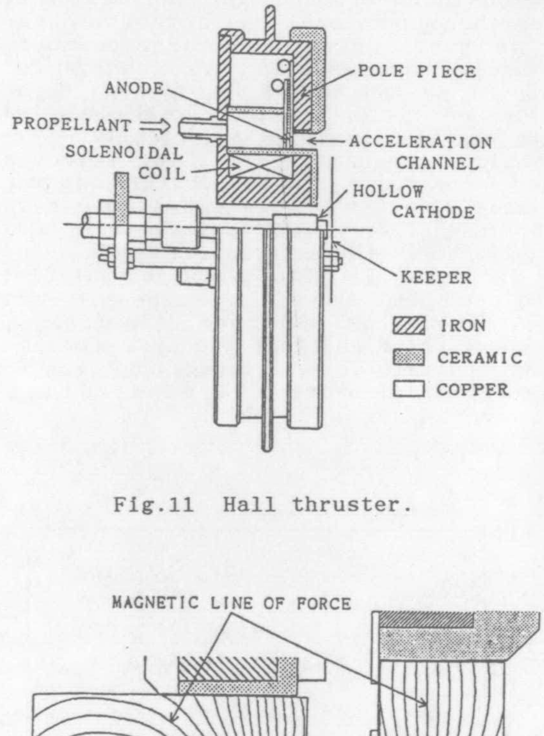


Fig.11 Hall thruster.

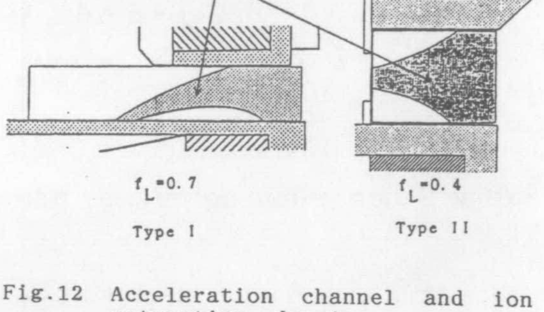


Fig.12 Acceleration channel and ion extraction domain.

channel modification, resulting in the increase in thrust efficiency.

Plasma characteristics and thruster performance in the thruster have been analyzed, using a one-dimensional plasma model in which the magnetic fields are applied perpendicular to the plasma flow. This model comprises equations of mass, energy, and electron current and is based on the assumption that ions produced by electron-neutral ionization collisions in the acceleration channel are electrostatically accelerated by an electric field, while electrons flow upstream to the anode by Bohm diffusion. The analytical results qualitatively agree with the experimental data and indicate that thrust efficiency can be raised by the reduction of ion loss fraction or by the increase of the propellant mass flow rate.

2.4 Kyushu University

In order to predict the performance characteristics of the arcjet thruster, the detail of the flow field inside the thruster is required. For this purpose, Department of Aeronautical Engineering, Kyushu University has conducted computational simulation of the thruster flow field, in cooperation with Central Research Laboratory, Mitsubishi Electric Corporation.

In a previous study[11], we applied the Euler equations to simulate the flow in the arcjet thruster, so that a boundary layer developed on the thruster nozzle wall and hence the heat transfer towards the nozzle wall were not considered. However, an actual arcjet thruster which is operated in space is radiation-cooled. Therefore, we have to consider the balance of the heat tranferred to the wall with irradiated energy from the wall. For this reason, we need to employ the Navier-Stokes equations as the governing equations. Thus, numerical similation of a low power DC arcjet thruster, which has the same geometry as used in laboratory experiments[12], has been performed using the Navier-Stokes equations. The flow equations are coupled with the electric field equations. The detail of the computation is described in this issue.[13]

As an example calculation, the computations have been done for the 1 kW arcjet thruster with 10 % efficiency. Argon has been considered as working gas. The contours of temperature are illustrated in Fig.13 , in which the maximum temperature region is shown by H .

