

Overview of Hall Thruster Activities at NASA Glenn Research Center

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Abstract: NASA Glenn Research Center is developing low, mid, and high power Hall thrusters for implementation on NASA science and exploration missions. NASA Glenn is working with the Busek Company on incorporating a life-extending discharge channel replacement innovation onto Busek's 600 W thruster to increase its xenon propellant throughput capability by a factor of 10. Under the sponsorship of NASA Science Mission Directorate's In-Space Propulsion Technology Program, NASA Glenn and Aerojet are developing a flight-like, 3.5 kW high voltage Hall accelerator with specific impulse magnitudes of up to 2,700 s and a xenon throughput capability in excess of 300 kg. Development of the power processing unit and xenon feed system for the 3.5 kW Hall system are also being carried out. Finally, under the support of NASA's Human Exploration and Operations Mission Directorate and NASA's Office of Chief Technologist, NASA Glenn is engaging in high power Hall thruster development activities that include: high power (20-50 kW) Hall thruster testing, fabrication and testing of high current cathodes, plasma diagnostics implementation, physics-based modeling with NASA JPL, structural and thermal Hall thruster modeling, and development of roadmaps and tools for high-power discharge module technologies.

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I. Introduction

Electric propulsion (EP) systems enable and enhance NASA's ability to perform scientific space exploration.¹ Planetary science of small bodies that include fly-by, rendezvous, and sample return missions from a diverse set of targets are directed by NASA's Science Mission Directorate (SMD). To augment its capability to perform these and other solar system exploration missions, NASA continues to develop advanced EP technologies.² For example, NASA missions have successfully employed EP systems in missions like Deep Space 1 (DS1) and Dawn.^{3,4} Recent small body mission studies indicate that nearly all of the small body missions of interest are enhanced with EP; and the majority of these small body missions are enabled by the use of EP.⁵ As further benefit, EP system performance can also significantly reduce spacecraft mass and, therefore, launch vehicle costs because of its high specific impulse capability when compared to chemical propulsion.

Hall thrusters are a type of EP thruster with performance that is well suited for a number of future NASA space missions. Low power (< 1 kW) Hall thrusters enhance and enable a number of small body and Radioisotope EP (REP) class science missions. Medium power (~ 3.5 -4.5 kW) Hall thrusters enhance and enable many NASA Discovery and New Frontier class science missions, while 4.5-10kW EP systems would be most applicable to NASA New Frontiers and Flagship class missions. High power (> 20 kW) EP systems are enabling and enhancing for time critical missions or missions requiring transportation of large payloads. A number of mission studies were performed highlighting the enhancing and enabling features of high power EP systems for reusable space tug applications for the transfer of payloads from LEO to GEO and for use in Mars mission scenarios.^{6,7,8}

The NASA Glenn Research Center (GRC) is presently developing sub-kilowatt (kW), 3.5 kW, and 20-50 kW class Hall thrusters in support of the aforementioned NASA missions. Most of these development tasks are being performed at NASA GRC. This paper will provide an overview of the Hall thruster development activities that are ongoing at NASA GRC.



Figure 1: Photograph of the Busek BHT-600 thruster.

configurations of the BHT-600 in NASA GRC's Vacuum Facility 8 (VF8). The first configuration is a new baseline design incorporating an improved magnetic field design that will have an expected lifetime that is at least 5 times that of the original BHT-600. The second configuration is the BHT-600XL (eXtended Life) which

II. Long Life Sub-Kilowatt Hall Thruster Development

Mission studies have found that a sub-kilowatt Hall thruster operating at power levels between 600 and 700 W with thrust efficiencies of $\sim 50\%$ and specific impulse of up to 2,000 s can enhance and enable a number of new science missions. These candidate missions include the Trojan Rendezvous and Tour, which is enhanced with a REP system,⁹ and the Centaur orbiter¹⁰ and Saturn Ring Observer¹¹ missions which are enabled by a REP system. The performance of the Busek Co. Inc. nominal 600 W Hall thruster, designated BHT-600, is well suited for these missions. Figure 1 shows a photograph of the thruster and Table 1 presents the operating parameters and performance levels of BHT-600. Although the BHT-600 performance is well suited for the aforementioned NASA missions, predicted thruster lifetime of $\sim 2,000$ hrs is insufficient to complete most of these missions.

NASA GRC and the Busek Co. Inc. are jointly working on incorporating the NASA GRC life-extending discharge channel replacement innovation into the BHT-600 thruster design to increase the xenon throughput capability of the thruster by a factor of 10. As part of this development effort, NASA GRC and Busek Co. Inc. will wear test two new

Table 1: BHT-600 operation and performance parameters

Discharge Input Power	600 W
Discharge Voltage	300 V
Discharge Current	2.0 A
Thruster Mass Flow	2.5 mg/s
Cathode Mass Flow	0.15 mg/s
Thrust	40.8 mN
Specific Impulse	1,600 s
Propulsive Efficiency	0.52

incorporates the NASA GRC discharge channel replacement innovation with an estimated thruster lifetime of 10 times that of the original BHT-600. Both wear tests will utilize a breadboard version of the Busek Company Inc. BPU-600 power processing unit, which was flown on the FalconSat-5 mission, and a flight-like xenon feed system.¹²

NASA GRC is also supporting two Small Business Innovative Research (SBIR) Program funded activities at the Busek Co. Inc. The first is development of a low-power, low-mass, and long-life Hall propulsion system. This new system will incorporate the BHT-600XL thruster, a simplified PPU that is based on Busek's patented multi-functional converter, and simplified xenon propellant feed system. The second project is developing a high-thrust efficiency, long-life, and high thrust-to-power (T/P) version of the BHT-600 thruster. Operation at thrust efficiencies of ~50% with a T/P of ~100 mN/kW is enabling to the Saturn Ring Observer flagship mission.¹¹

III. High Voltage Hall Accelerator System Development

NASA SMD's In-Space Propulsion Technology Project (ISPT) funds new EP system development for future NASA science missions.¹³ The two primary EP elements of this project are the development of NASA's Evolutionary Xenon Thruster (NEXT) ion thruster propulsion system¹⁴ and the development of a long-life High Voltage Hall Accelerator (HiVHAc)¹⁵ as a lower cost EP option for NASA Discovery class science missions.

