Development and Testing of a 4500 Watt Flight Type Hall Thruster and Cathode*†

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A flight design BPT-4000 Hall thruster has been completed and two engineering model thrusters built and tested. The thruster development effort is part of a Lockheed Martin Space Systems Company (LMSSC) and General Dynamics Space Propulsion Systems (GD-SPS) funded program to develop a Hall Thruster Propulsion System (HTPS) for use on geosynchronous satellites. The thruster is a high performance, dual-mode capable design enabling both a high thrust/power and a high specific impulse.

Performance characterization has been completed on both flightweight thrusters. The second unit has also been subjected to qualification level vibration and shock testing. It is currently undergoing endurance testing for 1,000 hours as part of a risk mitigation plan. The ultimate life of the thruster is projected to be greater than 7,000 hours at 4.5 kW. This paper describes both the flightweight design and extensive testing conducted to date. It also includes preliminary data from the on-going 1,000 hr risk mitigation test.

Introduction

In recent years, the increase in spacecraft power has led to the possibility of using electric propulsion devices for GEO orbit topping. For earth orbit missions, Hall thrusters provide the best combination of thrust level and I_{sp} to achieve reasonable trip times as well as significant mass and cost-savings compared to typical chemical systems.

Since 1994, General Dynamics Space Propulsion Systems (GD-SPS) has been developing Hall thruster systems targeted at the LEO and GEO marketplace. These efforts have included the development and integration of a Hall thruster system for the National Reconnaissance Office's (NRO) STEX satellite [1,2] and the design and test of a 4.5 kW laboratory breadboard system consisting of a power processing unit (PPU), a xenon flow controller (XFC), and a BPT-4000 Hall thruster. [3,4] GD-SPS has also conducted three successful accelerated life tests on the BPT-4000 and BPT-2000 thrusters which demonstrated lifetimes in excess of 6,000 hrs. These efforts culminated in October of 2000 when GD-SPS and Lockheed Martin Space Systems Company (LMSSC) entered into a long term agreement to jointly develop a 4.5 kW flight Hall thruster propulsion system (HTPS) for next generation LMSSC geosynchronous satellites. [5] As part of this effort, GD-SPS is developing and qualifying a flightweight BPT-4000 Hall thruster. This paper describes the status of that effort including details on

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the flightweight design, test results and qualification test plans.

The BPT-4000 is the latest in the GD-SPS family of thrusters. [4,6,7] The flightweight design is based on the successful 130 mm mid-diameter BPT-4000 lab thruster. It also leverages work done under funding from NASA-GRC on the development of multi-mode Hall thrusters to provide extended life capability and improved performance over a wide range of voltages. [8] Unlike previous Hall thrusters which are qualified for a single design point, the BPT-4000 will be qualified to operate at power levels from 3.0 to 4.5 kW and at both 300 and 400 V discharge voltages. This multi-mode capability provides maximum flexibility at the satellite level. For trip time reductions in the orbit raising phase, the thruster can be operated at low voltage for high thrust and then switched to high voltage, high I_{sp} operation for the stationkeeping phase of the mission to minimize propellant usage.

Since the HTPS program start in October of 2000, a flightweight engineering model design has been completed including structural and thermal analyses. The details of the design and analysis efforts are discussed in the first section. In early spring of 2001, two identical engineering model units (EDM #1 and #2) were successfully built and have been undergoing extensive environmental and performance testing this summer. The results of these tests are included in the second and third sections. The fourth section covers the test plan and results to date of a 1,000 hr extended duration life test of EDM #2. At the writing of this paper, approximately 200 hrs have been accumulated with no measurable (<2%) degradation in performance. It is expected that this test will be concluded in late October. The program will then proceed with the building of the qualification unit (EQM) which will undergo environmental testing as well as a 6,000 hr qualification life test. A basic outline of the plan going forward is included in the last section on future work.

