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Scaling of Ion Thrusters to Low Power

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Analyses were conducted to examine ion thruster scaling relationships in detail to determine performance limits, and lifetime expectations for thruster input power levels below 0.5 kW. This was motivated by mission analyses indicating the potential advantages of high performance, high specific impulse systems for small spacecraft. The design and development status of a 0.1-0.3 kW prototype small thruster and its components are discussed. Performance goals include thruster efficiencies on the order of 40% to 54% over a specific impulse range of 2000 to 3000 seconds, with a lifetime in excess of 8000 hours at full power. Thruster technologies required to achieve the performance and lifetime targets are identified.

Introduction

Analyses were conducted which indicate that 0.2 kW-class ion thrusters may provide performance benefits for near-Earth space commercial and planetary science missions.[1] Small spacecraft applications with masses ranging from 50 to 500 kg and power levels less than 0.5 kW were considered in this study.

A throttleable 0.5-2.3 kW 30 cm diameter xenon ion thruster and system are currently under development by the NASA Solar Electric Propulsion Technology Application Readiness (NSTAR) Program for use on planetary science spacecraft.[2] The system is rapidly approaching flight status and is scheduled to be used for primary propulsion on the New Millennium Deep Space-1 mission to be launched in July 1998.

The NSTAR system, however, may not be an optimal high specific impulse option for very small spacecraft, because of the inherent limited power, volume, and thermal control capacity available on-board. As such, an activity is being conducted to examine ion thruster scaling relationships to assess system requirements, performance limits, and lifetime expectations at input power levels below 0.5 kW.

Prior development efforts have brought low-power (sub-0.5 kW) ion thrusters to a high state of technology readiness, including an 8 cm mercury ion thruster[3] and the Hughes 13 cm xenon ion thruster.[4] However mercury propellant is not a viable option, and the Hughes thruster may not be optimal for small spacecraft from a performance and mass standpoint.

Current activities in on-board propulsion include development and testing of low-power ion thrusters and components, including low-flow rate hollow cathodes and efficient discharge chamber designs. A parallel effort to develop a breadboard power processor for operation in the 0.1-0.3 kW power range is on-going.[5] This paper discusses performance and lifetime expectations for low-power xenon ion thrusters, and the development status of thruster components and a 0.1-0.3 kW prototype ion thruster.

Mission Applications

Low-power electron-bombardment xenon ion thruster solutions were recently evaluated for near-Earth space commercial and science missions,[1] and for solar system exploration.[6] Two potential mission applications for a small ion thruster operating at approximately 0.3 kW include an Earth orbit magnetospheric mapping satellite constellation, and a geosynchronous north-south station keeping application.

In one mission study, projected xenon ion thruster efficiencies of approximately 40% to 54% were assumed. The projections result in an optimal specific impulse range of 2000 to 3000 seconds over an input power envelope of 0.1-0.3 kW.[1] For the reference missions, these performance levels yielded significant reductions in both propulsion system wet mass, and launch vehicle requirements, relative to the baseline chemical propulsion systems. Required thruster lifetimes ranged from approximately 3000 hours at 0.3 kW (geosynchronous stationkeeping) to nearly 14,000 hours at a mission-average input power of 0.2 kW (science mission), and total-impulse requirement of about 3.0×10^5 N-s.

Thruster Performance and Lifetime Goals

The performance levels assumed in the mission study[1] are believed to be reasonable goals based on component testing and technology projections conducted to date. Thruster performance targets, consistent with these mission requirements, are listed in Table I for thruster input power levels of 0.1 kW, 0.2 kW, and 0.3 kW.

Figure 1 displays published thruster efficiencies versus input power for several small thrusters,[4,7-10] as well as unpublished data for a 30 cm ion thruster.[11] All data are for xenon propellant, with the exception of the 0.05 kW point which was obtained from a 5 cm mercury ion thruster,[7] and the 0.12 kW point from an 8 cm mercury ion thruster.[3] Additionally, all data were corrected for thrust losses (associated with divergence and multiply-charged ions), and other fixed losses (notably, neutralizer and main cathode keeper), with the exception of the JPL 15 cm datum which did not include all neutralizer losses. Other thrusters, including the National Aerospace Laboratories (NAL) 14 cm thruster,[12-14] were not included because either the quoted efficiencies were uncorrected for thrust- and fixed-losses, or no direct reference to overall thruster efficiency could be located.

