

# Ion Engine and Hall Thruster Development at the NASA Glenn Research Center

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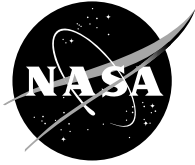
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## **ION ENGINE AND HALL THRUSTER DEVELOPMENT AT THE NASA GLENN RESEARCH CENTER**

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### **ABSTRACT**

NASA's Glenn Research Center has been selected to lead development of NASA's Evolutionary Xenon Thruster (NEXT) system. The central feature of the NEXT system is an electric propulsion thruster (EPT) that inherits the knowledge gained through the NSTAR thruster that successfully propelled Deep Space 1 to asteroid Braille and comet Borrelly, while significantly increasing the thruster power level and making improvements in performance parameters associated with NSTAR. The EPT concept under development has a 40 cm beam diameter, twice the effective area of the Deep-Space 1 thruster, while maintaining a relatively-small volume. It incorporates mechanical features and operating conditions to maximize the design heritage established by the flight NSTAR 30 cm engine, while incorporating new technology where warranted to extend the power and throughput capability.

The NASA Hall thruster program currently supports a number of tasks related to high power thruster development for a number of customers including the Energetics Program (formerly called the Space-based Program), the Space Solar Power Program, and the In-space Propulsion Program. In program year 2002, two tasks were central to the NASA Hall thruster program: 1.) the development of a laboratory Hall thruster capable of providing high thrust at high power; 2.) investigations into operation of Hall thrusters at high specific impulse. In addition to these two primary thruster development activities, there are a number of other on-going activities supported by the NASA Hall thruster program. These additional activities are related to issues such as thruster lifetime and spacecraft integration.

### **INTRODUCTION**

The success of the NASA Solar Electric Propulsion Technology Applications Readiness (NSTAR) program ion propulsion system on the Deep-Space 1 spacecraft[1] has secured the future for this propulsion technology for other NASA missions. The successful demonstration of the NSTAR ion thruster has provided future mission planners with an off-the-shelf 2.5 kW ion thruster. The 2.5 kW ion propulsion system on Deep-Space 1 performed flawlessly in space operating over 16,000 hours and processing in excess of 70 kg of xenon propellant.

While the NSTAR thruster is appropriate in terms of power level and lifetime for Discovery Class as well as other, smaller NASA missions, its application to large flag-ship type missions such as outer planet explorers and sample return missions is limited due to its lack of power and total impulse capability. Several missions under consideration for the Exploration of the Solar System, part of NASA's Space Science Enterprise, have identified a higher power, higher throughput capability, 5/10-kW ion propulsion system as an enabling technology. These missions include the Europa Lander, the Saturn Ring Observer, the Neptune Orbiter, and the Venus Surface Sample Return [2,3].

A NASA Glenn Research Center (GRC)-led team has recently been awarded a NASA Office of Space Science research project to develop the next generation of ion propulsion system. The successful proposal, for NASA's Evolutionary Xenon Thruster (NEXT), was developed by a team composed of GRC, the Jet Propulsion Laboratory (JPL), General Dynamics Space Propulsion Systems (GD-SPS),

Boeing Electron Dynamic Devices (BEDD), Applied Physics Laboratory, University of Michigan, and Colorado State University.

Additionally, NASA's ion propulsion program is also funded under the Energetics Program to develop fundamental technology for ion engine applications. Under this program, microwave discharge chambers and cathodes are being investigated as possible replacements for the hollow cathodes in ion engines operating at tens of kilowatts and requiring life to much more than 10,000 hours. Development of a low power (several hundred Watt) ion thruster to engineering model maturity is also being pursued. Propellantless cathode technology development, real-time, laser-based erosion diagnostics, discharge chamber modeling, and measurement of low-energy sputter yields are also funded under the Energetics Program.

The NASA Hall thruster program at the Glenn Research Center currently supports a number of tasks related to high power thruster development for a number of customers including the Energetics Program, the Space Solar Power Program, and the In-space Propulsion Program. High power (>>10-kW) thruster development is being emphasized because recent mission analyses have shown a need for higher power electric propulsion systems for both Earth orbital and deep space applications. However, these different applications require propulsion systems with different optimal characteristics. Earth orbital applications such as space tugs, spacecraft orbit insertion, etc. benefit from high thrust electric propulsion systems for timely space transfers. In contrast, deep space missions typically have large delta V requirements necessitating high specific impulse [4,5,6]. For recently considered power rich spacecraft architectures, both high power and high specific impulse propulsion systems are required [7]. The application of Hall thrusters to either mission type requires substantial technology advancements from state-of-the-art (SOA) thrusters.

In program year (PY) 2002, two tasks were central to the NASA Hall thruster program: 1.) the development of a laboratory Hall thruster capable of providing high thrust at high power; 2.) investigations into operation of Hall thrusters at high specific impulse.

Specifically the first task resulted in investigations of the issues associated with scaling a single thruster to power levels substantially in excess of the SOA. A 50-kW class Hall thruster was designed, built and operated over the range of 9 to 72 kW. The second task has focused on investigating factors critical to high specific impulse operation. This year the investigation has considered the role of magnetic field topography on high voltage operation.