2.5 Tohoku University

At Institute of Fluid Science (IFS), Tohoku University, a low power arcjet is fabricated mainly for the purpose of non-propulsive applications. It is used as a high-enthalpy wind tunnel which can simulate convective heat transfer or molecular kinetics under re-entry conditions. In particular, the sheath efforts on the convective heat transfer are being studied. Diagnostic measurements, i.e. the pressure and temperature distribution measurements, of the arcjet flowfield are also to be conducted.

Also, at IFS, theoretical studies on the acceleration mechanisms of an applied-field MPD thruster is being conducted in cooperation with the University of Tokyo.[14] Recently, a general formula for an applied-field MPD thruster, in which the effects of a generalized Hall acceleration, swirl acceleration and self-magnetic acceleration are taken into account, has been obtained. From this thrust formula, it is found that the generalized Hall acceleration is promising for high-I_{sp} propulsions.

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

IHI activity in the field of electric propulsion consists in developing EHT (Electro-thermal Hydrazine Thruster), DC arcjet and MPD arcjet.

2.6.1 EHT

Two flight models of EHT, as shown in Fig.14 , have been developed for NASDA's satellite ETS-VI, which will be launched by the 2nd flight of H-II in 1993 FY . The thrusters will be attached on east and west panels of the ETS-VI in order to conduct a space experiment of North-South station keeping. The thrust performance has been qualified to be 280 sec in specific impulse at 500 W . Life test of 200 hr was also executed.

2.6.2 DC arcjet

0.5 to 1 kW class DC arcjet system has been studied fundamentally in collabolation with Osaka University since 1984 and Institute of Space and Astronautical Science (ISAS) since 1989 . The studies have put emphasis on miniaturizing thruster, enhancing thrust and life performances and improving triggering performance. System study has been also conducted for future mission.

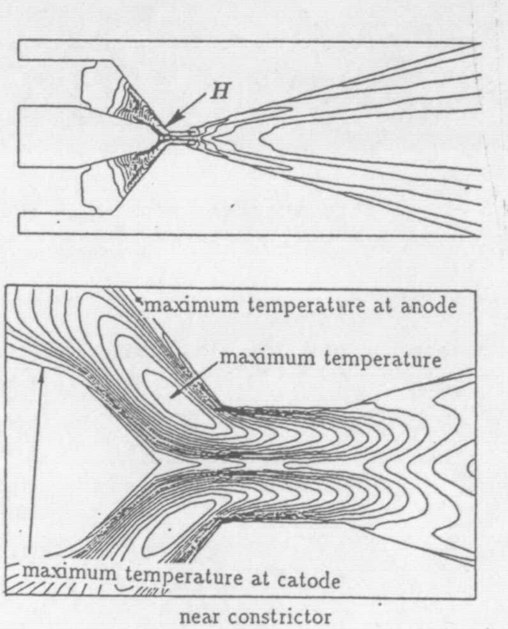


Fig.13 Temperature distribution in DC arcjet chamber.

2.6.3 MPD arcjet

IHI has been developing a proto-flight model (PFM's) of MPD thruster system for Space Flyer Unit - 1 (SFU-1) under contract with ISAS . PFM's of a thruster (HDS) and a propellant supply system (PSS) have been developed for shipping on next July. The major items of the development are EMI test, a design of hydrazine thermal control and a hydrazine dump test.[1]

The PFM of HDS was designed to get a weight saving by simplifying HDS EM, which is shown in Fig.15 . The EMI test was conducted by measuring electromagnetic noise radiated from HDS, which fired repeatedly in a vacuum chamber of Pyrex tube, and the noise level was found to be under an allowable level of SFU antenna located in the neighborhood of HDS. The thermal design has some difficulty in meeting NASA safety requirements " NSTS1700.7B" which requires a redundancy design of two fault tolerance in hydrazine thermal control system. The circuit redundancy necessarily results in a problem of overweight PSS . In the hydrazine dump test, it was confirmed that hydrazine of 500 g stored in PSS can be dumped out through a dump port and dried in a vacuum chamber.

