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**DEVELOPMENT OF HIGH POWER HALL THRUSTER SYSTEMS TO ENABLE THE NASA
EXPLORATION VISION**

**Jerry Jackson¹, Joseph Cassady², May Allen³, Roger Myers⁴
Todd Tofil⁵, Dan Herman⁶ and Eric Pencil⁷**

¹*AEPS Program Manager, Aerojet Rocketdyne, 8900 De Soto Ave., Canoga Park, CA USA 91304,
Jerry.Jackson@rocket.com*

²*Executive Director, Space, Aerojet Rocketdyne, 1300 Wilson Blvd #1000, Arlington, VA USA 22209,
Joe.Cassady@rocket.com*

³*AEPS Chief Engineer, Aerojet Rocketdyne, 11411 139th PI NE, Redmond, WA USA 98052,
May.Allen@rocket.com*

⁴*AEPS Technical Consultant, Aerojet Rocketdyne, 11411 139th PI NE, Redmond, WA USA 98052,
Roger.Myers@rocket.com*

⁵*Program Manager, NASA GRC, 21000 Brookpark Rd., Cleveland, OH USA 44135, Todd.A.Tofil@nasa.gov*

⁶*Project Lead Engineer, NASA GRC, 21000 Brookpark Rd., Cleveland, OH USA 44135,
Daniel.A.Herman@nasa.gov*

⁷*Project Lead Engineer, NASA GRC, 21000 Brookpark Rd., Cleveland, OH USA 44135, Eric.Pencil@nasa.gov*

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ABSTRACT:

The next phase of space exploration missions requires high power Solar Electric Propulsion (SEP) systems for large-scale science missions and cargo transportation. Development is underway at Aerojet Rocketdyne on Hall thruster systems that are intended to bracket the needs of future NASA SEP missions in support of space exploration. The Advanced Electric Propulsion System (AEPS) program is developing and qualifying a 13.3kW Hall thruster system to be demonstrated on the Power and Propulsion Element (PPE), which is intended to be the first element of a Lunar Outpost Platform – Gateway (LOP-G). The NextSTEP program is integrating a nested Hall thruster into a 100kW system and testing it for 100 hours. These two programs will provide a path to efficient in-space propulsion that will allow NASA to transfer the large amounts of cargo that is needed to support human missions – first to the moon and then on to Mars.

The Advanced Electric Propulsion System (AEPS) program is completing development, qualification and delivery of five flight 13.3kW EP systems to NASA. The flight AEPS system includes a magnetically shielded long-life Hall thruster, Power

Processing Unit (PPU) and a Xenon Flow Controller (XFC). The Hall thruster, developed and demonstrated by NASA, operates at input powers up to 12.5kW while providing a specific impulse over an estimated 2800s at an input voltage of 600V. The power processor is designed to accommodate an input voltage range of 95-140V, consistent with operation beyond the orbit of Mars. The integrated system input power is continuously throttleable between 3 and 13.3kW. Component level testing of the EP String has begun with prototype hardware.

The NextSTEP program is developing a 100kW Electric Propulsion (EP) system using a nested Hall thruster designed for powers up to 250kW, a modular power processor and a modular mass flow controller. While the program objective is to operate the integrated EP system continuously at 100kW for 100hrs to demonstrate thermal stability and support the development of system life time models, it builds on decades of experience with long-life Hall thrusters and the design is evolvable to a capability of 250kW. Design upgrades that demonstrate the 100kW EP system have been completed and tested.

Aerojet Rocketdyne (AR) is excited to support NASA as it extends human reach into deep space and believes that these programs will provide the propulsion to make such missions affordable and sustainable. These systems provide NASA with a

range of options to power its deep space transport vehicles. This paper presents the mission requirements for supporting the NASA exploration vision, as well as the status for the high power Hall thruster systems in development.

1. INTRODUCTION

The next phase of robotic and human deep space exploration missions is enhanced by high performance, high power solar electric propulsion systems for large-scale science missions and cargo transportation. A high-power Solar Electric Propulsion (SEP) element is integral to NASA's phased Mars exploration vision, illustrated in Figure 1 which presents an approach to establish an affordable evolutionary human exploration architecture.¹ Recent studies for NASA's Human Exploration and Operations Mission Directorate (HEOMD) and Science Mission Directorate (SMD) have demonstrated that SEP capability, with its substantially higher specific impulse (I_{sp}), can be enabling for both near term and future architectures and science missions.²

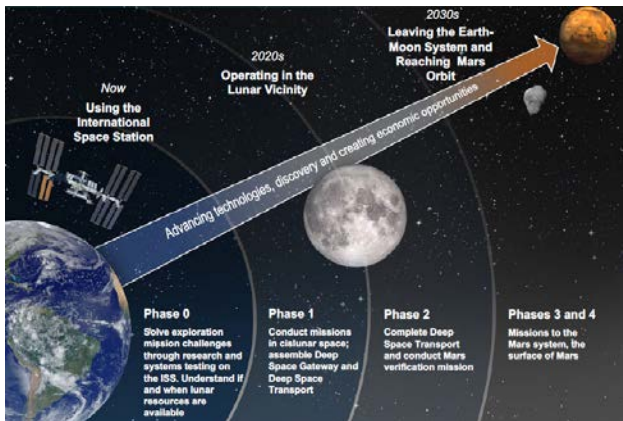


Figure 1. NASA EXPLORATION APPROACH

2. NASA EXPLORATION VISION

With the passage of the 2010 NASA Authorization Bill, NASA was directed by law to work toward putting humans in the vicinity of Mars by the 2030s. Since the US 2016 Presidential election, the new administration has stepped up their support and direction of the NASA vision to send humans beyond LEO. This has included a near-term focus on the Moon and operations in the lunar vicinity.

