

# In-Space Propulsion High Voltage Hall Accelerator Development Project Overview

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## ABSTRACT

NASA's Science Mission Directorate In-Space Propulsion Technology Project is funding the development of a high specific impulse long life Hall thruster. The goal of the high voltage Hall accelerator (HiVHAc) project is to develop a flight-like, engineering model (EM) Hall thruster that can meet future NASA science mission requirements. These requirements are met by a thruster that operates over an input power range from 0.3 to 3.5 kW, attains specific impulses from 1,000 to 2,700 seconds, and processes at least 300 kg of xenon propellant at full power. To demonstrate the HiVHAc project goal, two laboratory thrusters have been built and tested. The latest laboratory thruster, the NASA-103M.XL, incorporated a life-extending discharge channel replacement innovation and has been operated for approximately 5,000 hours at a discharge voltage of 700 volts. In 2007, NASA Glenn Research Center teamed with Aerojet to design and manufacture a flight-like HiVHAc EM thruster which incorporated this life-extending channel replacement innovation. The EM thruster was designed to withstand the structural and thermal loads encountered during NASA science missions and to attain performance and lifetime levels consistent with NASA missions. Aerojet and NASA Glenn Research Center have completed the EM thruster design, structural and thermal analysis, fabrication of thruster components, and have assembled and extensively tested one EM thruster. Performance and thermal characterization of the engineering model thruster has been performed for discharge power levels up to 3.5 kW. The results indicate discharge efficiencies up to of 63% and discharge specific impulse up to 2,930 seconds. In addition to the thruster development, the HiVHAc project is leveraging power processing unit and xenon flow system developments sponsored by other projects but that can apply directly to a HiVHAc system. The goal is to advance the technology readiness level of a HiVHAc propulsion system to 6.

## INTRODUCTION

Electric propulsion (EP) systems enable and improve NASA's ability to perform scientific exploration. NASA continues the development of advanced EP technologies in order to increase its capability to perform solar system exploration missions. To date, a number of NASA missions have successfully demonstrated and employed EP systems in missions like Deep Space 1 (DS1) and Dawn.<sup>1,2,3</sup>

NASA's Science Mission Directorate (SMD) In-Space Propulsion Technology Program (ISPT) manages the new electric propulsion system development.<sup>4</sup> The two primary electric EP elements of this program are the development of NASA's Evolutionary Xenon Thruster (NEXT) ion thruster propulsion system,<sup>5</sup> and the development of long life high voltage Hall accelerator (HiVHAc) as a lower cost electric propulsion alternative for future cost constrained missions.

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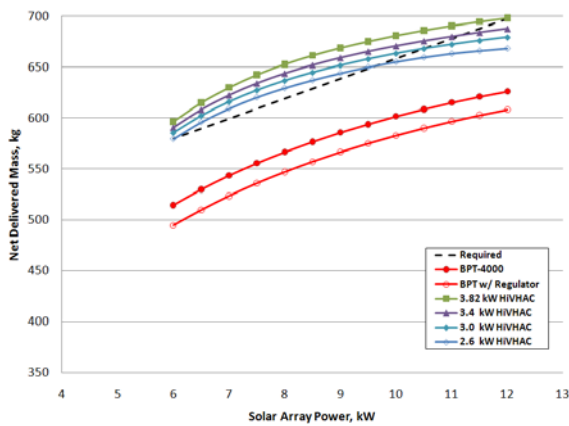
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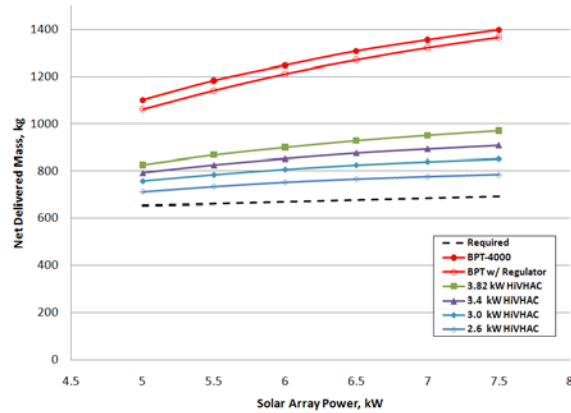
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A 2004 ISPT-sponsored refocus study<sup>6,7</sup> found that a Hall thruster system with performance characteristics similar to the HiVHAc thruster<sup>8,9</sup> resulted in substantial cost and performance benefits when compared to the NASA Solar electric propulsion Technology Application Readiness (NSTAR) and NEXT ion engine systems for certain types of NASA's Discovery-Class science missions.<sup>5,10,11</sup> More recently, mission studies were performed to compare the performance of the HiVHAc 3.5 kW thruster to Aerojet's flight BPT-4000 4.5 kW Hall Thruster Propulsion System (HTPS).<sup>12,13</sup> These studies were performed for four NASA Discovery-Class missions: Vesta-Ceres rendezvous mission (Dawn Mission), Koppf comet rendezvous, Near-Earth Asteroid Return Earth Return (NEARER) and Nereus sample return (NSR) mission. Results from the mission studies indicated that the HiVHAc thruster was able to close all the missions, whereas, the BPT-4000 thruster was unable to close the Dawn and Koppf comet rendezvous missions. For the Dawn, Koppf comet rendezvous, and NEARER missions the HiVHAc delivered mass at target was approximately 6-12% higher than the BPT-4000 thruster. The main reason for the higher HiVHAc delivered mass capability is its higher specific impulse (~2,700 sec) when compared to the BPT-4000 thruster (~2,100 sec).<sup>13</sup> Figure 1 shows the performance comparison for the Dawn mission. However, for the NSR mission, the BPT-4000 thruster significantly outperformed the HiVHAc thruster as shown in Figure 2.