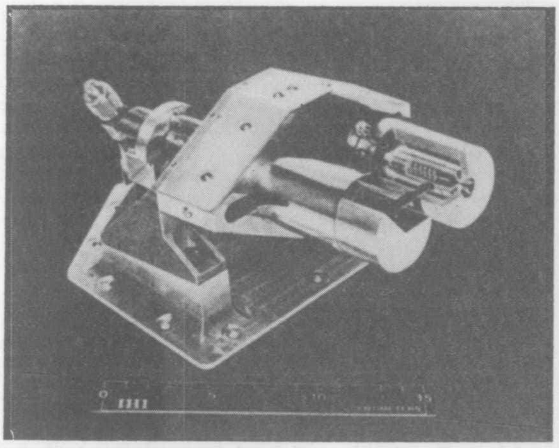


Fig.14 Electro thermal thruster (EHT).

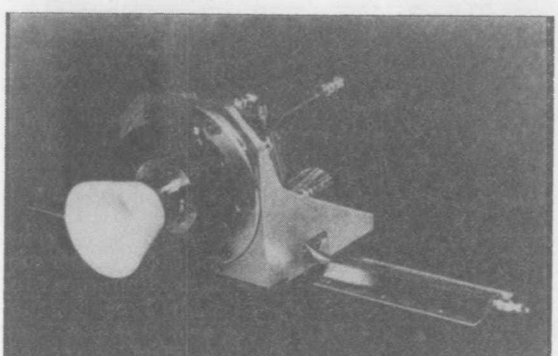


Fig.15 MPD thruster (HDC) EM.

After component AT, the MPD thruster system will be assembled on PLU (Payload Unit) panels and be submitted to a system AT carried out in ISAS .

As a next step, IHI starts a system study of 100 kW - class DC MPD arcjet with external magnetic field generated by a superconducting coil.

2.7 Mitsubishi Electric Corporation (MELCO)

MELCO has been developing the Power and Control System of MPD Arcjet for Electric Propulsion Experiment (EPEX) on the first flight of Space Flyer Unit (SFU) to be launched on the winter of 1994, under the contract of Institute of Space and Astronautical Science. The status of this program is in manufacturing of the flight hardwares.

The research of DC arcjet are going on to confirm performance prediction method by the numerical analysis.

3. Ion Propulsion

3.1 National Space Development Agency of Japan (NASDA)

NASDA has been developing a 20 mN class ion engine system (IES) for ETS-VI NSSK . The IES has four ion thrusters (TRS) . Two TRSs are clustered into a branket and installed on the east or west panel of the spacecraft as shown in Fig.16.

The thrust vector is canted by 30 deg. from the orbit normal axis so that the ion beam does not interfere with the spacecraft. Two TRSs, one of which is on the east panel and the other on the west panel, are operated at the same time in orbit. These results in the thrust of about 42 mN to the direction of orbit normal. The total operation time of 6,500 hours and on/off cycle numbers of 2,920 are required for ETS-VI IES .

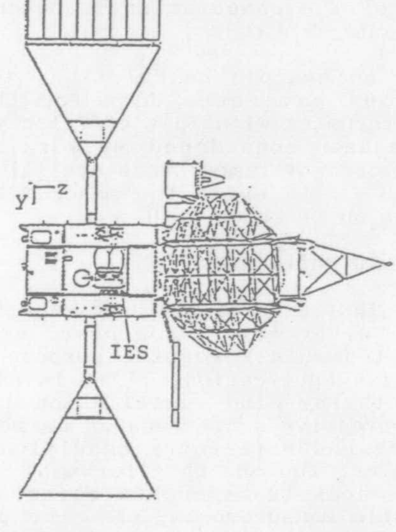


Fig.16 Location of IES on ETS-VI.