Flight Design Overview

The BPT-4000 flight type thruster assembly is designed to meet the following HTPS system requirements:

Table 1: HTPS Design Requirements

Requirement	Criteria	
Input Power	3.0-4.5 kW	
Input Voltage	300-400 V	
Thrust Thrust (cont.)	294 mN 254 mN 194 mN 168 mN	4.5 kW, 300 V 4.5 kW, 400 V 3.0 kW, 300 V 3.0 kW, 400V
I_{sp}	1844 s 2076 s 1769 s 1969 s	4.5 kW, 300 V 4.5 kW, 400 V 3.0 kW, 300 V 3.0 kW, 400 V
Total Impulse	> 4.6×10^6 N-s	
Cycles	> 6,300 starts	

As shown in Figure 1, the thruster consists of two integrated assemblies, an anode or accelerator and a hollow cathode.

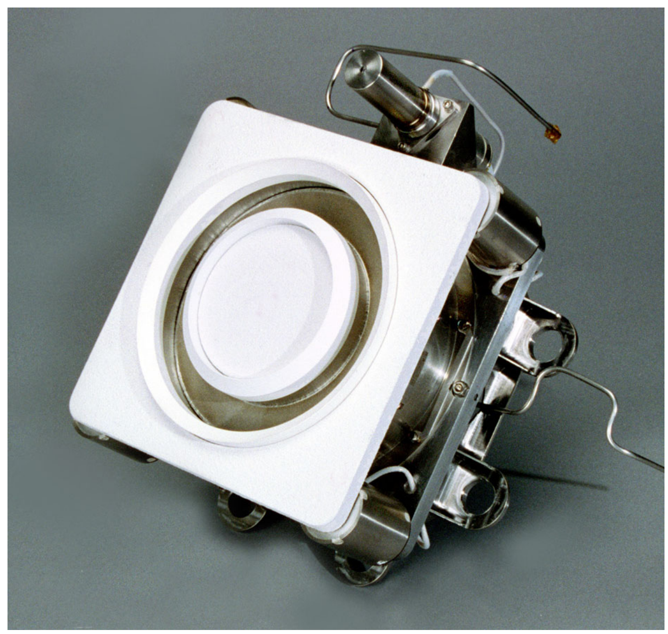


Figure 1: EDM #2 Flight Type Thruster Assembly

Cathode Design

Like the assembly, the cathode design is based on a long heritage. General Dynamics has designed and built hollow cathodes since 1994 to support testing with the GD-SPS 30-cm Ion Engine [9], the TsNIIMASH TAL D-55, and most recently, GD-SPS's BPT-2000 and 4000 Hall Thrusters. The technical basis for all of the designs is the NASA hollow cathode for the space station plasma contactor [10], which has demonstrated over 28,000 hours of life. To maintain the heritage of this demonstrated long life capability, GD-SPS has maintained the dimensions and materials of the components critical to cathode operation. The cathode tube and orifice plate configurations have been maintained, as has the same impregnant in the porous tungsten base of the emitter insert.

To reduce manufacturing cost, the NASA cathode design was subjected to an aggressive design-for-manufacturing analysis which resulted first in a prototype cathode [11], and then was refined to the current engineering development model. The current design has a reduced parts count, has a minimum of threaded fasteners, and minimizes the risk of poisoning the emitter during fabrication. The cathode is shown in Figure 2 below.

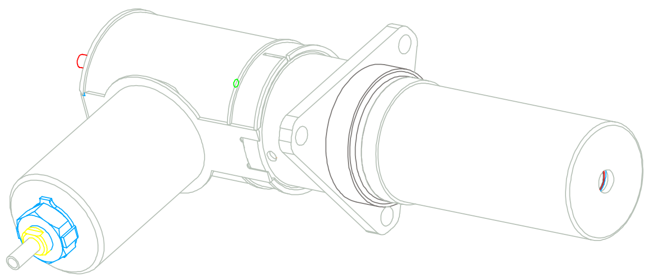


Figure 2: GD-SPS Hollow Cathode

The cathode tube is a brazed and welded assembly that extends from the cathode orifice plate to the propellant isolator. Heat for conditioning and ignition is provided by a swaged tantalum cable heater assembly. Internal ceramic insulators are used to electrically isolate the emitter tube from the body of the cathode and the keeper. In testing at GD-SPS, this propellant isolator has held off electrical potentials in excess of 3 kV over a pressure range of 26 to 21,328 Pa of xenon.

Accelerator Design

GD-SPS's flightweight accelerator design effort also focused on maintaining heritage while reducing manufacturing cost and simplifying the design. To maintain the high performance and long life capability of the laboratory design, the key anode and discharge channel geometries were carried over directly from the laboratory design. Design trades were conducted for the selection of materials, the magnetic coil design, and the gas distributor to ensure producible and reliable designs.