In addition to these two primary thruster development activities, there are a number of other on-going activities supported by the NASA Hall thruster program. These additional activities are related to issues such as thruster lifetime, high-power power processor architecture and spacecraft integration. Thruster lifetime issues were investigated by considering the impact of Hall thruster operation at high voltage on thruster lifetime. Breadboard power system tests were conducted with a single power converter that may be suited for high-power power processing unit (PPU) architecture studies. Impedance measurements of a Hall thruster were made in an effort to develop an electrical

model of a Hall thruster to enable optimum and cost effective PPU design. Finally, a new effort to consider the dynamic electrical behavior of Hall thruster discharges was initiated. This was deemed necessary due to the importance of electro-magnetic interference (EMI) concerns as a spacecraft integration issue. A detailed overview of each of these activities and their current status are included in this report.

## NASA'S EVOLUTIONARY XENON THRUSTER (NEXT)

NEXT, which is part of the Next Generation Ion Project managed by NASA's Marshall Space Flight Center, Huntsville, AL., will consist of two phases. The first is a demonstration of initial components within one year. The second phase, which NASA may exercise as an option, will last approximately 2.5 years and will advance the system's maturity to NASA Technology Readiness Level (TRL) 5 (component and/or breadboard validation in relevant environment), with significant progress towards TRL 6 (prototype demonstration in relevant environment), thereby achieving a stage of development where the technology can be provided to NASA users to incorporate into planned missions.

The NEXT system will consist of the following:

- A 40-cm diameter electric propulsion thruster (EPT) with a specific impulse of at least 4000 seconds, a specific mass > 3.6 kg/kW, efficiencies greater than the NSTAR thruster at all power levels, and a propellant throughput capability required to execute the design reference missions;
- a lightweight, modular power processing unit (PPU) with an efficiency and a specific power equal-to or better-than the NSTAR PPU;
- a highly flexible advanced xenon feed system (AXFS) which uses proportional valves and thermal throttles to significantly reduce mass and volume relative to the NSTAR feed system; and
- a simple, lightweight EPT gimbal.

Validation of NEXT will dramatically advance state-of-the-art ion propulsion technology, with significant gains in EPT capability, as enumerated in Table 1.

**Table 1 – EPT Attribute Comparison**

| Engine Attribute          | NSTAR               | NEXT      |
|---------------------------|---------------------|-----------|
| Max. Input Power, kW      | 2.3                 | Up to 8   |
| Throttle Range            | 4:1                 | Up to 8:1 |
| Max. Specific Impulse, S  | 3,170               | 74,050    |
| Efficiency @ Full Power   | 62%                 | 768%      |
| Propellant Throughput, kg | 83 design, 140 demo | >300      |
| Specific Mass, kg/kW      | 3.6                 | >3.6      |

The NEXT thruster approach leverages the investments made at GRC in fiscal year 2001 for the development of a next generation engine [8,9]. During this time, design analyses were performed, throttling tables were defined, two engine concepts were developed, and laboratory model (LM) engines were fabricated [9]. The design of the discharge chamber magnetic circuit, magnet retention scheme, and flake-containment approach was completed, and multiple 40-cm diameter ion optics and discharge cathode assemblies were fabricated.

Preliminary performance characterizations of a 40-cm LM engine were conducted for input power levels ranging from about 1.1 kW to 7.3 kW [9]. The efficiency at 7.3 kW was approximately 68% at 3,620 seconds specific impulse. Ion optics performance (purveyance and electron backstreaming; beam divergence and beam current density profiles) was documented, as were temperatures of critical components, including the discharge cathode and magnets. The 40-cm ion optics performance was comparable to the 30-cm NSTAR flight ion optics [10]. Test data show that discharge cathode and magnet temperatures for the 40-cm engine are within design limits and provide significant design margin up to at least 10 kW.

### **Thruster Design**

For NEXT, an NSTAR-derivative engine with 40 cm beam diameter was selected [11,12,13,14]. An engine of this size has twice the beam area of the NSTAR 30 cm (28 cm beam diameter) thruster. Doubling the beam area allows operation at significantly higher power while maintaining low voltages and current densities. Thus, potential complications associated with high-voltage electrode operations are avoided. At an input power of 4.7 kW, the engine would be operating at approximately the same operating voltages and beam current density as the 2.3 kW NSTAR thruster, and hence would be expected to yield the same operating lifetime, but producing twice the thrust.

The NSTAR thruster has demonstrated >195 kg propellant throughput in ground testing, which is substantially higher than the original design goal of 83 kg. A greater than 2 times increase in throughput capability for the 40-cm engine is anticipated (based on a 2 times increase in beam area, an improved flatness parameter, and an associated reduction in local charge exchange production [8]) which is sufficient to meet mission requirements. The engine capability is expected to reach >300-kg by virtue of an advanced, molybdenum ion optics design that includes an accelerator electrode that is thicker than that of NSTAR [10]. This is beyond the ‘single-engine-out’ requirement for proposed Neptune and Saturn missions.