Also shown in Figure 1 is a performance curve of the target efficiencies for the prototype ion thruster. Additionally, a projected performance curve for the JPL 14 cm thruster is shown.[6] As indicated in Figure 1, the prototype thruster efficiency targets and power levels are outside the present xenon ion thruster operational envelope. An important consideration of course is that improvements to state-of-the-art must be achieved to warrant investment in the development of a new thruster.

A thruster lifetime of $\geq$ 8000 hours at full power (0.3 kW) is targeted. This corresponds to a total impulse capability at full power of approximately 3.3×10^5 N-s. At the 0.3 kW power level, such a system would process a total of 11.0 kg of xenon in 8000 hours.

Thruster Scaling Considerations

Reducing the thruster beam diameter and thruster volume are important considerations for integration onto small spacecraft. For purposes of examining scaling relationships over the input power range of 0.1-0.3 kW, an 8-cm thruster beam diameter was selected for testing. The primary requirements are to achieve the aforementioned performance and lifetime goals. The considerations driving the thruster beam diameter include maximum acceptable beam current density, discharge chamber electrical efficiency, and operating discharge voltage.

The estimated performance of 8-cm ion optics, scaled from that demonstrated with 2-grid 30 cm optics,[15] yields a perveance-limited beam current consistent with the values indicated in Table I with about 100 volts total margin. The average beam current density varies from about 1.8 mA/cm^2 at 0.1 kW to about 4.1 mA/cm^2; or approximately 0.6-to-1.4 times that of the NSTAR thruster.

For the NASA NSTAR 30 cm thruster, the erosion of the molybdenum accelerator grid due to charge-exchange ions is one of the life limiting wear-mechanisms. If the internal discharge voltage of the 8-cm diameter thruster is limited to 28 V to mitigate internal erosion, then charge-exchange erosion of the accelerator grid is potentially the life-limiter. The relevant local measurement for this accelerator grid end-of-life mechanism is the bridge depth erosion in the grid center. The bridge is defined as the minimum eroded depth in the groove between two pits in the accelerator grid erosion pattern. Using life test data, a "grid erosion parameter" (or GEP) was proposed as a straightforward combination of measured parameters with a high correlation to the magnitude of the worst-case charge exchange erosion.[16] The GEP consists of the product of the accelerator grid impingement current, test time, and grid material sputter yield, divided by the beam area.

Recent in-situ erosion measurements from the NSTAR 2.3 kW Life Demonstration Test (LDT) indicate that the bridge erosion wear rates are less than 7 μm/khr,[17] yielding a conservative accelerator grid lifetime in excess of 29,000 hours (corresponding to a bridge erosion depth of 200 microns, or only 40% of the way through the electrode). The GEP was applied to the 8-cm thruster conditions identified in Table I, and then normalized to the estimated NSTAR thruster grid life at full power from the LDT data.

Accelerator grid life estimates for the 8-cm thruster versus input power are shown in Figure 2. As indicated, the normalized grid life varies from about 1.3x NSTAR at 0.1 kW to about 0.33x NSTAR at 0.3 kW. The results from the NSTAR LDT support an 8000 hour accelerator grid life capability for an 8-cm at the 0.3 kW condition, but further analyses are warranted.

Special consideration is warranted for the discharge chamber and neutralizer designs. As the thruster throughput is decreased, the discharge electrical efficiency decreases as reflected in the power required to produce an ampere of beam ion current.[18] This is because the neutral density in the discharge decreases, and hence the probability that energetic electrons will undergo inelastic collisions prior to being collected at anode surfaces decreases. The discharge losses for the thrusters identified in Figure 1 are displayed in Figure 3 as a function of input power. As indicated in general the discharge losses increase with decreasing input power. The targeted maximum discharge losses for the prototype thruster are also shown in Figure 3 and they range from approximately 333 W/A at 100 W to about 266 W/A at 300 W.