In 2004, mission studies found that a Hall thruster system with performance characteristics similar to the HiVHAc thruster 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 NASA Discovery class science missions.^{16,17,18,19}

Additional mission studies performed in 2009 evaluated the performance of the HiVHAc 3.5 kW thruster and a state-of-the-art (SOA) 4.5 kW flight Hall thruster.^{20,21} Four NASA Discovery class design reference missions (DRMs) were evaluated:

- Vesta-Ceres rendezvous mission (i.e., Dawn Mission) which has both time constraints and an incredibly high post launch ΔV , requiring both moderate thrust-to-power and a higher specific impulse than a conventional Hall thruster;
- Koppf comet rendezvous mission which has few constraints and is not thrusting in gravity wells (this favors a high specific impulse throttle table);
- Near-Earth Asteroid Return Earth Return (NEARER) mission; and
- Nereus sample return (NSR) mission which is a relatively low ΔV mission with time constraints, favorable for a higher thrust-to-power thruster.

Results from the mission studies indicated that the HiVHAc thruster was able to meet and exceed the needs of all the evaluated missions. For the Dawn, Koppf comet rendezvous, and NEARER missions, the HiVHAc-delivered mass was approximately 6-12% higher than the SOA Hall thruster.

The major elements of the HiVHAc propulsion system that are being developed and matured include the thruster, power processing unit (PPU), and xenon feed system (XFS). Figure 2 shows the major HiVHAc system components. HiVHAc thruster development is being performed by NASA GRC and Aerojet. For PPU development, the HiVHAc project has been leveraging and evaluating PPU developments that have been sponsored by industry and NASA's SBIR program but that can apply directly to a HiVHAc system. For XFS development, the HiVHAc project and Air Force Research Laboratory (AFRL) are furthering the development of an ISPT-funded Advanced Xenon Feed Module (AXFM) by VACCO Industries. The next sections of this paper will summarize the approach and path that the HiVHAc project is pursuing to demonstrate a Technology Readiness Level 6 (TRL 6) Hall Effect Thruster (HET) system for application in cost-capped NASA science missions.

A. Thruster Development

To demonstrate the HiVHAc project performance, throttleability, and lifetime goals, the NASA-77M and the NASA-103M.XL laboratory thrusters were built and tested. The NASA-77M demonstrated a 2.8 kW HET with a 14:1 power throttling range, operation at discharge voltages of up to 800 V, specific impulse levels between 922 and 2,911 s, and thrust efficiencies between 31% and 54%.¹⁹ The NASA-103M.XL (eXtended Life) incorporated an innovation that performs discharge channel replacement during thruster operation.^{22,23} Wear testing of the NASA-103M.XL thruster was performed to demonstrate the life-extending channel replacement innovation. The wear test demonstrated > 5,000 hours of operation, with a xenon throughput of ~100 Kg at a discharge voltage of 700 V.^{22,24}

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

engineering model (EM) thruster. The EM thruster design and manufacturing effort goal is to demonstrate a TRL 6 HiVHAc thruster. The EM thruster was designed to be throttleable with performance levels that meet or exceed levels achieved by the NASA-103M.XL thruster. The EM thruster design incorporated the life-extending channel replacement technology. In addition, the HiVHAc EM thruster was designed to survive structural and thermal environments for representative spacecraft and missions. An extensive analysis, design, manufacturing, and testing plan was devised and reported earlier to meet HiVHAc's requirements.²⁵ The design leveraged all the experience, knowledge, and lessons learned during the development of the NASA-77M and NASA-103M.XL thrusters in addition to incorporating all of Aerojet's experience in manufacturing the flight qualified BPT-4000 Hall thruster propulsion system. The NEXT environmental requirements (representative of NASA New Frontiers class mission) were used in the HiVHAc EM thruster design.²⁶ For the thermal requirements definition, the hot environment was based on a Venus flyby mission, and the cold environment was based on a distance of 4 Astronomical Unit (AU) from the Sun.²⁰

Extensive testing of the HiVHAc EM thruster (performance evaluation will be reported in the next section) revealed several areas that needed further refinement, and these included the magnetic circuit, thermal design, and discharge channel replacement mechanism. NASA GRC and Aerojet implemented design changes to the HiVHAc EM thruster, leveraging and incorporating all the lessons learned from earlier testing. The design changes included:

- a new magnetic circuit that preserved the HiVHAc EM thruster magnetic field topology while operating at lower electromagnet currents;
- a new anode isolator and anode mount design that greatly enhanced heat conduction from the anode assembly;
- new electromagnet designs that operate at lower temperatures;
- new boron nitride discharge channel configurations that are structurally more robust than the original designs; and
- a new discharge channel replacement mechanism that is simpler and more robust than original design.

The new HiVHAc thruster is designated HiVHAc EM-R (rework) and fabrication of the components necessary for performance testing has been completed. Functional and preliminary performance evaluations of the thruster has also been completed. Figure 3 shows a photograph of the HiVHAc EM-R after undergoing preliminary performance evaluation. Results of the preliminary performance evaluation will be presented in a future publication.

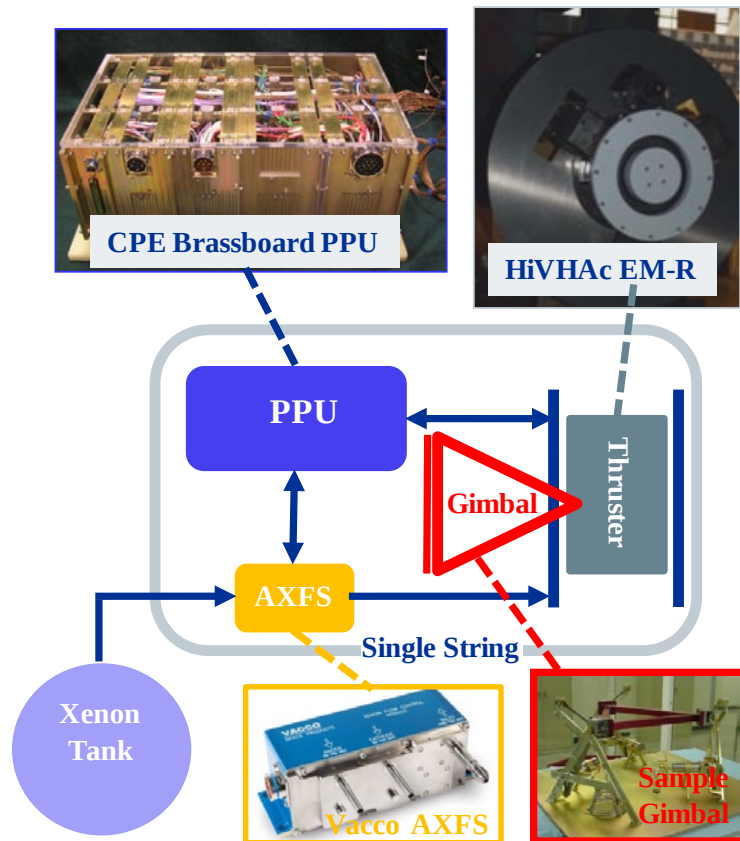


Figure 2: HiVHAc Propulsion system layout.