Figure 3 shows an isometric envelope view of the EDM accelerator design.

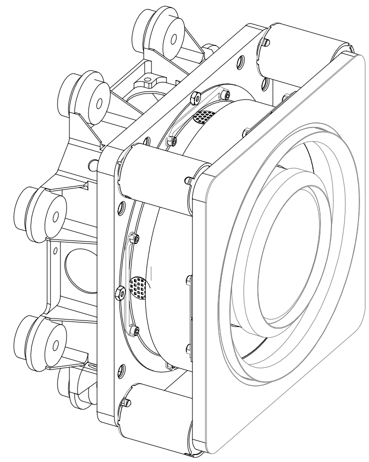


Figure 3: GD-SPS EDM Accelerator

As with a typical Hall thruster, the main structure consists of an inner pole and center core, a backplate, four outer cores, and an outer pole. The magnetic structure also includes a patented three piece magnetic shunt structure. [12]

The magnetic shunt and anode are supported by a metallic anode housing which surrounds the outside of the thruster. Xenon is introduced into the channel through a baffled gas distribution ring. The gas distribution ring is a patented design, which uses porous metal to provide highly uniform gas distribution across the full channel width. [12] Uniformity of gas injection was estimated using an ion pressure gauge on a rotating stage at a xenon mass flow rate of 8.0 mg/s. Typical variations in gas dynamic pressure are less than +/- 5% around the circumference as shown in Figure 4.

The radial magnetic field across the discharge channel is generated by four outer and one center electromagnet. The electromagnets are potted assemblies employing a high temperature ceramic-coated wire to allow them to operate above 250 °C where typical epoxy and polmer insulations melt.

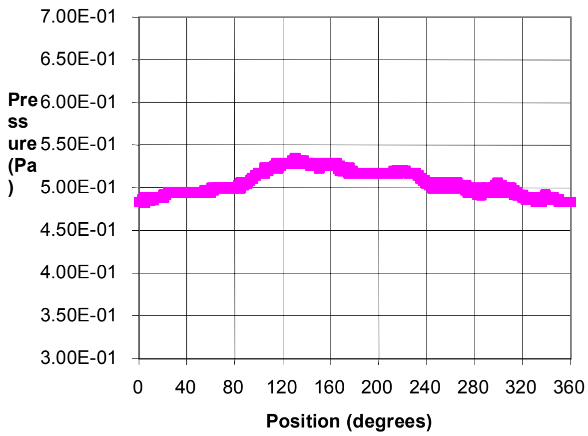


Figure 4: Gas Pressure Variations Around Circumference

Structural and Thermal Analysis
In concert with the mechanical design effort, GD-SPS performed full structural and thermal analyses on the flight design. Thermal analyses were performed using TMG to verify that the key magnetic components were kept below the Curie temperature. Thermal analysis results were also key in minimizing the power dissipation of the magnet coils and in holding the conducted heat transfer to the spacecraft to less than 15 W. Plasma heat load inputs were calculated from differential temperature measurements made on the BPT-4000 laboratory thruster and agreed very well with predictions based on thruster efficiency and magnetic field configuration.

A full thermal stress analysis was conducted using inputs from the thermal model. All stresses were well below the material and joint limitations over the full temperature range. Thermal cycle testing was also completed on the magnet coils and cathode heater transition joint to verify their integrity at temperature.

Structural analysis was performed using NASTRAN to determine the dynamic response to both sine and random vibration loads. Results showed that a stiff mounting structure was necessary to ensure that relative displacements would be small enough to prevent the ceramic insulator rings from breaking. Relative motion of the cathode was also a concern and analysis results led to a stiff cathode bracket mounting directly to the primary mounting structure rather than the backplate of the thruster.

