A Low Power Cylindrical Hall Thruster for Next Generation Microsatellites

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ABSTRACT

As the demand for highly capable microsatellite missions continues to grow, so too does the need for small, low power satellite technologies. One area which needs to be addressed is advanced propulsion systems capable of performing on-orbit maneuvers, station keeping and deorbit impulses with minimal propellant. A relatively low power Cylindrical Hall Thruster (CHT) is being developed at the Space Flight Laboratory (SFL) specifically to meet the low power and size requirements of microsatellite missions. The development of SFL's sub 200 Watt, 26 mm ionization chamber diameter Hall Thruster will permit more capable microsatellite missions while minimizing the required propellant and thruster mass. The aim is to produce a CHT qualified to Technology Readiness Level (TRL) 6 by the end of 2015. The first development phase focused on a highly configurable prototype design which facilitated optimization of performance parameters such as thrust, specific impulse and power consumption. The second phase incorporated lessons learned from prototype testing into the development of a proto-flight model. Subsequent work will involve packaging and qualifying a standalone flight unit including all electrical interfaces and the propellant feed system. Evaluations of alternative propellants such as Krypton and Argon against the baseline Xenon propellant will also be performed. This paper presents the test results from the low power Cylindrical Hall Thruster development campaign and discusses the status of the program and future plans.

INTRODUCTION

The Space Flight Laboratory (SFL) was awarded a contract under the Canadian Space Agency's (CSA) Space Technology Development Program (STDP) to develop an on-board propulsion system for micro and small satellite station keeping and de-orbiting. The mandate of the STDP was to encourage research and development by awarding a contract to develop and demonstrate technologies that would meet the current and future needs of the Canadian Space Programs. One among the twelve highlighted key technologies was microsatellite propulsion systems. As part of the development initiative, it was required that the key technologies be demonstrated to Technology Readiness Level 6 (TRL6), which by definition means developing a representative model or prototype system that has been tested in a representative operating environment.

SFL chose to address the STDP with two technology development initiatives that would be pursued in parallel. The first leverages the existing Canadian Nanosatellite Advance Propulsion System (CNAPS, see Figure 1) cold gas thruster technology of the CanX-4 & CanX-5 mission which successfully demonstrated microsatellite formation flying using this technology[1]. The CNAPS cold gas propulsion system is being upgraded into a low specific impulse, and high thrust monopropellant thruster. In accordance with requirements, the monopropellant thruster uses alternative non-toxic propellants as oppose to hydrazine.

The second development initiative, which is the subject of this paper, focuses on developing a low thrust, high

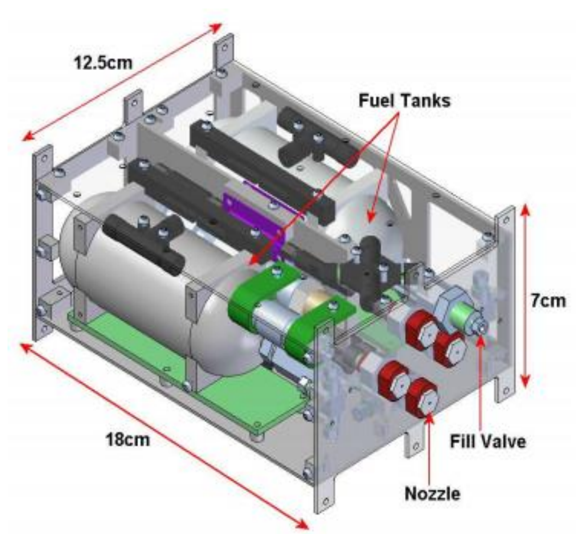

Figure 1: CanX-4 and CanX-5 Cold Gas Thruster Propulsion System

specific impulse propulsion system capable of efficiently operating for long durations of time in order to enable station keeping, orbit changes and end of life de-orbit impulse maneuvers. As a result, the initiative focuses on developing a low-power electric propulsion system that meets the requirements of future Canadian microsatellite and small satellite missions. This is to be Canada's first flight qualified electric propulsion system. In order to ensure the propulsion system being developed is relevant to future missions being planned within Canada, SFL collaborated with Canada's leading industrial space companies including COM DEV Ltd., MacDonald Detwiller and Associates Ltd., and Magellan Aerospace-Winnipeg.

REQUIREMENTS

The requirements for the propulsion program are derived from the needs of industrial partners and SFL to ensure that the thruster being developed is able to satisfy Canada's future microsatellite missions. A brief summary of the key requirements that drove the design of the Electric Propulsion System are found in Table 1.