The IES components were designed in detail and the designs had been successfully verified by the test using engineering models. Performance tests including an electromagnetic compatibility test and environmental tests such as thermal vacuum, vibration, shock and acoustic tests have been conducted at the component level and the subsystem level. The results show that the IES satisfies the requirements of ETS-VI and the interface with other subsystems meets well. The lifetime of the IES, particularly the TRS, has been partially confirmed using thrusters of a bread-board model and two development models.

At present, the final performance test of prototype/protoflight models is being conducted. The lifetime test of four engineering models is also under way.

3.2 National Aerospace Laboratory (NAL)

Research activities on xenon ion thrusters for auxiliary propulsion have been continued. Most of the efforts are presently focused on performance and endurance improvement of ring-cusp ion thrusters with a normal thrust of 25 mN at the beam voltage of 1 kV . The objective is to apply them to north-south stationkeeping of future geostationary satellites, hopefully those coming after ETS-VI

On performance improvement, an experimental investigation was made to examine the effects of discharge chamber configurations on thruster performance. Various configurations were constructed by changing the chamber diameter, the magnet positions, and the hollow cathode position. Figure 17 shows the 14-cm thruster, one of the configurations tested. Test results were compared mainly in terms of ion production cost and discharge voltage. Three cases of the test results are shown in Fig.18 , giving relatively better performance in the configurations investigated. These results as well as considerations on thruster endurance indicate that the 14-cm thruster is the most appropriate of these configurations for application to 2 ton class satellites.

A cyclic test of the 14-cm thruster was conducted to obtain data on thruster endurance. A 3-hr operation followed by a 1.5-hr non-operation was repeated. The test was terminated after 613 cycles with 1960 hr of run time. Post test analyses and grid mass loss measurements were conducted. In comparison with the results of the previous 1000-hr continuous operation test[15], the amount of metal flakes deposited in the discharge chamber was reduced, and the erosion of the screen grid was negligible. These are due to lower discharge voltage achieved in the present test. The accelerator grid was eroded as much as in the previous test. This indicates that the erosion rate may be acceptable, but some improvements on this erosion are desirable. Thruster performance was not constant during the test, but fluctuated cycle by cycle. Some causes are suggested in relation with the main hollow cathode, the ion accelerating system, or the discharge power supply, but more work is necessary to determine it. Practi-

