Preliminary Performance Characterization of the High Voltage Hall Accelerator Engineering Model Thruster

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Abstract: This paper presents preliminary performance characterization results of the High Voltage Hall Accelerator engineering model thruster. The engineering model thruster is designed to provide improved performance and a lower cost electric propulsion system for NASA Discovery Class Science missions. The engineering model thruster was designed, built, and manufactured by Aerojet and NASA Glenn Research Center. Testing of the engineering model thruster was performed at the NASA Glenn Research Center Electric Propulsion Laboratory. The engineering model thruster is designed to have a 12:1 throttle range with a maximum discharge power of 3,500 watts where it operates at a discharge voltage of 700 volts and a discharge current of 5 amperes. To date, the thruster has been operated over power levels between 281 and 2,502 watts corresponding to discharge voltages between 200 and 500 volts and discharge currents between 1.4 and 5 amperes. Thruster discharge efficiency varied between 0.35 and 0.60 and the discharge specific impulse varied between 1,173 and 2,442 seconds. The HiVHAc EM-1 thruster discharge efficiency and discharge specific impulse magnitudes were 6-20% and 5-13%, respectively, higher than their magnitudes for the HiVHAc NASA-103M.XL laboratory thruster.

Nomenclature

A	= Amperes
g	= Gravitational Acceleration, 9.81 m/sec^2
I_{beam}	= Ion Beam Current, A
I_{dis}	= Discharge Current, A
I_e	= Electron Current, A
I_{sp}	= Specific Impulse, sec
mg	= Milligrams
m_{anode}	= Anode Xenon Flow Rate, mg/sec
N	= Newton
P_{dis}	= Discharge Power, W
V_{avg}	= Mass Average Ion Voltage, V
V_{dis}	= Discharge Voltage, V
T	= Thrust, N
W	= Watts
η_{dis}	= Discharge Efficiency
η_I	= Current Utilization Efficiency
η_V	= Voltage Utilization Efficiency
Φ_{div}	= Cosine Divergence Loss Factor
Φ_{VDF}	= Velocity Distribution Function

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I. Introduction

NASA's science vision requires the scientific exploration of our planet, other planets and planetary bodies, our star system in its entirety, and the universe beyond. Through this vision the intellectual foundation for the robotic and human expeditions of the future is laid and formed.[1] Electric propulsion systems enable and improve NASA's ability to perform scientific exploration. As such, NASA continues the development of advanced electric propulsion 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 electric propulsion systems in missions like Deep Space 1 and Dawn.[2,3]

More recently, new electric propulsion technology activities are underway and are the responsibility of the In-Space Propulsion Technology Program (ISPT), as part of NASA's Science Mission Directorate (SMD).[1] The main focus of this program is NASA's Evolutionary Xenon Thruster (NEXT) ion thruster propulsion system,[4] however, ISPT is also developing the High Voltage Hall Accelerator (HiVHAc) thruster as a lower cost electric propulsion alternative for future cost constrained missions.

In 2004 the In-Space Propulsion Technology Program conducted a study to quantify the potential benefit of using the NEXT propulsion system which is nearing flight readiness, the NASA Solar electric propulsion Technology Application Readiness (NSTAR) ion thruster system which is flight qualified, and a Hall thruster propulsion system with characteristics based on the HiVHAc thruster.[5,6] This study considered New Frontier-Class science missions, that are currently cost capped at around $800 M, and Discovery-Class science missions, that are currently cost capped at around $450 M.[1] The Hall thruster propulsion system was considered as part of this mission study due to advancements in Hall thruster technology that had occurred during prior years. These advancements included increases in throttle-ability, specific impulse, and thruster efficiency[7] and the successful demonstration of Hall thruster propulsion systems for primary propulsion applications.[8,9] The proposed Hall thruster system evaluated during this technology mission assessment was a system envisioned to operate at power levels of 200 – 2,800 W while providing specific impulse ranging from 1500 to 2800 sec.[10] The results of this assessment were that a Hall thruster system with these performance capabilities and the ability to provide total impulses approaching that of ion thruster systems provided substantial cost and performance benefits relative to the other advanced electric propulsion technologies for certain types of NASA's Discovery-Class science missions.[5] As a result of this study, the development of a Hall thruster with these characteristics was included as part of ISPT's technology development portfolio. More recently, mission studies were performed to compare the mission performance of the HiVHAc NASA-103M.XL laboratory thruster to Aerojet's BPT-4000 state-of-the-art (SOA) 4,500 W Hall Thruster Propulsion System (HTPS).[11] These studies were performed for three NASA Discovery-Class science missions: Vesta-Ceres rendezvous mission (Dawn Mission), Koppf comet rendezvous, and Nereus sample return mission. Results from the mission studies indicated that the HiVHAc thruster was able to close all the missions and the HiVHAc delivered mass at target was 6-12% higher than the BPT-4000 thruster for the dawn and Koppf comet rendezvous missions. The main reason for the higher HiVHAc delivered mass capability is its higher specific impulse of ~2,800 sec when compared to the BPT-4000 thruster specific impulse of ~ 2,200 sec.[11,12]