Table 1: Propulsion System Key Requirements

Number	Requirement
1	The propulsion system shall impart a minimum ΔV of 100 m/s to the reference spacecraft
2	The reference spacecraft shall have a 150 kg dry mass, not including the mass of the thruster assembly
3	The propulsion system shall not require more than 300 W DC power while in operation
4	The propulsion system shall have a specific impulse greater than 1000 s at the design point
5	The targeted thrust range should be between 5 mN and 10 mN
6	The propellant shall be green

MICRO-PROPULSION PROGRAM STRUCTURE

The program has been organized into three phases, each having key development goals and test milestones to complete. Organized in this fashion, SFL mitigates technical risk while achieving a rapid development schedule. As of the time of writing, the program is in Phase Two.

Phase One

The first phase focused on designing and manufacturing a thruster prototype that would operate on laboratory power supplies with manually controlled propellant flow regulators. The thruster was built to be fully reconfigurable in order to allow for experimental optimization. The objective of this phase was to have a working prototype to perform basic functional testing which included the following:

1) Magnetic field characterization
2) Cathode testing
3) Experimental optimization of mass flow rate, magnetic field, and thruster ionization chamber length
4) Thrust and specific impulse measurements
5) Lifetime testing to observe thruster degradation

Phase Two

The second phase incorporates lessons learned from the prototype model to design a proto-flight model that is non-reconfigurable and representative of the flight model. The result is a design that reduced the mass by 76% and includes the proper mechanical and electrical interfaces in preparation for the next phase. Similar to Phase One, the end of this phase includes performance evaluation of the thruster.

Phase Three

In Phase Three, drive electronics including power regulators and control logic will be integrated in order to make a complete stand-alone system. At the end of the program the proto-flight model will undergo full flight qualification testing in order to bring the design to TRL6 status. Environmental testing will include:

1) Vibration testing including Sine Sweep, Shock and Random at specified space qualification levels
2) Thermal vacuum (TVAC) testing

DESIGN JUSTIFICATION

After a comprehensive review it was determined that a high specific impulse electric propulsion (EP) system capable of enabling significant on-orbit maneuverability and orbit changes would be beneficial for Canada's future microsatellite missions. Several EP technologies were evaluated for compatibility with the requirements along with their development risk. The outcome of the trade study determined that a Cylindrical Hall Thruster (CHT) was the best candidate as it is suitable for operation at power levels between 50 W to 300 W and has a mass and volume suitable for microsatellites.

Traditional annular Hall Thrusters (HT) which are characterized by an annular ionization chamber are mechanically simpler and produce higher thrust to power ratios than their counterpart – the Gridded Ion Thruster. Since microsatellites are generally power limited, a high thrust/power ratio is desirable. However, conventional HTs have drawbacks when miniaturized. When reducing the size of the ionization chamber for optimal low power operation, their annular geometry produces a high channel surface area to channel volume

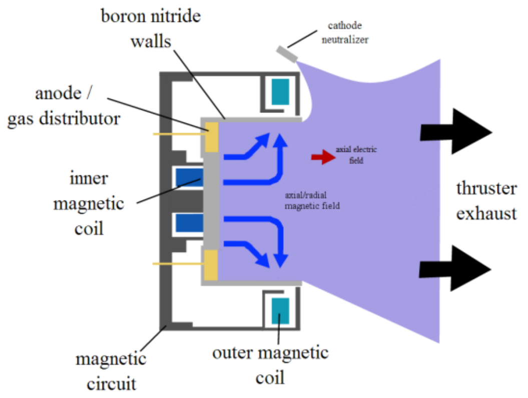

Figure 2: Schematic of a Cylindrical Hall Thruster

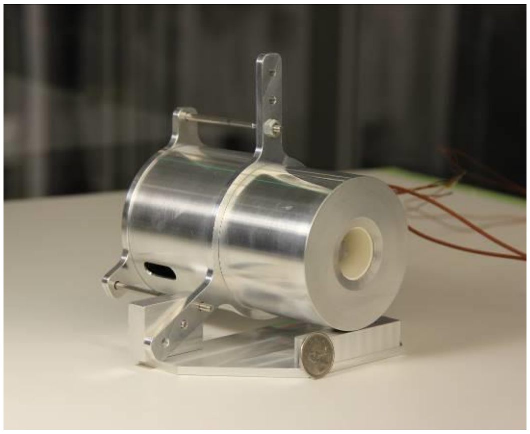

Figure 3: SFL's Prototype Cylindrical Hall Thruster on its Test Stand

ratio.[2] This increases the interaction of plasma with the channel walls resulting in heating and erosion of thruster parts, most prominently the critical inner parts of the coaxial channel and the magnetic circuit. Thus, compared to larger HTs, low power HTs are inefficient at 6-30 % efficiency at 0.1-0.2 kW compared to greater than 50 % efficiency at 1 kW.[3,4]