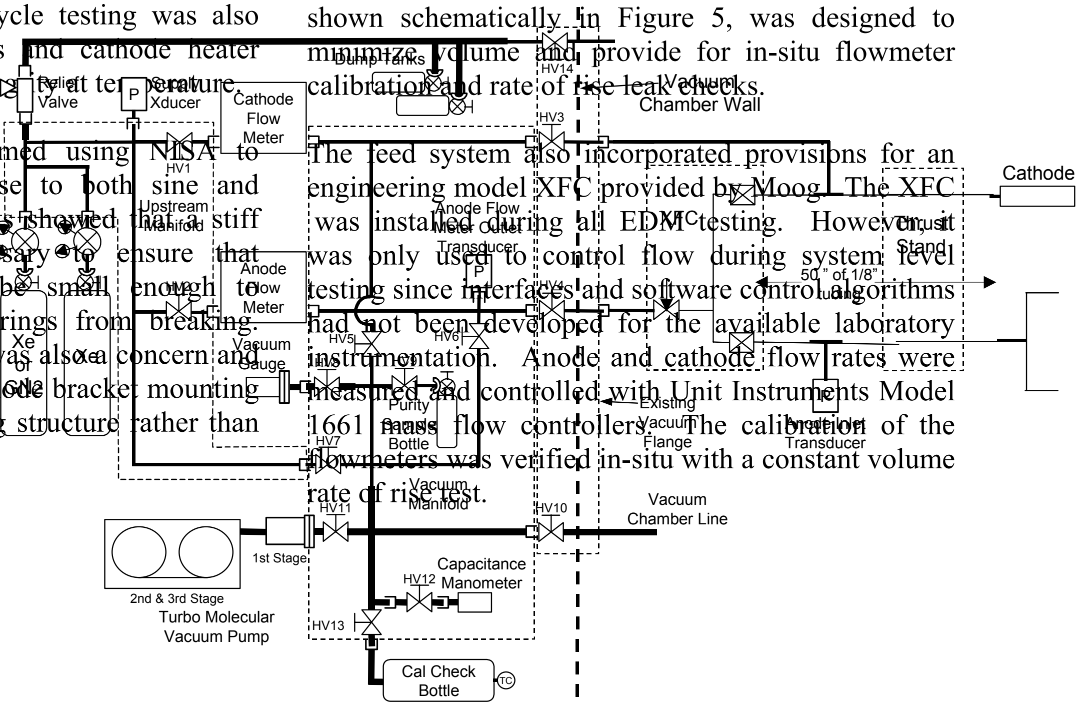
The final result of the analysis and design efforts was a simple, robust, flightweight thruster which has successfully passed all environmental testing and delivers high performance as is described in the following section.

Performance Testing

The performance and plume characteristics of both EDM thrusters were measured during extensive testing. Primary testing was conducted at GD-SPS in the Chamber 2 test facility. EDM #1 was also tested at the Aerospace Corporation to verify the GD-SPS facility and to obtain additional plume characteristic data. At both facilities, the performance showed good agreement with published performance on the laboratory BPT-4000 and was above the minimum HTPS specification requirements at all four target operating conditions. Plume data also showed good agreement with published data with 95% of the ion current enclosed in an angle of 42 degs at 4.5 kW, 300 V.

Test Facility
Test Chamber 2 in Redmond is a 2.1 m dia. x 7.2 m stainless steel vacuum chamber equipped with four cryopumps which yield a combined pumping speed of 75,000 L/sec on xenon. It is also equipped with a NASA-Glenn-style inverted pendulum thrust stand with a full scale range of 500 mN. It is operated in null-balance mode with auto-leveling to provide negligible zero drift due to thermal effects. To meet the stringent xenon purity levels required to achieve long cathode lifetimes, a new feed system was installed in the summer of 2000. The feed system, shown schematically in Figure 5, was designed to minimize volume and provide for in-situ flowmeter calibration and rate of rise leak checks.

The feed system also incorporated provisions for an engineering model XFC provided by Moog. The XFC was installed during all EDM testing. However, it was only used to control flow during system level testing since interfaces and software control algorithms had not been developed for the available laboratory instrumentation. Anode and cathode flow rates were measured and controlled with Unit Instruments Model 1661 mass flow controllers. The calibration of the flowmeters was verified in-situ with a constant volume rate of rise test.