The discharge electrical efficiency also decreases as the thruster diameter is decreased because of the reduction in primary electron containment length. To yield a constant propellant efficiency the discharge must be operated at successively higher voltages as the thruster diameter is decreased.[19] To minimize the screen (positive) grid erosion a maximum discharge voltage of 28 V at full power is targeted. This is consistent with past design criteria including that used in the development of the NSTAR 30 cm thruster.[20]

The increase in both discharge losses and operating voltage with decreased thruster size has two consequences. The increase in discharge losses reduces the thruster efficiency, and the increase in discharge voltage decreases the thruster life time due to the increase in the energy of ions striking cathode-potential surfaces.

A correlation has been established between discharge propellant efficiency and thruster input power,[18] and this was used in estimating prototype thruster performance. A linear increase in discharge propellant efficiency with input power is expected, and propellant efficiencies from about 78% at 100 W to 82% at 300 W are assumed for xenon thrusters.

The performance goals for the prototype thruster neutralizer include a 20 V keeper voltage and 15 V coupling voltage, at a keeper current of 100 mA and xenon flow rate of 36 eq. mA xenon (about 0.5 sccm). At this keeper current, a maximum ratio of 3:1 in total neutralizer emission current is required with/without beam extraction at 0.3 kW full power.

Figure 4 displays neutralizer flow rate (in equivalent milliamperes) versus neutralizer input power for several neutralizers. [3,4,7,8,10,21] As indicated, the typical xenon flow rates are of the order of 30 eq. mA, at neutralizer input power levels ranging from about 7 to 17 watts. Also shown is the performance target for the prototype thruster neutralizer. The intent is to develop a neutralizer operating at comparable flow rates, but at a substantially reduced input power.

Prototype Thruster Development

Ion Optics

Preliminary development work is focused on using a 2-grid molybdenum electrode configuration, with the same hole geometry as that used in the 30 cm NSTAR thruster ion optics. Two notable exceptions to the NSTAR geometry include of course the beam diameter (8-cm in this configuration), and the mounting system.

The mounting system used for the prototype small thruster optics differs from that of implemented on the NSTAR thruster in both material and configuration. This approach was motivated to reduce the fabrication cost, and to simplify optics assembly and electrode alignment. Provisions are made in the mechanical interface to the prototype thruster discharge chamber to accommodate other configurations, including carbon-carbon ion optics, as they become available.

Discharge Chamber

The thruster performance and lifetime goals necessitate that the discharge chamber operate at high values of electrical and propellant efficiency. As such, emphasis has been placed on modeling and testing of the discharge chamber magnetic circuit design to ensure that acceptable discharge losses and voltages are achieved.

Modeling efforts have included numerical simulation of discharge processes utilizing the magnetic field and plasma flow code developed by Arakawa and Ishihara.[22] Testing activities include mapping of magnetic field configurations and operation of the discharge to characterize the electrical performance and to quantify the extracted ion fraction.

Both divergent- and cusp-field circuits, using low-magnetic flux permanent magnets and high-magnetic flux rare-Earth permanent magnets have been examined. The advantages of low-magnetic flux magnets, such as Alnico, include high operating temperatures, low cost, and low magnetic fields external to the thruster. A disadvantage of this approach is that the magnetic field strength is generally too low to efficiently contain the primary electrons in a small-volume discharge. This is typically remedied by increasing the electron energies by using a physical impedance in the vicinity of the discharge cathode. However this results in the introduction of an additional cathode-potential erosion site in the discharge, and operation at high values of discharge voltage which exacerbate internal erosion.

The advantage of using a high-magnetic flux rare-Earth permanent magnet configuration is that it efficiently contains the primary electrons, and permits high efficiency discharge operation at low values of discharge voltage. An example of this is the NSTAR thruster ring-cusp discharge. It operates at 170-200 W/A, at approximately 90% discharge propellant efficiency, at a discharge voltage of less than 24 volts.[23]

The demonstrated performance capability of discharges using rare-Earth magnets in a ring-cusp configuration, potentially outweigh its disadvantages. As such, discharge chamber modeling and test activities to date have emphasized this design approach.

