# Research and Development of Electric Propulsion Thrusters in Japan

By
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**Abstract** : Cathode-less ion engines are on the Hayabusa asteroid explorer, and Kaufman-type engines are on the Engineering Test Satellite-VIII at present. A 5kW-class Hall thruster and a 200mN-class ion engine are under development. PPTs and laser micro-thrusters are prepared for the propulsion system of microsatellites. This paper reports the recent activities on electric propulsion conducted in Japanese universities, industries, and JAXA.

**Key words** : Electric Propulsion, Ion Engine, Hall Thruster, PPT, MPD, Micro Thruster

## 1. Space Missions

### 1.1 Hayabusa Asteroid Explorer (ISAS/JAXA)[1,2]

The Hayabusa space mission is focused on demonstrating the technology needed for a sample return from an asteroid and was launched in May 2003. The cathode-less electron cyclotron resonance ion engines propelled the Hayabusa asteroid explorer. Depending on the solar distance the ion engines were operated between 250W and 1.1kW in electrical power. It is identified that a single thruster generates the swirl torque 2x10-6Nm. It succeeded in rendezvousing with the asteroid Itokawa in September 2005 after a 2-year flight, producing a delta-V of 1,400 m/s, while consuming 22kg of xenon propellant and operating for 25,800 hours. The total accumulated operational time of the ion engines reaches 28,000hours at the end of May 2007. One of four thrusters, which has been most frequently used, reaches 12,000hours in space operation.

### 1.2 Engineering Test Satellite-VIII (JAXA)

The Engineering Test Satellite-VIII (ETS-VIII) launched in December 2006. It is a three-ton class geostationary satellite utilizing four Kaufman-type ion engines for North-South Station Keeping(NSSK). The major engine specifications are shown in Table 1. Although its thruster design was similar to those on ETS-VI launched in 1994 or COMETS launched in 1998, it was modified to extended its lifetime from 8,000 hours to 16,000 hours. The engine system was checked out in orbit in January 2007. After then, normal NSSK operations have been carried out almost every day since March 2007. The operation time is about six hours around the ascending node and descending node.

**Table 1 Ion engine Specifications.**

| | |
|---|---|
| Average thrust | >20mN |
| Average Isp | >2,200sec |
| Weight | 96kg |
| Operation time | 16,000hours |
| Total impulse | $>1.15\times10^6$Ns |
| Number of firing | 3,000cycles |
| Power consumption | <880W |
| Grid diameter | 12cm |

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## 2. Flight Model Developments

### 2.1 MELCO Hall Thruster[3]

A hall thruster and a power-processing unit (PPU) have been developed since 2003 as a national project under the contract of Institute for Unmanned Space Experiment Free Flyer (USEF) sponsored by Ministry of Economy, Trade and Industry (METI). The target specification is as follows: the thrust level is over 250mN, the specific impulse is over 1,500s under the PPU power consumption of below 5kW, and the lifetime is over 3,000hours. Mitsubishi Electric Corporation designed, fabricated and evaluated the thruster EM (Fig.1) and PPU EM.

**Fig.1 200mN class hall thruster EM**

The thruster EM showed the thrust level of 264mN, the specific impulse of 1,720s under the thruster input power of 4.56kW. The thrust ranges from 87mN to 293mN, and the specific impulse does from 1070sec to 1910sec under the input power range from 1.49kW to 5.46kW. The PPU EM consists of main power conditioners (PCs) such as anode PC, hollow cathode keeper PC, two electromagnetic coil PCs, and primary bus interface. It is designed for 100V regulated satellite bus. It showed over the total efficiency 93% at the output power range from 1.75kW to 4.5kW and at the anode voltage range from 250V to 350V.

### 2.2 Next-Generation Ion Engine (IAT, JAXA)[4-6]

The next-generation ion engine shown in Fig.2 performs very high performance over the wide thrust range from 80 to 200mN. The most important feature of the thruster is its large-sized ion extraction system, whose diameter is 35 cm. Current efforts are directed to evaluation of reliability and durability of the thruster. A life test of a main hollow cathode was started in March 2006. The high electron-emission capability of over 20 A and the anti-erosion graphite materials are its key features. In this test, the cathode was operated in a dummy discharge chamber whose geometrical and magnetic configuration were almost the same as the actual thruster. Except for some troubles in peripherals, the cathode operation itself has been very stable and favorable. Accumulated cathode-operation time reached 8800 hours in the mid of May 2007 and the operation will be continued until fatal troubles occur. In order to estimate the grid life by numerical simulations, beamlet formation should be described precisely. Ion-impingement current on an accelerator grid was measured using a specially made gridlet. The gridlet was composed of both barrel and downstream electrodes electrically isolated from each other. The results will provide reference data to verify and improve a numerical model.