The NEXT 40-cm engine incorporates design improvements beyond NSTAR. These improvements include:

- A discharge chamber with improved beam flatness and reduced discharge losses
- High-temperature stabilized rare-Earth magnets
- A compact propellant isolator
- Advanced ion optics design for longer life

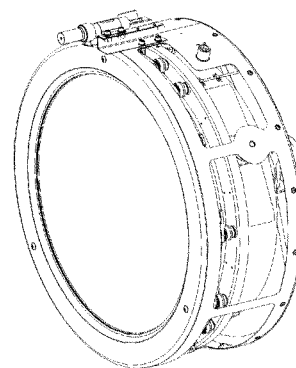
The mechanical integrity and design maturity of the 40-cm EM engine at the completion of the project first phase is expected to be superior to that of the engineering model of the NSTAR thruster (also manufactured by GRC) because it is being designed for the anticipated vibration environment.

Stainless steel is used in the construction of the discharge chambers for the 40 cm LM and EM engine designs fabricated at NASA GRC. Both 40 cm engine designs use ring-cusp magnetic circuits, with high-field strength permanent magnets for plasma containment [15]. A flake-retention scheme is employed in the discharge chamber, which also acts as a magnet retainer. The material, preparation, and installation

processes employed for the flake-retention system are identical to those implemented on the NSTAR 30 cm thruster. Both LM and EM 40 cm engines also incorporate a reverse-feed propellant injection process for the main plenum.

The discharge cathode assembly for the 40 cm engine is scaled from the NSTAR 6.4 mm diameter hollow cathode to accommodate the estimated emission current range, and uses similar design and manufacturing processes. The neutralizer is an enclosed-keeper hollow cathode assembly. Internal dimensions are adjusted to accommodate the higher emission current requirements for the 40 cm engine, and do so at reduced ratios of propellant flow rate-to-emission current, relative to the NSTAR neutralizer cathode.

Two design approaches are being pursued for the 40 cm ion optics. These include: NSTAR-type electrodes of increased beam diameter, and thicker accelerator-electrode geometry. Both are two-grid designs constructed of molybdenum. The electrode mounting system is scaled from the NSTAR design and also uses the same materials as implemented in the 30 cm thruster. Figure 1 shows an isometric perspective of the 40 cm EM version of the engine.



**Figure 1 – Isometric Drawing of the NEXT 40-cm EM Ion Engine.**

### **Thruster Manufacturing and Performance Evaluation**

NEXT's 40-cm engine will leverage the manufacturing processes of the GRC NSTAR ion thruster. Heritage from NSTAR includes critical processes used in subassemblies, such as the spun-form chamber components; cathode heaters and components; ion optics; propellant isolators; flake-containment mesh; and electrical wiring. The continuity between NSTAR and NEXT is intended to minimize the set of unknowns associated with a new mechanical design and maximizes the value of previous investments in ion engine technology.

Performance assessments will evaluate engine operation over its design input power throttling and specific impulse ranges. They will also provide information for service life and thermal modeling. Performance assessments will be conducted on all EM and PM engines in Phases 1 and 2. These will include component (i.e. discharge chamber, neutralizer, and ion optics performance) and engine performance.

The following beam parameters will also be documented:

- Beam flatness for grid service life modeling
- Beam divergence to determine thrust losses

- Beam plasma potentials for spacecraft interactions modeling
- Doubly-to-singly-charged ion current measurements to determine thrust losses
- Thrust vector stability

The selection criterion for the facilities proposed for engine performance assessments is to establish the highest-fidelity simulation required for each specific test to yield transportable results. For wear test evaluations of the engine, GRC's VF5 and VF6, the highest pumping speed and largest electric propulsion vacuum facilities in the country, will be used.

### Thruster Life Testing and Assessment

The NEXT 40-cm engine life capability will be validated through a combination of high-power wear tests and modeling. Previous ion engine wear tests revealed unanticipated wear mechanisms [16,17,18]. Experimental evaluations form the backbone of the NEXT validation process and are essential to life demonstration. The wear tests, *in situ* diagnostics, and real-time diagnostics will yield reliable life data with minimal time expenditure and cost. The lifetime modeling efforts will be implemented to reduce the experimental workload and enhance the confidence in the design life. A list of key life-limiting issues is given in Table 2, along with the techniques to be employed in the life assessment.