**Figure 1: Performance comparison for the Vesta-Ceres rendezvous (Dawn) mission.**



**Figure 2: Performance comparison for the Nereus sample return mission.**

In Hall thrusters, the erosion of the boron nitride discharge channel walls due to ion bombardment exposes the thruster's magnetic circuit elements. Exposing the magnetic circuit poles to ion bombardment changes their shape and results in a distorted magnetic field profile and thus reduced thruster performance. Eventually the ions will start impacting the magnetic circuit electromagnets and cause wire to wire shorting of the electromagnet turns. To be able to achieve and demonstrate greater than 300 kg of xenon throughput, insulator discharge channel erosion has to be mitigated as a life-limiting failure mode by assuring that the boron nitride discharge channel always shield the Hall thruster magnetic circuit elements (electromagnets and magnetic poles) from ion impingement.

The goal of the HiVHAc thruster development activity is to fully demonstrate an engineering model (EM) Hall thruster that has a throttling range of 12:1 for power levels between 0.3 and 3.5 kW,<sup>9</sup> has a thrust efficiency > 0.55 at full power and specific impulse > 2,700 sec at full power, and has a xenon throughput capability > 300 kg corresponding to an operational lifetime of >15,000 hrs.<sup>14</sup>

To demonstrate the HiVHAc project performance, throttleability, and lifetime goals, two laboratory thrusters the NASA-77M and the NASA-103M.XL, were built and tested. The NASA-77M operated with input powers between 0.2 and 2.8 kW and demonstrated a 14:1 throttle range, specific impulse levels between 922 and 2,911 sec and thrust efficiencies between 0.31 and 0.54.<sup>10</sup> The NASA-77M was also subjected to two 300 hours wear tests to assess discharge channel wear characteristics. Measured erosion profiles also provided insights into magnetic circuit design improvements that were implemented in the design of the HiVHAc EM thruster. The wear test results were also used to benchmark numerical erosion simulations and to gain insight into the erosion characteristics of high performance Hall thruster at voltages above 400 V. The erosion profiles of the NASA-77M boron nitride discharge channel walls indicated that

to achieve a xenon throughput of  $> 300$  kg, the thruster design had to mitigate channel erosion as a life-limiting failure mode. As such, the next laboratory thruster that was designed and tested, the NASA-103M.XL(eXtended Life), incorporated an innovation that performs discharge channel replacement during thruster operation. The discharge channel replacement mechanism results in more than three-fold increase in lifetime relative to current state-of-the-art (SOA) Hall thrusters. The performance of the NASA-103M.XL thruster was experimentally evaluated. The thruster's total efficiency varied between 0.33 and 0.55, and its specific impulse varied between 1,062 sec and 2,779 sec at power levels between 0.3 and 3.5 kW, respectively.<sup>15,16</sup> Wear testing of the NASA-103M.XL thruster was performed to demonstrate the life extending channel replacement mechanism, A 5,000 hour wear test at a discharge voltage of 700 V was performed.<sup>17</sup> Figure 3 shows a photograph of the thruster after 4,731 hours of operation. There are no plans to operate the thruster for any additional duration since the objectives of the wear testing of the NASA-103M.XL have been met.



**Figure 3: Photograph of the NASA-103M.XL thruster after 4,731 hours of testing at 700 V.**

This paper provides an overview of the flight-like, HiVHAc EM thruster development activity NASA GRC and Aerojet. An overview of recent results from performance and thermal characterization tests is presented. The functional, performance, environmental, and long duration testing plan of the HiVHAc EM thruster is outlined. Finally, evaluation of candidate options for system components of the HiVHAc thruster system are highlighted with emphasis on the power processing unit (PPU) and xenon flow system (XFS).