**Fig.2 BBM ion engine.**

### 2.3 Space debris removal systems using electrodynamic tether (EDT) propulsion (IAT/JAXA)[7]

EDT can be a totally propellant-free propulsion system if a bare tether is used as an electron collector and a field emission cathode as an electron emitter. We have been developing some components of the EDT system such as bare tethers, reel mechanisms, field emission cathodes using carbon nanotubes, and others aiming at demonstrating

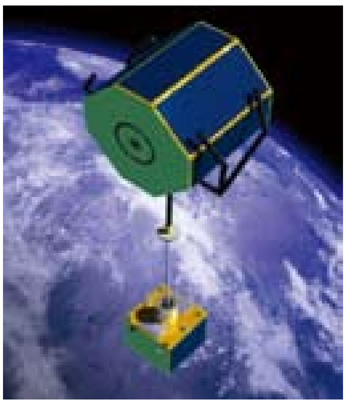


**Fig.3 Image of a debris removal system with EDT propulsion.**

on-orbit operation. Figure 3 shows an image of a debris removal system in near future.

### 2.4 Pulsed Plasma Thruster for Small Satellites (Tokyo Metropolitan University)[8,9]

Pulsed Plasma Thrusters have been developed for the application to 50kg-class small satellites. Following the R&D of a parallel electrode (rectangular) PPT, TMU has begun to study a co-axial PPT. Owing to its electromagnetic acceleration, a parallel electrode PPT shows the high specific impulse and very small impulse-bit, which would be advantageous for the attitude control and precise orbit control of satellites. On the other hand, a co-axial PPT, in which electrothermal acceleration is dominant, shows large impulse-bit (nearly 1mN) and high thrust efficiency. It is favorable for the station-keeping and de-orbit. Wide performance range (20$\mu$N-s impulse bit, 2,000s Isp to 1mN-s impulse bit, 14% thrust efficiency) has been achieved as shown in Fig.4.

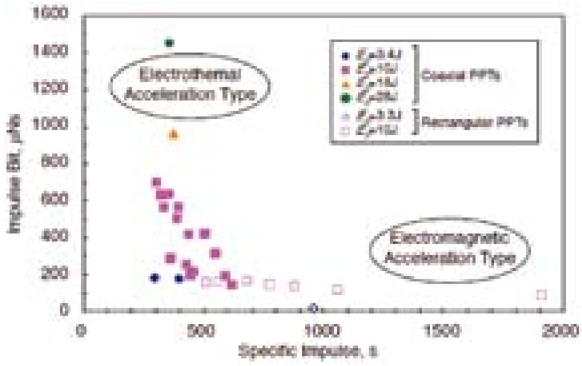



**Fig.4 Performance summary of electromagnetic / electrothermal type of PPT.**

### 2.5 Microwave Discharge Ion Thrusters (ISAS/ JAXA)[10,11]

In order to advance the technology of cathode-less microwave discharge ion engines known as the "$\mu$" family, two programs are being carried out: $\mu$20 and $\mu$10HIsp (Fig. 5). The former is a 20-cm diameter engine, and the latter is a higher specific impulse version of the 10-cm diameter $\mu$10. Table 2 summarizes the achieved and the target performance of the models.

**Table 2 Performance of "μ" series ion thrusters.**

| Items | $\mu$10 achieved | $\mu$20 target | $\mu$10HIsp target |
|---|---|---|---|
| Ion Prod. Cost, W/A | 230 | 200 | 230 |
| Beam Current, mA | 140 | 500 | 140 |
| Microwave Power, W | 32 | 100 | 32 |
| Beam Voltage, V | 1,500 | 1,200 | 15,000 |
| Specific Imp., sec | 3,000 | 2,800 | 10,000 |
| Thrust, mN | 8.5 | 27 | 27 |
| System Power, W | 350 | 900 | 2,500 |
| Thrust/Power, mN/kW | 22 | 30 | 11 |