Figure 3: HiVHAc EM-R thruster installed in VF12.

A.1 HiVHAc Engineering Model Thruster Performance Characterization

Discharge efficiency and specific impulse results of the HiVHAc EM thruster are presented in Figures 4 and 5, respectively, and include evaluating the thruster performance at additional operating conditions from what was reported earlier.²⁷ Figures 4 and 5 present performance results for thruster discharge currents of up to 10 A and discharge powers of ~ 3.7 kW. Testing at these new throttle points was made to explore thruster operation at high T/P. During testing, the facility background pressure in the thruster's plume did not exceed 5×10^{-5} torr (corrected for xenon). A peak thruster efficiency of $\sim 59\%$ was demonstrated when operating the thruster at a discharge voltage of 400 V at 3.5 kW. A Peak specific impulse of 2,720 s was demonstrated when operating the thruster at a discharge voltage of 700 V at 3.6 kW.

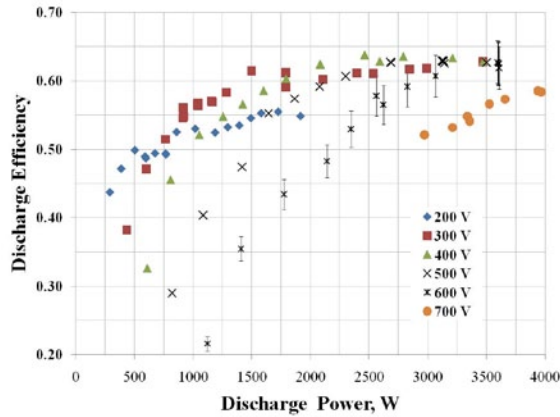


Figure 4: HiVHAc EM discharge efficiency vs. discharge power for discharge voltage operation between 200 and 700 V.

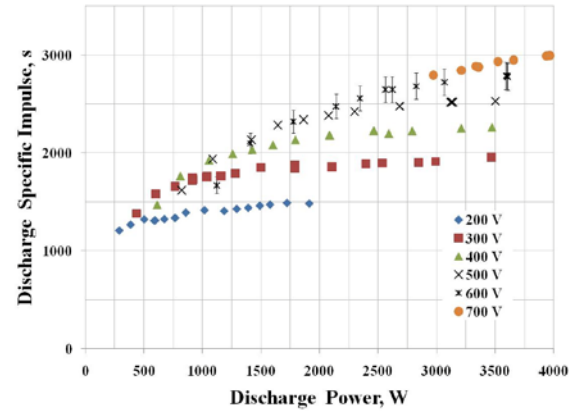


Figure 5: HiVHAc EM discharge specific impulse vs. operating power for discharge voltage operation between 200 and 700 V.

The recent HiVHAc performance results were used to generate a new HiVHAc throttle table that includes both high T/P and high specific impulse operation.¹⁵ The new throttle table will increase the HiVHAc thruster mission capture envelope. Finally, the above performance evaluation will soon be repeated for the HiVHAc EM-R thruster in Vacuum Facility 12 (VF12) at NASA GRC. VF12 is a cryopanel-pumped facility that will maintain background pressures of $< 2 \times 10^{-5}$ torr (corrected for xenon) during HiVHAc EM-R testing at the highest xenon flow rate. It is projected that the design changes incorporated in the HiVHAc EM-R will result in thruster total efficiency and specific impulse that match or exceed that of the HiVHAc EM thruster since the EM-R thruster preserved the EM magnetic circuit topology and discharge channel configuration while reducing the electromagnet power consumption. In addition, it is projected that the new throttle table for the HiVHAc EM-R thruster will include operation at a discharge voltage of ~ 700 V, which will result in a thruster specific impulse $> 2,700$ s. Finally, the thermal design changes incorporated in the HiVHAc EM-R design may result in sufficient thermal operating margin to allow the HiVHAc EM-R to operate at power levels above 3.6 kW and close to 3.9 kW.

An additional round of mission analysis was performed for the Dawn, Koppf CR, and NSR Discovery class design reference missions using the new throttle tables (see Figures 6, 7, 8).¹⁵ The HiVHAc thruster was again able to meet and exceed the needs of all the missions with improved performance for the NSR mission. The results of the three missions highlight the flexibility of the HiVHAc thruster to meet the needs of a wide range of Discovery class missions.

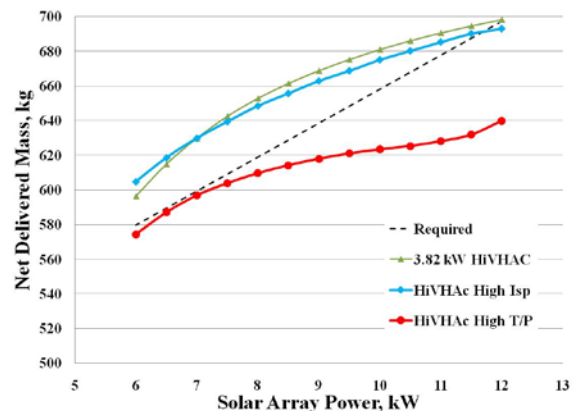


Figure 6: Performance comparison for the Dawn mission.

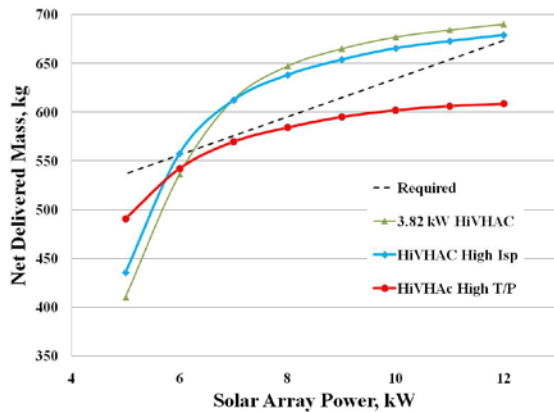


Figure 7: Performance comparison for the Kopff comet rendezvous.