## **HiVHAc SYSTEM COMPONENT DEVELOPMENT AND TESTING**

After the successful demonstration and validation of the life extending channel replacement innovation with the NASA-103M.XL laboratory thruster, NASA GRC teamed with Aerojet to design, manufacture, and test a HiVHAc EM thruster. The goal of the EM thruster design and manufacturing effort is to demonstrate a technology readiness level (TRL) of 6. The EM thruster is designed to be throttleable with performance levels that meet and exceed levels achieved by the NASA-103M.XL thruster. The EM thruster design and build incorporates the life-extending channel replacement innovation. In addition, the HiVHAc EM thruster is designed to survive structural and thermal environments for representative spacecraft/mission requirements such as the deep space reference mission identified for NEXT.<sup>5</sup> To achieve the HiVHAc EM thruster design and development goals an extensive analysis, design, manufacturing, and testing plan was devised as detailed in the subsequent sub-sections.<sup>18</sup> In addition to the thruster development, the HiVHAc project has been leveraging and evaluating PPU and XFS developments that have been sponsored by other projects but that can apply directly to a HiVHAc system. The goal is to evolve the HiVHAc propulsion system to TRL 6.

### **1. HiVHAc Engineering Model Thruster**

NASA GRC tasked Aerojet with designing, manufacturing, and assembling the HiVHAc EM thruster. The goal of the design was to leverage all the experience, knowledge, and lessons learned during the development of the NASA-77M, and 103M.XL thrusters in addition to incorporating all of Aerojet's experience in manufacturing the flight qualified BPT-4000 HTPS. The NEXT environmental requirements (both structural and thermal) were used in the HiVHAc thruster design.<sup>19</sup> As for the thermal requirements, the hot environment was based on a Venus flyby mission and the cold environment is based on a distance of 4 AU from the Sun.<sup>13</sup>

### **Design, Analysis, and Manufacturing**

Details of the design, analysis, and manufacturing approach of the HiVHAc EM thruster were presented in earlier references and were performed at Aerojet.<sup>18</sup> The magnetic circuit design and modeling

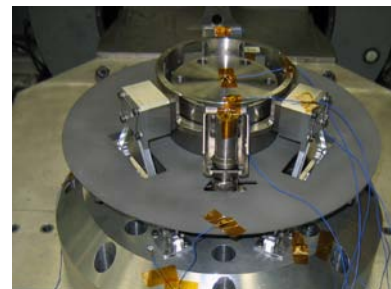
was performed at NASA GRC. Structural, thermal, and erosion modeling was performed at Aerojet. All modeling activities included use of high fidelity codes. The design and modeling tasks leveraged Aerojet's experience with the development of the BPT-4000 flight-qualified Hall thruster and results obtained from testing of the NASA-77M and the NASA-103M.XL laboratory thrusters.<sup>20</sup> Some key EM thruster design features include an integrated magnetic structure, a low cost anode design, a heritage 6.35 mm hollow cathode, a low cost propellant isolator, and a thermally efficient robust electromagnet design.<sup>16,18</sup> In addition, the gimbal/vibe simulator design leverages the low mass NEXT design by scaling down that design to accommodate mechanical interfaces to the thruster's circumference.

The EM thruster environmental structural and thermal requirements were derived from the NEXT thruster technical requirements document and are representative of the requirements for a NASA New Frontier-Class mission.<sup>19</sup> These requirements were deemed sufficiently stringent as to encompass future Discovery-Class science mission requirements. Structural analysis included assessments of the thruster and its mounting interface with a gimbal in order to support vibration testing with a mass simulator. Thermal analysis was performed to assure sufficient clearances between the various thruster components to allow for thermal expansion, to assure that appropriate materials are used, and to assure that the thermal load for the spacecraft interface is consistent with the expected thruster operating environment. Thermal modeling of the HiVHAc EM thruster indicated that the temperature of certain thruster components' will exceed their maximum design values. As such, thermal characterization of the NASA-103M.XL was performed to verify the thermal modeling results and indeed confirmed the thermal modeling findings. The elevated operating temperatures are due to the thruster power density that is approximately 1.5 times higher than SOA thrusters. To alleviate this thermal challenge, the HiVHAc EM thruster operating power could be de-rated but preliminary mission analysis confirmed that de-rating the thruster power would greatly reduce the mission capture of the HiVHAc thruster.<sup>13</sup> As such, during the design and analysis phases of the HiVHAc EM thruster, additional thermal modeling was performed to modify the thruster baseline design to reduce those components' temperatures. The design modifications included selection of component materials that enhance thermal conduction and heat flow from the highest temperature components, changes to the magnetic circuit, applying coatings to the thruster surfaces to enhance radiation, and the addition of a radiator.

