Overview of NASA's Electric Propulsion Development Activities for Robotic Science Missions

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Abstract: The In-Space Propulsion Technology Program (ISPT) is responsible for the development of electric propulsion technologies for NASA robotic scientific missions. The ISPT Program is managed at the NASA Glenn Research Center on behalf of the Science Mission Directorate. The objective of the Electric Propulsion project area is to develop near-term electric propulsion technology to enhance or enable science missions while minimizing risk and cost to the end user. Major hardware tasks include developing NASA's Evolutionary Xenon Thruster (NEXT), developing a long-life High Voltage Hall Accelerator (HIVHAC), and developing an advanced feed system. The objective of the NEXT task is to advance next generation ion propulsion technology readiness. The baseline NEXT system consists of a high-performance, 7-kW ion thruster; a high-efficiency, 7-kW power processor unit (PPU); a highly flexible advanced xenon propellant management system (PMS); a lightweight engine gimbal; and key elements of a digital control interface unit (DCIU) including software algorithms. This design approach was selected to provide future NASA science missions with the greatest value in mission performance benefit. The objective of the HIVHAC task is to advance the Hall thruster technology readiness for science mission applications. The task seeks to increase Hall thruster specific impulse, throttle-ability and operational life and thus broaden Hall propulsion system applicability to low-cost, deep space science missions. The primary application focus for the resulting Hall propulsion system would be cost-capped missions, such as competitively-selected, Discovery-class missions. The objective of the advanced xenon feed system task is to demonstrate novel manufacturing techniques that will significantly reduce mass, volume, and footprint size of xenon feed systems over conventional feed systems. This task has focused on the development of combined flow control and pressure control module, which regulates pressure from the propellant tank and provides two-channel flow control for Hall propulsion systems. Progress on current hardware development, recent test activities and future plans are discussed.

Nomenclature

ASOA	= Advanced State-Of-Art
AXFS	= Advanced Xenon Feed System
DCA	= Discharge Cathode Assembly

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DCIU	= Digital Control Interface Unit
EM	= Engineering Model
EMC	= Electromagnetic Compatibility
EMI	= Electromagnetic Interference
FCM	= Flow Control Module
GRC	= Glenn Research Center
HIVHAC	= High Voltage Hall Accelerator
HPA	= High Pressure Assembly
IPS	= Ion Propulsion System
ISPT	= In-Space Propulsion Technology
JPL	= Jet Propulsion Laboratory
LDT	= Long Duration Test
LPA	= Low Pressure Assembly
MFC	= Mass Flow Controller
mN	= milli-Newton
MTAT	= Multi-Thruster Array Test
NCA	= Neutralizer Cathode Assembly
NEXT	= NASA's Evolutionary Xenon Thruster
NRL	= Naval Research Laboratory
NSTAR	= NASA's Solar Electric Propulsion Technology Application Readiness
PAT	= Performance Assessment Test
PCM	= Pressure Control Module
PDR	= Preliminary Design Review
PM	= Prototype Model
PMS	= Propellant Management System
PPU	= Power Processor Unit
PV1	= Piezoelectric Valve 1
SBIR	= Small Business Innovative Research
SMD	= Science Mission Directorate
SOA	= State-Of-Art
TRL	= Technology Readiness Level

I. Introduction

NASA's Science Mission Directorate (SMD) conducts scientific exploration that is enabled by access to space. NASA's SMD is organized by mission themes that focus on investigations of the Earth, Sun, Solar System and Universe. The focus of solar system exploration is to extend humanity's knowledge about the solar system through robotic encounters to the other planets and their moons, to asteroids and comets, and to icy bodies in the outer reaches of our solar system. The progression of robotic missions is from observers to rovers to sample return missions. Each step brings us closer to the principal scientific goals: to understand our origins, to learn whether life does or did exist elsewhere in the solar system and to prepare for human expeditions to the Moon, Mars, and beyond.[1] Such ambitious goals exceed the capabilities provided by conventional technologies and will ultimately require improved spacecraft capabilities such as those obtained by advanced propulsion technologies.

Within SMD the In-Space Propulsion Technology (ISPT) Program is responsible for developing advanced propulsion capabilities to enable or to enhance science missions. NASA Glenn Research Center (GRC) is responsible for managing the ISPT Program for SMD. Previously ISPT has focused on the development of advanced chemical propulsion, aerocapture, and electric propulsion, while future investments will add to the portfolio the development of propulsion systems for sample return missions.[2] Sample return mission technologies include developing capabilities for planetary ascent vehicles, Earth-return vehicles, and Earth-entry vehicles.[3] Investments in electric propulsion have focused on completing the NEXT ion propulsion system, a throttle-able gridded ion thruster propulsion system suitable for future NASA Flagship, New Frontiers, and Discovery missions. In addition a novel Hall thruster concept is being developed to demonstrate a long-life, highly throttle-able thruster ideally suited for cost capped missions, like NASA Discovery missions. A novel feed system is being developed, which will significantly reduce the mass and size of future electric propulsion xenon feed systems. In addition to hardware development activities, ISPT performs mission analysis to assess system-level benefits in applying advanced propulsion technologies into robotic science missions.[4]

ISPT has targeted technologies for near-term, robotic, interplanetary science missions. Given the mission constraints, like available spacecraft power and performance requirements, ISPT has focused on the development of electrostatic thrusters, such as ion thrusters and hall thrusters. This paper will elaborate on recent hardware development activities within the ISPT electric propulsion portfolio as well as recent progress toward technology infusion.