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 300 and 3,500 W,[13] has a thrust efficiency > 0.55 and a specific impulse of 2800 sec at 3,500 W, and has a xenon throughput capability > 300 kg corresponding to an operational lifetime of >15,000 hrs.[13] Table 1 shows the HiVHAc throttle table.

This paper provides a brief overview of the two laboratory thrusters that were built and tested to demonstrate the throttling range, performance, and life time of a Hall thruster that can exceed NASA's science mission requirements. The paper describes the HiVHAc EM thruster design, analysis, and manufacturing process. Description of the vacuum facility, power console, propellant feed system, and thrust stand used in testing is also presented. Finally, the HiVHAc EM thruster anode discharge efficiency and specific impulse results are presented for power levels up to 2,500 W and are compared to those performance magnitudes of the NASA-103M.XL thruster.

Table 1. HiVHAc throttle table.

Discharge Voltage, V	Discharge Current, A	Discharge Power, W
200	1.43	286
250	1.73	432
300	2.02	607
350	2.32	812
400	2.62	1048
450	2.92	1312
500	3.21	1607
550	3.51	1931
600	3.81	2286
650	4.11	2669
700	4.40	3083
700	4.70	3291
700	5.00	3500

II. High Voltage Hall Accelerator Laboratory Thrusters

To demonstrate the HiVHAc project goals, two laboratory thrusters were built and tested; the NASA-77M and NASA-103M.XL. The NASA-77M was the first laboratory thruster built and tested, it was designed to operate with input powers between 200 and 2,800 W. The NASA-77M demonstrated a 14:1 throttle range, specific impulse levels between 922 and 2,911 sec and thrust efficiencies between 0.31 and 0.54.[10] The NASA-77M was also subjected to two 300 hour wear tests to assess discharge channel wear characteristics for boron nitride and boron nitride/silicon dioxide discharge channels.[13] The measured discharge channel erosion profiles confirmed the sensitivity of the erosion profiles to the magnetic field streamlines shape.[13] The measured erosion profiles also provided insights into magnetic circuit design improvements that were implemented in the design of the HiVHAc EM thruster. Finally, the wear test results were 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 second HiVHAc laboratory thruster that was designed, built, and tested was the NASA-103M.XL(eXtended Life). The NASA-103M.XL incorporates an innovation that enables a more than two-fold increase in lifetime relative to current 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 at power levels between 300 and 3,500 W, respectively. The thruster specific impulse varied between 1,062 sec and 2,779 sec at power levels between 300 and 3,500 W, respectively.[14,15] Beginning of life (BOL) performance results are presented in Figure 1 and indicate that thruster performance improved with increasing thruster power. After completion of the initial thruster performance characterization, wear testing of the NASA-103M.XL thruster was initiated. The primary goal of wear testing the NASA-103M.XL thruster was to demonstrate the life extending innovation. To date, the thruster has been operated for 4,731 hours at a discharge voltage of 700 V demonstrating its life extending innovation.[16] For the first 3,623 hours the thruster was operated at 3,500 W. For the next 1,108 hours the thruster was operated at a power level of 3.1 kW (V_d=700 V and I_d=4.4 A). Figure 2 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.