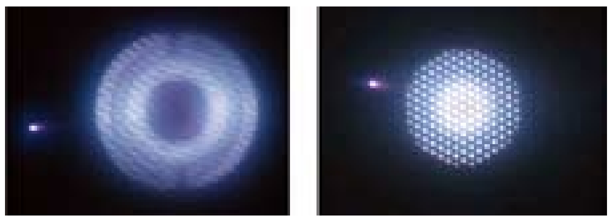


**Fig.5 Front view of μ20 and μ10HIsp.**

The performance of $\mu$20 was improved from the $\mu$10's level of 22mN/kW to 30mN/kW by the combination of a thin screen grid and an optimized gas injector & magnet layout. Aperture diameter of an accelerator grid was designed for improvement of propellant utilization efficiency. Evaluation of this new optics and the 1000-hour wear test of $\mu$20 system are planned in this summer.

The potential application of the $\mu$10HIsp is a solar-electric sail mission to Jupiter, which requires solar electric propulsion system with Isp of 10,000sec achievable using beam voltage of 15kV. Operation of the $\mu$10HIsp was demonstrated in the laboratory using high voltage insulation techniques such as a DC block, gas isolator and supporting elements. Although Isp of over 9,000sec was achieved, it has not reached 10,000sec yet because of relatively large drain current of 20mA to an accel grid. It is thought that the non-uniform ion current distribution causes direct impingement of ions to the accel grid due to cross over limit. By suppressing the impingement, 10,500sec Isp, 29mN thrust, 11.2mN/kW thrust power ratio and 58%

total efficiency would be expectable. A power processing unit for the $\mu$10HIsp is under development in corporation with NTSpace.

### 2.6 Project of Osaka Institute of Technology Electric-Rocket-Engine onboard Small Space Ship

The Project of OIT Electric-Rocket-Engine onboard Small Space Ship (POERESSS) was started. A 10-kg small satellite with electrothermal PPTs, named JOSHO, will be launched in 2010. An orbit raising and attitude control of JOSHO in LEO will be carried out by the PPTs. Their endurance test is under way.

An electrothermal PPT with a side-fed propellant feeding mechanism was tested. Initial thrust-to-power ratio of 43-48$\mu$Ns/J ( $\mu$N/W), Isp of 470-500sec and thrust efficiency of 11-12% with stored energy of 4.5-14.6J per shot, and a total impulse of 3.6Ns were obtained in a repetitive 10000-shot operation with a stored energy of 8.8J per shot.

### 2.7 Diode Laser Microthruster (The University of Tokyo)[12-14]

Two types of diode laser microthrusters with different propellants have been developed for 10cm class microsatellites. One provides total impulse of 30Ns within the propellant volume of $5 \times 5 \times 6\text{cm}^3$ with a single shot impulse of 650mNs. This thruster will enable a microspacecraft to orbit around mother spacecraft quickly. The other is a thruster of $3 \times 3 \times 2\text{cm}^3$, including a diode laser, optics, and micro-motor designed for reaction-wheel un-loading of a 1kg cube-sat developed by Tokyo Metropolitan College of Aeronautical Engineering, Japan. Its launch is scheduled in summer 2008 as a piggyback satellite of H-IIA rocket. (Fig. 6)

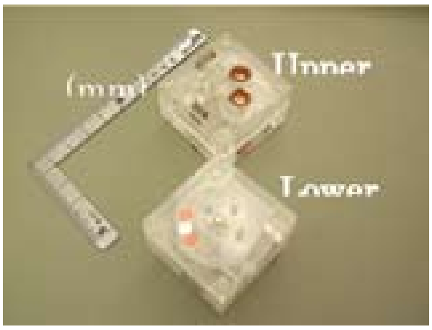


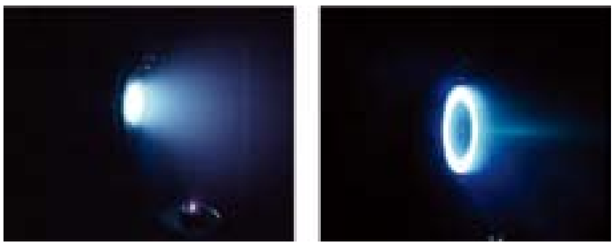

**Fig.6 EM of the laser ignition microthruster.**

## 3 Basic Researches

### 3.1 Osaka Institute of Technology

#### 3.1.1 Hall Thrusters

The cylindrical Hall thruster (CHT) is an attractive approach to achieve long lifetime especially in low power applications. Because of the larger volume-to-surface ratio than conventional coaxial Hall thrusters, CHTs are characterized by a reduced heating rate of thruster components and by a lower erosion rate of the channel. The 5.6-cm-diameter thruster TCHT-3B was operated, as shown in Fig.7(a), in the power range of 30-150W to examine the effects of magnetic field configuration. By adjusting the axial position where a large magnetic field is applied in the channel, TCHT-3B achieved a high efficiency of 18-39%.