II. NASA'S Evolutionary Xenon Thruster (NEXT) Ion Propulsion System

A. Description

The objective of the NEXT project is to advance next generation ion propulsion technology readiness. As shown in Fig. 1, the NEXT ion propulsion system (IPS) consists of a high-performance, 7-kW ion thruster; a high-efficiency, 7-kW power processor unit (PPU); a highly flexible xenon propellant management system (PMS); a lightweight engine gimbal; and key elements of a digital control interface unit (DCIU) including software algorithms.[5-8] The NEXT team consists of NASA GRC as technology project lead, JPL as system integration lead, Aerojet as Prototype Model (PM) thruster, PMS, and DCIU simulator developer, and L3 Communications ETI as PPU developer. This design approach was selected to provide future NASA science missions with the greatest value in mission performance benefit.

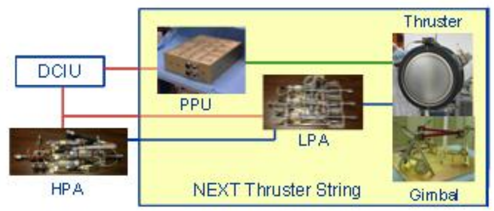

Figure 1. NEXT Ion Propulsion System Components

The NEXT thruster and other component technologies represent a significant advancement in technology beyond state-of-art (SOA) NSTAR thruster systems. NEXT performance exceeds single or multiple NSTAR (NASA's Solar Electric Propulsion Technology Application Readiness) thrusters over most of the thruster input power range. Higher efficiency and specific impulse, and lower specific mass reduce the wet propulsion system mass and parts count. The NEXT thruster xenon propellant throughput capability is more than twice NSTAR's, so fewer thrusters are needed. The NEXT power processor and propellant feed system technologies provide specific mass and performance benefits versus SOA technology, which translate into better science capability for a given spacecraft or mission. Comparisons of NEXT and SOA NSTAR performance characteristics are listed in Table 1.

Table 1. Design Performance Characteristics of NEXT vs. SOA Ion (NSTAR).

Characteristic	NEXT	SOA Ion
Thruster Power Range, kW	0.5-6.9	0.5-2.3
Throttle Ratio	>12:1	4:1
Max. Specific Impulse, sec	>4100	>3100
Max. Thrust, mN	236	92
Max. Thruster Efficiency	>70%	>61%
Propellant Throughput, kg	>300	157
Thruster Specific Mass, kg/kW	1.8	3.6
Max. PPU Efficiency	95%	92%
PPU Specific Mass, kg/kW	4.8	6.0
PMS Single-String Mass, kg	5.0	11.4

B. Recent Progress

The Prototype Model (PM1) thruster exhibited operational behavior consistent with its engineering model predecessors, but with substantial mass savings, enhanced thermal margins, and design improvements for environmental testing compliance.[9] A study of the thruster-to-thruster performance dispersions quantified a bandwidth of expected performance variations both on a thruster and a component level by compiling test results of five engineering model and one-flight-like model thrusters.[10] The thruster throughput capability was predicted to exceed 750 kg of xenon, an equivalent of 36,500 hours of continuous operation at full power.[11] The first failure mode for operation above a specific impulse of 2000 s is expected to be the structural failure of the ion optics at >750 kg of propellant throughput, 1.7 times the qualification requirement.[12] A review of life assessment predictions examining wear mechanisms at various throttle conditions was completed.[12] A Long-Duration Test (LDT) was initiated to validate and qualify the NEXT propellant throughput capability to a qualification-level of 450 kg, 1.5 times the mission-concept derived throughput requirement of 300 kg. As of July 31, 2011, an Engineering Model (EM) thruster has accumulated 37,073 hours of operation. The thruster has processed 633 kg of xenon and demonstrated a total impulse of 2.39×10^7 N-s; the highest total impulse ever demonstrated by any electric propulsion thruster.[13] An image of the LDT thruster in operation is shown in Fig. 2. Test results have been published for the performance, plume, and wear characteristics. Thruster performance parameters including thruster, input power, specific impulse, and thruster efficiency have been nominal with little variation to date.[13-14] Thruster plume diagnostics and erosion measurements have been obtained periodically over the entire NEXT throttle table. Observed thruster component erosion rates are consistent with predictions and the thruster service life assessment.[15]

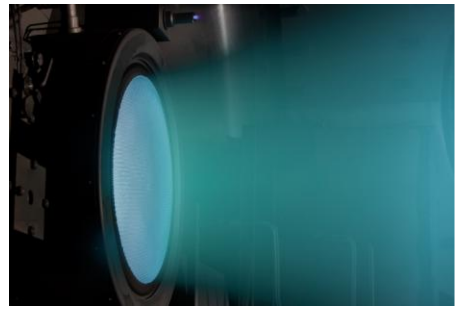

Figure 2. NEXT EM3 Long Duration Test Thruster.