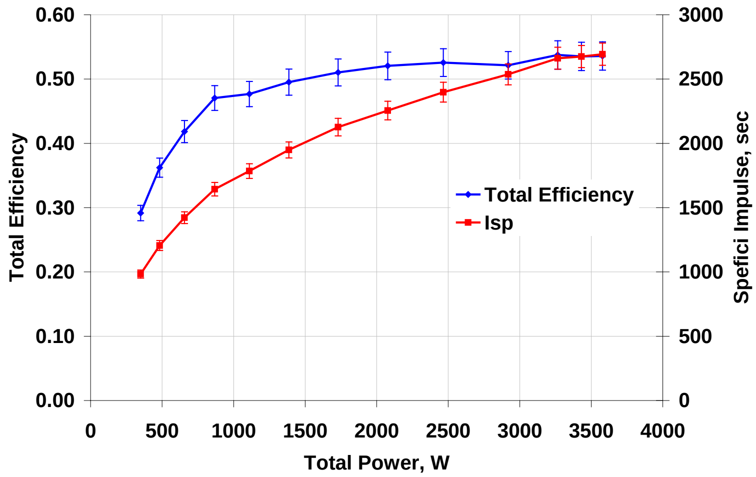

Figure 1. NASA-103M.XL BOL total efficiency and specific impulse as a function of thruster power.

Figure 2. Photograph of the NASA-103M.XL thruster after 4,731 hours of testing at 700 V.

III. Experimental Apparatus

After the successful demonstration and validation of the life extending innovation with the NASA-103M.XL laboratory thruster, NASA GRC teamed with Aerojet to jointly design, fabricate, 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.[1] The EM thruster design incorporates the life extending innovation with a projected xenon throughput capability greater than 300 kg. 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.[4] An extensive design, analysis, manufacturing, and test plans were devised to achieve the aggressive HiVHAc EM thruster requirements.[17]

A. HiVHAc Engineering Model Thruster

The HiVHAc EM thruster leverages all the experience, knowledge, and lessons learned during the development of the NASA- 77M and 103M.XL laboratory thrusters in addition to incorporating all of Aerojet's experience in building the flight qualified BPT-4000 HTPS. During the design of the HiVAHc EM thruster, various magnetic, erosion, structural, and thermal modeling tools were utilized.

Magnetic circuit design and modeling was performed at NASA GRC in consultation with Aerojet. The objective of the magnetic circuit design was to reduce the magnetic circuit complexity and parts count as well as reducing the cost of manufacturing the magnetic circuit components. Erosion modeling and life assessment of the EM thruster were made using an Aerojet developed code.[18] The code uses parameters such as discharge voltage and current, magnetic field magnitudes, and mechanical design characteristics to predict the thruster's discharge chamber erosion rates and profiles over time. As such, the code was implemented to predict the necessary channel parameters that would be required to achieve an EM thruster lifetime in excess of 15,000 hours.

The EM thruster structural and thermal requirements were derived from NASA's Evolutionary Xenon Thruster (NEXT) technical requirements document and are representative of the requirements for a NASA New Frontier-Class mission.[19] These requirements were deemed sufficiently stringent as to encompass future Discovery-Class mission requirements. 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.

The EM thruster design minimizes the mass, parts count, the use of tight tolerances, and the use of complex manufacturing processes without compromising the EM thruster's design fidelity. All these factors result in reduced thruster manufacturing costs. Some key EM thruster design features include: an integrated magnetic structure, a low cost anode design, a low cost propellant isolator, and a thermally efficient robust electromagnet design.[17] In addition, the gimbal design leverages the low mass NEXT design by scaling down that design to accommodate mechanical interfaces to thruster's circumference.

The EM thruster components were fabricated at Aerojet's Redmond facility where the flight BPT-4000 Hall thruster systems are built[20] and at NASA GRC. Components for two EM thrusters were fabricated. To date, one thruster has been assembled, designated EM-1, and has been delivered to NASA GRC. HiVHAc EM-1 is shown in Figure 3.

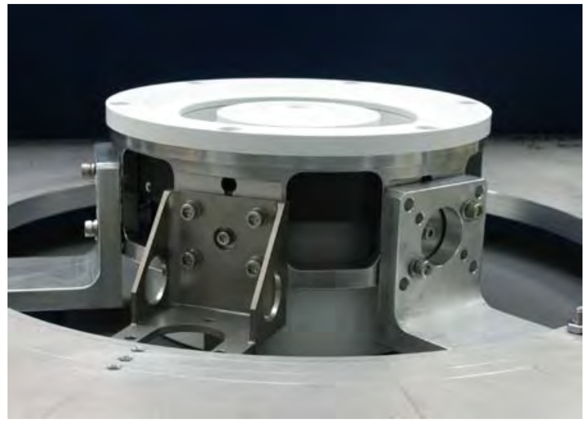

Figure 3. Photograph of the HiVHAc EM-1 thruster during assembly.