The EM thruster components were manufactured and fabricated at Aerojet's Redmond facility where the flight BPT-4000 Hall thruster systems are built.<sup>21</sup> Some manufacturing work occurred at NASA GRC to support the manufacturing schedule needs. Components for two EM thrusters were fabricated. To date, one thruster has been assembled and delivered to NASA GRC. A photograph of the HiVHAc EM-1 after testing is shown in Figure 4. Prior to thruster assembly, anode flow symmetry in the discharge chamber was verified for both anode manifolds (EM-1 and EM-2) at Aerojet. Then, vibration testing of a HiVHAc mass simulator unit was performed at Aerojet. Testing results confirmed the structural simulation predictions for the mass simulator geometry and verified the structural integrity of the hollow cathode assembly design and mounting scheme. Figure 5 shows a photograph of the HiVHAc mass simulator test setup at Aerojet



**Figure 4: HiVHAc EM thruster after testing.**



**Figure 5: HiVHAc thruster mass simulator test setup at Aerojet**

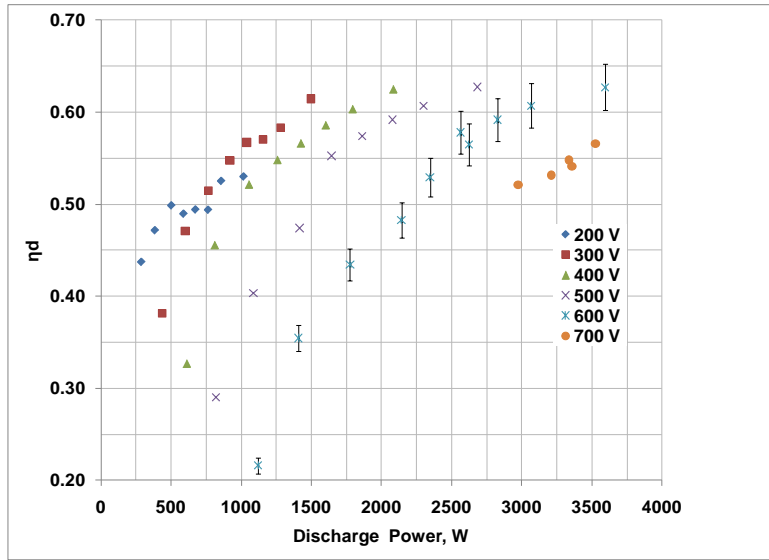
## **Performance and Thermal Characterization Tests**

Functional testing results and preliminary performance testing results of the EM thruster were reported in earlier references.<sup>22</sup> Testing at power levels above 2.5 kW was not achieved due to voltage breakdowns above 550 V and due to thermal concerns.

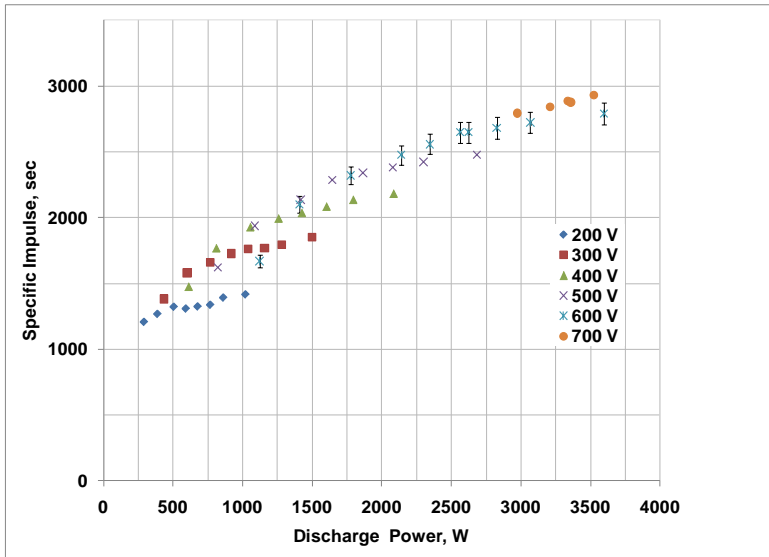
The preliminary round of thruster performance characterization (up to 2.5 kW) revealed the need to perform design modifications to the thruster's high voltage isolation schemes, discharge channel replacement mechanism, magnetic circuit, and thermal design. The design changes were necessary to ensure sustained thruster operation for the entire thruster operating power range and to ensure that the channel replacement mechanism deploys and functions as designed. To date, design changes to the high voltage isolation schemes and discharge channel replacement mechanism have been verified through extensive testing that started in the Fall of 2009 and continuing through March 2010. During these tests, thruster performance and thermal environment were mapped. In addition and concurrent with that testing, evaluation of different magnetic circuit configurations has also been performed. A laboratory hollow cathode unit was used during all tests.