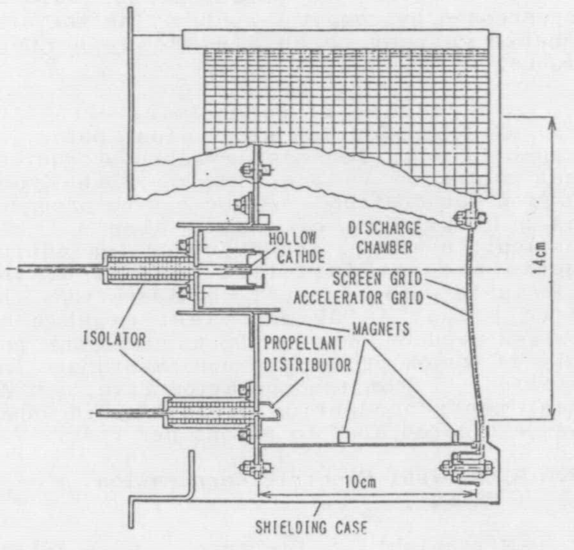
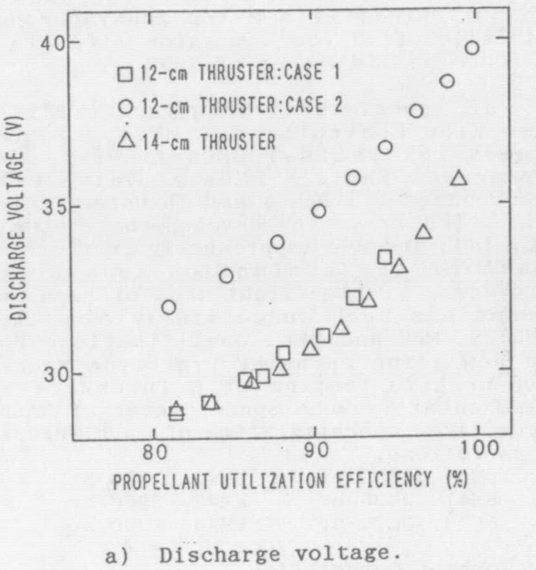
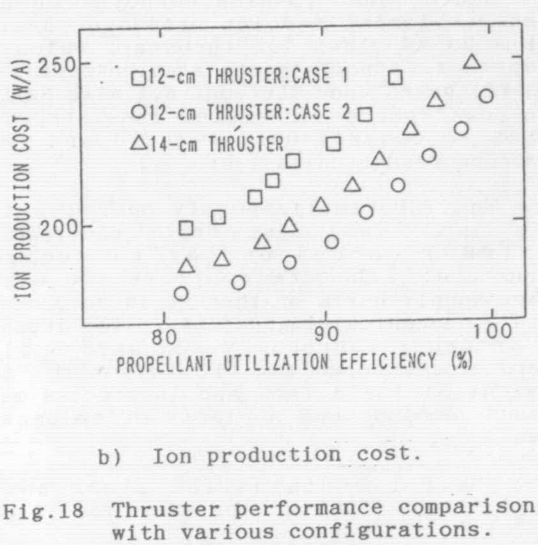


Fig.17 NAL 14-cm diameter ring-cusp xenon ion thruster



a) Discharge voltage.



b) Ion production cost.

Fig.18 Thruster performance comparison with various configurations.

cally, however, the fluctuation could be suppressed by computer-controlling the discharge current to obtain steady performance.

In advance of forthcoming extensive use of xenon for ion propulsion, potential capability in increasing xenon production was surveyed in cooperation with Kyodo Oxygen Corporation. Xenon can be produced as a by-product of oxygen plants. At present, however, xenon production equipments are not installed in most of the plants in Japan. Cost considerations indicate that an oxygen plant capable of oxygen production over 10 kNm^3/hr can pay off if xenon production equipments are installed. From the operation rate, potential xenon production capability in Japan was estimated at 7 to 9 tons per year.

3.3 Mitsubishi Electric Corporation (MELCO)

Mitsubishi Electric Corporation (MELCO) has been developing Ion Engine System (IES) for Engineering Test Satellite - VI (ETS-VI) under the contract of National Space Development Agency of Japan (NASDA) . IES is used as the auxiliary propulsion system for NSSK of ETS-VI, which is a 2-ton geosynchronous satellite of 10 years mission life, to be launched on the summer of 1993.

IES consists of Thrusters (TRSs), Mass Flow Controllers (MFCs), Power Processing Units (PPUs), Propellant Management Units (PMUs), Valve Drive Electronics (IVDEs) and Thruster Control Unit (TCU). The development status of TCU, IVDE and PMU is presently in the final phase of the evaluation as a flight hardware. Proto-flight test of each component has been successfully completed. For TRS, MFC and PPU, Qualification Tests are now going on using Prototype Models. 9500 hr Life Testing of 6 Thrusters are going on at Tsukuba Space Center of NASDA. Accumulated operating time of each Thruster is as follows.

DM#1	5660 hr	DM#2	7160 hr
EM#1	2630 hr	EM#2	2680 hr
EM#3	3070 hr	EM#4	1960 hr

3.4 Toshiba Corporation

Since 1983, Toshiba Corporation has been developing a power processor system for a 20 mN class ion thruster, which is adopted to a north-south stationkeeping of ETS-VI, based upon the contract with NASDA. The power processor system consists of a thruster control unit (TCU) and four power processor units (PPU) .