For a given thruster design - ion optics neutral transparency and discharge chamber length - there is a fixed neutral loss rate which is to-first-order independent of thruster operating condition, regardless of propellant flow rate.[24] The neutral loss rate, n_o, is expressed as

$$n_0 = J_b * (1/n_{ud} - 1), \quad A. \qquad (1)$$

where J_b is the beam current, and n_{ud} is the discharge propellant utilization efficiency. Only singly-charged ions are assumed in this simple model.

An examination of data from different thrusters[9,11,13,25,26] shows that the neutral loss rate increases with decreasing discharge chamber length, as illustrated in Figure 5. This is not unexpected since the neutral residence time in the discharge chamber decreases with decreasing effective length. The neutral loss rate data were normalized to account for the difference in thruster beam diameters and effective optics neutral transparencies.

Several observations are made from Figure 5:
(1) A dependency of neutral loss rate on discharge chamber length exists;

(2) Obtaining useful propellant efficiencies with xenon for discharge chamber lengths less than about 5 cm is problematic. For example, Figure 5 indicates that to obtain a 90% propellant efficiency for a 5 cm length would require operation at 0.9 ampere beam current; an excessively-high power density;

(3) The neutral loss dependency on thruster length reflects directly in the maximum propellant efficiency, and hence thruster efficiency. That is, in general, as the length of the thruster decreases, so does its efficiency;

(4) To achieve the discharge propellant utilization efficiency goals of about 0.78 at 0.10 kW and 0.82 at 0.3 kW requires that the neutral loss rate be $\leq$ 0.040 amperes.

From Figure 5, a minimum discharge chamber length of about 9.5 cm would be required to obtain these performance levels. Hence, the prototype small thruster design incorporates a chamber of this length, with appropriate margin. Additionally, a reverse-feed main plenum is used to increase propellant efficiencies at throttled power levels.

A prototype thruster discharge chamber is shown on a test stand in Figure 6. The design incorporates a partial-conic anode-potential discharge chamber constructed of non-ferromagnetic materials, and it uses a ring-cusp magnetic circuit.

Discharge and Neutralizer Cathodes

A critical area necessary to achieve the goals and performance levels identified in Table I include the development of low-flow rate xenon hollow cathodes. A program to develop efficient, low flow cathodes to support both low-power electric propulsion systems and a next-generation NSTAR 30 cm thruster, is in progress.

The cathodes under development for the prototype thruster are constructed from 3 mm diameter tube/electron emitter technology. The cathode tip orifice diameters for the discharge and neutralizer cathodes are sized to ensure stable long-life operation over the range of required emission currents. Also the aspect-ratio of the cathode tips are adjusted to yield a high ratio of emission current-to-flow rate. Figure 7 shows both a Space

Station cathode and prototype thruster cathode for size comparison.

For the discharge cathode the emission current requirement varies from about 1.0 A to 2.0 A over the 0.1 kW to 0.3 kW power envelope. At these conditions, the corresponding xenon flow rate varies from about 56 eq. mA to about 120 eq., mA maximum (assuming a 50/50-split in main plenum/discharge cathode flow rates).

The approach used for the neutralizer is to develop an efficient keepered-hollow cathode. The performance of one of the prototype cathodes, operated in a simple diode configuration, is shown in Figure 8, a plot of xenon flow rate versus total emission current. Also shown in Figure 8 are data for other published small thruster neutralizers.[3,4,7,8,10,12,21] As indicated, the prototype cathode operates at approximately the same flow rates as the other neutralizers over comparable emission currents.

The prototype cathode operates over approximately a 3.4:1 throttling range, from about 1.5 A down to 0.45 A. The cathode tip temperatures vary from about 1250 degrees C at the maximum emission current, down to about 840 degrees C. While this cathode represents state-of-the art, clearly additional improvements (factor of 2 reduction in flow rate and emission current) are needed to achieve the thruster performance levels identified in Table I.

Test Support Equipment

The following section briefly discusses the test support equipment developed to conduct performance and wear assessments of the prototype small ion thruster.