The modified HiVHAc EM thruster performance was evaluated for power levels from 0.3 to 3.5 kW. Tests were performed at discharge voltage values between 200 and 700 V. Performance test results are presented in Figures 6 and 7. Results in Figures 6 and 7, as would be expected, show increased discharge thrust efficiency and specific impulse as the thruster power is increased. At a given discharge voltage, the highest efficiency and specific impulse were attained at the highest operating discharge current. At a discharge current of 5.0 A, it is noted that performance improves as the discharge voltage is increased. Discharge efficiencies of 63% at 500V and 61% at 600 V were attained. However, at a discharge voltage of 700 V, the discharge efficiency was 57% which is 5% lower than at 600 V. This reduced performance at 700 V is attributed to magnetic saturation of the thruster's backpole which results in a magnetic field profile and magnitude in the discharge channel that is not optimized for 700 V discharge voltage operation.

Thermal characterization tests were performed at various thruster operating power levels between 0.5 and 3.5 kW. Temperature measurements of critical thruster components including the inner electromagnet, anode mount, anode isolator, and radiator were performed. Temperature measurements indicated that even with the inclusion of a radiator, operating at power levels of 3.0 kW and above, resulted in the inner electromagnet temperature exceeding 450 °C which was chosen as the maximum electromagnet operating temperature



**Figure 6: HiVHAc EM discharge efficiency vs. operating power for discharge voltage operation between 200 and 700 V.**

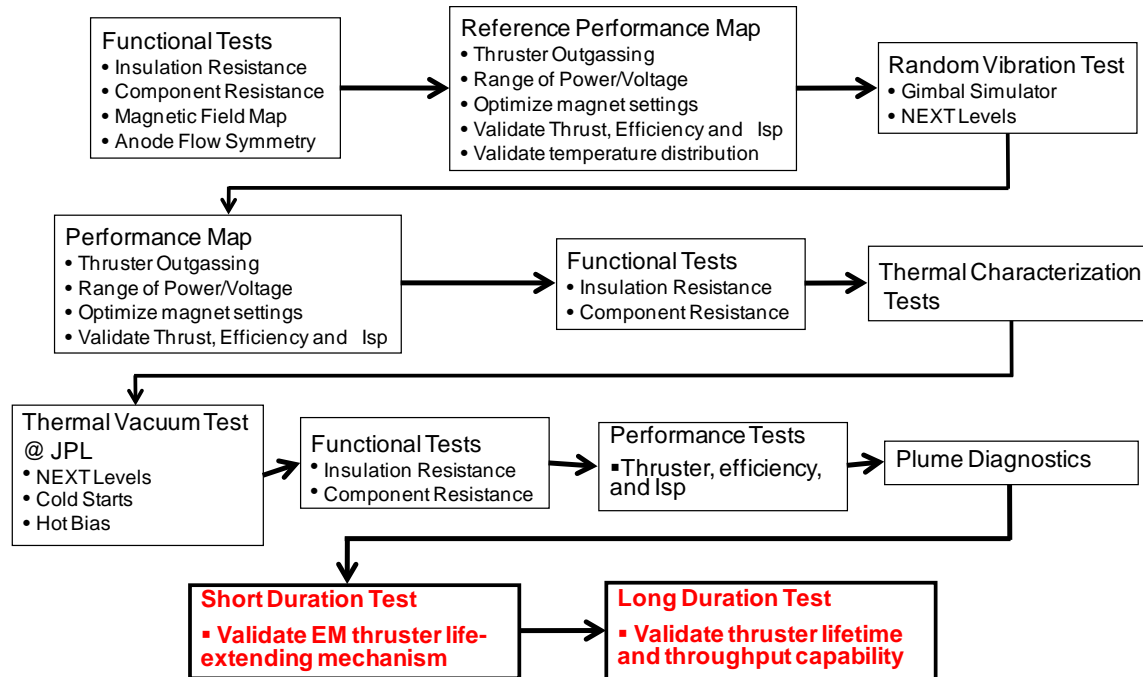


**Figure 7: HiVHAc EM discharge specific impulse vs. operating power for discharge voltage operation between 200 and 700 V.**

based on the electromagnet wire and insulation materials used. In addition, operating the thruster at a discharge voltage of 700 V required electromagnet current magnitudes that were higher than designed and resulted in an inner electromagnet temperature that was higher than 450 °C. Comparing measured components' temperatures to the thermal model predicted values indicated that the inner electromagnet temperatures were significantly higher than the thermal model prediction. However, the thermal model predicted temperatures for the other thruster components were within 10% of the measured values. This discrepancy in the prediction of the inner electromagnet temperature could be attributed to inaccuracies in predicting the self-heating profile of the inner electromagnet and might be due to erroneous boundary conditions. Additional magnetic circuit and thermal modeling is being performed to evaluate proposed design changes to the thruster magnetic circuit and thermal design to overcome the thruster's performance degradation at a discharge voltage of 700 V and to address the overheating of the inner electromagnet. These design changes will be verified during the performance acceptance mapping of the thruster.