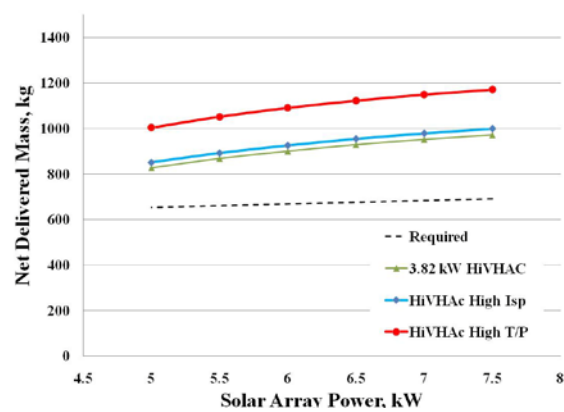


Figure 8: Performance comparison for the Nereus sample return.

A.2 HiVHAc Thruster Test Roadmap

The lessons learned from the HiVHAc EM extensive series of tests were incorporated in the design of the HiVHAc EM-R. The HiVHAc EM-R thruster has completed a series of functional tests, including insulation resistance, component resistance, and magnetic field mapping. Vibration testing in all 3 axes of a mass simulator unit representing the HiVHAc EM-R confirmed that the new boron nitride discharge channel configuration (for both inner and outer channels) can withstand the design vibration loads. Figure 9 shows a photograph of the mass simulator situated on an x-y shaker table at NASA GRC's Structural Dynamic Laboratory (SDL). The Test Readiness Review (TRR) for the HiVHAc EM-R was held on June 6. The Performance Acceptance test (PAT) of the HiVHAc EM-R will be performed in late August 2011 and is planned to be followed by a random vibration test of the thruster at NASA GRC's SDL or at Aerojet. Functional, performance, and thermal characterization tests will be performed after the random vibration test. These tests' purpose is to confirm that the thruster's operation and performance were not altered after the random vibration test. Thermal vacuum testing of the thruster will be performed at JPL and is planned for the fall of 2011. After completing the thermal vacuum test, functional and short duration (1,000-2,000 hours) testing will be performed at NASA GRC to confirm the thruster's operation and performance over an extended period of time and to confirm the discharge channel replacement mechanism operation during hot thruster firing.

After successfully completing the short duration testing a long duration test (LDT) will be initiated. The LDT will be performed at NASA GRC in VF12, which is a 3-m 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 8.9×10^{-8} torr (air). A base pressure of $\sim 1 \times 10^{-5}$ torr (air) was attained at a xenon flow rate of 125 sccm. The VF12 walls are 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 the same inverted pendulum thrust stand used for the PAT.²⁸ A number of diagnostics will be implemented during the LDT including: a pneumatically controlled quartz-crystal microbalance (QCM) six sensor head, pinhole cameras, quartz witness plates, Faraday ion flux probes, retarding potential analyzers (RPA), E×B probe, electrostatic probes, and an in-vacuum laser profilometer to monitor the discharge channel erosion during the LDT.

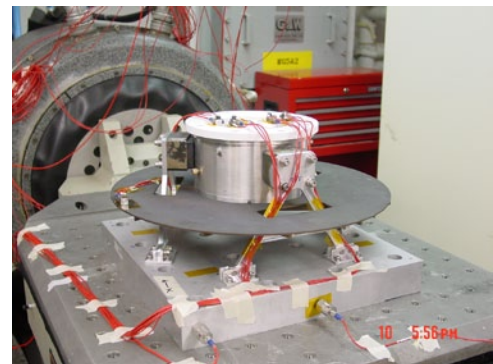


Figure 9: HiVHAc EM-R mass simulator unit undergoing testing at NASA GRC SDL.

B. Power Processing Unit Options

The HiVHAc PPU functional requirements are operating over a 0.3 to 3.8 kW throttling range and supplying output voltages between 200 and 700 V for input voltages between 80 and 160 V. Environmental requirements were derived from the NEXT thruster requirements documents.²⁶

NASA is looking at various options to perform critical designing and testing of PPU converter topologies, but this is dependent on funding availability. The near term plan is to leverage converter/PPU development through other projects. One option is to implement new discharge modules that are being developed by Aerojet.²⁹ Another option is to leverage Hall PPU developments within NASA's SBIR program. Three SBIR projects are developing wide range discharge modules for integration with HETs. The SBIR projects are the Busek Company Inc. "High Efficiency Hall Thruster Power Converter", Colorado Power Electronics (CPE) Inc. "Low Cost High Performance Hall Thruster Support System", and Arkansas Power Electronics International Inc. "Silicon Carbide PPU For Hall Effect Thrusters".

The highest maturity SBIR-related PPU technology development is a CPE designed and built PPU shown in Figure 10. The PPU contains two high voltage discharge modules, cathode heater and keeper power supplies, and two electromagnet power supplies. The 3.8 kW PPU can operate at input voltages between 80 and 160 V and is capable of output voltages between 200 and 725 V.³⁰ The unit's discharge modules use an innovative three-phase resonant topology capable of efficiently delivering full power over the wide input and output voltage ranges. Extensive testing of the CPE discharge modules was performed at NASA GRC prior to integrating the discharge modules with other PPU modules. Tests were performed with an electronic load and with the HiVHAc thrusters (NASA-103M.XL and EM); PPU tests were performed at atmospheric pressure and under vacuum. Module vacuum tests were conducted at base plate temperatures between 20 and 70 °C. The various discharge module test results verified their operation. Since integrating the discharge modules with the other power modules, the PPU has been powering the HiVHAc EM and EM-R thrusters. Latest testing of the CPE brassboard PPU includes a 400 hr uninterrupted test to confirm the extended duration operation of the unit. Figure 11 shows a photograph of the brassboard PPU installed in Vacuum Facility 70 (VF70) at NASA GRC. VF70 is a new vacuum facility that was designed for PPU vacuum testing. Integrated testing of other converter designs with the HiVHAc EM-R thruster will also be performed and assessed as hardware becomes available.