B. Hollow Cathode Assembly

The HiVHAc EM hollow cathode assembly (HCA) is based on the International Space Station (ISS) plasma HCA design and heritage. Modifications were made to the baseline configuration to optimize its performance with the HiVHAc thruster.[15] For this series of preliminary performance characterization, testing of the HiVHAc EM thruster was performed with a laboratory type HCA that had been employed during the wear testing of the NASA-103M.XL thruster. This decision was made to avoid the potential of contaminating the EM HCA emitter with oil from the oil diffusion pumps which could potentially limit its life when used in a long duration test. Figure 4 shows a photograph of the manufactured Aerojet HiVHAc HCA.

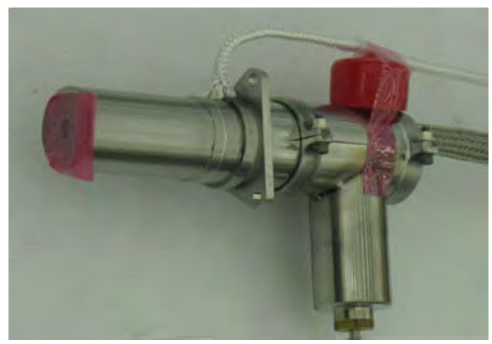

Figure 4. Photograph of the HiVHAc EM hollow cathode assembly.

C. Vacuum Facilities

The HiVHAc EM thruster performance characterization and evaluation was performed at NASA GRC Electric propulsion Laboratory VF-8. VF-8 is a 1.5 m diameter by 4.7 m long tank that is evacuated by four 0.9 m oil diffusion pumps with a pumping speed in excess of 160,000 liters per second (air). VF-8's interior was lined with graph foil and a graph foil beam dump was installed at the opposite end of the chamber to reduce the ion back sputter rate during thruster testing. During this preliminary performance evaluation, the HiVHAc EM-1 thruster was mounted in bell jar 2 as shown in Figure 5.

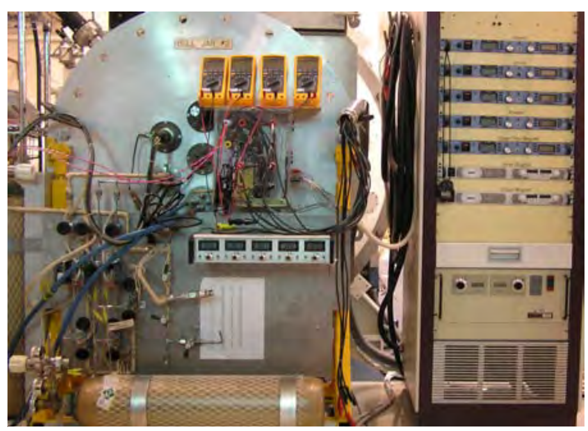

Figure 5. Photograph of VF-8 Bell jar 2, xenon feed system, and the power console used in EM-1 testing.

D. Power Supply and Data Acquisition

A laboratory power console consisting of a discharge supply and various auxiliary power supplies was used during the performance characterization of the HiVHAc EM thruster. The discharge power supply is capable of producing a constant voltage output ranging from 0 to 1,000 V at current levels between 0 and 6 A for a total output power capability of 6 kW. The auxiliary power supplies included two electromagnetic coils, a heater, and keeper supplies for the HCA. Figure 5 shows a photograph of the power console.

The data acquisition system used during the performance characterization of the HiVHAc EM thruster is a 22-bit multiplexed data logger with computer interface. The data logger monitored the voltages, currents, temperatures, propellant flow rates, chamber pressure, and thrust every second during performance testing.

E. Flow System

Two independent propellant mass flow controllers (MFCs) were utilized during the performance characterization and evaluation of the HiVHAc EM thruster. The anode xenon flow was supplied by a 100 SCCM MFC, whereas, the hollow cathode flow was supplied by a 25 SCCM MFC. A 99.999% pure xenon propellant was used during testing. The MFCs were calibrated before and after testing using a commercially available volumetric flow rate calibration system. The observed error between flow calibrations for xenon was no greater than 3% for the anode and 1% for the hollow cathode. Finally, prior to testing of the HiVHAc EM thruster, the anode propellant feed lines were baked out for 24 hours at ~120°C while under vacuum. The laboratory flow system can also be seen in Figure 5.