## **HiVHAc Thruster Test Roadmap**

Figure 8 outlines the HiVHAc EM thruster testing roadmap. The goal is to mature and qualify a HiVHAc thruster to TRL 6. Once the thruster passes all the functional, performance, environmental, and a short duration test to verify the operation of the life-extending channel replacement innovation, a long duration test (LDT) will be initiated at NASA GRC.

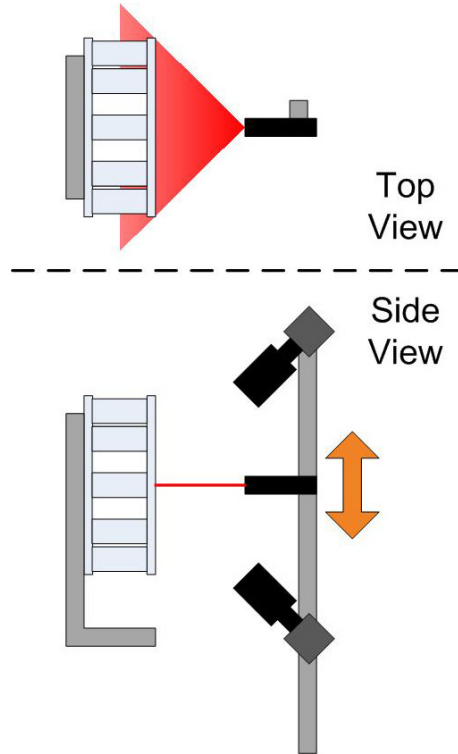


**Figure 8: Roadmap of planned tests of the HiVHAc EM thruster**

The LDT will be performed in VF-12, which is 3 m in diameter 9 m long cryopumped facility with a pumping speed of approximately 1,000,000 L/sec (air). Recent tests of VF12 indicate a base pressure of  $8.9 \times 10^{-8}$  Torr (air). A base pressure of  $3.1 \times 10^{-5}$  Torr (air) was attained at a xenon flow rate of 125 sccm. VF-12 walls are being lined with 1.3 cm thick graphite paneling to reduce the back-sputtered material flux to the thruster and test support hardware. The thruster will be mounted on an inverted pendulum thrust stand as previously used.<sup>23</sup> A number of diagnostics will be implemented during the LDT including:

- A pneumatically controlled quartz-crystal microbalance (QCM) six sensor head assembly which provides for continuity for long duration tests and allows for rotating to a backup position should a crystal fail. The QCM sensor head will be mounted next to the EM thruster to help quantify the backsputter rate and will be microscopically analyzed upon conclusion of the LDT,

- Pinhole cameras will be mounted next to the thruster exit plane to determine source of backspattered material and will be microscopically analyzed upon conclusion of the LDT,
- Quartz witness plates will be installed along the length chamber walls and will be microscopically analyzed upon conclusion of the LDT,
- Ion flux probes will be mounted on x-y translation tables. The probes will help quantify the thruster's beam divergence and how the ion flux profiles vary with channel erosion,
- Retarding potential analyzers (RPA) to measure the ion energies in the thruster plume,
- E×B probe which will be used to measure the near-field and far-field single to double ion ratio
- An in-vacuum laser profilometer to monitor the discharge channel erosion during the LDT. The laser profilometer operates by first creating a sheet of laser light that is perpendicular to the exit plane of the thruster. The intersection of this sheet of light with the thruster channel contour creates a bright line of reflected light on the surface of the thruster. The reflected light is captured by one of two cameras. A custom LABview program applies an intensity-weighted algorithm to the images combined with calibration data from a pre-measured sample to find the shape of the lit section. This procedure will be repeated at regular spatial intervals to construct a three-dimensional map of the thruster channel. The entire diagnostics will mounted on a single vertical motion stage, the laser head and cameras are retracted into a cover when not in use to prevent back sputtered material buildup on the optics. Previous use of in-vacuum laser profilometers include a two-axis laser-range-finder diagnostics used during the NSTAR LDT<sup>24</sup> and a multi-axis multi-camera telemicroscopy tool developed by Alta Spa.<sup>25</sup> Figure 9 shows a schematic of the laser profilometer setup for the HiVHAc LDT.