The NEXT Prototype Model (PM1) was incorporated in a short duration wear test, in which the thruster was operated for a total of 1680 hours and processed 30.5 kg of xenon. Overall ion engine performance, which includes thrust, thruster input power, specific impulse, and thrust efficiency, was steady with no indications of performance degradation. The ion engine was also inspected following the test with no indication of anomalous hardware degradation.[16]

The NEXT PM1 thruster was subjected to qualification-level environmental testing to demonstrate compatibility with environments representative of anticipated mission requirements. Thruster functional testing was performed before and after the vibration and thermal-vacuum tests. Random vibration testing, conducted with the thruster mated to the breadboard gimbal was executed at 10.0 Grms for two minutes in each of three axes. Thermal-vacuum testing included a deep cold soak of the engine to temperature of -168 °C and thermal cycling from -120 °C to +215 °C.[16] Thermal development testing of the PM1 thruster was conducted to assist in developing and validating a thruster thermal model and assessing the thermal design margins.[17] An ion thruster thermal model has been developed for the latest PM design to aid in predicting thruster temperatures for various missions. This model has been correlated with a thermal development test on the NEXT PM1 thruster with most predicted component temperatures within 5-10 °C of test temperatures.[18] The PM1 thruster was subjected to the series of qualification-level environmental tests as shown in Fig. 3. Post-test performance assessment test and inspection have shown that thruster performance was nominal and unchanged throughout the test and at post-test conditions, which completes environmental test validation of the PM1 thruster.[19]

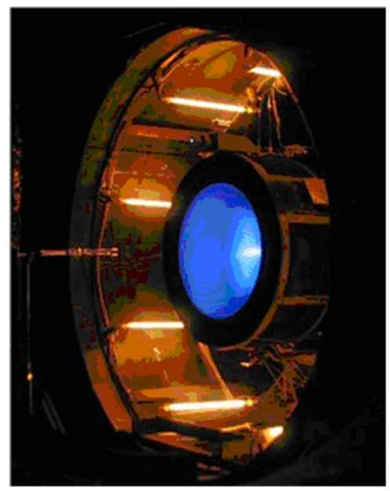

Figure 3. NEXT PM1 Environmental Test.

The NEXT Engineering Model PPU is a modular design capable of very efficient operation through a wide voltage range because of innovative features like dual controls, module addressing, and a high current mode. The modular construction of the PPU resulted in improved manufacturability, simpler scalability, and lower cost relative to Dawn designs. The efficiency was measured in benchtop tests to range from 94.1 to 82.6 percent over the input power range.[20] The EM PPU demonstrated most requirements and stable operation with a thruster. However, throughout the development tests, the PPU experienced a series of unexpected problems and component failures. Formal failure investigations were conducted to identify the root cause of these failures and recommend corrective actions.[21]

The EM Propellant Management System (PMS) delivers low pressure gas to the thruster from a supercritical xenon supply source, and it consists of a High Pressure Assembly (HPA) and a Low Pressure Assembly (LPA).[8] The PMS provides independent xenon flow control to the thruster main discharge, and discharge and neutralizer cathodes. Aerojet completed manufacturing of the EM PMS elements, including 2 HPAs (one flight-like) and 3 LPAs (one-flight like). All assemblies have completed functional testing, and both flight-like HPA and LPA assemblies successfully completed qualification-level vibration and thermal vacuum testing.

Other thruster components under development include high voltage isolators, and heaters. The high-voltage isolator has been undergoing a life test to quantify leakage currents at full voltage to verify operation of this component over anticipated life. To date the NEXT isolators have accumulated over 26,800 hours of operation. Measurements indicate a negligible increase in leakage current during the test. Cyclic testing of multiple heaters was initiated to validate these modified fabrication processes while retaining high reliability heaters. Multiple heaters have been cycled to failure giving a service life twice that established for qualification space station plasma contactor heaters.[22]

A multi-thruster array test (MTAT) was beneficial to address thruster and gimbal-specific questions that drive the configuration of the IPS components as shown in Fig. 4. This MTAT utilized multiple engineering model (EM) NEXT ion thrusters as well as laboratory power consoles and laboratory propellant feed systems to operate multiple thrusters simultaneously. The engineering demonstration portion of MTAT[23] focused on the characterization of performance and behavior of the individual thrusters and the array as affected by the simultaneous operation of multiple ion thrusters. The MTAT physics effort focused on the characterization of the plasma environment generated by the simultaneous operation of multiple ion thrusters. The interaction of this plasma environment with the spacecraft and the thrusters themselves plays an important role in the determination of spacecraft configuration, acceptable array operating condition, and array lifetime. Published papers document ion beam characterization,[24] array local plasma,[25] electron flowfield characteristics of the plume,[26] and neutralizer coupling characteristics.[27]

Figure 4. NEXT Multi-Thruster Array Test.

A System Integration Test has been completed in which the PM thruster, EM PPU, EM PMS, and DCIU simulator were operated together in a single-string configuration. The purpose of the test is to demonstrate functionality and characterize operational capabilities of a complete string of NEXT components. The single-string configuration was operated over 17 throttle points to demonstrate compatibility of the individual components over the range of operating conditions. A multi-thruster configuration was also operated to demonstrate three-string functionality of the PMS.[28]

An area in which further NEXT work has been needed is that of precise plume, particle, and field characterization. A non-reimbursable Space Act Agreement (SAA) was drafted by NASA and The Aerospace Corporation to establish a collaborative measurement program intended to examine the plume, particle, and field environments of the latest generation NASA ion propulsion technology. A series of measurements has been completed to verify basic characteristics of NEXT operation and expand on the available public-domain and internal databases regarding NASA technology and its potential use on non-NASA spacecraft systems.[29] Figure 5 shows the NEXT thruster installed in the vacuum facility at The Aerospace Corporation. Among the work elements planned are in-depth EMI/EMC, plume particle and plasma probe, optical emission and laser diagnostic measurements. This work is of considerable relevance to future spacecraft integration of the subject thrusters.