To mitigate these issues, Princeton Plasma Physics Lab (PPPL) proposed and developed a Hall Thruster with a cylindrical ionization chamber[5] as shown in Figure 2. This configuration reduces the surface area to volume ratio thus limiting electron transport and ion losses.[6] A 2.6 cm-dia CHT was studied at PPPL, in the 50-300 W range it showed to have efficiencies of 15 % - 32 % and thrust between 2.5-12 mN.[6] Having smaller wall losses, it was concluded that the CHT should have lower ionization chamber wall erosion and heating.[2] The advantages of the CHT for small low power applications coupled with the limited availability of flight qualified models made this type of EP system an ideal candidate for development by SFL.

PROTOTYPE DESIGN

The prototype was designed to serve as a learning platform in order to develop a refined proto-flight model later on. Consequently, design features allowing the re-configuration of the proto-flight model were included early on in the design phase. To start, the magnetic field is generated by a set of electromagnets as opposed to permanent magnets. This allows the magnetic field to be fine-tuned inside the ionization chamber. Secondly, the ionization chamber depth was adjustable by means of a threaded central magnetic core. Lastly, the propellant flow rate was regulated with an off-the-shelf mass flow controller (0-10 ± 0.01 SCCM). The goal was to find the ideal operating parameters that would maximize specific impulse and thrust between 50 and 200 W of discharge power.

A photograph of the 26 mm diameter SFL prototype CHT is shown in Figure 3. Weighing 1.6 kg (excluding cathode) the thruster is oversized for what is needed. Much of the mass is attributed to the use of electromagnets which tend to be heavier and bulkier than permanent magnets. Within the thruster, a steel magnetic core strengthens and shapes the magnetic field inside the ionization chamber. Several materials were evaluated for the magnetic core trading permeability and temperature rating.

The anode and ionization chamber are insulated with Boron Nitride (BN) ceramic which was selected for its low sputtering yield properties and good electrical isolation properties. The anode was fabricated from multiple machined pieces that were later assembled and laser welded in order to secure the pieces together, prevent gas leaks and ensure electric conductivity. The anode is the high positive voltage pole opposing the cathode and provides the propellant feed. For even propellant distribution inside of the ionization chamber a baffled anode design was implemented. The propellant is chocked and expanded through holes in the manifold and then distributed through a set of baffles. The baffles provide a means of evenly diffusing the propellant azimuthally in the ionization chamber and slowing down its axial speed. The propellant is fed through a welded tube behind the anode which also serves as the point of electrical connection for the discharge supply.

In order to avoid ITAR restricted products and reduce development risk, an industrial cathode was procured. This cathode is nominally used with industrial plasma sources, and is not optimized for spaceflight. It is not equipped with a heat shield and has a minimum operating flow rate which grossly surpasses space grade cathodes. Consequently, lower specific impulse was expected. The proto-flight thruster has been designed for compatibility with almost any flight qualified cathode.

PROPELLANT

In order to conform to the requirements of CSA's STDP for green propulsion systems, it was decided that the EP system propellant would be traditional noble gases such as Xenon, Krypton and Argon. The propellant that performs the best is one which exhibits properties of low ionization energy and high molecular mass. As shown in Table 2, it is evident that Xenon offers the best performance and this is apparent by its extensive use in previous missions. However, Xenon is approximately 3.5 times more expensive then Krypton and 240 times more expensive then Argon. Consequently, for initial testing and life time testing of SFL's CHT, Argon was used, while Xenon was used for qualification and performance testing. Ideally, it would be desirable to obtain a thruster configuration that would be able to operate with multiple propellants in order to offer the customer more options that trade performance versus cost.

Table 2: Propellant Properties

Margin	Xenon	Krypton	Argon
Atomic Number	54	36	18
Molecular Mass (g/mol)	131.29	83.80	39.95
Melting Point (°C)	-112.0	-157.0	-189.0
Boiling Point (°C)	-107.0	-153.0	-185.7
Ionization Energy (eV)	12.12	14	15.75
Mass/Ionization Energy Ratio (g/eV mol)	10.83	5.99	2.54

EXPERIMENTAL APPARATUS

Given SFL's experience in microsatellite technology design and flight qualification, most of the testing infrastructure was already in place to perform the developmental and qualification testing of the CHT in house. The test apparatus (Figure 4) and the modifications required to the vacuum chamber are described as the following.