Discharge power was supplied by a 600 V, 18 A Sorensen Model DCR600-18A power supply. Three additional power supplies were used for the magnets, cathode keeper, and cathode heater. The cathode keeper supply was only used to ignite the cathode during thruster testing except in the cases where cathode current oscillations were monitored periodically to assess cathode operating mode. The cathode heater supply was also only used to condition the cathode and to heat the emitter to ignition temperature. Once the cathode was ignited, the heater was turned off. The thruster magnets were wired in series as is typical for flight thrusters but were separated from the discharge. The thruster body was grounded to the chamber via the cable shielding. Voltage was sensed at the thruster power cable connector interface and current was measured on the positive side of each power line. An L-C filter box was used between the thruster and the power supply interfaces to isolate the power supplies and provide a simulated power processor (PPU) like interface.

A computer-based data acquisition system was used to continuously monitor voltage, current, flow, and chamber pressure. This computer was also used to monitor discharge current and adjust flow as necessary to keep the thruster at a constant power.

During extended duration testing, this same computer was also used to start and stop the thruster at the prescribed duty cycle rate and monitor power limits to prevent damage to the thruster in the case of a facility or thruster anomaly.

Ion current measurements were made in the perpendicular plane with both collimated and uncollimated Faraday cups during all testing. The setup was similar to that described previously. [7] Subsequent to the report, a new motor and rotation arm were installed to provide a smoother motion so that post-test data filtering was unnecessary. A second data acquisition channel with a lower range amplifier was also added to provide higher accuracy data angles greater than 70 degrees.

Thruster Performance Data

The performance of both EDM thrusters met or exceeded predictions based on previous lab thrusters achieving vacuum corrected efficiencies of greater than 57% at the 4.5 kW level. Figures 6 and 7 show plots of thrust and total vacuum corrected I_{sp} versus power level from 1.5 kW to 4.5 kW. All of the measurements were taken with the cathode flow rate set to 7% of the anode flowrate. The plots show the typical tail off in efficiency as power is reduced due to lower propellant utilization. The variation at all operating conditions tested was less than 4% in mass flow rate, thrust, and I_{sp} between units.

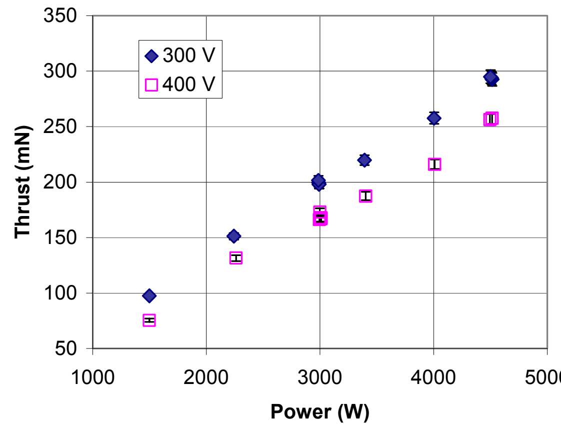


Figure 6: Thrust vs Power (300 and 400 V traces)

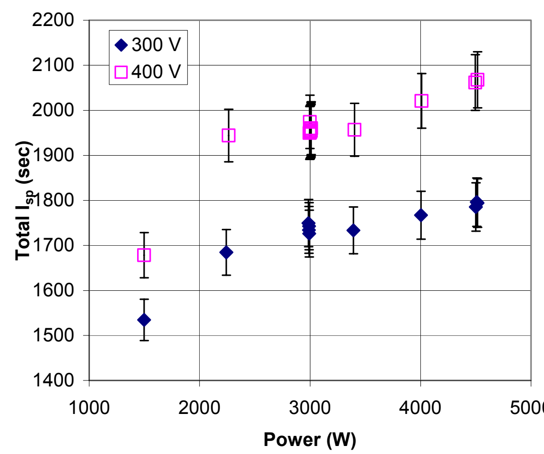


Figure 7: I_{sp} vs Power (300 and 400 V traces)

Chamber pressure corrections were applied based on linear extrapolation of the anode mass flow rate to 0 Pa pressure. The slope used (27.2 mg/s-Pa) was based on the average of data obtained on both BPT-4000 laboratory thrusters and the EDM #2 thruster in GD-SPS's test facility at several power and voltage levels. Data for anode flow rate versus pressure were obtained by bleeding xenon into the tank and monitoring mass flow rate with the thruster operating in constant power mode. An average was used due to the relatively large uncertainty in the measurement. Despite the large magnitude, the uncertainty in the correction factor only translates to a 0.5% increase in uncertainty in total efficiency.