**Table 2 – EPT Life-Limiting Issues**

| Component                          | Life-Limiting Phenomenon  | Technique  |
|------------------------------------|---|--|
| Discharge and Neutralizer Cathodes | <ul style="list-style-type: none"> <li>• Depletion of low-work function material</li> <li>• Keeper and cathode erosion leads to structural failure</li> <li>• Keeper shorts to the cathode</li> </ul> | <ul style="list-style-type: none"> <li>• Wear tests</li> <li>• Component level tests</li> <li>• Real-time diagnostics</li> <li>• Plasma model</li> <li>• Insert Chemistry model</li> <li>• Probabilistic Failure Analysis (PFA)</li> </ul> |
| Accel Grid                         | <ul style="list-style-type: none"> <li>• Aperture enlargement leads to electron backstreaming</li> <li>• Structural failure due to erosion</li> <li>• Accel and screen grid short together</li> </ul> | <ul style="list-style-type: none"> <li>• Wear Tests</li> <li>• Real-time diagnostics</li> <li>• <i>In situ</i> diagnostics</li> <li>• Plasma model</li> <li>• PFA</li> </ul>   |
| Screen Grid                        | <ul style="list-style-type: none"> <li>• Removal of web leads to: direct impingement on the accel grid, and structural failure</li> </ul>   | <ul style="list-style-type: none"> <li>• Wear Tests</li> <li>• Real-time diagnostics</li> <li>• Plasma model</li> <li>• PFA</li> </ul>   |
| Propellant Isolators               | <ul style="list-style-type: none"> <li>• Increasing leakage current with time</li> </ul>  | <ul style="list-style-type: none"> <li>• Wear tests</li> <li>• Component level tests</li> </ul>  |

### Thruster Development Status

Fabrication of LM 40 cm engines has been completed, and detailed performance characterizations are on-going at NASA GRC. An EM 40-cm engine design (see Figure 1) based on the most promising LM concept has been assembled and is also under test. The design of several components for the EM engine—including the spun-formed, partial-conic discharge

chamber, hollow-cathodes, and ion optical system—are derived from NSTAR engine technology.

Table 3 provides performance data for the LM version of the 40 cm engine operating on xenon propellant for a power range from about 1.1 to 6.9 kW, where  $V_{bps}$  is the beam power supply voltage. As indicated, the specific impulse and efficiency varies from about 4060 seconds and 0.69 at 6.9 kW, down to about 2300 seconds and 0.51 at 1.1 kW.

**Table 3 – EPT Performance**

| V, $V_{bps}$ | F, mN | $I_{sp}$ , sec | $P_{in}$ , kW | Eff. |
|--------------|-------|----------------|---------------|------|
| 5.92 mg/sec  |       |                |               |      |
| 1800         | 238   | 4060           | 6.90          | 0.69 |
| 1570         | 222   | 3750           | 6.06          | 0.67 |
| 1400         | 211   | 3570           | 5.50          | 0.67 |
| 1180         | 192   | 3255           | 4.70          | 0.65 |
| 4.65 mg/sec  |       |                |               |      |
| 1800         | 182   | 3950           | 5.29          | 0.67 |
| 1570         | 170   | 3690           | 4.67          | 0.66 |
| 1400         | 160   | 3480           | 4.21          | 0.65 |
| 1180         | 148   | 3200           | 3.64          | 0.64 |
| 1020         | 137   | 2970           | 3.21          | 0.62 |
| 3.43 mg/sec  |       |                |               |      |
| 1800         | 135   | 3970           | 3.98          | 0.66 |
| 1570         | 126   | 3700           | 3.51          | 0.65 |
| 1400         | 119   | 3490           | 3.17          | 0.64 |
| 1180         | 109   | 3200           | 2.74          | 0.63 |
| 1020         | 102   | 2980           | 2.43          | 0.61 |
| 2.17 mg/sec  |       |                |               |      |
| 1800         | 81.4  | 3770           | 2.44          | 0.62 |
| 1570         | 75.7  | 3510           | 2.15          | 0.60 |
| 1400         | 71.5  | 3310           | 1.95          | 0.60 |
| 1180         | 65.7  | 3040           | 1.69          | 0.59 |
| 680          | 49.6  | 2300           | 1.11          | 0.51 |

## ION ENGINE DEVELOPMENT ACTIVITIES

### High-Power Engine and Component Technologies

The long-range goal of this activity is to develop and demonstrate ion engine operation at 30 kW input power levels and above, at specific impulse levels of interest to support human exploration of the solar system. A near term objective is the demonstration of ion engine operation in the range of 10 to 30 kW, at 4000 seconds specific impulse.

As part of this activity, development work in high-current hollow cathodes was initiated at NASA GRC. This included the design and fabrication of high-current hollow cathode assemblies [19,20]. A laboratory model 100-A class cathode operated stably over a range of emission currents from about 14 A to 100 A on xenon propellant, although thermal-redesign is required to yield adequate life time at >50 Amperes emission current.

### NSTAR Derivative Technologies

The activities under this element are focused on development of component technologies for NSTAR-derivative high-throughput ion engines.

### Ion Optics

The primary objective of this effort is to develop and validate ion optics designs which yield a > 2x increase in life



over that demonstrated with the Deep Space 1 ion engine optics technology. This includes efforts in both metallic (molybdenum and titanium) as well as carbon-based (carbon-carbon, and pyrolytic graphite) ion optics designs.

In the metallic optics work, evaluations of both thick-accelerator-grid (TAG) electrode molybdenum designs, and titanium designs are underway. TAG molybdenum ion optics of 30 cm diameter have been manufactured in-house and characterized over, and beyond, the Deep Space 1 power throttling envelope [21]. The TAG ion optics utilize a 50% thicker accelerator grid (relative to the Deep Space 1 design) in an attempt to double the optics service life. The TAG design has demonstrated comparable perveance to that obtained with flight ion optics for the NSTAR Deep Space 1 engine [21].