Power Supplies

Performance assessments of the cathodes and the discharge chamber are conducted using commercial power supplies. Operation of the prototype thruster with beam extraction will be conducted using a power console originally developed for the NASA 30 cm thruster.[27] A breadboard power processor for the small thruster is also under development and will be integrated with the thruster as it becomes available.[5]

Propellant Management

An inert gas feed system was designed and constructed for performance and life time assessments of small xenon ion thrusters. The requirements imposed on the feed system included:

(1) In-situ propellant flow rate calibration capability for verifying the accuracy of the flow rate readings during the course of thruster life testing to within 2% of the reading. In-situ flow calibration is achieved by the volumetric method. The pressure drop and temperature of a known volume of gas upstream of the flow controller are monitored with time and compared to the flow controller reading. The known volume is sized to achieve an accuracy within +/- 2%. This procedure is accomplished without varying the upstream pressure to an extent that will affect the flow controller performance, and hence, may be performed while the thruster is operating.

(2) Control of the propellant flow rates to within 0.05 sccm (1% of full-scale) for the cathode, main plenum, and neutralizer.

Summary

An activity is being conducted to examine ion thruster scaling relationships to determine system requirements, performance limits, and lifetime expectations at input power levels below 0.5 kW. This was motivated by mission studies indicating the potential advantages of developing a low-power high specific impulse propulsion option.

For purposes of examining scaling relationships over the input power range of 0.1-0.3 kW, a prototype thruster with 8-cm beam diameter is in development. Performance goals over this power range are 38% efficiency at 2000 s specific impulse to about 54% efficiency at 3000 s specific impulse. Activities include design and testing of components, including low-flow rate hollow cathodes and efficient discharge chamber designs. A discharge chamber design and magnetic circuit have been selected, and low-flow rate cathodes are in test.

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Table I - Thruster Performance Targets

Parameter	0.1 kW	0.2 kW	0.3 kW
Thrust, mN	4.0	8.1	11.3
Efficiency, %	38	48	54
Specific Impulse, sec	2000	2500	2950
Discharge Voltage, V	28	28	28
Discharge Current, A	1.05	1.71	1.90
Discharge Flow Rate, eq. mA	113	200	243
Discharge Losses, W/A	333	300	266
Discharge Chamber Propellant Eff.	0.78	0.80	0.82
Screen Voltage, V	800	960	1200
Beam Current, mA	88.2	160	200
Accelerator Voltage, |V|	200	240	300
Accelerator Current, mA	0.44	0.80	1.00
Neutralizer Keeper Voltage, V	20	20	20
Neutralizer Keeper Current, A	0.1	0.1	0.1
Neutralizer Coupling Voltage, V	15	15	15
Neutralizer Flow Rate, eq. mA	36	36	36

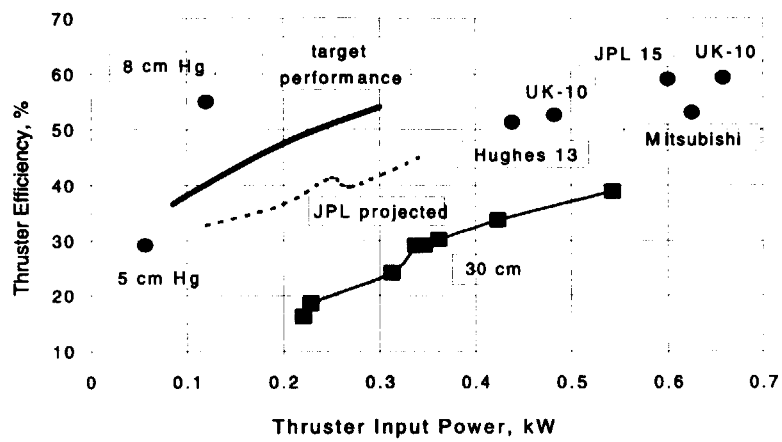

Figure 1. Thruster efficiencies versus inpu power.