Vacuum Chamber Upgrade for Pumping Xenon

To avoid reduced lifetime and performance degradation, the maximum pressure in the vacuum chamber while operating the thruster should be less than 1×10^{-4} Torr. Given that the expected ingress of Xenon during testing is 0.70 mg/s, the pumping capacity of the chamber needs to be at least 12,000 L/s of Xenon. Traditional methods of pumping Xenon can incur large cost from procuring the equipment and operating it.[8] Cryopumps for instance require liquid nitrogen and the size of the pump is relatively large compared to SFL's 0.75 m vacuum chamber (Figure 5). After considering the results of Garner, C.E., et al[8], an alternative method was selected to pump Xenon given that the majority of the equipment required was already acquired. The method used a cryogenic cold head located at the back of the chamber. The cold head effectively acts as a cryopump and traps residual gas on the scavenger plate within the chamber. On the other hand, since the system has a closed liquid helium loop, some of the disadvantages of traditional cryopumps are eliminated.

To achieve pumping requirements, the scavenger plate must be maintained at a temperature below 60 K in order to maintain a Xenon partial pressure below 1×10^{-4} Torr. The cold head installed on SFL's chamber (Figure 6) is capable of maintaining the scavenger plate at 51 K which results in a steady state pressure of 1.58×10^{-5} with 0.70 mg/s ingress of Xenon. The scavenger plate was polished in order to reduce the amount of heat radiated to the plate. The measured emissivity of the plate is 0.025 ±0.02.

Flight Qualification Environmental Testing

During the final phase of the program, the proto-flight model will undergo full qualification testing of the entire propulsion system (Power Processing Unit and Thruster). This test plan will demonstrate the thruster's ability to work in the relevant operational environment specified by the initial requirements. In order to simulate the environment the test campaign will use SFL's 2.5 m Thermal Vacuum Chamber (TVAC) and vibration table. The TVAC chamber will also serve as a backup chamber for thruster operations.

RESULTS AND DISCUSSION

The preliminary test results from SFL's low power Cylindrical Hall Thruster are presented below.

Magnetic Characterization

Before operating the thruster in vacuum, measurements of the magnetic field in the ionization chamber were taken and compared to magnetic simulations of the magnetic circuit. The magnetic field simulations were performed in open source software. Measurements of the magnetic field strength in the thruster ionization chamber along its bore were performed with a Tesla meter. To measure the magnetic field incrementally along the bore, the CHT was mounted on a mill table