**Figure 9: Schematic of Laser profilometer setup for HiVHAc LDT**

## **2. Power Processing Unit Options**

The PPU is a critical component of an EP system. The HiVHAc PPU functional requirements are that it can operate over a wide power throttling range of 0.3 to 3.8 kW, over a range of output voltages between 200 and 700 V and output currents between 1.4 and 5 A as the input voltage varies over a range of 80 to 160 V. Additional environmental requirements were derived from the NEXT thruster requirements documents.<sup>19</sup>

Currently, NASA is looking at various options to perform some critical design and testing of PPU converter topologies dependent on funding availability. The near term plan is to leverage converter/PPU development by other projects where possible and applicable.

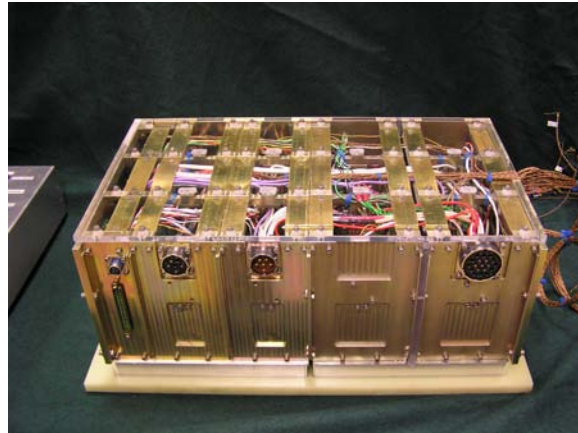
One option for developing a HiVHAc PPU is modifying the design of the BPT-4000 PPU so that it can be used to power both the HiVHAc and the BPT-4000 thrusters.<sup>26</sup> In its current configuration, the BPT-4000 PPU contains two parallel discharge modules that operate from a regulated input voltage of 70 V and up to an output voltage of 400 V. Design modifications have to be made to enable operation over wider input voltage ranges required for the HiVHAc system and to increase the output voltage to 700 V. Efforts would be synergistic with Aerojet and Air Force research Laboratory (AFRL) PPU development.

Another option is to develop a HiVHAc PPU that is a new custom design. Within NASA's small business innovative research (SBIR) program, there are three projects that are developing wide range discharge modules for integration with Hall thrusters. The SBIR projects are: the Busek Company Inc. "High Efficiency Hall Thruster Power Converter", Colorado Power Electronics Inc. "Low Cost High Performance Hall Thruster Support System", and Arkansas Power Electronics International Inc. "Silicon Carbide PPU For Hall Effect Thrusters".

Figure 10 shows a photograph of a brassboard PPU that was designed and built by CPE under a previous SBIR project.<sup>27</sup> The brassboard PPU contains two high voltage discharge modules, cathode heater and keeper power supplies, and two electromagnet power supplies. The discharge supply in this unit uses an innovative three-phase resonant topology capable of efficiently delivering full power over the wide input and output voltage ranges. The resonant transitions result in very high conversion efficiency for the entire operating range while the three-phase topology inherently reduces input and output filtering requirements.<sup>27</sup>

Earlier test results of the of the CPE discharge modules with the NASA-103M.XL thruster indicated that a discharge module efficiency of ~96% was achieved at a thruster power of 3.5 kW for an input voltage of 80 V. Additional test results were reported in reference<sup>28</sup>

Recent testing of the discharge modules was performed with an electronic load to assess the discharge module performance over a wide range of operating conditions. Testing was performed for input voltages of 80, 120, and 160 V for output voltages between 200 and 600 V. Testing at 700 V was performed with the NASA-103M.XL thruster. Results indicate improved discharge module efficiency with increased power. At an input voltage of 80 V the discharge module efficiency ranged from 90 to 96% as the output power was varied between 0.3 and 3.8 kW. At an input voltage of 120 V the discharge module efficiency ranged from 86 to 95% as the output power was varied between 0.3 and 3.6 kW. Finally, at an input voltage of 160 V the discharge module efficiency ranged from 82 to 94% as the output power was varied between 0.3 and 3.6 kW. As



**Figure 10: Photograph of the Colorado Power Electronics brassboard PPU**

such, the discharge module efficiency decreased as the input voltage was increased from 80 to 160 V. Future testing of the CPE brassboard PPU will include integrated vacuum testing with the HiVHAc EM thruster with different base plate temperatures. Integrated testing of other converter designs with the HiVHAc EM thruster will also be performed and assessed as hardware becomes available.

### **3. Xenon Flow System Options**

The HiVHAc project goal is to use a low cost light-weight XFS. A number of XFSs are available for integration with the HiVHAc thruster including the Moog flight qualified BPT-4000 XFS (TRL 9),<sup>29</sup> and the Aerojet manufactured NEXT thruster XFS which is comprised of a high and low pressure assemblies (TRL 6).<sup>30</sup> NASA's ISPT project has been investing in advanced xenon feed system technologies to significantly reduce the cost, mass, and volume while increasing the reliability over the SOA alternatives. VACCO industries was competitively selected to develop an AXFS under a NASA Research Announcement solicitation and completed their effort with the delivery of an advanced XFS and hot-fire demonstration.<sup>31</sup> The VACCO AXFS represents a dramatic improvement over the NSTAR flight feed system and also represents an additional 70% reduction in mass, 50% reduction in footprint, and 50% reduction in cost over the baseline NEXT XFS.