The TCU simultaneously operates two PPUs and two valve drive electronics (IVDE) corresponding to two ion thrusters. The TCU controls the eight power supply units of the PPU in accordance with a sequential logic flow. The TCU has an internal redundancy and uses 8-bits micro processor as the CPU to control the sequential logic flow and to process magnitude commands and a plenty of telemetry data.

The PPU contains the eight power supply units for the thruster operation. The high voltage power for the screen grid and accelerator grid are generated by means of a switching regulator which consists of boost choppers regulator, a current resonant inverter and a voltage multiplying rectifier. The power supply units achieve the power efficiency over 85 % .

The critical design review of the power processor system was completed on February of 1990. And flight hardwares of TCU (1 PFM) and PPU (1 PM, 1 PFM and 3 FM) are under fabrication. After fabrication and component test, combination test of TCU with PPU and combination test of TCU with IVDE and propellant management unit (PMU) are performed.

In application to the primary propulsion system for an orbit transfer vehicle, it is required to develop a high thrust ion engine. A cooperative project between NAL and Toshiba Corporation has begun to develop a 30-cm diameter ring-cusp xenon ion engine with a thrust of 150 mN and a specific impulse of 3,500 sec at an exhaust beam energy of 1 keV.

The laboratory model of mark 1 using accelerator grids made of stainless steel with convex curvature functioned successfully. Performance tests showed an ion beam production cost of 178 eV/ion at the propellant utilization efficiency of 90 %.[18]

The engine of mark 2 shown in Fig. 19 is the same as the engine of mark 1 with the exception of the accelerator grids. The grids are made of molybdenum with the concave curvatures for the purpose of continuous operations. The performance tests showed the ion beam production cost of 188 eV/ion at the propellant utilization efficiency of 90 % .

3.5 University of Tokyo

A computer code for cusp ion thrusters has been developed in order to not only calculate discharge chamber performance but also to provide its design data. This code can be used to calculate both ion production cost and propellant utilization for a cylindrical discharge chamber having any axisymmetric magnetic field configuration. It contains four calculation sections; namely ones that are used to 1)

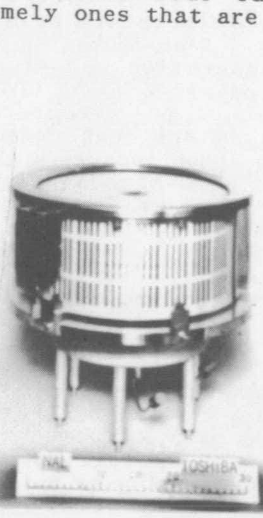


Fig.19 Primary ion engine of mark 2.

to compute primary electron confinement length, 2) to compute a fraction of ions produced that are extracted as an ion beam, 3) to compute the transparency for ions to the grids, and 4) to compute plasma properties such as ion density, neutral atom density, and electron temperature. In the first section, a charged particle kinetic model using the Monte Carlo method[17] was employed to compute orbits of primary electrons in the discharge chamber and then to count their average confinement length before they escape to anodes. In the second, a plasma flow model in which ion flow rates in directions across and along magnetic field lines are assumed to be described by Bohm and ambipolar diffusion equations, respectively, is used.[18] Applying the Finite Element Method to the model, one can calculate ion density distribution, ion flow rates to the surrounding walls and the grids, and then obtain the ion beam fraction. In the third, a two-dimensional optics code is used to compute ion beam trajectries and then to obtain the transparency for ions to the grids. In the last section, combining these with basic conservation equations of mass, charge, and energy, one can compute the plasma properties and then calculate the discharge performance. Details of the model and calculation procedure are presented in this conference.