F. Thrust Stand

An inverted pendulum thrust stand was utilized to measure the thrust magnitudes of the HiVHAc EM thruster. The thrust stand had an accuracy of 1.5% full scale for forces ranging from 1 mN to several hundred mN of thrust. The operation and theory of the inverted pendulum thrust stand were described in detail in References 21. The thrust stand was operated in a displacement configuration. The thrust stand was also equipped with a closed loop inclination control circuit, which utilized a piezoelectric element to minimize thermal drift during performance testing. The thrust stand was calibrated before and after each performance mapping period with 3 in-situ calibrated masses on a pulley system.

IV. Performance Characterization Results and Discussion

The performance of the HiVHAc EM-1 thruster was characterized at discharge voltages of 200, 300, 400, and 500 V for anode flow rates between 1.81 and 5.48 mg/sec. Operation at higher discharge voltages was not attained due to intermittent voltage breakdowns at 600 V which necessitated testing stoppage to avoid possible damage to the thruster. These voltage breakdowns were caused by overheating of a Torlon insulator that was placed around the anode power lead. A design fix has been implemented, and testing at 600 and 700 V is planned for the near future. During testing, the thruster's operating parameters of anode and hollow cathode flow rates, inner electromagnet current and voltage, outer magnetic current and voltage, cathode keeper current and keeper voltage, cathode coupling voltage, and thrust were monitored and recorded.

For this initial round of performance characterization, testing was initiated at a discharge voltage of 200 V. At a given discharge voltage, the anode flow rate was increased from 1.81 mg/sec to its maximum value of 5.48 mg/sec, thus increasing the magnitude of the discharge current to its maximum value of approximately 5 A.

At each discharge voltage and anode flow condition, minimization of the discharge current was performed by adjusting the inner and outer electromagnet current magnitudes. Once the minimum discharge current was attained the thruster's discharge current was allowed to stabilize for approximately 5 minutes at which time the thruster's operating parameters were recorded. Finally, prior to test initiation and upon test conclusion thrust stand calibrations were performed.

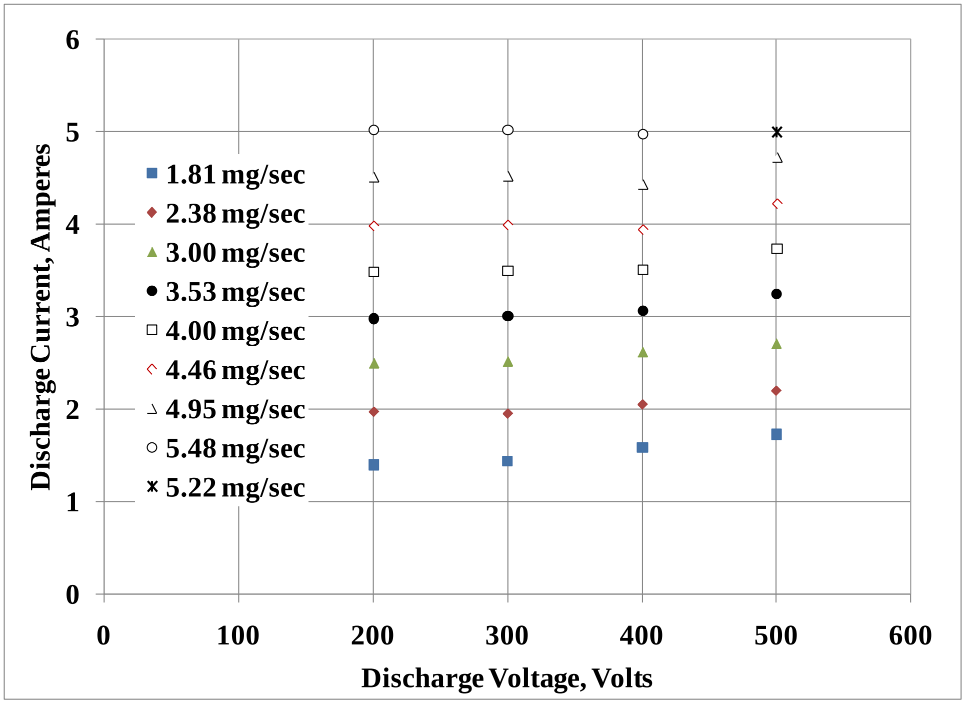

Figure 6. Discharge current profile for the various discharge voltage and anode flow rate operating conditions

In this paper only discharge efficiency and discharge specific impulse results are being reported. Total thruster efficiency and specific impulse results will not be presented in this paper since the HCA used in this round of testing was not the EM unit and since the thruster was not allowed to thermally stabilize at each test condition. Table 2 in the Appendix tabulates the HiVHAc EM-1 thruster operation and performance parameters.