**(a) TCHT-3B** **(b) TALT-2**

**Fig.7 Photographs of Hall thruster operations.**

The effects of magnetic field topography and discharge channel structure on performance were experimentally examined using a 1-kW-class anode-layer Hall thruster TALT-2 as shown in Fig.7(b). The performance enhancement using a divergent-type hollow anode was confirmed under the optimum channel length and various magnetic field topographies created by magnetic shields and a radial trim coil. As a result, thrust efficiency was enhanced to 57% with the divergent-type hollow anode at a discharge voltage of 400 V and a xenon mass flow rate of 3.0mg/s.

### 3.1.2 Electrothermal PPT with multiple cavities

An electrothermal PPT with multiple cavities (discharge rooms) was proposed. It has an igniter and holes for inducing discharges in all cavities. This mechanism was used as a substitute for the propellant feeding mechanism to use a large amount of propellant. A PPT with two cavities showed an initial impulse bit per stored energy of 75$\mu$Ns/J with a stored energy of 14.6J and a lower decreasing rate of impulse bit in a repetitive operation than a conventional electrothermal PPT. Furthermore, a PPT with three cavities was successfully operated as shown in Fig.8. This inducing mechanism is applicable to a PPT with larger number of cavities for a higher total impulse. Unsteady phenomena, such as discharge in the circuit, heat transfer to the PTFE, heat conduction inside the PTFE, ablation from the PTFE surface and plasma flow were simulated numerically. The result showed the existence of considerable amount of ablation delaying to the discharge. However, it was also shown that this phenomenon should not be regarded as the late time ablation (LTA) for electrothermal PPTs because neutral gas ablated delaying to the discharge generates most of total pressure and impulse bit.

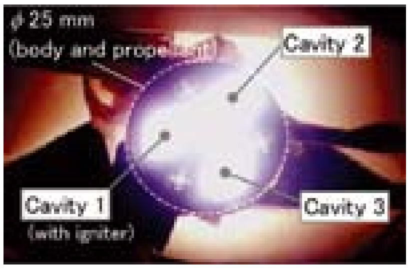



**Fig.8 Discharge in a multi-cavity electrothermal PPT.**

### 3.2 Kyushu University

Miniature microwave discharge ion engines (Fig.9) are candidates for use as miniature propulsion systems. However, their thrust performance has been far inferior to conventional ion thrusters. For understanding the mechanism of plasma production and loss, internal plasma structure of this was measured and numerically analyzed. These results showed that there is an optimum magnitude of magnetic field due to the tradeoff between a magnetic confinement and a microwave-plasma coupling.

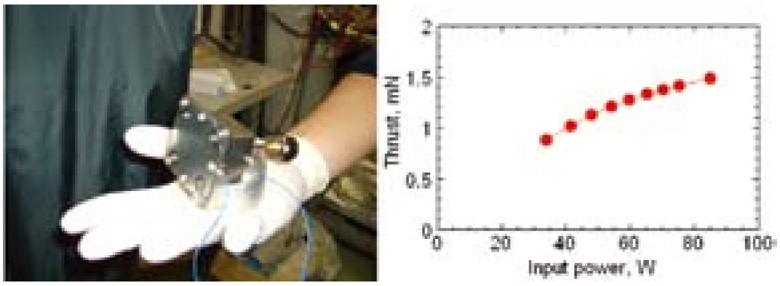



**Fig.9 Miniature microwave discharge ion engine and its performance.**

### 3.3 Tokyo University of Agriculture and Technology[15,16]

An electrodeless MPD thruster using a compact helicon source with electromagnetic plasma acceleration has been studied for applications to future electric propulsions. They already confirmed the helicon plasma of 1013cm-3 inside a glass tube (Fig.10) and examined the principle of electrode-less electromagnetic acceleration by RF antennae.