Titanium ion optics of 30 cm diameter have also been manufactured and performance characterized [22]. The potential advantages of titanium, over molybdenum, include a lower volumetric sputter erosion rate, reduced cost of manufacturing, and reduced optics assembly mass. The performance of these optics compare favorably to those of the NSTAR design as well. Additional work to be performed includes detailed evaluations of the transient electrode gap change of titanium optics under thermal load.

Carbon-based ion optics are also under design and manufacturing, with activities including fabrication of carbon/carbon composite grids using multiple vendors with alternate processes, and fabrication of pyrolytic graphite grids of 8 cm (Figure 2) and 30 cm diameter from a commercial supplier [23]. Graphite carbon has a very low sputter yield and coefficient of thermal expansion; the latter property is nearly zero in some situations. This is important in order to maintain precise geometric alignment between screen and accelerator aperture holes. High performance ion thrusters typically have large grid span-to-gap ratios which are vulnerable to thermal distortion. The high geometric stability of carbon materials may permit larger diameter thruster designs while still maintaining high beam density. Alternately, it may be possible to significantly increase the beam density of low power ion thrusters, improving applications on small spacecraft.



**Figure 2 – 8-cm Diameter Pyrolytic Graphite Ion Optics (shown with 8-cm engine).**

### **Cathode Lifetime**

The requirements on hollow cathode life are growing more stringent with the increasing use of electric propulsion technology and demand for greater total impulse capability. The life limiting mechanism that determines the entitlement lifetime of a barium impregnated thermionic emission cathode is the evolution and transport of barium away from the emitter surface. In concert with a robust hardware development effort at NASA GRC [19,24], a model is being developed to study the process of barium transport and loss from the emitter insert in hollow cathodes [25].

The coupled thermochemistry/diffusion model has been developed for xenon hollow cathode operation and it shows good qualitative agreement with observable phenomena in hollow cathodes. The model has been used initially to examine the evolution and loss of Ba from the insert impregnate. Model results have also been compared to qualitative experimental evidence of chemical processes within the cathode insert.

### **Flake Retention**

Grit-blasted wire mesh on stainless steel backing is commonly used in ion engine discharge chambers to ensure adherence of sputtered coatings. Spalled coatings or flakes pose some risk in shorting the high-voltage ion optics. If spalling does occur, the fine wire mesh ensures the spalled material will have sufficiently small dimensions so electrode gaps cannot be bridged. A 8,200 hour ground test of a 2.3 kW ion engine had no problems that could be attributed to spalled flakes [1]. This test had a xenon throughput of 88 kg. Post-test analyses indicated the coatings and flakes were primarily molybdenum and were in the 2  $\mu\text{m}$  to 12  $\mu\text{m}$  thickness range. The Deep Space One ion engine has operated in zero-g conditions for over 16,000 hours, and no problems have been encountered due to spalling of the chamber coatings.

### **Sub-Kilowatt Ion Propulsion**

The goal of this activity is to develop lightweight (< 3.0 kg combined mass), sub-kilowatt thruster and power processor technology. Performance goals include 50% efficiency at 0.25 kW, representing a 2x increase over state-of-the-art.

The sub-kilowatt ion propulsion activity includes both an in-house hardware development element for the thruster and power processor, as well as a contracted system element. At NASA GRC, the fabrication and performance assessment of a small (0.25 kW class) laboratory model thruster with an 8 cm beam diameter has been completed [26,27,28]. Also at NASA GRC, first- and second-generation breadboard power processors have been fabricated and successfully integrated with the 8 cm thruster [29].

### **Selected Fundamentals**

#### **Low-Energy Sputtering**

The objective of this activity is to develop a fundamental understanding of low energy sputtering processes, and to measure low energy sputtering yields of materials relevant to ion thrusters. This activity is being conducted jointly with Tuskegee University. A long range goal is to quantify the sputter yields of relevant refractory metals under ion bombardment, down to the threshold energies. A near-term

goal is the quantification of the sputter yields of Ta, Ti, and Mo under Xe ion bombardment down to 100 eV.

To date, low energy (< 500 eV) xenon ion sputtering yields were measured of molybdenum and tantalum targets [30]. The sputtering was carried out in an ultra high vacuum chamber at NASA GRC. The targets were bombarded with a xenon ion beam generated by an ion gun capable of producing a beam current of approximately 1  $\mu$ A. The sputtered atoms were captured on a semi-cylindrical aluminum collector strip, which was subsequently removed from the vacuum chamber and analyzed with a Rutherford Backscattering Spectrometer. The sputter yield was then obtained from the atomic density and thickness of the sputtered film.

### **Cathode High Energy Ion Formation**

The production of energetic ions in a hollow cathode discharge is well documented [31,32,33,34]. These energetic ions have been detected at energies well in excess of the discharge voltage. Energetic ions are capable of eroding not only ion thruster cathode and keeper electrodes, but also discharge chamber surfaces such as the screen grid. Cathode production of energetic ions may be a thruster life-limiter particularly for long duration missions. The mechanism behind the production of these energetic ions remains unresolved [34]. Proposed mechanisms include multiply-charged ion processes, potential hills, charge-transfer effects, and z-pinch acceleration [34,35]. Though the theories explain the generation of energetic ions given the presence of certain mechanisms, compelling experimental evidence to support any of the theories remains elusive.