Figures 8 and 9 show total efficiency and thrust-to-power as a function of voltage at 2.25 kW (half of the nominal power density). The thruster shows significantly improved performance at the lower voltages compared with published data on other flight type thrusters with efficiency only falling off significantly below 200 V. The in-house discharge power supply was limited to 15 A, which prevented testing at high power and low voltage. The trends however remain the same as the power is increased with the efficiency increasing slightly due to higher propellant utilization. The slight tailoff in efficiency at 450 V is also not seen a higher powers and is believed to be a result of very low gas density and therefore poor propellant utilization efficiency. As in the previous plots, all data were taken with cathode flow at 7% of the anode flow rate.

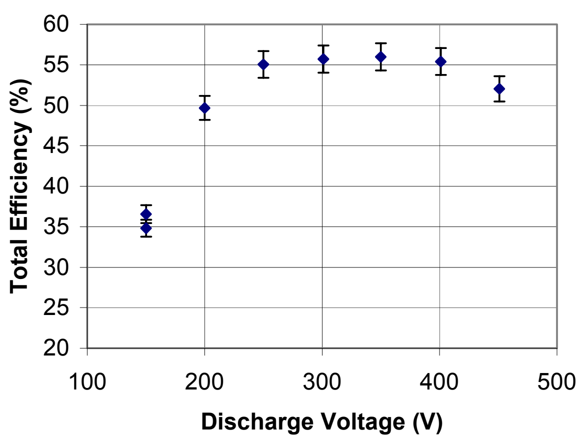


Figure 8: Total Efficiency vs Voltage at 2.25 kW

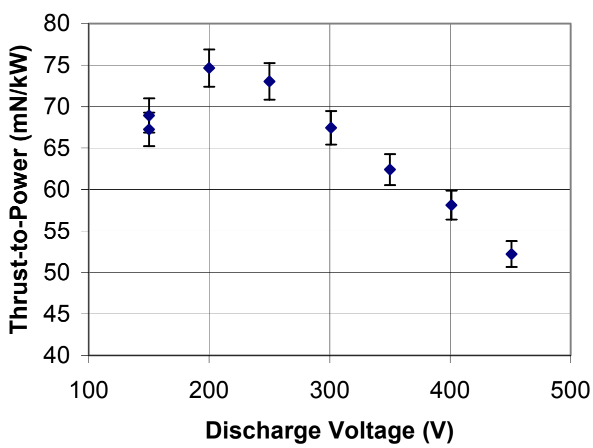
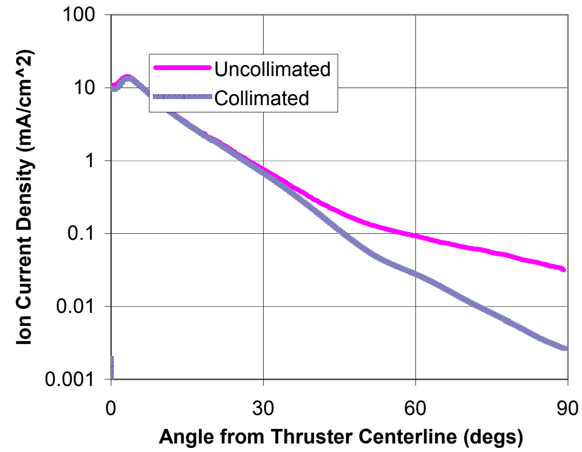


Figure 9: Thrust to Power vs Voltage at 2.25 kW



Testing was also conducted to verify the magnetic field strength necessary for optimum performance. Figure 10 shows a plot of anode efficiency versus magnet coil current for 3.0 kW, 300 V, 4.5 kW, 300 V, and 4.5 kW, 400 V. The plot shows the expected behavior where the optimum field strength is a strong function of power level decreasing as discharge power is decreased.