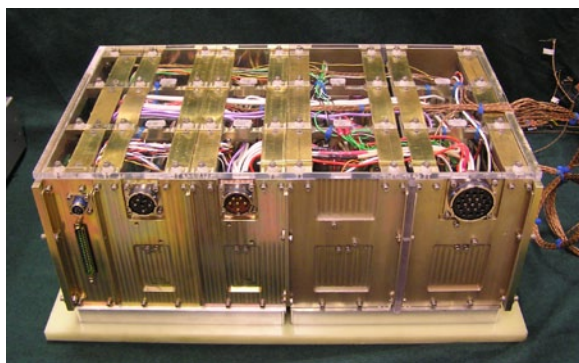


Figure 10: Photograph of the Colorado Power Electronics brassboard PPU.



Figure 11: Photograph of the CPE brassboard PPU inside VF70.

C. Xenon Feed System

In 2008, the HiVHAc thruster was hot-fire tested with VACCO's first generation XFS developed under the ISPT project.³¹ 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 AXFM for integration with the HiVHAc propulsion system and other EP devices of interest to the Air Force. The HiVHAc project plans to use the VACCO ChEMS. The AXFM is a low-cost, light-weight, low-power consumption XFS, which represents a dramatic improvement over the NSTAR flight feed system and also an additional 70% reduction in mass, 50% reduction in footprint, and 50%

reduction in cost over the baseline NEXT XFS.³¹ The AXFM is designed to be a two channel electronic flow controller with a series redundancy to protect against leakage. It includes integral pressure and temperature sensors. The unit is designed to withstand and comply with the vibration, thermal, and shock loads environments for NASA missions. The AXFM compliance with the flow accuracy, power consumption, vibration environment, shock environment, thermal environment, and minimum and maximum inlet pressure operation will be demonstrated by test. Figure 12 presents a layout and picture of the AXFM. Table 2 lists some of the AXFM specifications. Finally, VACCO will deliver the AXFM to NASA GRC around March of 2012. The AXFM will then be incorporated in the HiVHAc EM-R thruster performance, short duration testing, and LDT tests.

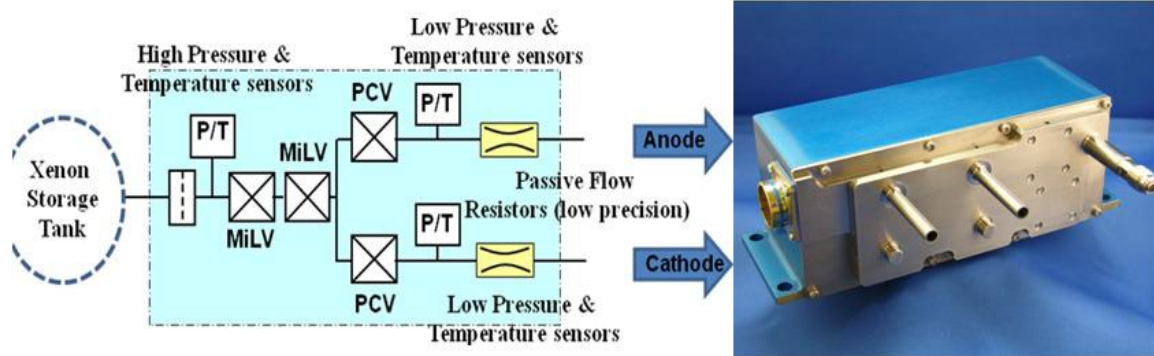


Figure 12: VACCO AXFM layout and photograph.

Table 2: VACCO AXFM Specifications

Inlet Pressure Range	10 to 3000 psia
Anode Flow Range	0 to 120 sccm Xenon
Cathode Flow Range	0 to 6 sccm Xenon
Flow Accuracy	$\pm 3\%$ of Full Scale (closed loop)
Internal Leakage	10×10^{-3} scch GHe
External Leakage	1.0×10^{-6} sccs
Lifetime	10 years, 7,300 cycles, 100% margin
Mass	< 1.25 kg
Power Consumption	< 1 W steady state
Size (W×H×D)	19.5 cm × 7 cm × 7.5 cm

IV. High Power Hall Thruster Development

National interest in high power EP systems has been renewed. In 2010, NASA's Human Exploration Framework Team (HEFT) concluded that the use of a high power (i.e. on the order of 300 kW) Solar electric propulsion (SEP) system could significantly reduce the number of heavy lift launch vehicles required for a human mission to a near earth asteroid.³² Hall thrusters are ideal for such applications because of their high power processing capabilities and their efficient operation at moderate specific impulses, which leads to reduced trip times for such missions.³³ Recent developments of high specific mass, smaller volume solar array technologies such as the Fast Access Spacecraft Testbed (FAST) concentrator array³⁴ will enhance the capability of higher power EP systems by reducing the power system size and mass.³⁵

NASA's Human Exploration and Operations Mission Directorate Enabling Technology Development and Demonstration (ETDD) Program is focused on developing, maturing, testing, and demonstrating the technologies needed to reduce the cost and expand the capability of future space exploration activities. The ETDD program content includes performing foundational research and studying of the requirements and potential designs for advanced, high-energy in-space propulsion systems. These high energy propulsion systems are intended to support deep-space human exploration and reduce travel time between Earth's orbit and future destinations for human

activity. This would enable a new space transportation capability via a SEP Stage. The SEP Stage could enable cost effective missions within Earth orbit, near earth objects (NEOs), and deep space robotic science missions. Although, the ETDD program has recently transitioned to the NASA's Office of Chief Technologist (OCT), the program content remained and is still focused on developing and maturing the high power propulsion technologies needed to enhance the agency's capabilities to explore and move large payloads in space. NASA GRC's fiscal year 2011 high power Hall development activities are focused on:

- testing high power Hall thrusters (NASA-300M and NASA-457M-v2);
- designing, fabricating, and testing a high current cathode assembly with an emission current capability >100A;
- developing plasma diagnostics for Hall thrusters;
- performing Hall thruster physics based modeling with JPL;
- performing Hall thruster structural and thermal modeling; and
- developing roadmaps and tools for high-power discharge module designs.

The following sections will provide an overview of the above activities.