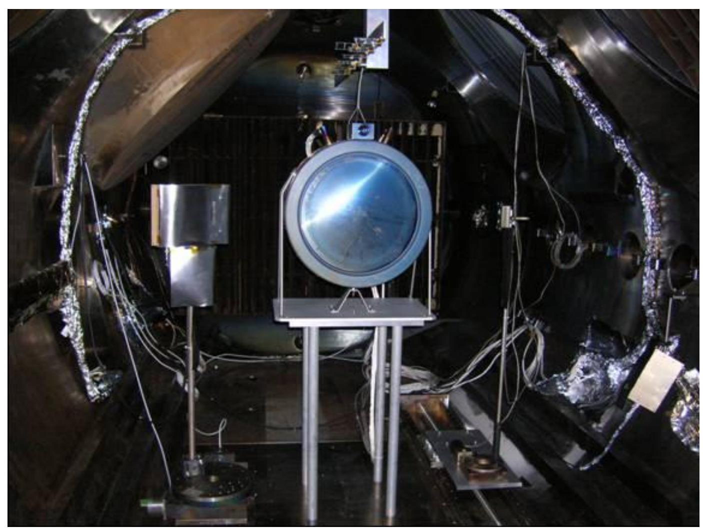

Figure 5. Photograph of NEXT thruster installed in The Aerospace Corporation vacuum facility.

The NEXT evaluation at Aerospace also includes measurement of ion beam flux and divergence, charge state ratios, charge exchange ion flux, plume optical emission spectrum and absolute flux, radio frequency and microwave absolute emission spectrum plus time-domain emissions, carrier wave attenuation and phase effects, plume erosion and molybdenum contamination effects, absolute thrust and thrust correction factors. Plume characterization tests with the NEXT ion thruster were performed using the EM and PM thrusters. Examinations of the beam current density and xenon charge-state distribution as functions of position on the accelerator grid have been completed.[30] The angular dependence of beam current was measured at intermediate and far-field distances to assist with plume modeling and to evaluate the thrust loss due to beam divergence. Thrust correction factors were derived from the data.[30] Transmission and phase noise measurements were made through the plume of an EM NEXT ion thruster.[31] Attenuation measurements were taken at multiple operating points at frequencies between 1 and 18 GHz. Attenuation was observed between 1 and 3 GHz and scaled with plasma density.[31] Phase noise spectra were also taken. Direct thrust measurements have been made on the NEXT PM ion thruster using a standard pendulum style thrust stand constructed specifically for this application.[32-33] Values have been obtained for the full 40-level throttle table as well as for a few off-nominal operating conditions.[32-33]

A particle-based model with a Monte Carlo collision model has been developed by Wright State University (WSU) to study the plasma inside the discharge model of the generic ion thruster. This model tracks five major particle types inside the discharge chamber in detail: xenon neutrals, singly and doubly charge xenon ions, secondary electrons and primary electrons.[34] Both electric and magnetic field effects are included in the calculation of the charged particle's motion. Validation of this computational model has been made with comparisons to the NSTAR discharge chamber. Comparison of numerical simulation results with experimental measurements were found to have good agreement.[34] The model has been applied to the NEXT discharge chamber design at multiple thruster operating conditions.[35-39]

C. Future Plans

NEXT project activities have brought next-generation ion propulsion technology to a mature state, with existing tasks completing the majority of the NEXT product technology validation. Functional and appropriate qualification-level environmental tests of the EM PPU are anticipated as the PPU recovery efforts conclude. The thruster life test will continue to accumulate additional throughput on the EM hardware beyond the project goal of 450 kg. ISPT funding for the LDT continues through FY12, with the aim of demonstrating up to 750 kg of xenon throughput. A Project Validation Review will be conducted at the conclusion of Phase 2 in late 2012, during which implementation risks will be identified and prioritized. A framework for that review has been published,[40] which provides the Technology Readiness Level (TRL) definitions, hardware maturity, and relevant environment definition.

III. High Voltage Hall Accelerator (HIVHAC) Thruster

A. Description

The focus of the HIVHAC thruster development task has been to develop a 3.5 kW Hall thruster with increased specific impulse, throttle-ability and lifetime to broaden Hall propulsion system applicability to deep space science missions. The primary application focus for the resulting Hall propulsion system would be cost-capped missions.[41] The project is led by NASA GRC teamed with Aerojet and JPL. The needs of many targeted robotic science missions exceed the throughput capability achievable without advanced development of Hall electric propulsion systems. Several different approaches to increasing HIVHAC performance and propellant throughput have been evaluated. Improved thruster performance and throttle-ability have been achieved by employing a new magnetic circuit design and by operating the thruster at higher power densities compared to state-of-the-art. Increased thruster lifetime has been achieved by employing a life extending innovation that mitigated channel erosion as a life limiting mechanism.

This throughput capability must be achieved at a discharge voltage of 700 Volts. The high voltage operation allows the thruster to operate at specific impulses much higher than conventional Hall thrusters. A comparison of the HIVHAC thruster to a conventional, SOA Hall thruster is shown in Table 2. To demonstrate the performance and lifetime improvements over SOA, a laboratory-model thruster, designated NASA-103M.XL, has been fabricated, tested, and demonstrated critical functionality to proceed into the engineering model development phase. The NASA-103M.XL development approach was a less traditional approach to extending thruster

Table 2. Performance Characteristics of HIVHAC vs. SOA Hall (BPT-4000).

Characteristic	HIVHAC	SOA Hall
Thruster Power Range, kW	0.3-3.6	0.3-4.5
Throttle Ratio	12:1	15:1
Operating Voltage, V	200-700	150-400
Specific Impulse, sec	1000-2800	710-2100
Thrust, mN	24-150	22-260
Efficiency	0.32-0.60	0.25-0.58
Propellant Throughput, kg	>300	450

lifetime with the potential of enabling lifetimes in excess of 15,000 hours and xenon throughputs in excess of 300 kg.