For future evolutionary propulsions, TUAT also has started a study of propellant-less propulsion using photon pressure as the motive force. A precision thrust stand discriminating 0.1$\mu$N is being developed so that a 300W tungsten filament (as a quasi-black body radiation) with a parabolic mirror demonstrates 1$\mu$N assuming 100% conversion efficiency directly from electrical to thrust power.

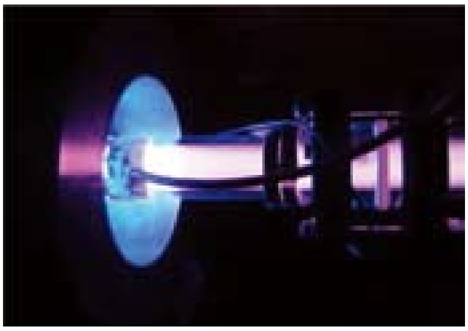


**Fig.10 Compact helicon plasma in blue-mode in a 2.5cm i.d glass tube.**

### 3.4 Kyoto University[17-20]

The microplasma source was made of a dielectric tube 10mm long and 1.5mm in diameter, producing plasmas in the pressure range from 5 to 50kPa. Optical emission spectroscopy and Langmuir probe measurement showed that the emission intensity, electron density, and rotational temperature of mixed $N_2$ increased with increasing microwave power (Pin=1 − 10W), frequency (f = 2, 4GHz), and dielectric constant ( $\varepsilon$ d ≈3.8, 6, 12 − 25) of the tube. The rotational temperature was found to increase toward the micronozzle to achieve high thrust performance. At Pin=2 − 10W with an Ar gas flow rate of 50sccm, the electron density and rotational temperature obtained were ne ≈ $10^{12}$ − $10^{14}$cm − 3 and Trot ≈700 − 1800K, respectively. These were consistent with the results of numerical analysis.

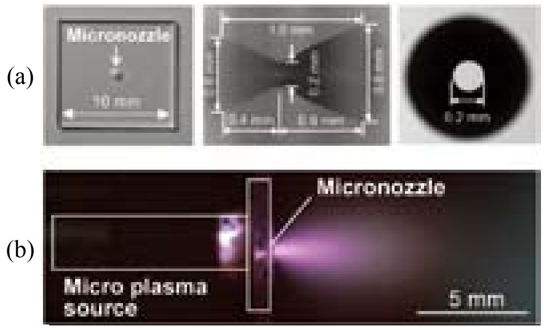



**Fig.11　(a) Microscope image of the micronozzle, and (b) supersonic Ar plasma jet plume into vacuum (f=4 GHz, $P_{in}$=5W).**

The micronozzle was fabricated in a 1-mm-thick quartz plate, where the nozzle had an inlet, throat, and exit diameter of 0.6, 0.2, and 0.8mm, respectively, as shown in Fig.11(a). Plasma discharges gave an elongated plume of supersonic plasma jet, as shown in Fig.11(b), downstream of the nozzle exit into vacuum.

### 3.5 Shizuoka University[21-25]

#### 3.5.1 Studies on Electrodynamic Tethers

Shizuoka University is supporting the IAT/JAXA electrodynamic tether deorbit system with analytical and experimental studies. The hybrid tether system in which the electrodynamic tether is used as the orbit restoration system of rotational momentum tether orbit transfer system, is also studied. The results of the mission analysis at present show that the EDT-momentum tether hybrid system can reduce the total system mass by almost 30% compared with an ion thruster-momentum tether combination for the same missions. We are also studying about the new interplanetary transportation system using the electrodynamic tether with a magnetic coil (we call this system "Mag-Tether") and showed that this system was more effective for the travel beyond the Jupiter than a mag-sail in the initial study.

#### 3.5.2 Interplanetary propulsion system using stardust as propellant

For the travel to outer planets like Jupiter and Saturn, the total mass of the transportation system can be extraordinarily reduced if we can use the interplanetary materials as propellant. We pay attention to the stardust in space as an interplanetary material. The stardust abundantly exists in the asteroid belt and is easier to be taken than gases in space. To use the stardust as propellant, it is necessary to accelerate effectively solid micro-