In response to this direction NASA has laid out an ambitious set of missions in the 2020s to establish the capabilities required for exploration in deep space. This begins with the launch of the SLS and Orion crew vehicle on a test flight without crew in

2019, followed by the first launch of the SLS / Orion system with crew in 2021/2022. At the same time, NASA is planning to assemble a Lunar Orbiting Platform– Gateway (LOP-G) in a near rectilinear halo orbit (NRHO) around the Moon. The first element of the LOP-G is the Power and Propulsion Element (PPE), which is scheduled for launch in 2022. The PPE will include a 50 kW-class Solar Electric Propulsion (SEP) system to be used for orbit placement and repositioning. Although the Space Launch System remains central to NASA's plans for exploration, the agency will engage partners through nontraditional arrangements to enable various launch options for the LOP-G elements.

Aerojet Rocketdyne has partnered with NASA to develop the 13 kW Advanced Electric Propulsion System (AEPS) thruster strings that will be the primary propulsion for the PPE. Details of the SEP demonstration are still being finalized and depend on whether the PPE is launched as a co-manifested payload on the SLS or as a separate launch on a commercial launch vehicle. However, in either case it is NASA's intention to perform a thorough in-space test of the entire SEP system including AEPS and the advanced solar arrays. Among the capabilities to be demonstrated is refueling of the xenon tanks to allow for additional thrusting and extend the life of the LOP-G. The LOP-G lifetime is planned for 15 years and it is likely that there may be several orbital adjustments made over that time. As shown in Figure 2, the SEP system is capable of moving the entire LOP-G stack from NRHO to a low lunar orbit and then back again. Such orbital maneuvering capability opens up the possibility of much broader support for lunar surface operations for international and commercial partners.

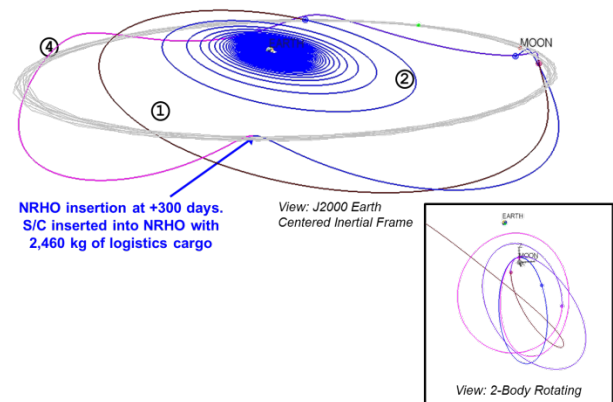


Figure 2. LOP-G orbit transfer maneuvering

Longer term, NASA is considering options for delivery of cargo and crew to Mars orbit. While final decisions have not been made, both SEP or SEP-

chemical hybrid vehicles are being considered for both applications. In both cases, the 13kW AEPS thruster strings are the baseline electric propulsion system, though options for higher power Hall thruster systems are also being evaluated.

3. ADVANCED EP SYSTEM (AEPS)

The development of a 13.3 kW Hall thruster system, led by the NASA Glenn Research Center (GRC) and the Jet Propulsion Laboratory (JPL), began with the internal development of a high-power Hall thruster³ and power processing unit⁴. This technology development work has since transitioned to Aerojet Rocketdyne via a competitive procurement selection for the Advanced Electric Propulsion System (AEPS) contract. The AEPS contract includes the development, qualification, and delivery of five 13.3 kW Hall thruster flight strings, including the qualification string.

3.1 AEPS System

The AEPS System EP string consists of a magnetically-shielded Hall thruster, Power Processing Unit (PPU), Xenon Flow Controller (XFC), and associated electrical harnessing between these components. The PPU can process up to 13.3 kW of power to support the thruster output and operations (See Table 1). The Hall thruster uses xenon propellant and up to 12.5kW of power from the PPU to provide over 589 mN of thrust. The XFC is a flow system that is controlled by the PPU and can operate at feed pressures as high as 3000psi in a spacecraft xenon pressure regulator failure scenario. The AEPS program has completed the System Requirement Review (SRR); the EP string, thruster, PPU and XFC Preliminary Design Review (PDR); engineering model development; and early system integration testing. Engineering Development Unit (EDU) hardware fabrication is in work to support component and system level testing.

Total Input Power	Discharge Voltage	Thrust	Specific Impulse	Total System Efficiency
13.3 kW	600 V	589 mN	2800 s	57%
11.1 kW	500 V	519 mN	2600 s	55%
8.9 kW	400 V	462 mN	2300 s	54%
6.7 kW	300 V	386 mN	1900 s	52%

Table 1. AEPS Performance Requirements

The AEPS System is designed to function in several operating modes. Given a set of commands from the

spacecraft, the system is able to run autonomously with closed-loop control and fault monitoring programmed into the PPU software. The system returns telemetry and can accept updates to the software and data tables during mission operations.

System level testing will follow component level testing. The system testing includes electrical checks and a full system performance test to ensure the integrated system operates as expected. A system level radiated emissions characterization is planned as part of the development effort. Environmental testing will be completed at the component level.

3.2 AEPS Power Processing Unit

The AEPS PPU leverages the work performed by NASA GRC on a brassboard power processor that was utilized in the integration testing of the HERMeS thruster.⁴ The brassboard High Power 120/800 V Power Processing Unit (HP 120/800 V PPU) was required to have all of the functionality to operate a Hall thruster, including the auxiliary power, master control board, telemetry, and filters. The unique aspect of this development was the wide range (95 to 140 V) of the input voltage for the PPU. The test results of the HP 120/800 V PPU helped to guide the design of the AEPS PPU.

The AEPS PPU (Figure 3) provides discharge power to the thruster as well as system control and other associated functions. The PPU provides up to 30 A of discharge current over a variable voltage output range of 300 to 400 V and 20.8A from 400V to 600V. Other features include XFC control, cathode heater power, cathode keeper power, inner magnet power, outer magnet power, telemetry, system health monitoring, and spacecraft communications.