B. Recent Progress

Wear tests of the NASA-103M.XL thruster have validated and demonstrated the life extending innovation as a means to mitigate discharge channel erosion as a life limiting mechanism in Hall thrusters. Test priorities have focused on the wear test of the laboratory thruster to validate the lifetime extending innovation to demonstrate throughput capabilities of the design. The thruster, shown in Fig. 6, has been operated in excess of 4700 hours (100 kg of xenon throughput) at a discharge voltage of 700 V.[42] The thruster has demonstrated a throttle range of 12:1 and a maximum nominal power of 3.5 kW. At 3.5 kW the thruster has demonstrated a performance of 55% total efficiency and 2780 seconds total impulse, and a predicted lifetime exceeding 15,000 hours.[43] In addition, measured erosion profiles at the various time segments have been used to validate numerical simulation of the discharge channel erosion.

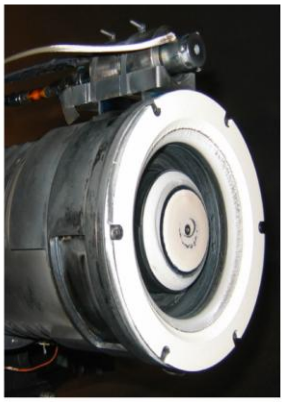

Figure 6. Photograph of the NASA-103M.XL thruster after 4,731 hours of testing at 700 V.

Due to the successful demonstration of the NASA-103M.XL thruster during the wear test, the HIVHAC effort has proceeded with the engineering model (EM) design.[44] NASA GRC teamed with Aerojet to design, manufacture, and test a HIVHAC EM thruster. The HIVHAC EM thruster design leverages Aerojet's experience with the development of the BPT-4000 flight-qualified Hall thruster. In addition, the EM thruster design incorporates the lessons learned from the NASA-77M and the NASA-103M.XL laboratory thrusters. The EM thruster design minimizes the mass, part count, the use of tight tolerances, and the use of complex manufacturing processes without compromising the EM thruster's design fidelity in order to reduce thruster manufacturing costs. 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.[44-45] Extensive testing of the EM thruster revealed that a design iteration with the magnetic circuit, thermal operating environment, and channel replacement mechanism operation was needed. Design changes included a new magnetic circuit, a new anode isolator, new electromagnet designs, new boron nitride discharge channel configurations and a new channel advancement mechanism.[46] The new thruster design has been designated HIVHAC EM-R (rework) thruster. Components for the thruster have been fabricated and the thruster has been assembled for performance testing in late August 2011. The thruster has undergone functional and preliminary performance characterization. Figure 7 shows a photograph of the EM-R thruster.

Figure 7. HIVHAC EM-R thruster istalled in VF12.

Preliminary performance mapping of the EM thruster at various operating conditions was performed at NASA GRC. Preliminary characterization results indicate that the HIVHAC EM thruster discharge efficiency varied between 0.32 and 0.60 and the discharge specific impulse varied between 1157 and 2665 sec for thruster operation at discharge power between 321 and 3664 W, respectively. Figures 8 and 9 presents plots of the EM discharge efficiency and specific impulse at various discharge power levels.[46] It is projected that the design changes incorporated in the HIVHAC EM-R thruster will result in total efficiency and specific impulse that match or exceed that of the HIVHAC EM thruster,[46] thus demonstrating the effectiveness of the design updates.

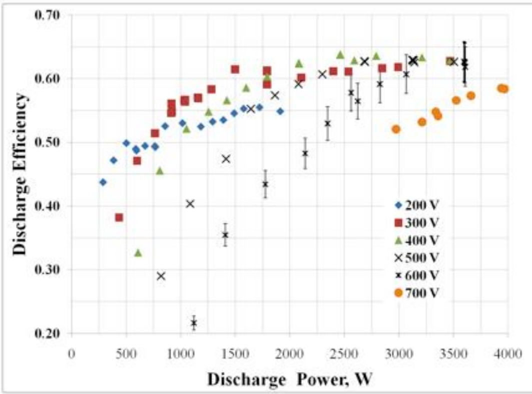

Figure 8. HIVHAC EM discharge efficiency vs. discharge power for discharge voltage operation between 200 and 700 volts.

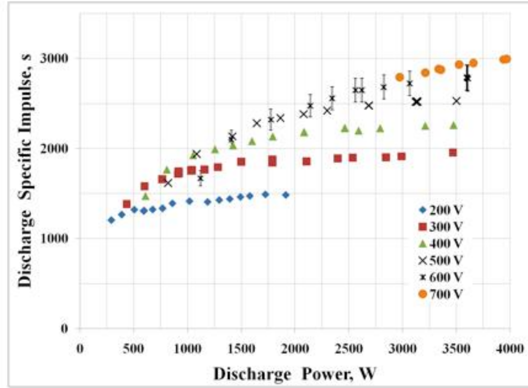

Figure 9. HIVHAC EM discharge specific impulse vs. operating power for discharge voltage operation between 200 and 700 V.