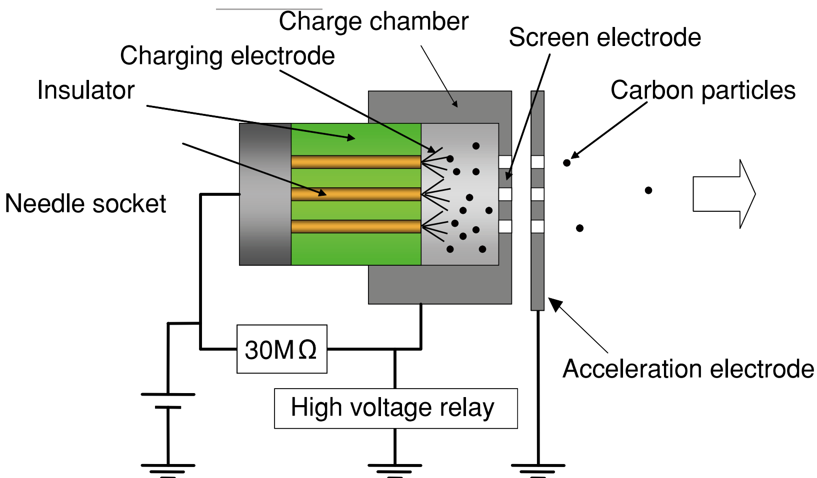



**Fig.12　Conceptual figure of stardust accelerator.**

particles.

We apply the contact charging and the electrostatic acceleration to charge and accelerate solid micro-particles. The experimental study of stardust accelerator (Fig.12) is now undergoing to improve its charging and accelerating performance of the solid micro-particles, and to find the effective way to supply the solid micro-particles.

### 3.6 Tokyo Metropolitan University[27-29]

Evaluation on hydrogen peroxide and another propellant reaction process with discharge plasma without catalyst is being carrying on. The objective of this study is to estimate the effects of discharge plasma to the chemical reaction process for the catalyst-less green propellant RCS thruster.

Evaluations of an electrodynamic tether system using a hollow cathode, another electron emission devices, and bare tether, have been carrying on. Understanding the contacting process with simulated space plasma, it enables various applications of electrodynamic tether system such as orbit raising, de-orbit, station keeping and power generator.

Fundamental studies on hollow cathode discharge phenomena and electron bombardment ion thruster performance improvement are also keeping on. RF ion thruster with RF neutralizer is also evaluated.

### 3.7 National Defense Academy

A visualized ion thruster (VIT) was designed and fabricated for evaluation and validation the numerical analysis models, and for the fundamental/educational understanding of an ion thruster. The shape of VIT is two-dimensional rectangular parallelepiped, and the plasma is produced by direct current discharge. The electron produced within an electron source is emitted to the discharge chamber through the keeper bridge plasma. Xenon propellant is ionized by electron bombardment and extracted by a grid system. It was confirmed that the plasma sheath formed near the grid slit was convex against the grid when the applied screen grid potential was low, and was concave when the potential was high. In addition, it was also confirmed that the electrical connection to the anode/anodes changed the discharge path and influenced the thruster performance.

### 3.8 Tokai University[30-32]

We have been conducting microfabrication of micro-arcjets or -plasmajets with ultra-violet lasers, and development rectangular DC micro-arcjets of various sizes operated under 5W. The micro-arcjet nozzle was machined in a 1.2mm thick quartz plate. For an anode, a thin Au film ( ~100nm thick) was coated in vacuum on a divergent part of the nozzle. As for a cathode, an Au film was also coated on inner wall surface. In operational tests, a stable discharge was observed for mass flow of 0.4 mg/sec, input power of 4W. In addition, microfabrication of the micro array-nozzle (Fig.13) with UV lasers was conducted.

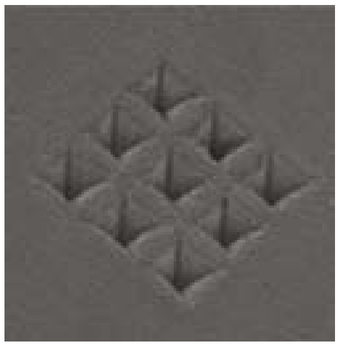


**Fig.13 SEM image of a micro-arcjet array: nozzle element height 500μm, spacing 100μm.**

Moreover, its preliminary thrust characteristics were compared with the single-nozzle as shown in Fig.12. Significant increases of the thrust and Isp with mass flow can be seen in the array-nozzle case. To elucidate influence of the interaction of exhaust multi-jets on internal flow of the micro array-nozzle, numerical simulation was conducted using a DSMC code. As a result, pressure at the nozzle exit increases through the interaction of exhaust-jet boundaries. This effect must increase static pressure of boundary layers in internal nozzle flows. The boundary layer thickness can be reduced by increasing pressure. Mach numbers drop between the jet-boundaries, and then the exhaustjets are not expanded at the nozzle exit, or rather confined. These effects will reduce losses derived from viscous losses of internal nozzle flow and under-expanding flow of exhaust jet.