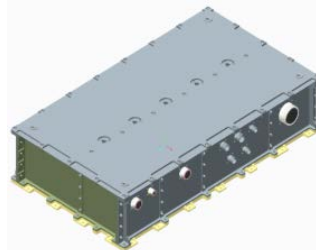


Figure 3. PPU Enclosure

Unique capabilities of the PPU include its high efficiency target of 95% with a wide input voltage range of operation with a baseplate temperature range of -15 to 50°C. With the mass burden associated with increased input voltage range and built-in digital control and interface functionality, the PPU has a specific power (0.3 kW/kg) comparable to the current state-of-the-art (SOA) range of 0.25 – 0.33 kW/kg. Specific performance parameters are enumerated below:

- Input Power: 13.3kW input (14kW contingency)
- Input voltage: 95V – 140V, Unregulated
- Output Voltage: 300V to 600V
- Output Current: 20.8A (400V to 600V), up to 30A for 300-400V
- Output Power: 12.5kW at 600V
- Dimensions: 900mm X 518mm X 20mm
- Mass: ~50 kg

The PPU block diagram in Figure 4 shows signal and power flow. The PPU provides 12.5 kW of regulated electric power to an electric thruster. The PPU controls the thruster output and operations. The XFC is a low pressure flow system that is also controlled by the PPU.

The PPU includes the following list of major component subsystems:

- System Control Board (SCB)
- Discharge Supply Unit (DSU)
 - Discharge Master Controller (DMC)
 - Power Module (Qty 4)
 - Output Filter
- Auxiliary Power Supply Assembly
 - Inner/outer Magnet, Heater, Keeper
- Xenon Flow Control Board (XFCB)
- Housekeeping Power Supply Board (HPSB)
- Input Filter Assembly
- Output Filter Assembly
- Cabling and Enclosure

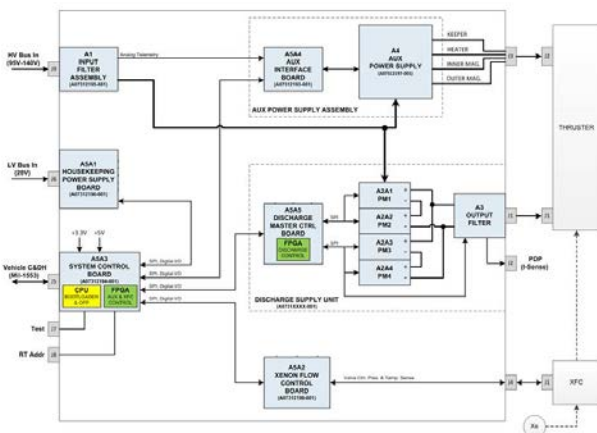


Figure 4. Power Processing Unit Block Diagram

The SCB controls the system and interfaces with the spacecraft communication bus. Through this bus, commands from the spacecraft are received and telemetry from the PPU is transmitted along with timestamp information. The SCB controls the outputs of the PPU by sending commands to the various power supplies in the system (auxiliary

power supplies and discharge supplies). It synchronizes the auxiliary power supplies with the DSU. The SCB monitors sensor information via the internal communications bus. The SCB also provides self-test and fault detection functions, including a watchdog timer function.

The DSU supplies the discharge voltage and current to the thruster. It has a DMC board controlling four Power Modules (PM) in a series/parallel combination. Two sets of series PMs are in parallel with each other. The DMC receives commands from the SCB and translates them into voltage levels for the individual power modules. The DMC provides short circuit protection for spark events. Additionally, the DSU employs zero volt switching and a lossless snubber for increased efficiency.

The Auxiliary Interface Board (AIB) interfaces with the auxiliary power supplies, which provide power to the Cathode Heater, Keeper, Inner Magnet, and Outer Magnet. The XFCB provides closed-loop control of the XFC. The HPSB provides low voltage power to the boards and modules in the PPU.

The input filter assembly provides EMI/EMC filtering and power quality compliance to ensure compatibility with the S/C power bus. In addition to filtering the power input, this board also provides input voltage and current telemetry.

The output filter assembly provides a low impedance source at the thruster breathing modes while minimizing the voltage ripple. In addition to filtering the breathing mode currents from the DSU, the output filter provides DC and AC output voltage and current telemetry. The AC voltage and current characteristics provide thruster performance health monitoring.

When low voltage power is applied, the housekeeping power supply board is activated and provides housekeeping power to the SCB. Once powered, the SCB performs initialization checks and waits for spacecraft commands. Following verification that the SCB high voltage bus is active, the system enters startup, conditions the system for operation, and can then start the thruster.

The SCB continuously monitors the health of the entire EP string, allowing on-going diagnostics for changing conditions over the mission life. This monitoring includes reporting warning or fault level status and discharge voltage and discharge current ripple for the thruster to the spacecraft.

During operation, the SCB provides set-point commands for the heater to condition the cathode, the ignition command to the keeper, and the

discharge current set-point. The magnet supplies provide magnetic field strength necessary for high performance. The SCB commands the DSU to the desired starting voltage and current limit set-point, commands the XFC flow using pressure feedback to set the initial drive, and closes the thrust control loop with current from the DSU output. Figure 5 shows the discharge supply output voltage and current capability from 3kW up to 12.5 kW.