Figure 6 presents the discharge current magnitudes for the different applied discharge voltages at the various anode flow rates that were tested. Note that at a discharge voltage of 500 V, the discharge current value of 5 A was attained at a flow rate of 5.22 mg/sec not 5.48 mg/sec as is the case for the discharge voltage of 200-400 V. Results in Figure 6 show that as the discharge voltage is increased from 200 to 500 V, the change in the discharge current for a given flow rate was small, this is an indication that the magnetic field configuration effectively limited and restricted axial electron transport as the discharge voltage was increased.[10] For anode flow rates of 4.46 mg/sec and higher, the discharge current magnitude at 400 V is the lowest, indicating that additional current minimization could have been achieved by adjusting the applied magnetic field at 200, 300, and 500 V. Optimizing the current minimization is critical for achieving high current and propellant utilization because it results in the highest achievable discharge efficiency as can be seen from equations 1-3[22,23]

$$\eta_{dis} = \eta_V \eta_I \Phi_{div} \Phi_{VDF} \tag{1}$$

$$\eta_V = \frac{V_{avg}}{V_{dis}} \tag{2}$$

$$\eta_I = \frac{I_{beam}}{I_{dis}} = \frac{I_{beam}}{I_{beam} + I_e} = \frac{1}{1 + \left(I_e \middle/ I_{beam} \right)} \tag{3}$$

A. *Discharge Efficiency*

The HiVHAc EM-1 discharge efficiency was calculated using equation 4 below

$$\eta_{dis} = \frac{T^2}{2\dot{m}_{anode} P_{dis}} \tag{4}$$

Discharge efficiency results of the HiVHAc EM-1 thruster as a function of discharge power are presented in Figure 7 and indicate that the HiVHAc EM-1 thruster discharge efficiency varied between 0.35 and 0.60 for thruster operation at discharge power between 281 and 2,502 W.

In general, results in Figure 7 show that as discharge voltage is increased from 200 to 400 V, discharge efficiency increased for discharge power above 1 kW. Discharge efficiency magnitudes at a discharge voltage of 500 V did not follow that trend and their values were lower than at 400 V. As stated earlier, it is speculated that at a discharge voltage of 500 V, optimum current minimization was not fully realized, which resulted in the lower thruster performance. In addition, the internal voltage breakdown problems encountered at 600 V might have been occurring at 500 V and could have contributed to the degraded thruster performance.

As indicated earlier, the EM thruster design was based on the NASA-103M.XL design but it included design modifications to the magnetic circuit to enhance the thruster's performance and discharge channel erosion profiles. Comparison of the EM-1 thruster discharge performance versus the NASA-103M.XL performance is also presented in Figure 7. In general, it is observed that the discharge efficiency of the EM-1 thruster was 6 to 20% higher than that of the NASA-103M.XL. Since discharge efficiency is also a measure of how efficiently the input discharge energy is converted to kinetic ion energy, as can be seen from equation 3, it is speculated that the improved discharge efficiency of EM-1 thruster relative to the NASA-103M.XL thruster is due higher current utilization (more efficient ionization and reduced electron current) and lower ion beam divergence as can be seen from equations 1-3.