### 3.9 The University of Tokyo[33-38]

#### 3.9.1 Hall Thruster

Sheath structures in an anode-layer thruster was computed using the fully kinetic 2D3V Particle-in-Cell and the DSMC methods. The ion production current in an anode hollow is found to decrease with magnetic flux density: At the low magnetic flux density, ion production current in the anode hollow is high and an ion sheath was created on the anode surface, contributing to the stable discharge.

A two-dimensional dual-pendulum thrust stand (Fig.14) had been developed to measure a thrust vector of a thruster with steering mechanism. Its measurement errors were less than 0.25mN (1.4%) in the main thrust direction and 0.09mN (1.4%) in its transversal direction. The steering angle of thrust vector of $\pm 2.3$deg was successfully measured with the error of $\pm 0.2$deg.

Xe density profiles in a plume were measured using laser absorption spectroscopy for the development of a plume shield. The maximum total number density was $3.9 \times 10^{19} m^{-3}$ at the channel exit. Then, the number density decreased by one order at 200mm away from the exit.

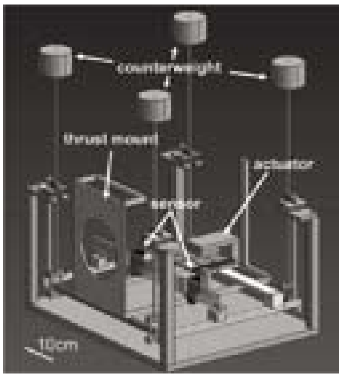



**Fig.14 2Ddual-pendulum thrust stand.**

#### 3.9.2 Pulsed Plasma Thruster

Liquid propellant is fed using a pulsed injector. A spark plug initiator is synchronized with the liquid injection. The single shot impulse was measured using a thrust stand with the resolution of 1.0$\mu$N. As a result, Isp of 3,000sec was accomplished by throttling the propellant down to 3$\mu$g.

In an ablative PPT, the acceleration processes were observed by high speed photography (Fig.15). Monochromatic images showed that high density, ablated neutral gas stayed near the propellant surface, and only a fraction of the neutrals was converted into plasma and electromagnetically accelerated, leaving the residual neutrals behind.

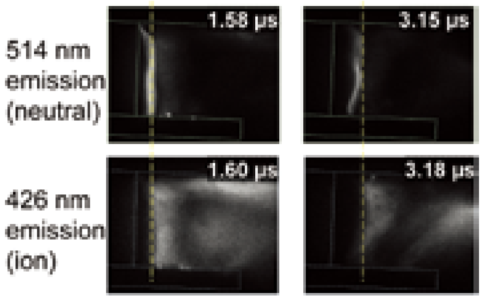



**Fig.15 High speed photo of ablative PPT (ion and neutral emissions).**

### 3.10 Tohoku University

A high power magneto-plasma-dynamic thruster (MPDT) operated with a magnetic nozzle has been investigated in detail. Experiments are performed in the HITOP device, which consists of a large cylindrical vacuum chamber (diameter 0.8m, length 3.3m) and 17 magnetic coils. An MPDT shown in Fig.16 is operated quasi-steadily (1ms) with He gas as a propellant. Axial magnetic field is applied to the MPDT and exhaust velocity is measured by a spectrometer in order to clarify the optimum field structure for the applied-field MPDT.

**Fig.16 An MPDT with an extended structure for optical measurement.**

Experiments of both ion cyclotron resonance heating (ICRH) and acceleration in a magnetic nozzle are also performed. This research is related to the VASIMR-type thruster, in which thrust and specific impulse can be changed with constant electric power. When ICRF (ion-cyclotron-range of frequency) waves are excited by a helically-wound antenna, ion temperature Ti drastically increases during the RF pulse. Perpendicular component to the magnetic field of ion energy decreases, whereas parallel component increases along the diverging magnetic field. This indicates that the increased thermal energy is converted to flow energy in a diverging magnetic nozzle. The ion acceleration along the field line is clearly observed in both He and H2 gases. The exhaust energy can be controlled by input RF power only.

## 4 Acknowledgements

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