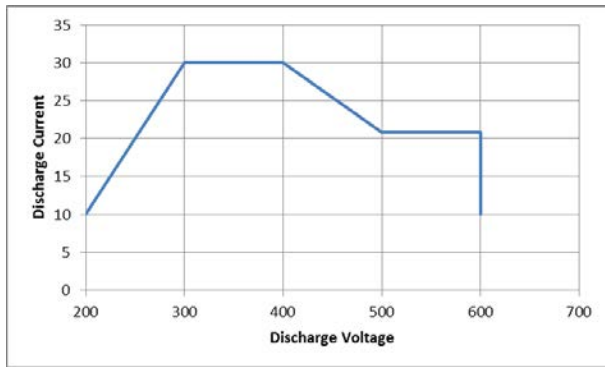


Figure 5 Discharge Current Vs. Voltage

The DMC can adjust the output voltage to a resolution of less than 0.5 V, which provides a continuously variable output voltage from 300 to 630 V. Figure 6 shows a sample of this capability during the early system test. The full range will be verified during design verification testing in 2018. The PPU will go through extensive design verification testing with an engineering model including environmental testing at qualification levels and the first flight production unit will complete full qualification testing with the first flight production unit in 2019.

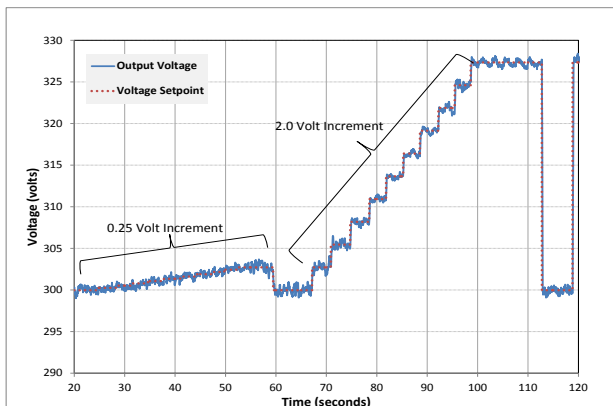


Figure 6. PPU Output Voltage Control

3.3 AEPS Hall Thruster

The AEPS thruster design is based largely on the NASA Hall Effect Rocket with Magnetic Shielding

(HERMeS) Technology Development Unit (TDU). The TDU has demonstrated operation at discharge currents in excess of 30A and discharge voltages up to 800 V.⁴ The TDU is a magnetically shielded, 12.5 kW thruster with a center-mounted cathode and carbon pole covers. The cathode is electrically tied to the conductive pole covers of the TDU. With this novel electrical configuration and effective magnetic shielding, the TDU has demonstrated the low erosion rates necessary to meet the mission requirement of 23,000 hours of operation.⁵

The performance capability of the AEPS thruster is expected to be the same as the NASA HERMeS TDU. Critical features such as the gas distributor and magnetic circuit design of the HERMeS TDU have been maintained in the AEPS thruster. Total efficiencies up to 68% and specific impulses up to 2800s⁶ should be achievable for AEPS and will be demonstrated during up-coming development tests.

The AEPS thruster design has evolved from the HERMeS TDU in order to improve dynamic stress capability, reduce thermally-induced stresses, ensure long life, and improve manufacturability. Many structural modifications to the design have been implemented in order to increase the environmental robustness of the design to survive flight environmental requirements. The thruster is designed to survive qualification shock levels up to 1000 g's and qualification random vibration levels up to 11.4 g_{rms}.

Detailed thermal analysis guided many thermal improvements which enabled the elimination of the thermal radiator used in the HERMeS TDU design. The AEPS thruster design is capable of operation in deep space between 0.8 AU and 1.7 AU while mounted to a spacecraft interface between -100°C and +150°C. In proposed spacecraft configurations, the thruster radiates only 22W to the spacecraft when operating at discharge power levels of up to 12.5 kW. The thruster is maintained above its minimum qualified temperature via integrated heaters and temperature sensors. These thermal components are operated by the spacecraft's thermal management system. Materials and coatings have been selected to ensure that the thruster can provide a minimum of 5,000 starts. Shielding and material selection will ensure that the thruster survives deep space solar radiation for up to 15 years.

In order to meet the requirements, the AEPS thruster is rated to operate at discharge currents up to 25A and discharge voltages up to 630V. As described in the previous section, operational power

from the PPU is provided via the Discharge and Auxiliary Power Cable Assemblies. The thruster is a maximum 516 mm in diameter and 204 mm tall with a mass that is predicted to be less than 50 kg. The thruster is designed to produce a first natural frequency greater than 70 Hz. Figure 7 below shows a CAD image of the AEPS thruster and the three electrical harnesses integrated into the thruster.

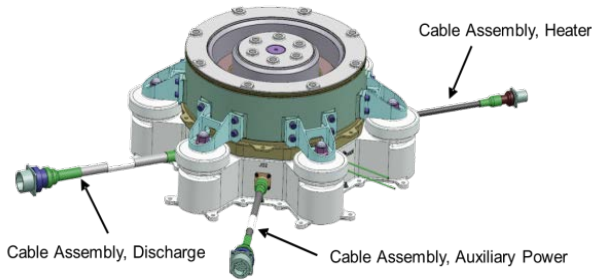


Figure 7. CAD image of AEPS Hall Current Thruster showing integrated harness assemblies

As part of the current program, all of the tooling and test equipment necessary to fabricate and test flight AEPS thrusters are being developed. This support equipment will be verified during manufacturing and test of two EDU thrusters. Procurement and manufacturing of the two EDU thrusters is underway at AR's Redmond, Washington facility. Materials testing was performed to validate several materials that were selected. This data improved our understanding of the material capabilities and design margins. Process development and qualification of all manufacturing process is in work to support the assembly of the thruster.

Development testing is planned to take place in Redmond, at NASA GRC, and at JPL. EDU thrusters will undergo dynamic and thermal qualification level environmental testing, hot-fire performance and wear testing, plasma characterization, and radiated emissions characterization testing in 2018. Wear test data along with testing of critical subcomponents will enable the AEPS team to update and refine thruster models that will be used to predict the life of the thruster. Qualification testing of the first flight production unit is planned for 2019.