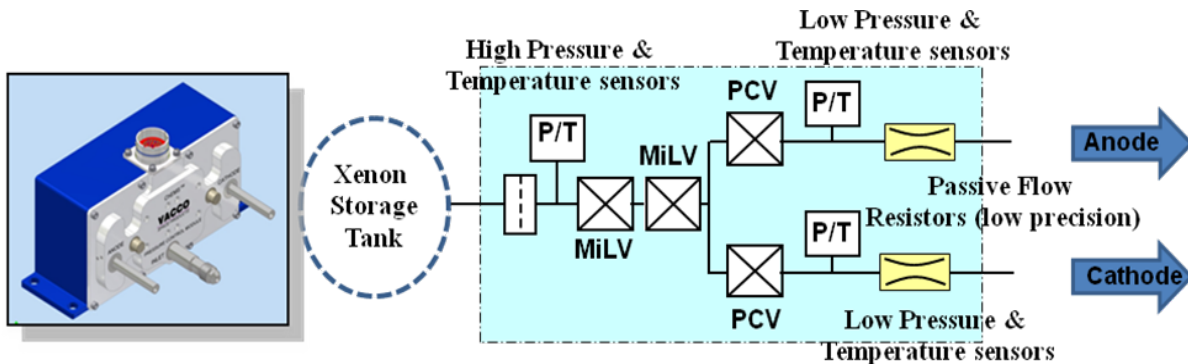
HiVHAc thruster hot-fire testing with the VACCO AXFS was performed in 2008. Thruster and XFS testing was performed for three thruster-XFS configurations to verify the XFS integrated operation with a Hall thruster. Tests results indicated an identical thruster performance for all three configurations.<sup>31</sup>

As a result of the successful testing of the HiVHAc thruster with the VACCO XFS, NASA GRC and AFRL are acquiring a flight-like VACCO xenon flow control module (XFCM) for integration with the HiVHAc thruster LDT. The goal is to use the LDT as an opportunity to qualify the VACCO flight-like XFCM over extended operation. The XFCM is designed to be a two channel electronic flow controller with a series redundancy to protect against leakage and integral pressure and temperature sensors. A Preliminary design review of the XFCM was held in June of 2010 and XFCM unit delivery is expected in December of 2010. Table 1 lists some of the specifications on the XFCM unit and Figure 11 shows a schematic of the XFCM unit.



**Table 1: VACCO XFCM Specifications**

Inlet Pressure Range	10 to 3000 psia
Anode Flow Range	0 to 100 sccm Xenon
Cathode Flow Range	0 to 6 sccm Xenon
Flow Accuracy	±3% of Full Scale (closed loop)
Internal Leakage	$10 \times 10^{-3}$ scch GHe
External Leakage	$1.0 \times 10^{-6}$ scs
Lifetime	10 years, 7,300 cycles, 100% margin
Mass	1.25 kg
Size (W×H×D)	19.5 cm × 7 cm × 7.5 cm



**Figure 11: VACCO XFCM layout**

**CONCLUSIONS AND PLANNED FUTURE WORK**

The objective of the HiVHAc thruster development activity is to fully demonstrate an HiVHAc EM thruster that has a throttling range of 12:1 for power levels between 0.3 and 3.5 kW, has a thrust efficiency > 0.55 at full power and specific impulse > 2,700 seconds (sec) at full power, and has a xenon throughput capability > 300 kg corresponding to an operational lifetime of >15,000 hrs. One EM HiVHAc thruster has been assembled and tested. The thruster design incorporates the life-extending channel replacement mechanism while being able to withstand the structural and thermal loads encountered during NASA science missions. Performance characterization tests indicate improved EM thruster performance of 10-15% over the NASA-103M.XL laboratory thruster. HiVHAc EM demonstrated discharge efficiencies up to of 63% and discharge specific impulse up to 2,930 sec. Magnetic circuit and thermal issues need to be resolved in order to attain sustained thruster operation at 3.5 kW. On-going magnetic circuit and thermal modeling and testing is resolving the magnetic and thermal issues. The modified design will be verified during HiVHAc performance acceptance mapping. Evaluation of candidate options for a PPU and XFS for a HiVHAc low-cost high-efficiency Hall propulsion system is on-going. As resources become available the development of high specific impulse Hall system would be pursued for NASA science missions.

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