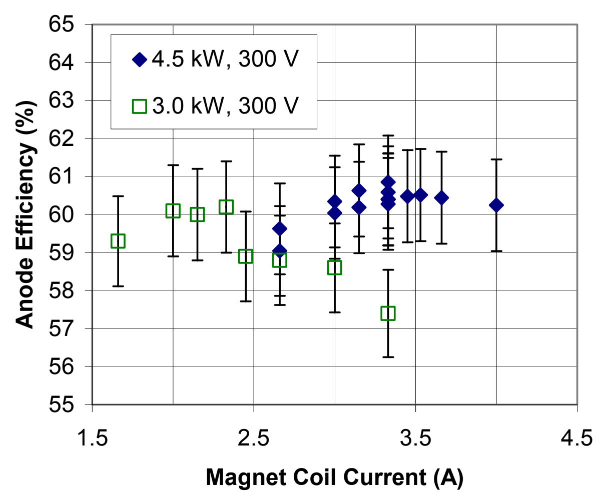


Figure 10: Anode Efficiency vs Magnet Coil Current

Plume Data

Limited plume characterization data were also taken on the thruster at GD-SPS to confirm the measurements made at the Aerospace Corporation [13] on the laboratory thruster and allow changes in plume shape to be monitored over life. Plume measurements were also made on EDM #1 at the Aerospace Corporation. These measurements showed similar trends to those data published on the laboratory thruster. Figure 11 shows an example of the collimated and uncollimated ion current measurements versus angle at 4.5 kW, 300 V. Based on the collimated Faraday cup measurements the angle enclosing 95% of the ion current is calculated to be 42°.

Figure 11: Ion Current vs Angle at 4.5 kW, 300 V

Environmental Testing

In addition to the extensive performance mapping discussed in the previous section, GD-SPS subjected the second EDM unit to both vibration and pyroshock testing. The sine and random vibration testing in all axes was conducted at GD-SPS to the levels shown in Tables 3 and 4.

Table 3: Random Vibration Levels

Axis	Freq (Hz)	PSD Level (g^2/Hz)	Overall Acceleration	Duration (sec)
All	20 50-125 260-1000 2000	0.096 0.4 0.2 0.05	17.8	180

Table 4: Sine Vibration Levels

Axis	Freq (Hz)	Acceleration (g)	Sweep Rate
All	10-24 24-36 36-55 55-100	0.5 in DA 15 15 7	2 Oct/min

For the test, the thruster mounting structure was bolted to a magnesium fixture. The cathode and anode propellant lines and pigtail cable were tied down to the fixture in a manner simulating a spacecraft interface mounting. Unit response was monitored with seven accelerometers mounted directly to the unit as well as with accelerometers mounted on the fixture to monitor input. Detailed pre and post electrical and magnetic field checkouts and visual inspections showed no changes in the thruster after vibration. After the vibration, the thruster was successfully hot fired.

The pyroshock test was conducted at NTS with an ordnance induced shock. The shock levels are given in Table 5.

Table 5: Pyroshock Levels

Axis	Freq (Hz)	Level (g)

All	100	100
	300	420
	600	1000
	1000	2000
	2400	5000
	3000	5400
	4000	6000
	5000	6000
	10000	5100

The thruster was mounted to the same fixture as used during vibration testing. Tri-axial accelerometers were mounted on the top of the fixture and on the unit to monitor responses. Prior to testing, the thruster and vibration isolators were chilled to -100°C. Just prior to the shock event, the mounting interface temperature was -40°C.

In addition to verifying survival at -100°C, GD-SPS is planning to complete hot-fire cold start tests from a thruster soak temperature of -100°C after the conclusion of the 1,000 hr life test. All of these environmental tests will also be repeated on the qualification unit prior to the start of the 6,000 hr life test.

Life Testing

After passing environmental testing, a 1,000 hr life test was commenced on EDM #2. The life test will reverify the >7,000 hr life capability of the BPT-4000 thruster design that was demonstrated previously on the laboratory model thruster. The life test consists of three segments: a 600 hr portion at 4.5 kW, 300 V, a 200 hr portion at 3.0 kW, 400 V, and a 200 hr portion at 4.5 kW, 400 V. A total of 1,000 starts will also be accumulated over the course of the three segments. During each segment, a combination of both long (20+ hrs) and short duration (<30 min) burns will be completed to simulate both orbit transfer and station keeping modes of operation. At the conclusion of each segment, a full performance map over the range of operating conditions as well as cathode characterization will be conducted. The chamber will then be opened for insulator ring erosion measurements. Full measurements of both the inner and outer insulator ring profiles will be made at eight clock position a on GD-SPS's coordinate measuring machine (CMM). The results will be compared to the erosion model predictions shown in Figure 12 to verify the erosion predictions over the 6,000 hr qualification test.