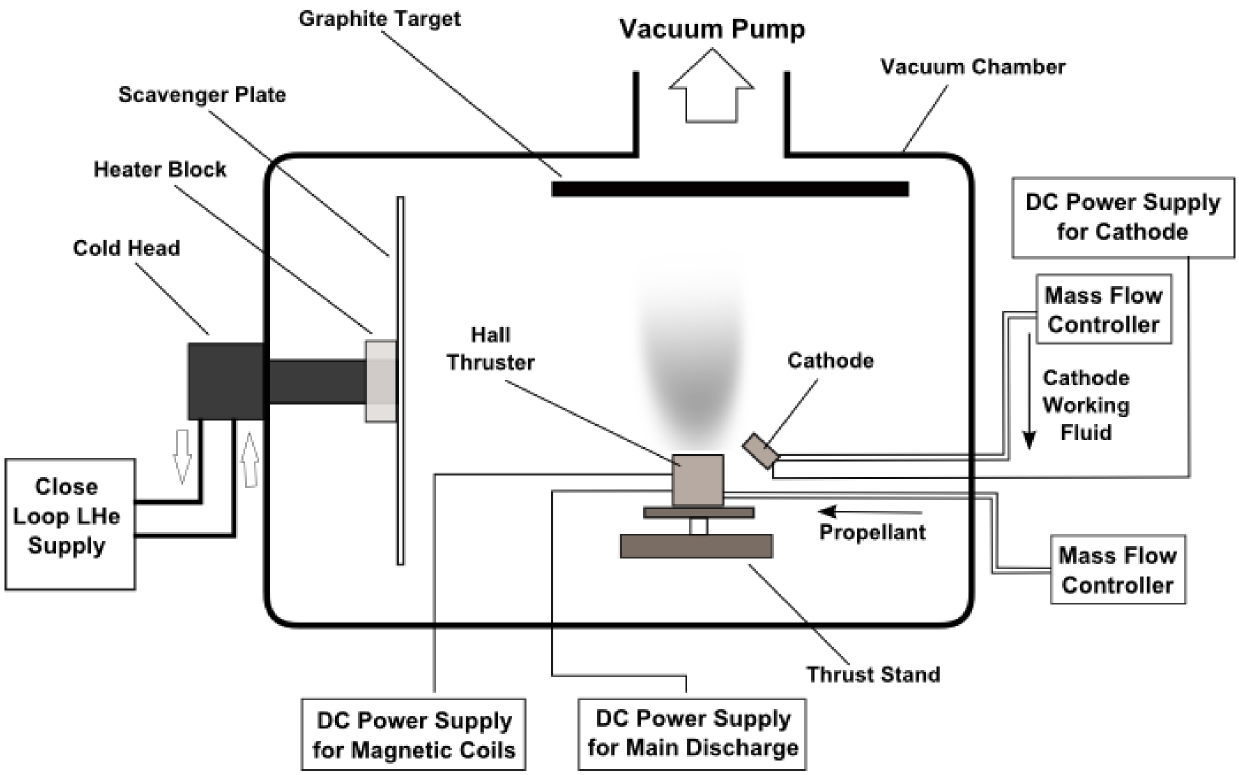

Figure 4: Vacuum Chamber Configuration During Performance Determination

Figure 5: SFL's Primary Micropropulsion 0.75 m Vacuum Chamber

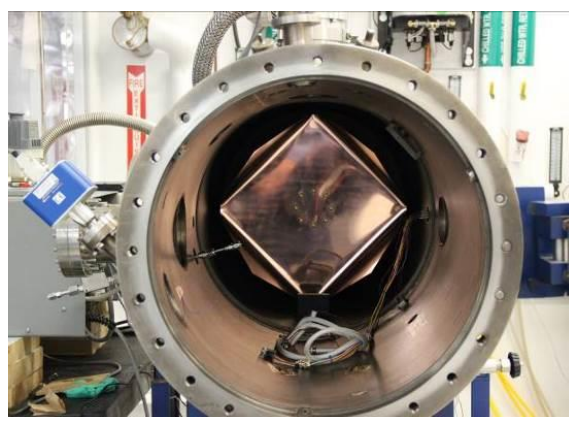

Figure 6: Cryogenic Cold Head Mounted on the Back of the Chamber with the Polished Copper Scavenger Plate

with the Gauss probe suspended centrally in the ionization chamber (Figure 7). The mill table allowed a translation of the CHT relative to the probe in increments of a thousandth of an inch. Around 70 measurements were taken between the anode and the exit plane of the ionization chamber. Given that the prototype utilizes electromagnets, measurements were taken with a few different magnetic configurations. Results show that the simulation was able to predict the magnetic field strength in the ionization chamber to within 10 %. Slight discrepancies were observed near the anode. It is assumed this is due the high concentration of magnetic field lines in this region and the fact that the Gauss probe has a cross sectional area large enough that the measurement is not taken at a finite point.

The B-N curve in the simulation software had to be adjusted to match the properties of the mild steel magnetic core material in order to achieve a simulation that better matched the experimental results.

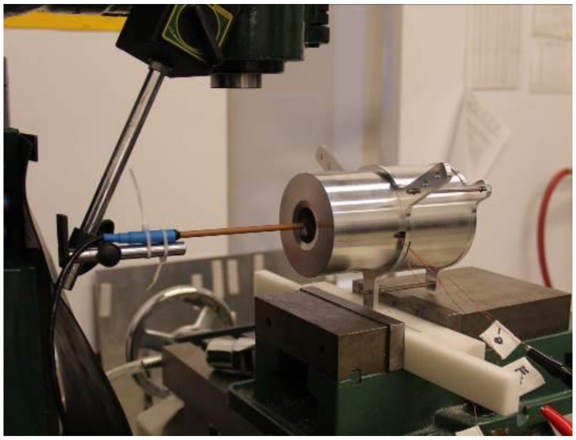

Figure 7: Magnetic Testing Performed on Mill Table

CHT Performance

Performance of SFL's 26 mm Cylindrical Hall Thruster operating on Argon and Xenon was measured over a range of discharge power from 50 W to 300 W. Thrust was measured at anode flow rates of 0.17 mg/s to 0.32 mg/s and 0.31 mg/s to 0.50 mg/s for Argon and Xenon respectively. The cathode flow rate was maintained at 0.30 mg/s for Argon and 0.20 mg/s for Xenon. The cathode flow rates were specified by the manufacturer. Further reduction of mass flow rate could shorten the lifespan of the cathode. Consequently, it has not been optimized for operating with the CHT and lower specific impulse was expected.