3.6 Institute of Space and Astronautical Science (ISAS)

The Institute of Space and Astronautical Science (ISAS) has started to develop the microwave ion thruster, YOSHINO series.[19] The off-resonance discharge microwave ion thruster YOSHINO-I, which has a microwave cavity, was characterized as low thruster efficiency and short life time. The quartz glass in the microwave cavity is an important component to transpire the microwave and to wall-stabilize the plasma. The performance and the life time of the off-resonance ion thruster are limited by the quartz glass, which is contaminated with the sputtered materials. In order to breakthrough the above defects, the microwave ion thruster YOSHINO-II, which produces the plasma by means of the electron cyclotron resonance (ECR) discharge, was designed.[20] The 5.9 GHz microwave power launched into the discharge chamber propagates to the surface of the permanent magnets, where the high energy electrons are produced by ECR. The plasma confinement by the ring cusp magnetic field removes the quartz glass from the discharge chamber so that the limitation of thruster life-time is resolved completely. The xenon beam extraction test demonstrated the ion production cost 700 eV/ion when the propellant utilization efficiency is 90 % as shown in Fig.20.

3.7 Osaka University

Microwave ion sources have many attractive characteristics as follows: (1) no electrode, i.e., no erosion; (2) easy electrical breakdown of gases; (3) no warming-up; (3) simple configuration and system. Microwave ion sources have been studied to develop a high-performance ion thruster.[9,20] An experimental ion thruster with conventional off-resonance microwave discharges consists of a cylindrical resonant cavity and a thin discharge chamber located in its flont plate. Microwave of 2.45 GHz was introduced into the resonant cavity, in which TM_{011} electromagnetic-field mode was excited. Electric field patterns measured on the inner wall of the resonant cavity showed that a modified TM_{011} mode was expected to be excited in the resonant cavity near the theoretical resonant cavity length with high coupling efficiencies over 90 % . In ion beam extraction, the lowest ion production cost of 836 W/A and the highest propellant utilization of 53.1 % were achieved. At the second stage, we newly design a microwave ion thruster using ECR (Electron Cyclotron Resonance) plasmas or Whistler-wave heated plasmas , as shown in Fig.21 , and start parametric studies on discharge chamber lengths, magnetic field configurations and cavity resonance modes to obtain fundamental operational characteristics.

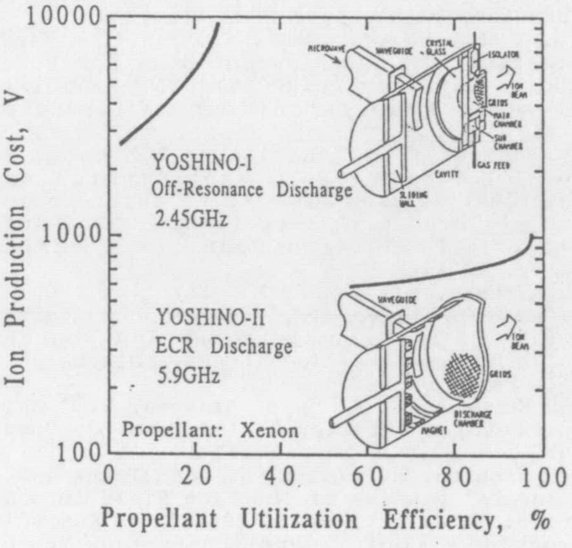


Fig.20 Performance of microwave ion thruster " YOSHINO " series.

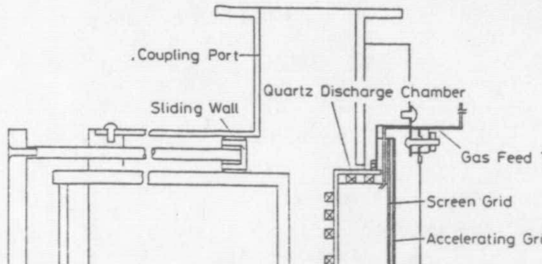


Fig.21 Cross section of ECR - discharge ion thruster with resonant cavity.

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