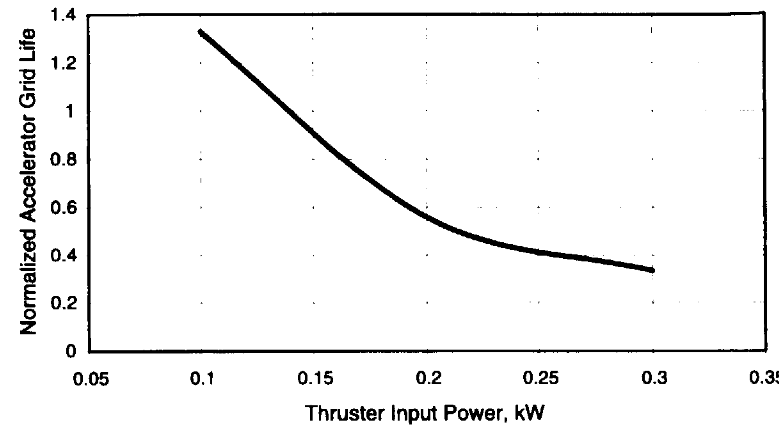

Figure 2. Accelerator grid life versus input power normalized to NSTAR thruster.

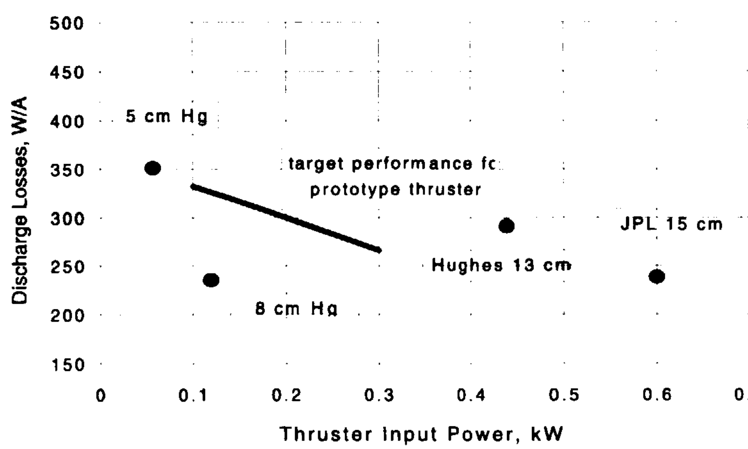

Figure 3. Discharge losses versus input power.

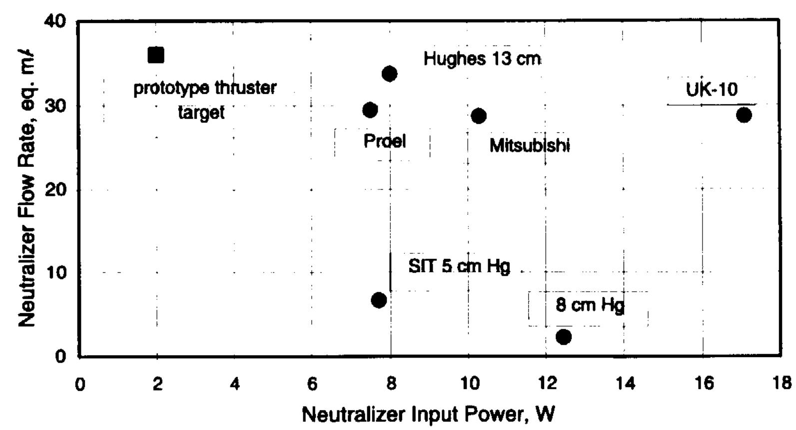

Figure 4. Neutralizer flow rate versus neutralizer input power.

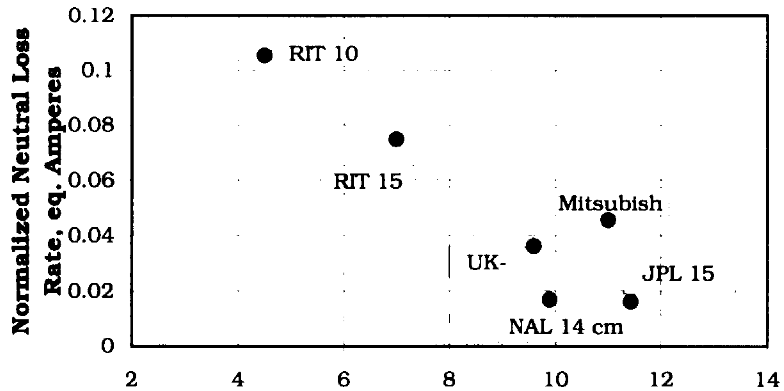

Figure 5. Neutral loss rate versus discharge chamber length.