A. High Power Hall Thruster Testing

Two candidate high power Hall thrusters were tested during 2010, the NASA-300M and NASA-457Mv2, which are 20 kW and 50 kW class thrusters, respectively. Extensive performance evaluation of the NASA-300M thruster was conducted, and results will be summarized in this paper. As for the NASA-457Mv2, only a preliminary performance characterization was conducted and the preliminary results will also be summarized in this paper.

A.1 NASA-300M

The NASA-300M was designed and fabricated under the support of the ESMD Exploration System Research and Technology (ESR&T) Program in 2005. The NASA-300M design is a scaled version of the NASA-457Mv2.³⁶ The NASA-300M design incorporated lessons learned from the development and testing of the NASA-457M,³⁷ NASA-400M,³⁸ and NASA-457Mv2 thrusters.³⁶ The goal of the design was to minimize thruster size while optimizing the magnetic field and plasma lens to attain improved performance. The NASA-300M nominal design specifications were a discharge power of 20 kW, a discharge voltage range of up to 600 V, a discharge current of up to 50 A, and a magnetic circuit that has a magnetic field topology similar to the NASA-457Mv2.

Testing of the NASA-300M was performed with xenon and krypton for power levels between 5 and 20 kW. Detailed performance characterization results have been reported earlier and can be found in Reference 39.

Xenon propellant testing results indicated that at 20 kW the thruster produced a peak thrust of 1.13 N. Table 3 presents the average T/P at the various operating discharge voltages. As is expected, the T/P magnitudes drop with increasing discharge voltage due to the lower flow rates necessary to achieve a given discharge power. Results in Table 3 also indicate that at a discharge voltage of 300 V the thruster may be capable of producing ~1.25 N of thrust at 20 kW. Table 4 presents discharge and total thrust efficiency and specific impulse values at maximum operating discharge power for the various discharge voltages. Results in Table 4 indicate that a peak total thrust efficiency of ~67% was achieved at a discharge voltage of 400 V and a peak total specific impulse of 2,916 s was demonstrated at a discharge voltage of 600 V. Figure 13 shows a photograph of the NASA-300M operating on xenon at a power level of 20 kW.

Table 3. Average xenon Thrust-to-Power at various discharge voltages for the NASA-300M.

V_d , V	Average T/P, mN/kW
200	68
300	63
400	56
500	50
600	47

Table 4. Xenon discharge and total specific impulse and thrust efficiency magnitudes for various discharge voltages at peak operating power for the NASA-300M.

Peak P_d , kW	V_d , V	$(I_{sp})_d$, s	I_{sp} , s	η_d , %	η , %
10	200	1,709	1,590	57	52
15	300	2,173	2,011	67	62
20	400	2,635	2,445	73	67
20	500	2,931	2,701	73	66
20	600	3,154	2,916	71	65



Figure 13: Photograph of the NASA-300M operating on xenon at 20 kW.

Krypton propellant testing results indicated that at 20 kW the thruster produced a peak thrust of 0.90 N. Table 5 presents the average T/P at the various operating discharge voltages. As is expected, the T/P magnitudes in Table 5 are lower than the values presented in Table 3 for xenon due to krypton's lower atomic mass. Table 6 presents the demonstrated discharge and total thrust efficiency and specific impulse values at maximum operating discharge power for the various discharge voltages. Results in Table 6 indicate that a peak total thrust efficiency and specific impulse of ~63% and 3,220 s, respectively, were achieved at a discharge voltage of 600 V.

Table 5. Average krypton Thrust-to-Power at various discharge voltages for the NASA-300M.

V_d , V	Average T/P, mN/kW
200	56
300	50
400	43
500	42
600	40

Table 6. Krypton discharge and total specific impulse and thrust efficiency magnitudes for various discharge voltages at peak operating power for the NASA-300M.

Peak P_d , kW	V_d , V	$(I_{sp})_d$, s	I_{sp} , s	η_d , %	η , %
10	200	1,744	1,619	47	42
15	300	2,468	2,290	60	56
20	400	2,906	2,695	64	59
20	500	3,129	2,909	64	59
20	600	3,479	3,223	68	63

A.2 NASA-457Mv2

The NASA-457Mv2 is the second version of a 50 kW-class thruster whose development was initiated in 2000 for the NASA Space Solar Power Concept and Technology Maturation Program to enable space solar power systems and other high power spacecraft.^{37,40} The first laboratory version of the 50 kW thruster, designated NASA-457M, had a 457 mm outer diameter ceramic discharge chamber and a centrally-mounted high current hollow cathode. It was fabricated and tested with xenon in 2002 at power levels up to 72 kW and at discharge voltages between 300 and 650 V. Discharge specific impulses were measured from 1750 to 3050 s with anode efficiencies between 46% and 65% on xenon.³⁷ Later tests with this thruster were conducted with a krypton propellant over a similar power range but with discharge voltages up to 1000 V to achieve even higher specific impulses.⁴¹ This thruster was further tested at low discharge voltages (100-150 V) to evaluate performance at the low specific impulse range of 800-1500 s for high thrust-to-power applications of near-earth space.⁴²

The successful test campaign with the laboratory model NASA-457M led to the development of a higher fidelity version of this thruster, labeled the NASA-457Mv2. This work was funded by NASA's In-Space Transportation Program.³⁶ This newer version retained the same discharge chamber critical dimensions, however the magnetic circuit was designed for improved performance and reduced mass. In addition, the new mechanical design eliminated deficiencies with respect to anode mounting and electrical isolation, concentricity, and thermally induced mechanical interferences. Thruster fabrication was completed in 2004 (see Figure 14 for a photograph of the thruster) and initial checkout testing was initiated thereafter. Unfortunately, thruster development funding was terminated prior to the completion of performance testing.

Performance testing of the NASA-457M v2 was re-initiated this year with funding from NASA's ETDD Program. Testing was conducted in NASA's Vacuum Facility 5 (VF5), tank pressures next to the thruster were less than 4×10^{-5} torr (corrected for xenon) during testing. The thruster was mounted on an inverted pendulum thrust stand. Thruster power was varied from 5 to 25 kW over a 200-500 V discharge voltage range and the cathode emission current was limited to 50 A for the initial and preliminary test set. Preliminary performance test results indicate that at a total thruster operating power of 25.2 kW and a discharge voltage of 500 V, the NASA-457Mv2 generated 1.28 N of thrust and had a total thrust efficiency and specific impulse of 63% and 2,520 s, respectively. For similar operating conditions (26.3 kW and 500 V), the NASA-457Mv2 generated 1.17 N of thrust had a total thrust efficiency and specific impulse of 55% and 2,350 s. This preliminary performance characterization indicates that performance gains were attained with the NASA-457Mv2 when compared to the NASA-457M.