The relationship between thrust and discharge power for Argon and Xenon are specified in Figure 8 & 10. As expected, the relationship is linear and Xenon outperformed Argon. The length of the ionization chamber was maintained the same while operating with

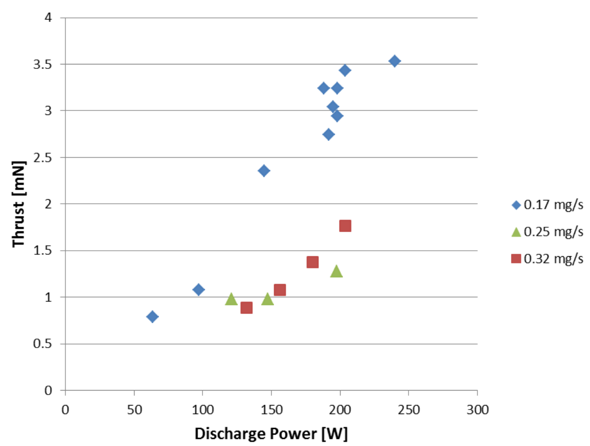

Figure 8: Characteristics and Performance (Argon) for a Range of Anode Mass Flow Rates: Thrust

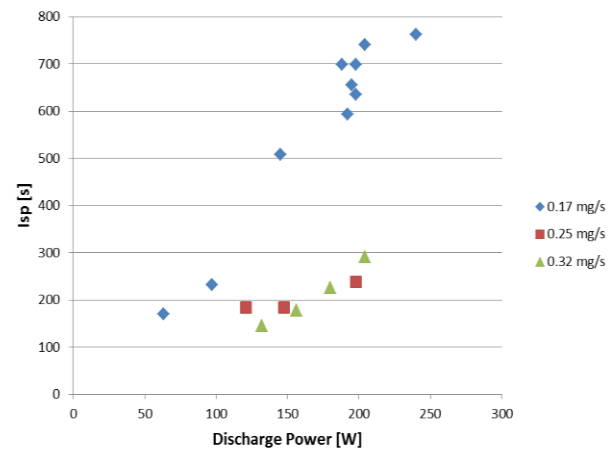

Figure 9: Characteristics and Performance (Argon) for a range of anode mass flow rates: Specific Impulse

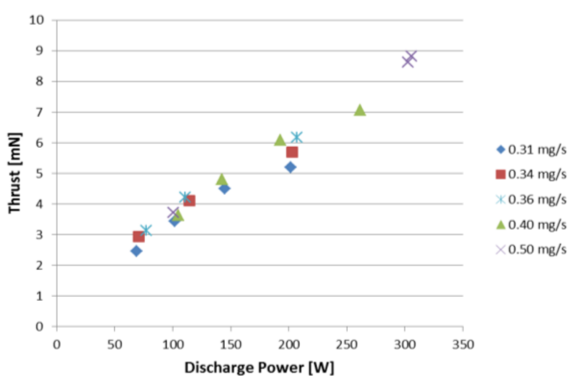

Figure 10: Characteristics and Performance (Xenon) for a Range of Anode Mass Flow Rates: Thrust

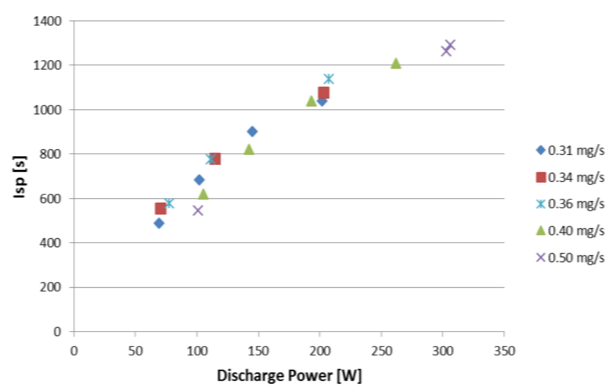

Figure 11: Characteristics and Performance (Xenon) for a Range of Anode Mass Flow Rates: Specific Impulse

