

The Ion Propulsion System for the Solar Electric Propulsion Technology Demonstration Mission

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Abstract: The Asteroid Redirect Robotic Mission is a candidate Solar Electric Propulsion Technology Demonstration Mission whose main objectives are to develop and demonstrate a high-power solar electric propulsion capability for the Agency and return an asteroidal mass for rendezvous and characterization in a companion human-crewed mission. The ion propulsion system must be capable of operating over an 8-year time period and processing up to 10,000 kg of xenon propellant. This high-power solar electric propulsion capability, or an extensible derivative of it, has been identified as a critical part of an affordable, beyond-low-Earth-orbit, manned-exploration architecture. Under the NASA Space Technology Mission Directorate the critical electric propulsion and solar array technologies are being developed. The ion propulsion system being co-developed by the NASA Glenn Research Center and the Jet Propulsion Laboratory for the Asteroid Redirect Vehicle is based on the NASA-developed 12.5 kW Hall Effect Rocket with Magnetic Shielding (HERMeS0 thruster and power processing technologies. This paper presents the conceptual design for the ion propulsion system, the status of the NASA in-house thruster and power processing activity, and an update on flight hardware.

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I. Introduction

FOR missions beyond low Earth orbit, spacecraft size and mass can be dominated by onboard chemical propulsion systems and propellants that may constitute more than 50 percent of the spacecraft mass. This impact can be substantially reduced through the utilization of Solar Electric Propulsion (SEP) due to its substantially higher specific impulse. Studies performed for NASA's Human Exploration and Operations Mission Directorate and Science Mission Directorate have demonstrated that a 50kW-class SEP capability can be enabling for both near term and future architectures and science missions.¹ A high-power SEP element is integral to the Evolvable Mars Campaign, illustrated in Fig. 1, which presents an approach to establish an affordable evolutionary human exploration architecture. To enable SEP missions at the power levels required for these applications, an in-space demonstration of an operational 50kW-class SEP spacecraft has been proposed as a SEP Technology Demonstration Mission (TDM). In 2010 NASA's Space Technology Mission Directorate (STMD) began developing large, deployable photovoltaic solar array structures for high-power electrical power production and high-power electric propulsion technologies.²⁻⁷ The maturation of these critical technologies has made mission concepts utilizing high-power SEP viable.

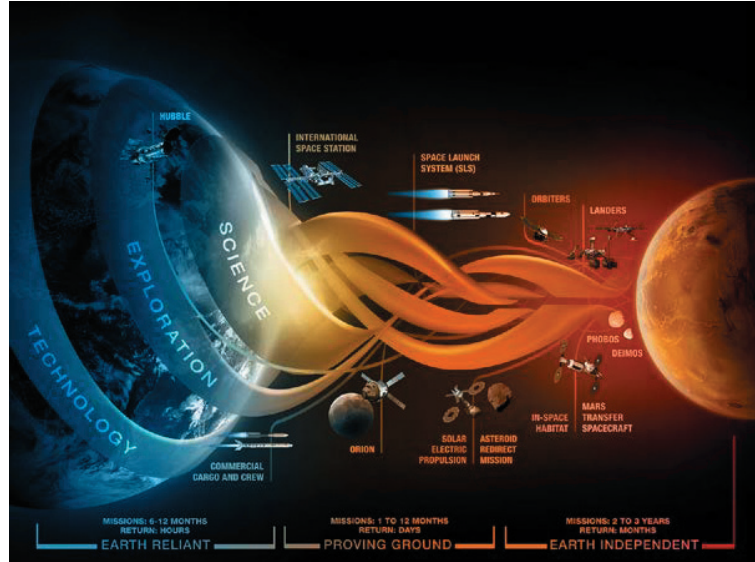


Figure 1. Evolvable Mars Campaign (EMC) graphical depiction.

The Asteroid Redirect Robotic Mission (ARRM) is an SEP TDM concept that utilizes an SEP spacecraft to return up to 20 metric tons (with a largest dimension of up to 6 m diameter) of asteroidal mass from the surface of a larger asteroid, to a stable orbit around the Moon for subsequent access by a human crewed mission.⁸⁻¹² The Asteroid Redirect Vehicle (ARV) concept, shown in Fig. 2, is a 50kW-class spacecraft having a total mass of 15 metric tons including up to 10,000 kilograms of xenon propellant. ARRM would satisfy the SEP TDM objectives while simultaneously providing a transportation capability with the potential for follow-on NASA applications either as is or with further modifications referred to as block upgrades.²

The Ion Propulsion System (IPS) being co-developed by the NASA Glenn Research Center and the Jet Propulsion Laboratory for ARRM will be used for heliocentric transfer from Earth to the target asteroid, orbit capture at the asteroid, transfer to a low orbit about the asteroid, a planetary defense demonstration after retrieval of the asteroidal mass from the larger asteroid, departure and escape from the asteroid, the heliocentric transfer from the asteroid to lunar orbit, and insertion into a lunar distant retrograde orbit. In addition, the IPS will provide pitch and yaw control of the spacecraft during IPS thrusting. The required total impulse provided by the IPS is 3×10^8 N-s, by far the largest total impulse ever provided by an on-board propulsion system for a planetary spacecraft.