Future development activities with the NASA-300M and NASA-457Mv2 should include performing complete thruster thermal characterizations, measuring plasma properties in the near- and far-field plume of the thruster, modeling of thruster operation using JPL developed physics based models to estimate its propellant throughput capability, and performing structural and thermal modeling of the thruster.



Figure 14: Photograph of the NASA-457Mv2.

B. High Current Cathode Development

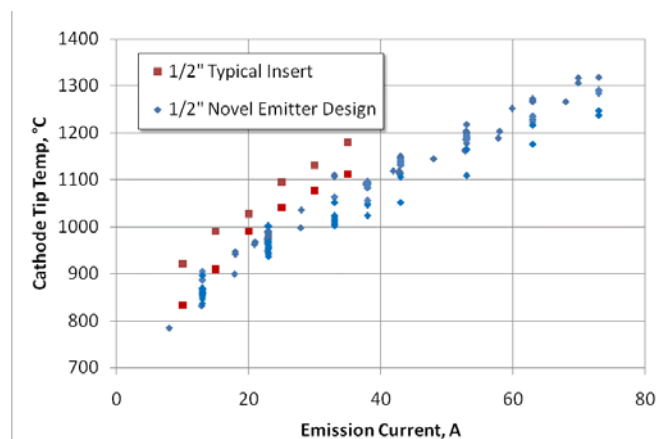


Figure 15: Comparison of 1/2" diameter cathode tip temperatures with standard and novel emitter designs.

NASA GRC is developing high current (>100 A), long-life hollow cathode assemblies (HCAs) for high power Hall thrusters. Under this task, laboratory HCAs are being designed and will be tested to emission current levels up to 150 A. The HCA designs will utilize impregnated porous tungsten thermionic emitters and will investigate implementation of novel emitter configurations. In addition, studies will evaluate various cathode orifice plate and keeper orifice plate configurations. Once an optimized configuration is selected, a flight-like unit will be fabricated, tested, and subjected to a long duration test.

Preliminary tests were performed on a novel emitter configuration that has enhanced emission area. As shown in Figure 15, this configuration allows the same 1/2" diameter cathode to operate about 50-100 °C cooler than the current SOA designs, yielding longer life for the emitter. This allows for much higher current cathodes in compact configurations.

C. Plasma Diagnostics Development

Advanced plasma diagnostics are being developed to measure thruster plasma properties that reveal the source of performance inefficiencies, characterize thruster and component life, validate numeric models, and characterize the interaction of the thruster with the test facility and potential spacecraft. Plasma measurements in the plume will consist of swept Faraday probes, stationary retarding-potential analyzers (RPA's), and stationary ExB probes. These far-field diagnostics permit the evaluation of thruster ion beam divergence and spacecraft interaction, ion energy

distribution, and relative populations of ion charge states. To perform detailed plume characterization in VF5, a large 4 degree-of-freedom probe positioning system was installed. The positioning system is capable of 5.6 m axial, 3 m radial, and 0.6 m vertical travel range, and a rotary table positioned on the vertical axis. The detailed plasma diagnostics measurements will allow us to understand potential spacecraft-plume interactions and facility impacts on high-power Hall thruster testing.

Near-field and internal plasma measurements provide the best data to resolve the causes of performance inefficiencies, and to validate thruster performance and service life models. The probe positioning system can be reconfigured to obtain near-field measurements immediately downstream of the thruster exit plane. Electrostatic probe measurements consisting of Langmuir, emissive, and ion saturation probes will provide near-field plasma number density, electron temperature, and plasma potential. The data is essential to validate the Hall2De physics-based model results.⁴³ A high-speed actuator will also be implemented to provide plasma measurements inside the Hall thruster channel. The internal measurements provide the best data for physics based model validation.

Several techniques are being implemented to assess thruster erosion rates. Optical emission spectroscopy will be used to obtain spatially resolved BN erosion rates. Multi-layer coating inserts are being considered to measure channel erosion rates at discrete locations. Internal and near-field laser-based measurements are being considered to obtain ion velocimetry near the BN channel and in the acceleration zone, Xe I and Xe II excited state density measurements, and near-field plasma density measurements.

D. Physics Based Modeling

To reduce flight risk as well as time and cost during the development and flight qualification of high-power Hall thrusters, NASA JPL and GRC will employ physics-based numerical simulation, plasma and wear diagnostics, and thruster testing by combining unique capabilities at the two centers. Modeling of the NASA-300M and NASA-457Mv2 will be performed. The NASA-300M was modeled first because its performance, with propulsive efficiencies as high as 67%, has been fully characterized.

A first series of numerical simulations of a 20-kW class Hall thruster have been completed to establish the groundwork for the development of high-power, high-performance, long-life Hall thrusters. The numerical simulations have been performed with Hall2De, a 2-D axisymmetric, magnetic-field-aligned computational mesh (MFAM) plasma solver developed at JPL.^{43,44} It has recently been exported to GRC. Detailed results from the computational work will be presented in a companion paper.⁴⁵ Simulation results performed a first assessment of the erosion rates in the existing design of the NASA-300M. Figure 16 shows the 2-D contours of the number density in the thruster's acceleration channel. The simulation results indicated that the ionization zone in NASA-300M is located far upstream of the channel exit and that significant ion focusing is achieved due largely to the applied magnetic field topology. This combination leads to erosion rates along the channel insulators that do not exceed 1 mm/kh. Recent findings on erosion physics and related wear mitigation techniques suggest it may be possible to modify the magnetic circuit and channel geometry of this thruster to achieve "magnetic shielding," a technique that seeks to reduce erosion rates at the channel walls by more than one order of magnitude. Extended simulations that include assessment of thermal loads to the thruster surfaces, plasma diagnostics and model validation, thruster modifications and testing constitute the focus of this effort in the next several months.