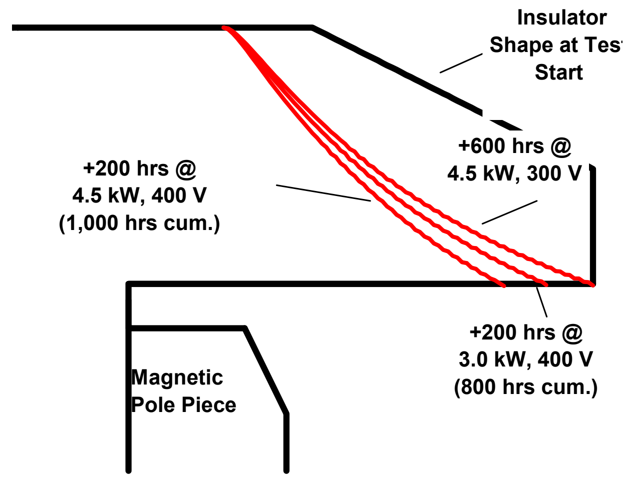


Figure 12: Predicted Insulator Erosion Profiles Over 1,000 Hr Life Test

At the writing of this paper, 200 hrs of the first 600 hr segment have been completed and it is expected that the test will be completed in late October. To date, there have been no measurable (<2%) change in thrust or I_{sp}. Cathode orifice plate temperature and cathode ignition voltage have also shown no change indicating as expected that cathode is healthy. The orifice plate temperature was measured using a single-color optical pyrometer through a vacuum viewport looking at an approximately 45 degree angle face on. During early testing, a Type K thermocouple was also attached to the emitter tube through a slot in the keeper. The pyrometer showed good agreement with the measured temperatures. Ignition voltage was measured by current limiting the keeper power supply to 2 A and then rolling up the voltage at a rate of approximately 2 V/sec until the current jumps to 2 A.

At the conclusion of the 1,000 hr life test, a full disassembly and inspection of both the thruster and the cathode is planned.

Future Work

At the conclusion of the 1,000 hr test, additional development testing will be conducted on the first EDM unit. This testing will include the aforementioned cold-start testing where the thruster will be soaked to a -100 °C temperature and then the discharge ignited and ramped to 4.5 kW. Thrust vector alignment measurements will also be made. EDM #1 is also scheduled for system level EMI and integration testing at the Aerospace Corporation. This testing will be conducted with an EDM PPU and flight type cabling to measure radiated EMI and signal transmission.

Prior to initiating the qualification phase of the program, a Critical Design Review (CDR) will be held at GD-SPS to review the full design. At the conclusion of the CDR, GD-SPS will initiate the build of the qualification unit. The unit subjected to a full ATP sequence including hot fire and vibration testing, qualification level vibration and shock testing, and thermal cycle testing before beginning the extended duration life test. A total of 4.6 million N-s of impulse will be accumulated during the 6,000 hr life test along with >6,300 cycles. The impulse will be split approximately equally between 4.5 kW, 300 V, 4.5 kW, 400 V, and 3.0 kW, 400 V operation. During the entire test duration, the thruster will be operated with an EDM or ETM (Engineering Test Model) PPU and the qualification XFC.

Conclusions

GD-SPS has successfully designed, built and tested a flightweight, dual mode, high performance Hall thruster for use on GEO and LEO satellites. Targeted applications include orbit insertion, on-orbit stationkeeping, reposition maneuvers, and de-orbiting. The BPT-4000 thruster provides the user a wide range of operational flexibility and maximum benefit by allowing operation over a wide power and voltage range. The flightweight thruster delivers high performance with total efficiencies exceeding 57% at 4.5 kW. It has also been successfully tested for heavy launch vehicle vibration and shock environments. The first engineering model is currently undergoing a 1,000 hr risk mitigation life test. After 200 hrs, no degradation in performance has been measured. As part of the HTPS development program [5], during the 1,000 hr test, the thruster will be operated with a PPU to verify system level compatibility and control algorithms. Additional development testing will be conducted after the 1,000 hr test followed by a complete design review which will complete the engineering development model phase of the program. This phase will be followed by the build and test of the qualification thruster. The qualification test program will include environmental testing followed by a 6,000 hr life test in GD-SPS's Chamber 2 test facility.

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