Concurrent with the wear test activity is a hollow cathode development activity aimed at providing a hollow cathode that operates in spot mode with minimal propellant and power consumption while producing the required life. A variety of cathode configurations were built and evaluated. The variations included studying the effects of cathode orifice plate throat length, emitter inner diameter, keeper plate orifice diameter, and cathode-keeper gap on hollow cathode performance and operation. Results indicated that changes to the cathode-keeper gap had the most profound effect on stable cathode operating conditions.[47]

In addition to the thruster development, the HIVHAC project has been evaluating PPU and xenon feed system XFS developments options that have been sponsored by other projects but that can apply directly to a HIVHAC system. The goal is to advance the TRL level of a HIVHAC Hall thruster propulsion system to level 6 in preparation for a first flight.

The functional requirements of a HIVHAC PPU are that it can operate over a wide power throttling range of 300 to 3,800 W, over a range of discharge voltages between 200 and 700 V and output currents between 1.4 and 5 A as the input varies over a range of 80 to 160 V. Two PPU development options have been identified as possible paths to developing a HIVHAC PPU. One option is to modify the BPT-4000 PPU so that it can operate over an 80-160 V input voltage range, provide the required output discharge voltage and current to power the HIVHAC thruster. Another option is to develop a HIVHAC PPU that is a new custom design like the one being developed by Colorado Power Electronics under a Small Business Innovative Research (SBIR) contract with NASA GRC. Two brassboard 2 kW discharge power supply modules that operate in parallel have been fabricated and bench tested. Integration tests with the NASA-103M.XL thruster were conducted at NASA GRC to characterize the performance of the discharge power supply modules. Test results indicate that for the 80 V input case, the efficiency is higher than 0.95 for output power levels higher than 1.8 kW and higher than 0.94 for outputs higher than 1.0 kW. Small reductions are observed at 120 and 160 V inputs. Test results are presented in Fig. 10. Additional details pertaining of the discharge power supply module testing are presented in Reference 48. As for the xenon feed system, at least three technology options are available. One option is to use the flight qualified BPT-4000 Moog feed system. Another option is to employee the TRL 6 NEXT feed system that is being built by Aerojet. A third option is to employ the VACCO advanced xenon feed system (AXFS), which provides significant mass and performance savings over the first two options. The VACCO AXFS has completed integrated system testing with the NASA-103M.XL thruster and advanced the modules to TRL 6. Additional details about the AXFS development will be included in the next section.

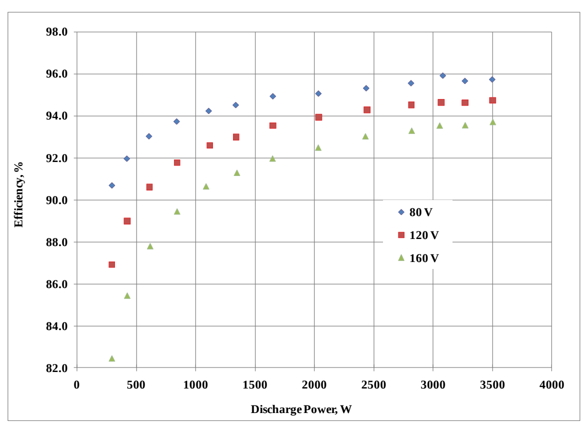

Figure 10. Discharge power supply module efficiency at various input voltages.

C. Future Plans

HIVHAC EM-R thruster testing will continue in FY11/12. Performance Acceptance Tests will be conducted in 2011. Environmental testing will include both vibration and thermal vacuum operation. Vibration testing and thermal-vacuum testing of the EM-R thruster is planned in the late 2011 time frame. Performance characterization tests and plume characterization tests will be conducted periodically to confirm thruster performance after both environmental tests. Once all these tests are successfully completed the EM-R thruster will undergo short-duration wear testing, which would be followed by a long-duration wear test to validate and demonstrate its sustained performance, throttle-ability, and throughput capability. Candidate component evaluations will continue as a prelude to a selection process for a full system development. Evaluation tests include flight-ready hardware, such as the BPT-4000 thruster and PPU elements, which will be tested beyond the conditions that these components were originally designed and qualified, such as operation for extended durations at high specific impulse operation.

IV. Advanced Xenon Feed System (AXFS)

A. Description

The Advanced Xenon Feed System (AXFS) task was funded to develop feed system components based on a novel diffusion bonding manufacturing technique. The task has been led by VACCO Industries and seeks to improve the reliability of electric propulsion feed systems while significantly decreasing mass and volume over conventional xenon feed system technologies.

To improve reliability, the entire system is both series and parallel redundant as shown in Fig. 11. An initial study of the reliability analyses completed at a component level has shown this configuration to have an expected lifetime approaching 30 years; exceeding all mission requirements. The mass of the proposed system is also substantially lower than conventional technologies. The Flow Control Module (FCM) has a mass of 650 grams and the Pressure Control Module (PCM) has a mass less than 730 grams. A three thruster system has an estimated mass of 2.71 kg or an 80% mass reduction over conventional technologies.

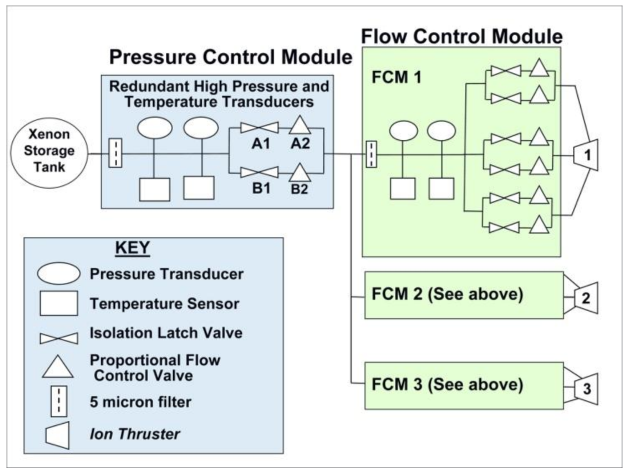

Figure 11. Advanced Xenon Feed System Configuration

B. Recent Progress

VACCO delivered an integrated AXFS with significant qualification level testing and hot-fire testing was completed in early 2009. Altogether, the project delivered two Flow Control Modules (FCM), one Pressure Control Module (PCM) and one controller operated by control software installed on a portable computer. Preceding the hot-fire testing, the FCMs and PCM underwent functional and environmental testing at VACCO and the Naval Research Laboratory (NRL) respectively.[49] Environmental testing showed no variation between pre and post-test operation.