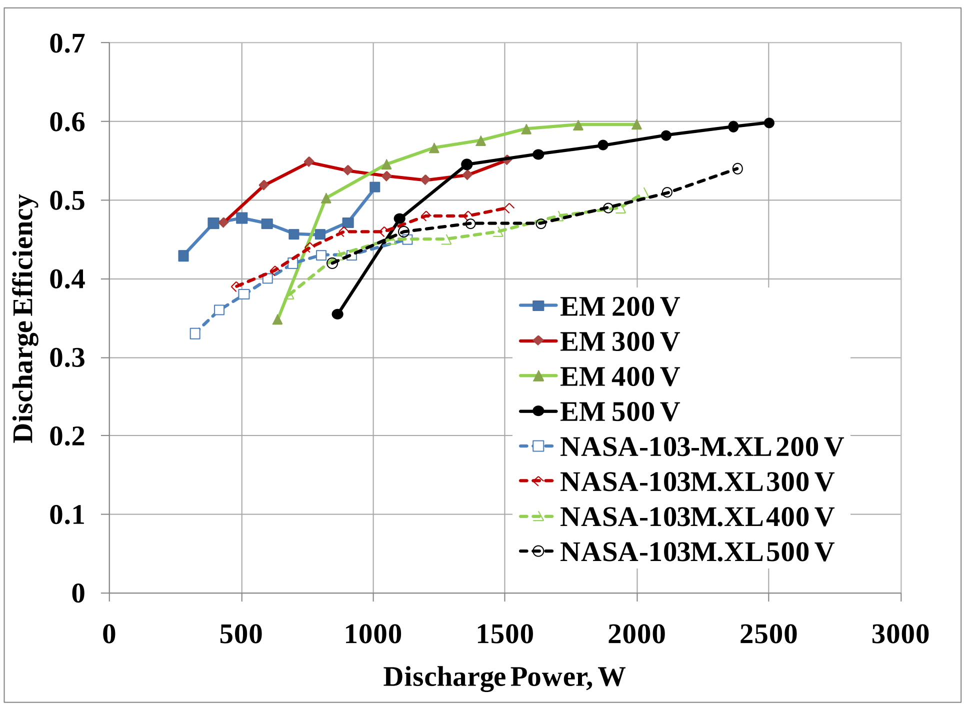

Figure 7. Discharge efficiency results for the HiVHAc EM-1 and NASA-103M.XL thrusters

B. *Discharge Specific Impulse*

The HiVHAc EM-1 discharge specific impulse was calculated using equation 5 below

$$I_{sp} = \frac{T}{g\,\dot{m}_{anode}} \tag{5}$$

Discharge specific impulse results for the HiVHAc EM-1 thruster as a function of discharge power are presented in Figure 8 and indicate that the HiVHAc EM-1 thruster discharge specific impulse varied between 1,173 and 2,443 sec for thruster operation at discharge power between 281 and 2,502 W.

Results in Figure 8 show that as the discharge voltage is increased from 200 to 500 V, discharge specific impulse increased for a given anode flow rate as would be expected since specific impulse is proportional to the square root of the discharge voltage.

Comparing the discharge specific impulse performance of the HiVHAc EM-1 and NASA-103M.XL thrusters, shown in Figure 8, indicates that discharge specific impulse magnitudes for the EM-1 thruster were approximately 5 to 13 % (except at lowest anode flow rate) higher than their values for the NASA-103M.XL thruster. As stated earlier, this is mainly attributed to higher propellant ionization and utilization efficiencies.

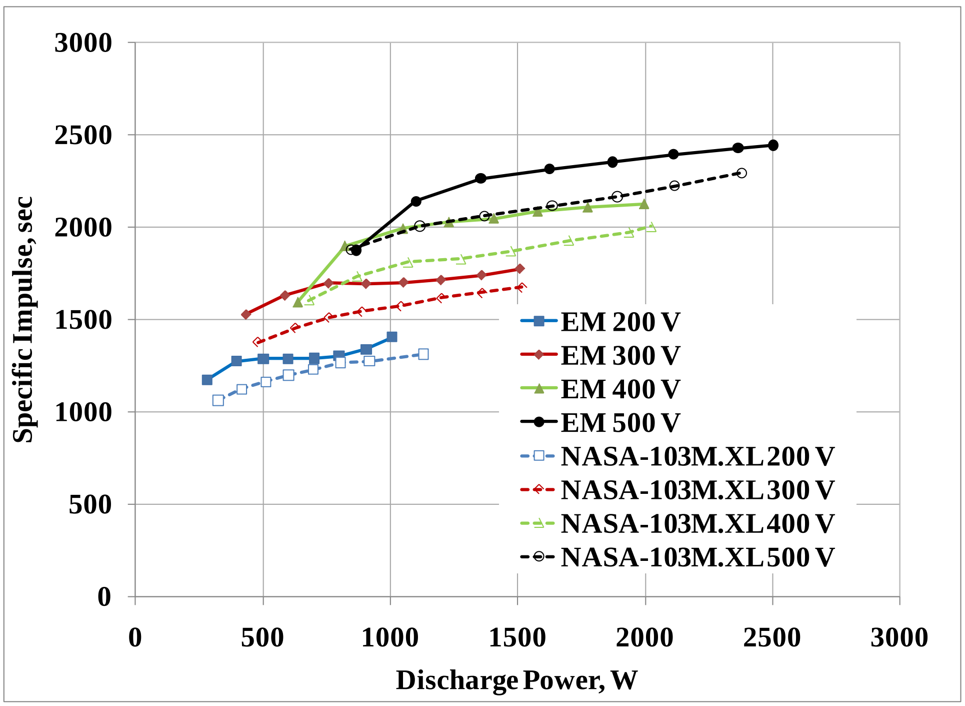

Figure 8. Discharge specific impulse results for the HiVHAc EM-1 and NASA-103M.XL thrusters