In order to address and better understand this issue, an experimental investigation of the formation of high energy ions in high-current hollow cathodes has been initiated at NASA GRC. The goal of this investigation is to characterize the energy distribution and estimate the magnitude of the energetic component of the emitted ions generated by a high current hollow cathode.

### **Propellantless Cathode Technology**

The goal of this activity is to develop a propellantless cathode technology appropriate for electric propulsion applications. Low-power electric thrusters, spacecraft plasma contactors, and electrodynamic tether systems need electron emitters that require very low gas flow rates or zero gas flow to perform their functions. In order to ensure that very low-power colloid thrusters, Field Emission Electric Propulsion devices, ion engines, Hall thrusters, and gridded vacuum arc thrusters are reasonably attractive systems, a propellantless neutralizer cathode is highly desirable. NASA GRC is currently investigating the suitability of field emitter array and ferroelectric emission cathode technologies for electric propulsion thrusters.

## **HALL THRUSTER DEVELOPMENT ACTIVITIES**

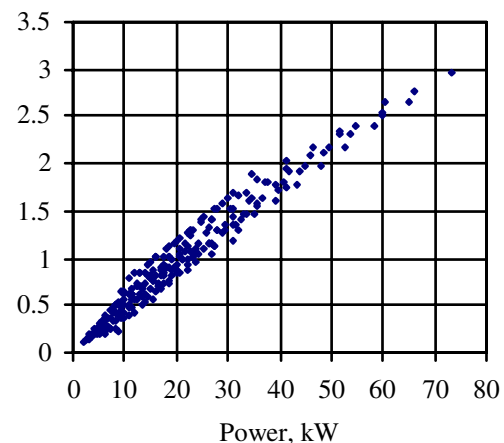
### **High Power-High Thrust Performance**

The NASA Glenn Research Center Hall thruster program developed the laboratory model NASA-457M thruster (Figure 3). The thruster was designed and fabricated to nominally operate at 50 kW and 100 A. The initial test program focused on establishing the operational characteristics and performance

of the thruster at conservative power densities to identify any potential design deficiencies and to evaluate the viability of the scaling relationships used to design the thruster. Subsequent tests investigated thruster performance over an extended range of operating conditions. The results, as can be seen in Figure 4, demonstrated operation up to 72 kW and 3 N of thrust. At the design point of 500 V, 100 A discharge specific impulse and efficiency were 2747 seconds and 63%, respectively. Total specific impulse and efficiency with a non-optimized cathode were 2528 seconds and 56%, also at 500 V, 100 A. Thruster operation was stable over all conditions; the maximum power demonstrated thus far being limited only by available power supplies. Manzella, et al. has a complete description of the thruster and test results [36].



**Figure 3 – NASA-457M (large thruster in the center), NASA-120M (small thruster in center), SPT-100 (left), T-160 (right).**



**Figure 4 – Demonstrated Thrust (in Newtons) and Power Range of the NASA-457M Hall Thruster.**

Future efforts will focus on the effect of an auxiliary magnetic trim coil incorporated in the NASA-457M. Other NASA GRC funded efforts have recently shown how similar trim coils can enable efficient operation over a wide range of operating conditions [37,38].

Previous tests on smaller thrusters at NASA GRC have shown maximum discharge efficiencies at discharge voltages of 500 to 800 V depending on the flow rate [39,40]. Operation up to 1000 V and 100 kW will be investigated in order to establish maximums with regard to efficiency and specific impulse for the NASA-457M. These results will be incorporated into the findings from the high specific impulse work to be described subsequently.

### **50 kW Hall Thruster Development**

The 50-kW Hall thruster program is sponsored by the In-Space Program and is developing a 50 kW-class Hall propulsion system through contract with industry. In February 2001, NASA released a request for offer (RFO) for the design and development of a 50 kW Hall thruster that could minimally provide 2.5 N of thrust at 50 kW using xenon as a propellant with a target thruster efficiency of at least 60%.

Proposals were received, evaluated and Phase I design awards were made to General Dynamics and Busek Co. Due to funding limitations, only the General Dynamics Phase II fabrication option was exercised. General Dynamics offered two concepts including an SPT-type device and a SPT/TAL hybrid device. These proposed devices have been designed with an aggressive power density to enhance performance and reduce thruster mass. Typical Hall thruster power densities are nominally 55 W/cm<sup>2</sup> whereas the thrusters being considered will have a nominal power density of 110 W/cm<sup>2</sup>.

In PY2002, funding constraints limited the scope of this effort to the fabrication of these devices. In PY2003, the functionality and performance of both thrusters will be evaluated. The thermal design will be verified and feasibility of the elevated power density assessed.

### **High Specific Impulse Performance**

The Hall thruster program at NASA GRC has in recent years awarded several contracts to extend the operational envelope of Hall thrusters from the current SOA. These efforts have included investigations of operating thrusters at specific impulses ( $I_{sp}$ ) as low as 1000 seconds [37] and as high as 4100 seconds [39,40,41].