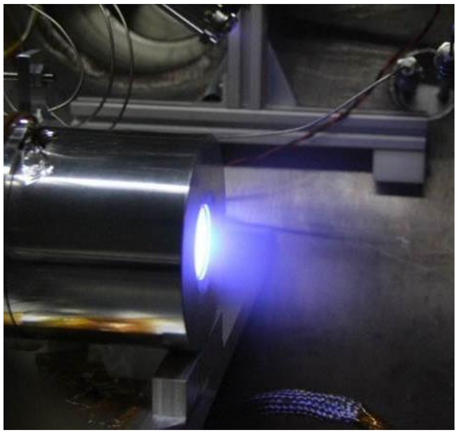

Figure 12: SFL's CHT Operating with Argon

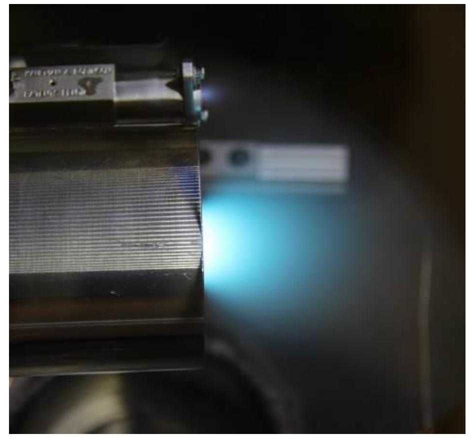

Figure 13: SFL's CHT Operating with Xenon

Argon and Xenon. According to calculations, it is assumed the performance of Argon could have been increased if the ionization chamber length was optimized for Argon. It was observed that the Argon plume divergence was larger then that of Xenon; the non-ideal ionization chamber depth could have been a factor. The thrust obtained when operating Xenon was double that of Argon at a similar discharge power. At 300 watts a maximum thrust of 8.8 mN was observed for Xenon and 3.6 mN at 250 W for Argon.

As expected, the specific impulse is greater with Xenon then Argon. For Xenon, at 69 W the Isp was 490 s at an anode flow rate of 0.31 mg/s while at 306 W, 1294 s with 0.50 mg/s. For Argon stable operation at 63 W and an anode mass flow rate of 0.17 mg/s was achieved with an Isp of 169 s. At 240 W and 17 mg/s an Isp value of 761 s was achieved.

All specific impulse values include the propellant mass flow rate of the cathode in addition to the anode, although the small amount of thrust created by the cathode is not included in the thrust measurement. As others have shown, [9] this cathode requires a higher flow rate (0.2 to 3 mg/s in Xe) which is relatively significant compared to typical anode flow rates (0.5 to 1 mg/s in Xe).[7] In addition, the mass of the cathode is greater than comparable cathodes used with other similar thursters.[8] It is assumed that better performance characteristics could be achieved with a purpose build spacecraft cathode.

The anode efficiency of the CHT ranges from 9 % - 26 % between 50 W and 200 W for Xenon and 3 % - 17 % for Argon. The propellant utilization for this 26 mm CHT is estimated to be about 25 % higher than comparable conventional Annular Hall thrusters. Similar results were observed in other Cylindrical Hall thruster.[6] These results correlate closely with the predicted performance calculated from a detailed parametric model. High propellant utilization observed in CHT allows them to operate at lower discharge voltages where otherwise annular Hall thrusters could not sustain discharge.

Thruster Deterioration

As part of qualification and developmental testing, the CHT will undergo a lifetime testing of 1000 hours continuous operation at 200 watts. Sputtering and wall erosion is expected to occur yet should not interfere with thruster operation. Most concerning is the Boron Nitride (BN) ionization chamber wall which if eroded substantially would expose the electromagnets and aluminum structure which has a much higher sputtering yield than BN.

A preliminary assessment of the thruster condition was done after operating the thruster for 10 hours, with no detectible degradation of wall thickness observed. Sputtering was observed on the front aluminum face of the thruster as shown in Figure 14. In addition, the BN ceramic used to insulate the central magnetic core at the center of the anode ring had cracks propagating in it. It is assumed that the cracking occurred from the differential expansion rate of the Boron Nitride ceramic and the mild steel magnetic core located behind it. Large temperature fluctuations cause the mild steel to expand a greater rate than the ceramic causing local stress regions. The subsequent proto-flight model will incorporate improved clearances and dimensional tolerances between parts in order to compensate for thermal expansion. Additionally, the front face of the proto-flight model will have a BN shield to reduce sputtering.