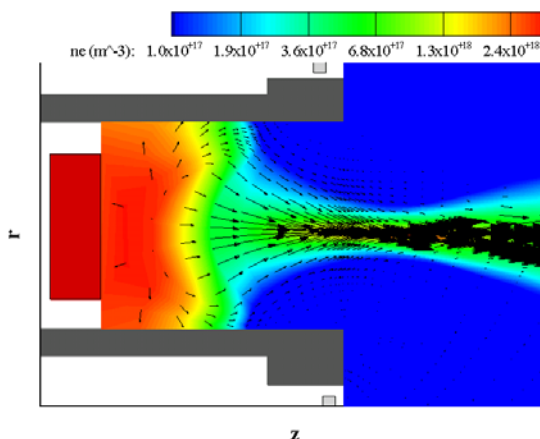


Figure 16: Two-dimensional contours of the number density in the NASA-300M discharge channel.⁴⁵

E. Structural and Thermal Modeling

Advancing Hall thrusters to flight design quality requires ensuring structural and thermal integrity of the thruster unit. A Structural analysis is currently being performed on the NASA-300M to evaluate the response of the thruster under launch acceleration and vibration loads. Shock events, as would result from the firing of pyrovalves, are also being evaluated. Results from this analysis and a similar analysis previously performed on the NASA-457Mv2 thruster will be verified from vibration testing scheduled to occur next year. Thermal modeling of the NASA-300M is also being characterized to evaluate the thruster under deep space cold conditions and also under worst-case hot conditions—in LEO with solar and Earth albedo heat fluxes while firing. This analysis will help reduce the risk of thermally-induced loads on the thruster design. The NASA-300M is planned to be tested with thermocouples to establish thermal characterization to help verify the thermal model. The lessons learned from the structural and thermal analyses and testing will aid in the understanding and improvement of existing thruster designs and also provide guidelines for future thruster development.

F. Discharge Module Roadmap and Tool Development

The development of high power electric propulsion systems imposes particular challenges to the PPU. The real challenge is not if a PPU can be built but if it can be made efficient, lightweight and cost effective enough for the application. All electric propulsion flight and EM PPUs developed to date process less than 7 kW of power.^{29,46} The larger power supplies within these units consist of multiple DC-DC converters, ranging from 0.5 to 2.3 kW of power, operating in parallel to supply power to the thrusters. The modular approach maintains power losses and electrical stress under control. However, these have a limited power processing capacity because the radiation-hardened, flight-rated power transistors currently available limit the amount of power that can be efficiently processed by a single converter. One viable option is a DC-DC converter and can process multiple kilowatts. This will reduce the number of modules, parts count, complexity and cost of the PPU and still yield the benefits of a modular approach. However, in order to implement this type of design, high power components, including transistors and diodes, will be necessary.

Power processing units developed to date operate from an input voltage of no more than 160 V. Future high-power spacecraft will likely have high-voltage solar arrays and power distribution systems to reduce conduction losses and mass. This further complicates the design of a high power PPU because key components like SOA radiation-hardened, flight-qualified silicon MOSFETs are available at maximum ratings of only 500 V and less than 20 A, which is only sufficient to develop kW-class converters. To develop a high power PPU, new flight-qualified, radiation-hardened, high-power, and high-voltage semiconductors will be necessary.

For the last few decades, the silicon MOSFET has been the transistor of choice for DC-DC converters. Although the current selection of flight-qualified parts is very limited, many commercial-grade parts are available for high-voltage and high-power ratings. Also, very recently, silicon carbide (SiC) and gallium nitride (GaN) solid-state technologies have successfully been developed for commercial-grade, enhancement-mode MOSFETs. Although still not available at optimum voltage and power levels for the application, these parts appear to have excellent performance and rival silicon MOSFETs. A development effort for high power semiconductors for a high-power PPU must consider these parts.

Another challenge of a high-power PPU is developing an optimal electrical design that yields maximum efficiency and minimum mass, parts count, complexity, risk and cost. A comprehensive trade study including power level, transistor choice, switching frequency, circuit topology and input/output voltage is underway. To facilitate this, models and software tools are currently under development at NASA GRC to predict converter mass and switching and conduction losses for various design solutions. Component testing, including silicon MOSFETs, IGBTs, SiC MOSFETs and GaN MOSFETs is conducted to obtain performance data, switching and conduction losses at a variety of switching frequencies, voltages and currents, to validate the efficiency models for the power transistors. Once perfected, these software tools will be critical to select the best transistors and converter topologies for future high power PPU applications from tens to hundreds of kW.

V. Summary

NASA Glenn Research Center is developing low, mid, and high power Hall thrusters for implementation in NASA science and exploration missions. NASA Glenn is working with the Busek Company on incorporating a life-extending discharge channel replacement innovation onto Busek's BHT-600 600 W thruster to increase its xenon propellant throughput capability by a factor of 10. This new thruster will enhance NASA's ability to perform a number of radioisotope electric propulsion missions.

Under the sponsorship of NASA's Science Mission Directorate In-Space Propulsion Technology Project, NASA Glenn and Aerojet are developing a flight-like 3.5 kW high voltage Hall accelerator with specific impulse magnitude capability up to 2,700 s and a xenon throughput capability in excess of 300 kg. The HiVHAc EM-R thruster will soon undergo performance acceptance and environmental testing. At the conclusion of these tests, short duration testing will be performed to confirm the life extension technology operation during thruster hot-firing. Afterwards, a long duration test will be initiated. For the PPU development, NASA GRC is leveraging HET PPU developments by Aerojet and within NASA's SBIR program. The highest maturity SBIR program produced PPU is a brassboard unit that was designed and built by Colorado Power Electronics. The PPU is currently undergoing testing at NASA GRC. Finally, for the xenon feed system, NASA GRC is acquiring a VACCO AXFM unit that represents a great improvement over SOA XFS with its low-cost, low-mass, and low-power consumption.

Finally, under the support of the NASA's Human Exploration and Operations Mission Directorate and NASA's Office of Chief Technologist, NASA GRC is engaging in high power Hall development activities that include: high power (20-50 kW) Hall thruster testing, high current cathodes fabrication and testing, plasma diagnostics implementation, physics based modeling with NASA JPL, structural and thermal Hall thruster modeling, and development of roadmaps and tools for high-power high-discharge module designs. To date, NASA GRC has completed the performance characterization of the NASA-300M, performed preliminary performance characterization of the NASA-457Mv2, designed and initiated the fabrication of a high current cathode, implemented new plasma diagnostics capabilities in VF5, performed physics-based simulations of the NASA-300M, and developed tools and roadmaps for designing and testing high power discharge modules and components.

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