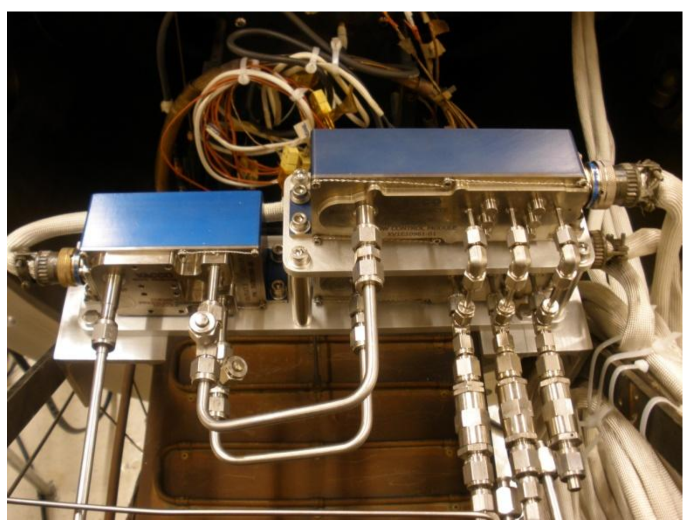
Figure 12. AXFS mounted on the HIVHAC thrust stand.

Functional testing and hot fire testing of the VACCO system was performed at NASA GRC.[50] Initial atmospheric testing of the VACCO AXFS involved calibrating at two different input pressures of 45 and 50 psi during VACCO AXFS operation in pressure control closed loop mode. Initial calibration tests indicated consistent xenon flow rates for a given VACCO XFS commanded pressure. Next, the VACCO AXFS was installed inside a vacuum chamber bell jar and was mounted on the HIVHAC thruster thrust stand as shown in Fig. 12. A number of configurations were tested and are outlined in Fig. 13.

In configuration 1, the output of the FCMs was directed outside of the vacuum chamber and was connected to the xenon calibrator unit. The HIVHAC thruster performance was baselined using the existing laboratory xenon flow system which utilized two mass flow controllers (MFCs). Figure 13 shows the normalized thrust during HIVHAC thruster operation with the MFCs. During that test, the VACCO AXFS xenon flow rate was measured at different FCM1 PV1 and FCM2 PV2 pressure settings with the calibrator unit to verify that the FCMs output xenon flow rate matched the calibration curve results and that xenon flow rate remained constant during thruster operation. Results of that FCM1 PV1 test series are presented in Table 3. Table 3 results show that as the thruster discharge power was increased from 288 W to 3456 W, the measured xenon flow rate for a given VACCO AXFS FCM1 PV1 setting varied by less than one percent. This testing validated the vacuum operation of the AXFS and demonstrated the ability to accurately control flow at the various thermal environments within the chamber.

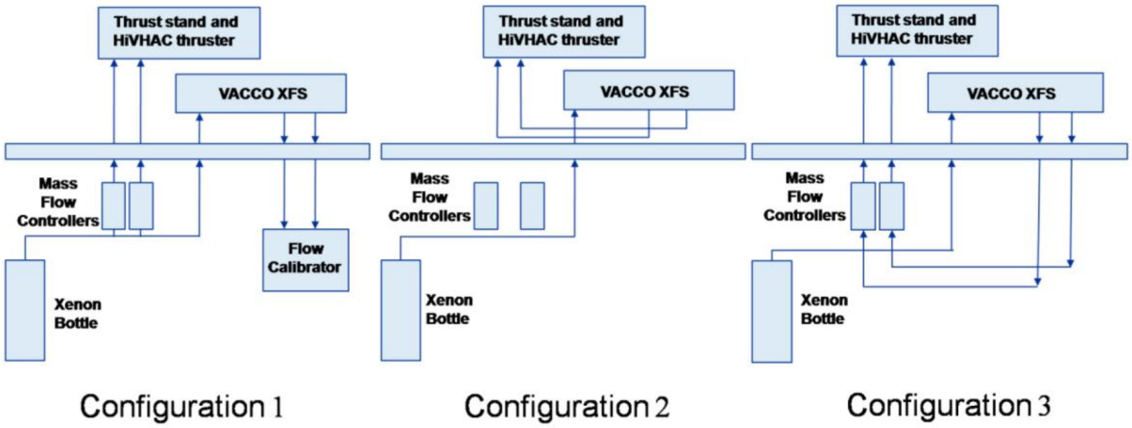

Figure 13. Testing configurations of the AXFS.

Table 3. Flowrate measurements vs. pressure set point at various power levels.