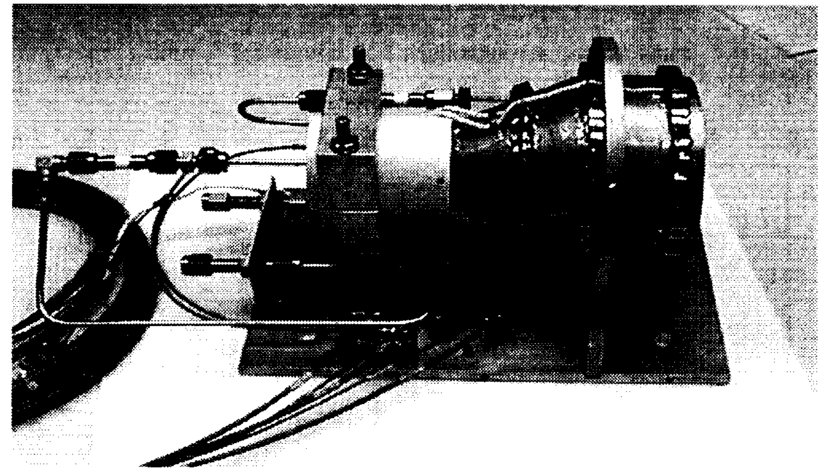

Figure 6. Prototype thruster discharge chamber on test stand.

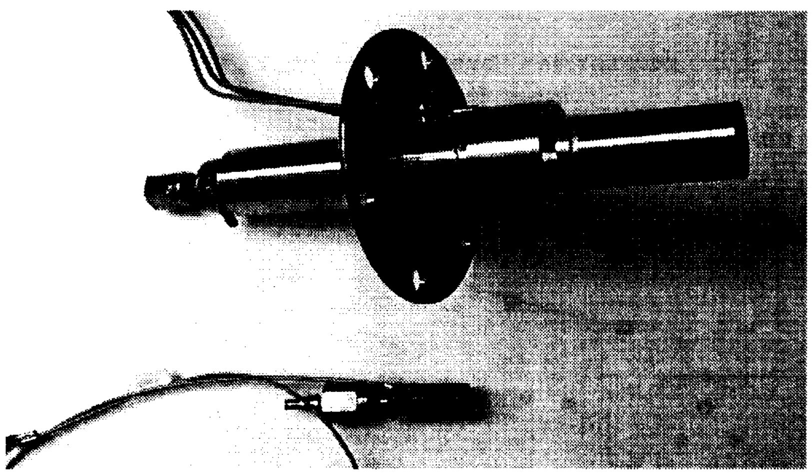

Figure 7. Space Station and prototype thruster cathodes.

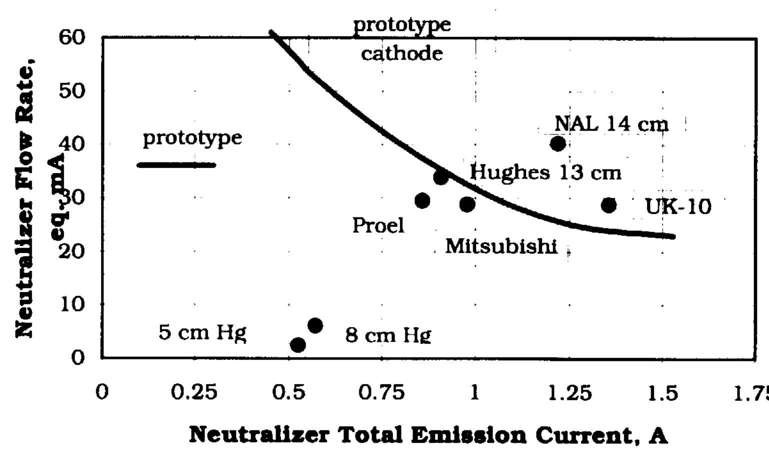

Figure 8. Neutralizer flow rate versus emission current.

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