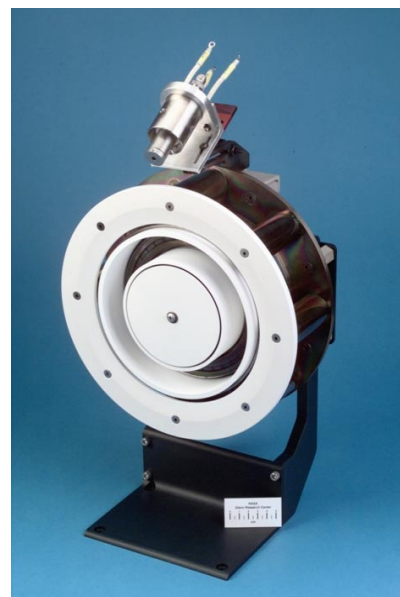
Extending the operational envelope of Hall thrusters will benefit the two most prevalent trends in the community today, multi-mode and high power thrusters. For multi-mode thrusters, developing an understanding of the processes involved has the potential to lead to Hall thrusters that operate at  $I_{sp}$ 's and efficiencies comparable to ion thrusters while offering the benefits of operating the same thruster in high thrust mode at reduced  $I_{sp}$  (~1000 seconds). For high power thrusters, developing the design knowledge to extend the efficient operating range could be vitally important in controlling waste heat. A 50 kW Hall thruster utilizing standard design philosophies will most likely be about 60% efficient, which translates to as much as to 20 kW of waste heat that must be radiated or conducted away from the thruster. Clearly, the ability to increase the efficiency of a high power Hall thruster

beyond the SOA would be beneficial to spacecraft integration issues.

In the high  $I_{sp}$  regime, contracts were awarded in previous years to investigate Hall thruster performance characteristics at  $I_{sp}$ 's greater than 3000 seconds. The first supplier was the Boeing Corporation, which through subcontract to TsNIIMASH delivered the D-80 anode layer thruster [40]. The Atlantic Research Corporation supplied the second thruster, through subcontract to Fakel. This device was a magnetic layer thruster designated the SPT-1 [39]. The third supplier was the Busek Co., which built the BHT-HD-1000, also a magnetic layer thruster [41]. The thrusters have been operated at maximum voltages of 1700 V, 1250 V, and 1000 V for the D-80, SPT-1 and BHT-HD-1000, respectively. The D-80 demonstrated an  $I_{sp}$  range of 1600 to 4100 seconds, the SPT-1 a range of 1600 to 3700 seconds and the BHT-HD-1000 a range of 1600 to 3300 seconds. Each thruster exhibited a range of voltages where the anode efficiency was maximized. This voltage was between 500 to 800 V depending on the mass flow rate. Precise evidence for this peak efficiency was not identified, but it was likely attributable to either increased electron current due to insufficient magnetic trapping or increased production of multiply-charged species due to the high voltages.

Building on these efforts, the objectives for NASA PY2002 and beyond are to: 1) identify the physical mechanisms that result in maximum efficiency with voltage, and 2) develop the design tools necessary to extend efficient operation in excess of the 800 V ceiling recently observed. To achieve these goals, internal GRC efforts are underway as well as investigations through a grant with the University of Michigan [38,42] and through an NRA with MIT and Busek [43].

In conjunction with the University of Michigan, a 5 kW high voltage thruster designated the NASA-173M V.1 has been constructed and is shown in Figure 5. Initial testing has concentrated on characterizing the performance and identifying design variables most likely to increase efficiency. A follow on thruster designed entirely at GRC, the NASA-173MV.2, is currently in fabrication and scheduled for testing in Fall 2002.

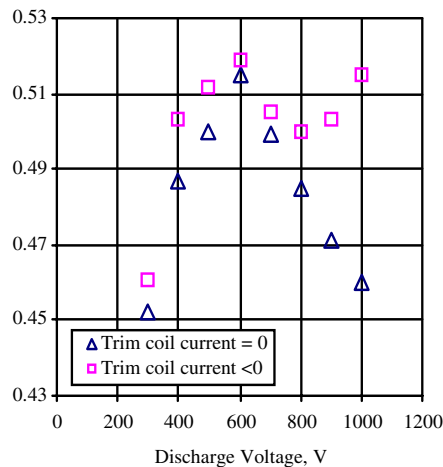


**Figure 5 – The NASA-173M V.1.**



Recent results from the NASA-173M V.1 operating at 300 to 1000 V and 5 mg/s are shown in Figure 6 [38]. The figure shows two sets of data, with and without a trim coil energized. When there is no trim coil, only the inner and outer coils are used (the traditional design), the previously observed peak efficiency is found. With the trim coil energized, the efficiency is maintained at >50% above 600 V, demonstrating that efficient Hall thruster operation is possible at high voltage when the magnetic field is tailored for the operating point. Similar investigations at higher flow rates, and correspondingly higher efficiencies, are planned for the NASA-173M V.2 in coming months.