Thruster Discharge Power, W	PV1 Set Point, psig	Measured VACCO AXFS Xenon Flow Rate, sccm
288	10	20.55
	30	41.76
	43	54.12
609	10	20.51
	30	41.72
	43	54.07
1029	10	20.49
1646	10	20.45
	43	54.06
2438	30	41.62
3456	30	41.63

The next test series, configuration 2, involved using the VACCO AXFS to supply the xenon to the HiVHAC thruster anode and cathode. In this test series, the output of the VACCO AXFS FCMs was connected to the thrust stand's xenon input connectors inside the vacuum chamber. Initial tests were performed with the VACCO AXFS being operated in closed loop pressure control mode. In that mode the VACCO AXFS was commanded to a prescribed pressure which corresponded to a given xenon propellant flow rate as determined by the earlier obtained calibration curves. Configuration 2 test results indicated that thruster performance was consistent with the baseline thruster performance and that the discharge current held constant for a given FCM1 PV1 pressure setting. Next the VACCO AXFS control software was modified to enable operation in current control close-loop mode while still in configuration 2. To accomplish this, new control software was created to perform thruster operation in closed-loop current control mode. In current control mode the VACCO AXFS was commanded to a given discharge current setting and the anode flow rate was varied by the control software to maintain the set discharge current value. Results of that test are presented in Fig. 14. Comparing the thruster performance during MFC and VACCO AXFS current control mode operation indicates almost identical thruster performance. Finally, to be able to obtain xenon flow measurement while operating the VACCO AXFS in current control closed loop mode necessitated making changes to the xenon propellant tubing scheme. Those changes included routing the VACCO AXFS FCMs output to outside the vacuum chamber and onto the inlet of the anode and cathode MFCs of the original laboratory xenon feed system. As such the MFCs were used to meter the xenon flow rate output of the VACCO AXFS. In-situ calibrations were performed for the new xenon feed scheme. Results of that test are presented in Fig. 14 and again show an almost identical performance to the original tests conducted with the laboratory xenon feed system, which utilized two MFCs.

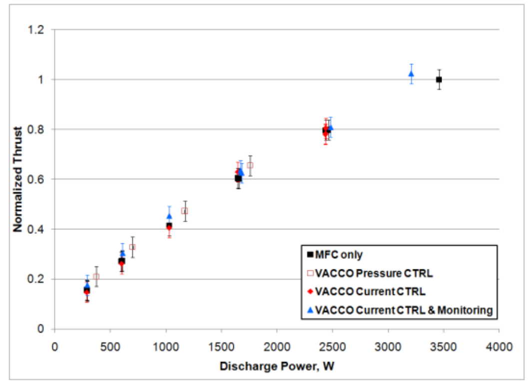

Figure 14. Performance characterization of Hall thruster with various XFS configurations.

C. Future Plans

The AXFS was developed as a highly reliable low-cost feed system alternative for the NEXT Ion Propulsion System and required closed loop control of downstream pressure transducers to achieve precision flow control. However, using the same technology can also allow for a simplified Hall thruster module with an expected mass of only 1.25kg. Additionally, the Hall thruster module can provide closed-loop control for the anode with feedback from the thruster discharge current. This operation was demonstrated during the AXFS hot-fire testing, and demonstrated stable steady-state flow control. Based on this concept, a Hall thruster xenon flow control module was developed with a schematic of the pressure control module with added temperature and pressure sensors downstream of the piezoelectric flow controller in a single module. A joint investment from NASA and the AFRL is maturing the Hall module XFCM through formal flight qualification. Delivery of the qualification unit is anticipated in March 2012. An acceptance tested flight unit has also been procured and is scheduled for delivery in December 2012. The schematic of the Hall module is shown in Fig. 15 and a photograph of the assembled unit in Fig. 16.

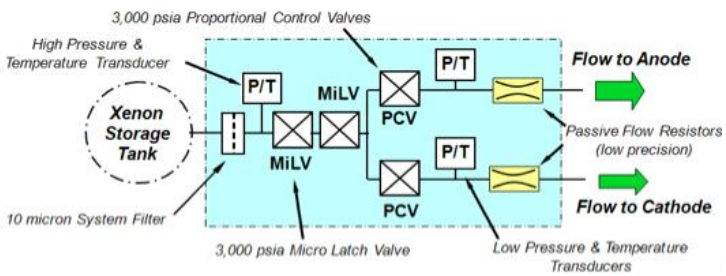

Figure 15. Hall Module XFCM schematic

Figure 16 Hall Module Qualification unit.

V. Conclusions

Major hardware development tasks within the In-Space Propulsion Technology Project include NEXT Ion Propulsion System, HIVHAC Hall thruster, and an advanced xenon feed system. NEXT project activities have brought next-generation ion propulsion technology to a mature state, with existing tasks completing the majority of the NEXT product technology validation. Functional and qualification-level environmental tests of key system components are anticipated to be completed in the near future. Cost incentives, which have been provided in NASA Announcement of Opportunities, will help address concerns about first user costs. The HIVHAC task is meeting its goals to increase specific impulse, throttle-ability and lifetime to broaden Hall propulsion system applicability to low-cost, deep space science missions. The HIVHAC thruster has demonstrated a throttle range of 12:1 and a maximum nominal power above 3.6 kW. At 3.6 kW the thruster has demonstrated a performance of 60% total efficiency and 2665 seconds total impulse, and a predicted lifetime exceeding 15,000 hours. An Engineering Model thruster was completed and was tested in thermal vacuum. Design improvements were identified and incorporated into the reworked EM thruster, which will undergo performance and environmental tests in FY11/12. The advanced xenon feed system task is developing an integrated flow control module for a Hall propulsion system, which will undergo acceptance and qualification testing. Efforts under each of the development tasks focus on advancing technology readiness for flight infusion.

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