3.4 AEPS Xenon Flow Controller

VACCO has been providing electric propulsion components and feed systems for over 20 years. In order to minimize the size and mass of future xenon feed systems, VACCO developed and qualified a Xenon Flow Control Module (XFCM) based on their

Chemically Etched Micro Systems (ChEMS™) technology.

VACCO teamed with Aerojet Rocketdyne on AEPS



Figure 8. XFCM

with the development responsibility for the Xenon Flow Controller (XFC), an application engineered version of VACCO's fully-qualified XFCM (Figure 8). The result is a highly-integrated, compact, low

mass subsystem that provides:

- 10 Micron propellant filtration
- Propellant heater for off-nominal conditions
- A Micro Latch Valve for propellant isolation
- Independently throttleable flow to both the Anode and Cathode
- Flow rate feedback

With a minor adjustment to the previously qualified XFCM design, it is capable of supporting a wide range of flow regimes and electric thrusters well

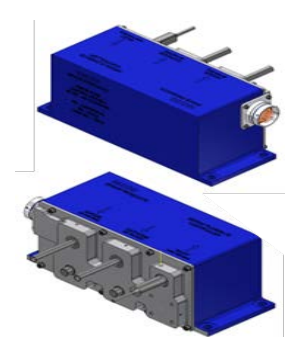


Figure 9. AEPS XFC

beyond 12.5 kW. The AEPS XFC (see Figure 9) is based upon a modified version of the XFCM. It weighs less than 2 kg and measures less than 8 x 8 x 20 cm, with one inlet tube and two outlet tubes. It consists of two major sub-assemblies; a manifold and an enclosure.

The manifold is welded against external leakage with inlet and outlet tubes welded to the outside and the functional components mounted on the inside. Inter-connecting flow paths between components are machined into the manifold eliminating the need for tubing. The enclosure provides environmental protection for the electrical components and four mounting holes for structural attachment. An electrical connector provides for the electrical interface to the PPU.

Propellant enters the XFC through an inlet tube equipped with an integral 10 micron etched disc filter. Flow then passes through an electric heat exchanger that is only required for extreme off-nominal operation. Flow from the heat exchanger passes through a micro latch valve that, when latched closed, provides the first interrupt against

internal leakage. When latched open, flow from the micro latch valve splits into an anode and cathode branches. Both branches are identical except for their flow restrictors. Each branch contains a proportional flow control valve (PFCV), a 100kRad tolerant pressure and temperature transducer, an integrated 40 micron outlet filter/flow restrictor. Flow is controlled by modulating the PFCV to regulate pressure upstream of the flow restrictor. When unpowered, the normally-closed PFCV closes and seals, providing a second interrupt against internal leakage. See the XFC diagram in Figure 10.

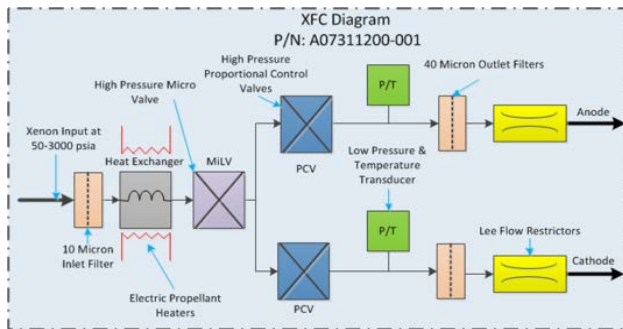


Figure 10. XFC Schematic

Inlet pressure is regulated by the spacecraft to a nominal 40 psia. Under off-nominal conditions (e.g. failed spacecraft regulator), inlet pressure can be as high as 3,000 psia. For this reason, the XFC is rated for a maximum design pressure of 3,000 psia up to the inlet of the pressure control valve with proof pressure of 4,500 psia.

The XFC provides a flow rate of 8 to 23 mg/second of xenon with the ability to set the cathode to anode flow split through independent control of the anode and cathode proportional flow control valves. Cathode flow will be between 4 and 10 percent of the anode flow. The XFC design has less than 1.0×10^{-4} sccs of internal helium leakage and less than 1×10^{-6} sccs of external leakage at the high inlet pressure of 3,000 psia. Each XFC will be calibrated to provide a flow rate measurement accuracy of $\pm 1.25\%$ at the inlet pressure of 40 psia and over a temperature range of 20°C to 50°C. The XFC provides off-nominal operation at 3,000 psia at derated accuracy and set-point capability in the event of a spacecraft regulator failure.

The XFC design has already been qualified to vibration and shock levels in excess of the AEPS vibration levels of 8.1 grms for acceptance testing, 11.4 grms for qualification and shock levels of up to 2000 g's. The XFC is specifically designed for the long durations required by future exploration missions. To that end, the internal components are

designed for infinite fatigue life. Component fatigue life has been verified by analysis, except for the proportional actuators, which will be verified by third-party testing. The XFC is capable of a throughput greater than 1770 kg of xenon. An EDU will be fabricated and will go through qualification-like testing in late 2018. Full qualification testing will be performed on the first flight production unit in 2019.

The NextSTEP (Next Space Technologies for Exploration Partnerships) XR-100 program was awarded to Aerojet Rocketdyne in January of 2016. Under the contract, the Aerojet Rocketdyne team will complete the development of the XR-100, a 100 kW Hall Thruster System, including a 200 kW thruster; critical elements of a 100 kW modular Power Processing Unit (PPU); and elements of the modular xenon feed system. The contract includes system integration testing, and will culminate with a NASA TRL 5 demonstration of a 100 kW system for 100h at thermal steady-state. The specific goals of the NextSTEP project are shown in Table 2.