Work beyond PY2002 will focus on characterizing the plasma parameters of the NASA-173M thrusters using several diagnostics to assess what effects changes in the thruster are having on performance. These diagnostics will include the use of ion current density probes, and energy diagnostics such as ExB probes or ion energy analyzers. Internal plasma diagnostics using a high-speed probe system at the University of Michigan are also planned.



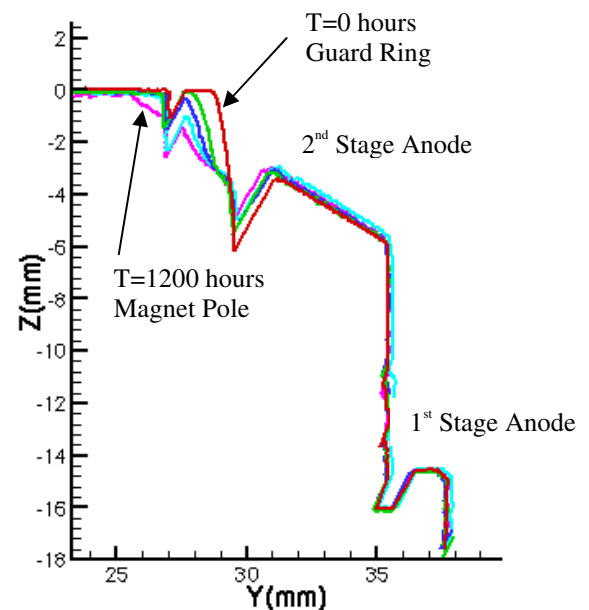
**Figure 6 – Efficiency versus voltage for the NASA 173M V.1 operating at 5 mg/s.**

### High Voltage Thruster Life

In 1999, NASA GRC began investigating Hall thrusters with specific impulses greater than the 1500 to 1700 seconds of SOA thrusters [39–42,44]. This increase in specific impulse over conventional xenon Hall thrusters was achieved by increasing discharge voltages from 300 V to as much as 1700 V. Operation at these elevated discharge voltages substantially increased the kinetic energy of the ions produced during thruster operation. At any discharge voltage, some small fraction of the ions accelerated by the thruster impinge upon the thruster itself, eroding critical components, eventually causing the thruster to wear out. While experiments have shown this in the laboratory at voltages of 300 to 500 V, the effect of discharge voltages in excess of 500 V on thruster lifetime has not been investigated.

The objective of the current investigation was to evaluate the effect of high voltage operation on thruster erosion. Specifically, a D-80 anode layer thruster was operated at 700 V and 4 A. A 700 V discharge voltage was selected because a

previous performance investigation demonstrated maximum efficiency at voltages between 500 to 800 V [40]. A 4 A discharge current was selected for extended operation because this was the maximum value that could be achieved while maintaining the desired steady state thermal conditions. The anode efficiency and specific impulse at this operating point were 57% and 2625 seconds, respectively. While efficiency over 70% and specific impulse over 4000 seconds were demonstrated at higher discharge voltages and currents, high thruster temperature prohibited operation for extended periods of time. The thruster was operated for a total duration of 1200 hours in this investigation. Erosion of critical thruster components was evaluated every 300 hours. The thruster exhibited asymmetric erosion that exposed the magnetic poles between the 600 and 900 hour increment (Figure 7). Even as the magnetic poles were exposed the robustness of the thruster was demonstrated by the continued nominal operation of the thruster. Jacobson [45] has a complete description of the thruster and test results.



**Figure 7 – D-80 Hall Thruster High Voltage Erosion Cross-Sections.**

### SUMMARY

NASA's Glenn Research Center has been selected to lead development of NASA's Evolutionary Xenon Thruster system, as follow-on to the highly-successful Deep-Space 1 system. The system is envisioned to incorporate a lightweight 40-cm diameter electric propulsion thruster, a lightweight, modular power processing unit, a lightweight, low-volume, highly flexible advanced xenon feed system, and a simple, lightweight gimbal. During this two-phase program, the thruster will be developed to prototype model, with manufacturing of this unit conducted by General Dynamics Space Propulsion Systems. General Dynamics will also manufacture the AXFS. Boeing Electron Dynamic Devices will manufacture the PPU.

To date, fabrication of laboratory model 40 cm engines has been completed at GRC, and detailed performance

characterizations are on-going at NASA. An engineering model 40-cm engine design has been assembled at GRC and is also under test. Typical engine performance at 6.9 kW is 4060 seconds specific impulse, 238 mN thrust, and 69% efficiency.

The NASA Hall thruster program currently supports tasks for a number of customers including the Energetics Program, the Space Solar Power Program, and the In-space Propulsion Program. In program year 2002 the following was accomplished as part of these tasks: 1) A laboratory Hall thruster was built and tested to 72 kW and 3 N of thrust; 2) A 5.5% increase in thruster efficiency at high voltage was demonstrated by modifying the magnetic field profile; 3) High voltage Hall thruster lifetime was quantified; 4) A 96.5% efficient breadboard power module was demonstrated; 5) A technique for measuring the impedance of a Hall thruster was demonstrated; and 6) A plan for investigating the dynamic electrical behavior of Hall thrusters was presented.

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