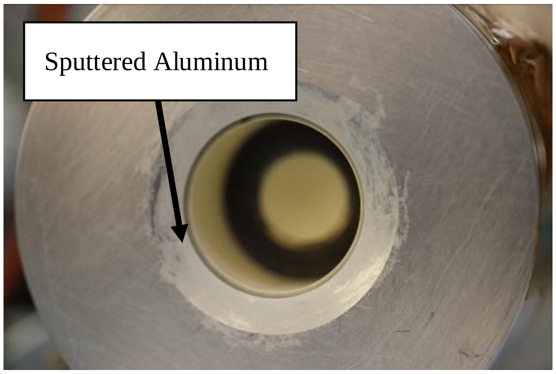

Figure 16: After 1 hour of Operation Sputtering was Observed on the Aluminium Front Face of the Thruster

The early signs of deterioration did not affect the performance of the thruster. In addition, the initial cracking did not appear to propagate after 10 hours of operation. However, a discoloration of the BN was observed as shown in Figure 15. The amount of wall erosion after 10 hours of operation appeared to be negligible.

PROTOFLIGHT MODEL

A flight representative proto-flight model is being developed in parallel with the later stages of Phase One testing. From the performance characterization results, an optimal configuration for magnetic field and ionization chamber depth has been selected. The magnetic circuit of the proto-flight model was fabricated and tested with similar methods mentioned in earlier sections. Results show that the measurements compared relatively well to the simulation results. Neodymium magnets were selected as they are

Figure 17: Cracks Propagating in Ceramic Insulation due to Thermal Expansion

significantly lighter and smaller than electromagnets which greatly reduced the mass and volume of the prototype CHT. Neodymium has a lower demagnetization temperature then other commonly used magnetic material such as SmCo magnets. However, the thruster has been designed with thermal consideration in order to maintain the magnets below their curie temperature.

The proto-flight model adopted a passive thermal management strategy that minimizes the thermal loads to the spacecraft bus by radiating the majority of heat to space. The Boron Nitride (BN) ionization chamber and anode have been thermally isolated from the body of the thruster and the magnets. This will allow the anode and BN to achieve higher operating temperature thus enabling more efficient radiative heat transfer to space.

Internal requirements dictated that the prototype thruster shall weight less than 700 grams and fit within

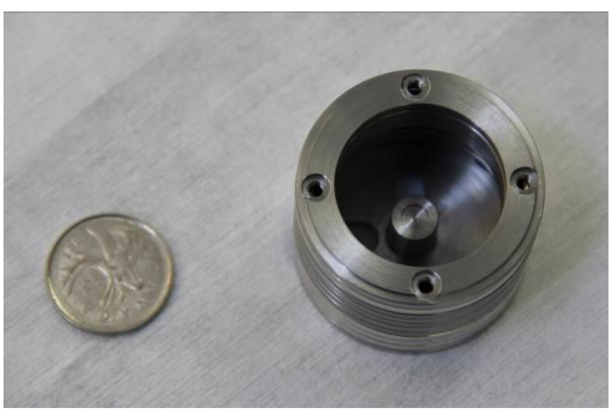

Figure 14: Size Comparison Between a Quarter and the Proto-flight Magnetic Circuit

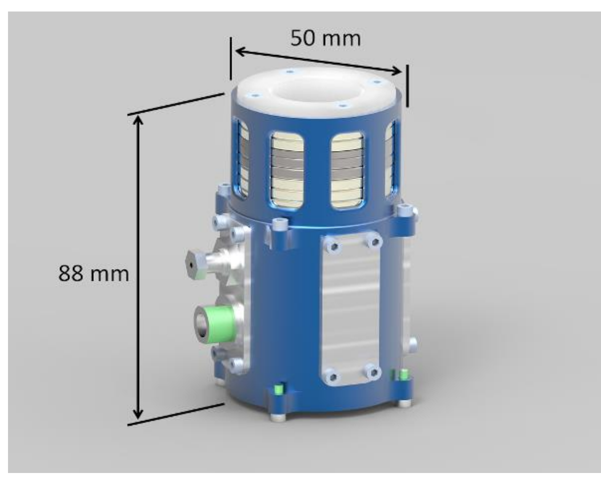

Figure 15: Rendering of Proto-flight Thruster

a 1L volume. The current design shown in Figure 17 is estimated to be around 400 grams. This mass does not include the power processing unit, drive electronics, cathode and propellant feed system. The intention is to flight qualify the prototype thruster, therefore it is being designed to endure the structural and vibration launch loads and the thermal environment of space. The propellant tank is not part of the scope of this development program given that the propellant tank volume is mission dependent.

CONCLUSION

The Space Flight Laboratory has successfully designed and tested its first 26 mm Cylindrical Hall Thruster with promising results that has met all performance requirements to date. At the design point of 200 watts, the CHT produced 6.2 mN of thrust with a specific impulse of 1139 s with Xenon. Future work will include continuation of lifetime testing on the prototype model and completion of the build and flight qualification of the proto-flight model.

Acknowledgments

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