V. Conclusions and Planned Future Work

Preliminary performance characterization of the HiVHAc EM-1 thruster was performed at NASA GRC. Characterization was performed at power levels between 281 and 2,502 W corresponding to discharge voltages ranging from 200 to 500 V and discharge currents ranging from 1.4 to 5 A. Performance characterization results indicate that the thruster's discharge efficiency varied between 0.35 and 0.60 and the discharge specific impulse varied between 1,173 and 2,442 sec. Comparing the HiVHAc EM-1 thruster discharge performance to the NASA-103M.XL performance indicates that EM-1 performance was higher than that of the NASA-103M.XL due to an improved magnetic circuit which resulted in higher propellant and current utilization efficiencies along with lower ion beam divergence. Since the NASA-103M.XL thruster performance was used to baseline the mission benefits that a HiVHAc thruster would achieve over

SOA, the HiVHAc EM-1 thruster would achieve even greater mission benefits since it performance is superior to that of the NASA-103M.XL thruster. Near term planned work includes the following tasks:

- Vibration testing of the EM-1 thruster at Aerojet,
- Functional and performance testing of the EM-1 thruster for discharge power levels up to 3,500 W at NASA GRC,
- Thermal characterization of the thruster's various components including the anode assembly, the electromagnet, and thruster housing,
- Plume diagnostics in the thruster plume to measure the plume properties in the thruster plume and to help assess and elucidate the thruster propellant and current utilization efficiencies, and
- Thermal vacuum testing.

Upon conclusion of the above tasks, the HiVHAc thruster will be readied for and will undergo a long duration test to validate the life extending innovation and assess the thruster's performance throughout the long duration test.

VI. Appendix

Table 2. HiVHAc EM-1 operation parameters and performance

Discharge Voltage, V	Discharge Current, A	Anode Xenon Flow rate, mg/sec	Thrust, mN	Discharge Power, W	Discharge Efficiency	Discharge Specific Impulse, sec
200.4	1.40	1.82	20.9	280.6	0.43	1173.3
200.4	1.97	2.38	29.7	394.8	0.47	1273.9
200.5	2.50	3.00	37.9	501.3	0.48	1287.4
200.5	2.98	3.53	44.5	597.5	0.47	1286.8
200.5	3.49	4.01	50.6	699.7	0.46	1287.9
200.6	3.98	4.46	57.0	798.4	0.46	1302.7
200.6	4.51	4.95	65.0	904.7	0.47	1338.1
200.6	5.02	5.48	75.5	1007.0	0.52	1404.1
300.0	1.44	1.82	27.2	432.0	0.47	1525.5
300.1	1.95	2.38	38.0	585.2	0.52	1628.8
300.1	2.52	3.00	49.9	756.3	0.55	1695.0
300.2	3.01	3.53	58.5	903.6	0.54	1692.4
300.2	3.50	4.01	66.8	1050.7	0.53	1699.9
300.2	3.99	4.46	74.9	1197.8	0.53	1712.7
300.3	4.52	4.96	84.6	1357.4	0.53	1739.4
300.3	5.02	5.48	95.4	1507.5	0.55	1774.2
400.4	1.59	1.81	28.3	636.6	0.35	1595.5
400.4	2.05	2.38	44.3	820.8	0.50	1898.9
400.5	2.62	3.00	58.6	1049.3	0.55	1991.5
400.5	3.07	3.53	70.1	1229.5	0.57	2026.6
400.6	3.51	4.02	80.6	1406.1	0.58	2047.2
400.6	3.94	4.46	91.2	1578.4	0.59	2084.3
400.6	4.43	4.95	102.3	1774.7	0.60	2106.9
400.8	4.98	5.48	114.3	1996.0	0.60	2124.8

499.9	1.73	1.82	33.4	864.8	0.35	1873.2
499.9	2.20	2.38	49.9	1099.8	0.48	2140.2
500.0	2.71	3.00	66.6	1355.0	0.55	2262.2
500.1	3.25	3.53	80.0	1625.3	0.56	2312.8
500.2	3.74	4.01	92.4	1870.7	0.57	2352.0
500.3	4.22	4.46	104.7	2111.3	0.58	2393.8
500.4	4.72	4.95	117.8	2363.9	0.59	2426.9
500.4	5.00	5.22	125.0	2502.0	0.60	2442.6

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