Metric		Objective
Requirement	TRL 5 demonstration power	100 kW
	TRL 5 steady state operation time	100 h
Objective	Specific Impulse	~2,000-5,000s
	In-space lifetime capability	>50,000 h
	Operational lifetime capability	>10,000 h
	System efficiency	>60%
	Power per thruster	100 kW
	System kg/kW	<5 kg/kW

Table 2. NASA NextSTEP Program Objectives

The modular nature of these technologies enables system scalability to 200kW with the current thruster design and multi-megawatt power levels with scaled thruster design while meeting the long-term system performance and mass objectives of the project. To perform this contract, there is close collaboration with teammates at Aerojet Rocketdyne, the University of Michigan, the NASA Jet Propulsion Laboratory, and the NASA Glenn Research Center.

4.1 NextSTEP System

The XR-100 system provides margin on the performance requirements for Mars cargo transportation. The modularity of the system components features the largest throttling capability of any Hall thruster system to date, with seven different firing configurations and power levels ranging from 2 kW to 200 kW.⁷ Additionally, the X3 is capable of operating over a range of discharge voltages of 200 to 800V.

The XR-100 system uses a PPU capable of delivering 100 kW of power to the Nested Hall Thruster (NHT) and of providing closed loop control of the mass flow controllers. The XR-100 feed system uses an Aerojet Rocketdyne proprietary Propellant Management Unit (PMU) and modular Mass Flow Controller (MFC) design for accurate flow control in a closed-loop system . The spacecraft feed system would include propellant tanks which feed the PMU. The PMU includes latch valves and filtering required to meet range safety and mission assurance requirements. Conditioned propellant from the PMU is fed into the MFCs that allow for system scaling and flow rate tailoring for each Hall thruster channel so that it can be optimized by channel for a range of operation. The baseline propellant is xenon gas, which has been shown to be safe, dense, and long-term storable as compared to cryogenic propellants. The NHT is designed with concentric rings and can operate those rings in any combination. This design is very compact relative to other high power EP systems.

The modular architecture of the XR-100 design has several advantages. The PPU approach enables a distributed PPU architecture with scalability to higher powers that allow vehicle designers the flexibility to spread the PPU module mass and thermal loads for optimal placement on spacecraft, even for spacecraft with different physical designs. The MFC modules allow for optimized flow for each channel of the NHT. This modularity allows for optimized performance and efficiency at several firing configuration and all components have a path for scaling to higher power. Finally, the overall maturity of the NHT approach provides for a low-risk solution to meet high thrust EP demands.

As part of the NextSTEP program, a preliminary set of system requirements were generated and are continually updated as the development of the XR-100 matures. Integrated system testing is one of the key activities that help refine the system requirements. The program team successfully conducted a 10kW integrated system test in January 2018. The test demonstrated stable operation up to 10kW at both 400 and 800V input to the NHT with closed loop control of the MFCs. The team is now preparing for a 100kW demonstration to be performed later this year.

4.2 NextSTEP Power Processing Unit (PPU)

The NextSTEP PPU is required to power three thruster power channels (the inner, middle and outer) providing 13.6kW, 32kW and 55.2kW respectively for a total of 100kW. While the final

PPU will incorporate low power supplies for the cathode, heaters and magnets, the current focus of the NextSTEP PPU effort is the discharge supply since this is the most challenging part of the PPU. The discharge supply uses a modular design that can support multiple configurations for a single Discharge Supply Unit (DSU) or multiple DSUs in parallel and power each of the 3 NHT channels. This architecture offers greater flexibility and can easily expand to greater power levels. Shown in Figure 11, the Discharge Supply system inner channel will have a single DSU, the middle channel will have three DSUs, and the outer channel will have four DSUs; requiring 8 DSUs in total for the 100kW EP system. This architecture will be used for the 100h 100kW test, which is planned for 2018.

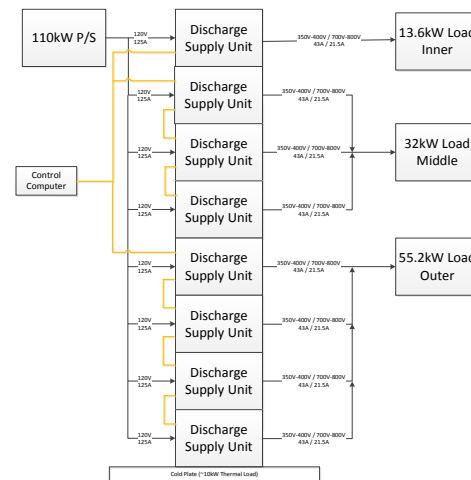


Figure 11. DSU Architecture

Each DSU consists of a Discharge Master Controller (DMC), input filter (IF), output filter board (OFB) and four power modules (PM). The PMs each provide up to 400V and 3.45 kW of power and can be configured in parallel mode or a series/parallel mode. The configuration selection is accomplished with selection switches. The DMC communicates with laboratory control computer (Space Craft C&DH simulator) receiving set point command and providing telemetry for display and data logging. The DMC controls each PM with base the S/C set-point commands. The input filter has a disconnect switch to provide a controlled power sequence during multiple DSU configuration testing.

During the 1st year, a single DSU was designed, built and tested. Each DSU is capable of delivering 13.8kW of power from an input voltage of 95V to 140V. The controls architecture was devised to allow multiple DSUs to operate in parallel on a single thruster anode channel with the returns tied to the one common cathode. The DSU is capable of

delivering 13.8kW at 350V-400V to maximize thrust or 700V-800V to maximize specific impulse (I_{sp})

The testing successfully verified over 10kW operation at output voltages between 400V and 800V.⁸ After this successful demonstration, lessons learned were incorporated into the design to build a 100 kW EP system.

A Mass Flow Control (MFC) valve driver was developed during 2016. Integrated flow control testing of a single valve demonstrated successful closed-loop operation of the MFC. In 2017 the MFC valve driver was expanded to drive five valves required for the 100 kW TRL-5 test. The spacecraft simulator will provide closed loop control between the xenon flow and discharge current for each of the three discharge channels.

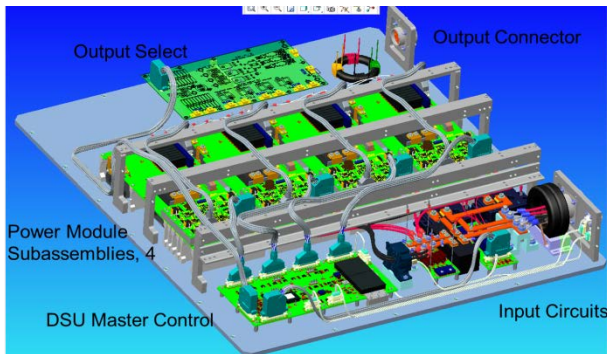


Figure 12. DSU Model with 4 power modules

In 2017, several modifications were made from the first breadboard unit to a demonstration unit. The control system was updated to allow operation of multiple DSUs in parallel. Some of the magnetics were modified to reduce excessive power losses and improve thermal management for the 100h test in vacuum. Thermal modeling and analysis was completed to verify proper thermal configuration of the magnetics updates. Figure 12 shows the final design model.

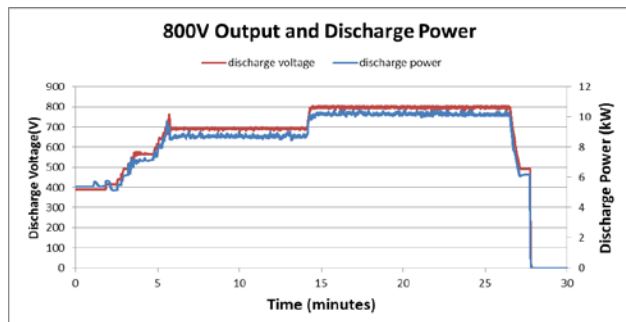


Figure 13. DSU Output at 800V

An integrated system test of a single DSU and mass flow controller with the NHT and feed system was performed at the end of 2017. The test included a 100 kW integrated system demonstration with eight DSUs powering the three NHT channels. Figure 13 shows the output of the DSU when operating at 800V discharge.

4.3 NextSTEP Nested Hall Thruster (NHT)

The NHT uses a novel approach to scale up in power by adding channels that circumscribe the centrally mounted cathode. Each channel is independently controllable enabling throttleability in thrust and power. NHTs have been identified as an important next step in electric propulsion technologies by enabling improved mass and footprint scaling with increased propulsive power¹⁸.

The X3 NHT builds on the extensive research and development conducted at University of Michigan coupled with experience and flight heritage of government and commercial partners. The X2 NHT was developed by the University of Michigan as a proof-of-concept device for nesting two concentric channels as a means to increase power and throttleability while limiting the increase of thruster mass and footprint. Following the successful demonstration of the X2,^{9,10} the University of Michigan developed the three-channel X3 NHT in collaboration with AFRL, NASA GRC, and NASA JPL. The X3 design heritage traces to the X2 NHT and H6 Hall thruster, and leverages lessons learned in developing the NASA high-power single-channel Hall thrusters (i.e. NASA-457M, -400M and -300M).



Figure 14. NHT installed at Univ. of Mich. Test facility (left); 30kW hotfire test (right)

Photographs of X3 installed in University of Michigan's LVTF (left) and operating three channels simultaneously at 30 kW (right) are shown in Figure 14.

The X3 NHT incorporates a high-current, Lanthanum Hexaboride (LaB6) hollow cathode developed by JPL.¹¹ JPL's novel cathode design has two external

gas injectors in addition to the traditional internal flow passage. The external injectors require an additional flow controller in both the PPU and MFC, but provide potential life and performance benefits for the propulsion system.

The X3 NHT is roughly 1 meter in diameter and 10 cm deep and weighs about 250 kg. The X3 was designed to process 200kW without additional radiative surfaces, yielding a potential specific power of 1.25kg/kW. Thermal analysis of the NHT was conducted at JPL as part of the NextSTEP program. The objective of the thermal analysis was to assess the thruster and cathode thermal margins at a total discharge power of 100 kW at 200 A. Modeling results showed positive margin given that the NHT was originally designed for 200 kW.

Plasma and erosion modeling is being performed at JPL with the objective of predicting erosion rates and operating life of a multichannel Hall thruster for the first time.¹² Preliminary modeling results show discharge channel erosion rates similar to those of other high power NASA Hall thrusters, yielding an operating life time of several thousand hours. The thruster magnetic circuit will be upgraded to significantly reduce discharge channel erosion in order to reach the NextSTEP program goal of 10,000 h of operational life. Implementation of AR's zero erosion design¹³ with JPL's magnetic shielding modelling¹⁴ will facilitate operating life times greater than >10,000 h. Aerojet Rocketdyne's XR-5 Hall thruster already demonstrated 10,000 hours of operation in 2009 during an extension of the qualification life test, so these design methodologies are well understood.¹⁵

In August 2017, the X3 demonstrated continuous 80 kW operation for two hours and demonstrated continuous 100 kW operation for about 10 minutes as part of a risk mitigation test conducted by NASA. 100 kW operations were halted due to a thermal expansion issue. A design solution has been identified and will be implemented in support of the 100 kW, 100 hour continuous demonstration test in mid-2018. Also of note, the X3 demonstrated 250 A total discharge current during three channel operation at 100 kW using the LaB6 hollow cathode provided by JPL. This represents the highest discharge current ever achieved in a Hall thruster.

The X3 demonstrated very good performance at lower power where thermal expansion was not yet an issue. The X3 achieved performance is comparable to the highest 500 V performance demonstrated by the HERMeS thruster, a state-of-the-art NASA thruster that has already incorporated

magnetic shielding.¹⁶ Total thruster efficiency was 66% and a total specific impulse was 2580 seconds while operating three channels concurrently at a total discharge current of 125 A and a discharge voltage of 500 V. The operating duration at 100 kW was insufficient to optimize performance but was invaluable in preparation for the 100 hour test.

Future work includes design improvements identified in the risk mitigation test in support of the demonstration of 100h of continuous operation at 100 kW in 2018. This test will demonstrate the ability of the X3 NHT to operate at steady-state hot temperatures without any additional cooling surfaces. Data from this test will help correlate existing thermal and plasma models, which will then be used to estimate maximum operating limits and lifetime capability.

Future design improvements have already been planned. Future design iterations will implement a more capable cathode design developed JPL¹⁷ and upgrades to the thruster magnetic circuit discussed above. Other design improvements include upgrades to the structural and thermal capabilities of the thruster enabling successful testing of qualification-level loads imposed by dynamic and thermal environments expected in future missions. Potential follow-on work in 2019 will help to create a design reference mission for this project. These improvements will enable raising the Technology Readiness Level (TRL) of the NHT.

4.4 Mass Flow Controller

The NextSTEP feed system consists of a Mass Flow Controller (MFC) and a Propellant Management Unit (PMU). The PMU regulates pressure of the xenon from the tank to the MFC while the MFC controls the flow rate of xenon to each channel of the NHT.

The MFC has five Proportional Flow Control Valves (PFCV); one PFCV for each for the three anode channels and the two cathode channels of the NHT. An absolute pressure transducer provides a pressure signal for of each flow circuit. This signal can be used for closed loop flow rate control and for telemetry to monitor flow rate throughout the mission.

The PFCVs are voice-coil linear actuators that provide full proportional flow rate control. The PMU electronic regulator steps the tank pressure of 2000 psia down to 40 psia. The MFC valves operate at a regulated pressure of 40 psia.

Lee restrictors are sized for the maximum flow rate demands of each NHT channel. The target flow is achieved through a closed loop Proportional–Integral–Derivative (PID) circuit in the MFC control board of the PPU that adjusts the input current which changes the back pressure to Lee flow restrictors resulting in adjustments to the flow rate.

An objective of the MFC and PMU design was to be simple and easy to fabricate thus reducing the production costs. Based on AR proprietary designs, the voice-coil PFCVs utilize design features that allow the detail parts to be manufactured with wide dimensional tolerances. There are also no valve body weldments typical of aerospace solenoids and no need for tight stroke and load adjustment during assembly.

The PMU and MFC utilize the same internal components realizing further cost advantages even within a single MFC and PMU assembly. The same PFCV design can serve many functions; a proportional flow controller, an electronic regulator and a service valve.

The net result is that the MFC and PMU can be assembled at a single station, with minimal tools and completed within a few hours. The MFC is treated as a single component and the PFCVs are assembled into the MFC or PMU from detail parts.

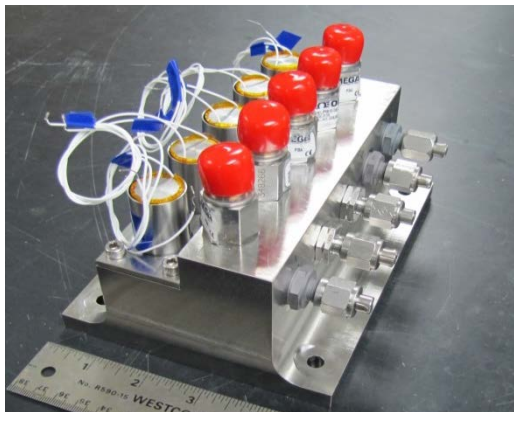


Figure 15, Prototype MFC

The prototype MFC, shown in Figure 15, has a machined manifold. Flight assemblies may even integrate MFC and PMU into a single additively manufactured manifold. Additive manufacturing enables contours out “dead mass” in the manifold to reduce the overall mass.

The performance of the PFCV design was demonstrated on a bench running open loop tests to map the flow performance of this design against input current. The MFC successfully demonstrated precise and stable closed loop flow control with the control board. The prototype 5-valve MFC was then tested in a 10kW integrated system hot fire test. The MFC was controlled by MFC control board and fed propellant to the NHT. Due to the test setup, the MFC was run at higher current than its original design value causing the PFCV to overheat and the valve to be damaged. Up to this point the MFC demonstrated stable closed loop flow operation.

Future work includes refurbishment and sizing of the 5-valve MFC for the higher flow rates of a 100kW hot fire followed by demonstration of stable operation in an integrated system hot fire test at 100kW.

NASA’s plans for deep space exploration call for the use of high power solar electric propulsion systems for orbit transfer and control of large spacecraft including the LOP-G and missions to Mars. Aerojet Rocketdyne is developing two high power electric propulsion systems, the Advanced EP System (AEPS) and the XR-100 NextSTEP system, to meet these mission requirements.

The AEPS program has completed the first phase of the design process through the Preliminary Design Review and successfully completed an early integration system test demonstrating the required control and throttling capabilities of the thruster, PPU and XFC. The component teams have incorporated design improvements to the respective heritage/baseline designs to better meet the program and mission needs. Engineering Development Unit hardware manufacture is underway with extensive component testing and system level operation planned for 2018. The program is on track to complete the component and system qualification testing in 2019 as well as delivery of five flight EP strings.

The NextSTEP XR-100 project is on schedule to perform a 100kW system demonstration test in 2018 in order to attain a maturity level of TRL-5.

These two programs will provide a path to efficient in-space propulsion that will allow NASA to transfer the large amounts of cargo that is needed to support human missions – first to the moon and then on to Mars.

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