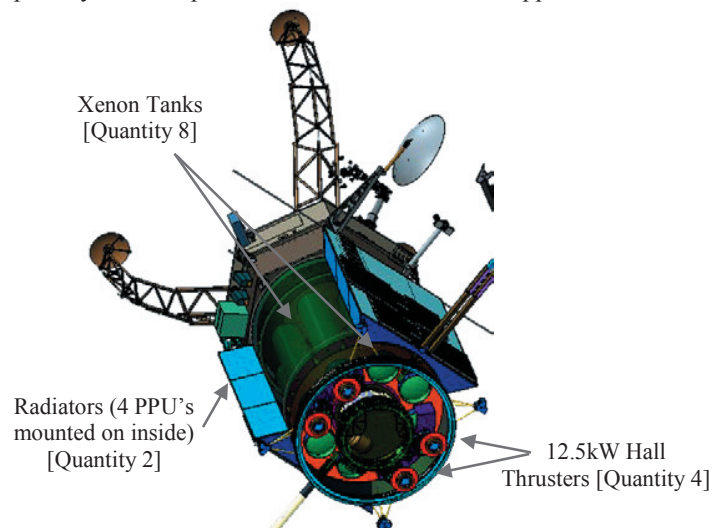


Figure 2. Asteroid Redirect Vehicle (ARV) Conceptual Design (arrays not shown).

II. Ion Propulsion System Description

The key driving requirements for the ARRM IPS, shown in Table 1, are that it must be single fault tolerant while processing up to 10,000 kg of xenon, providing a total impulse of 3×10^8 N-s over an input power range of 6.67 to 40 kW with input voltages ranging from 95 to 140 V and a mission duration of eight years.

Table 1. Key ARRM Requirements for the Ion Propulsion System.

Capability	Value
Total system power	40 kW
Maximum specific impulse	3000 s
Xenon throughput	10,000 kg
String fault tolerance	Single
Solar range	0.8 – 1.9 AU
Input voltage range	95 – 140 V

A. IPS Functional Description

The ARRM IPS is comprised of the following four elements:

1. Flight Thruster (FT).
2. Power Processor Unit (PPU).
3. Xenon Feed System (XFS).
4. Mechanical Integration Hardware (MIH) including cabling.

The technology investments by the Space Technology Mission Directorate has been critical in establishing the viability of the IPS for ARRM. A risk assessment of the ARV spacecraft bus revealed that four of the top six technical risks reside within the IPS. The electric propulsion investment and component development plans were formulated to address these technical risks to assure mission success. These technical risks are:

- Hall Thruster Lifetime Qualification: The mission development schedule for ARRM precludes wear testing on the ground for the durations conventionally required. This necessarily means exceptions to the standard electric propulsion life qualification approach.¹³ An approach to certifying the life of the thruster is being developed that relies on physics-based modeling and limited testing of the primary wear-out failure modes. This approach is designed to mitigate the risk that non-traditional life qualification required by mission schedule will result in reduced mission capability.
- Hall Thruster Ground Test Facility Effects: The performance, stability, thermal, and wear characteristics of electric thrusters measured during ground-based tests are influenced to some degree by the limitations of ground test facilities. Some of these limitations include finite operating pressures, volume constraints, the influence of grounded vacuum chamber walls, thermal environments that are non-representative of space, and backscatter from facility equipment and chamber walls. The development plan for the Hall Effect Rocket with Magnetic Shielding (HERMeS) thruster includes efforts to quantify facility effects and to develop designs or methodologies to mitigate them such that the performance and life of HERMeS may be reliably predicted during in-space operation. This approach is designed to mitigate the risk that performance or wear characteristics could significantly differ between the qualification in ground-based facilities and in-flight.
- PPU Development Schedule: Electric thruster PPU development is typically the pacing activity for an EP system development and testing has, in some instances, encountered issues that required significant additional resource to address (e.g. Dawn¹⁴). This risk is exacerbated by the brevity of the duration for qualification and flight unit fabrication required to meet the target launch date for ARRM. There is a risk that the PPU development will encounter similar problems that will have cost and schedule impacts. This risk is being mitigated by performing the in-house NASA PPU development, reducing the complexity of the PPU, simplifying requirements where possible, and contracting for engineering model hardware development and flight units early in the mission design cycle.
- New Xenon Tank Development: New tank development and testing issues can also require significant additional resources and/or result in reduced mission capability if problems are encountered. To mitigate this risk, a seamless, aluminum-lined tank is being developed that utilizes existing manufacturing tooling for the conceptual ARV. Recent efforts have also been focused on development of a pathfinder xenon tank to address the new xenon tank development risks.

The 2-axis thruster gimbal assembly work breakdown is included in the Structures and Mechanisms System for the conceptual ARV. A high-level block diagram, shown in Fig. 3, indicates that the conceptual ARV IPS includes four flight electric propulsion (EP) strings, a xenon feed system, and the interconnecting cabling and tubing. Each EP string consists of a Hall thruster, a PPU, a xenon flow controller, and the interconnecting cable harnesses. The xenon feed system consists of eight xenon tanks, a propellant management assembly, and the xenon flow controllers. Each EP string is operated independently of the others by the spacecraft. Single fault tolerance is maintained by block-redundancy at the EP string level with internal redundancy for the xenon feed system components outside of the EP strings.

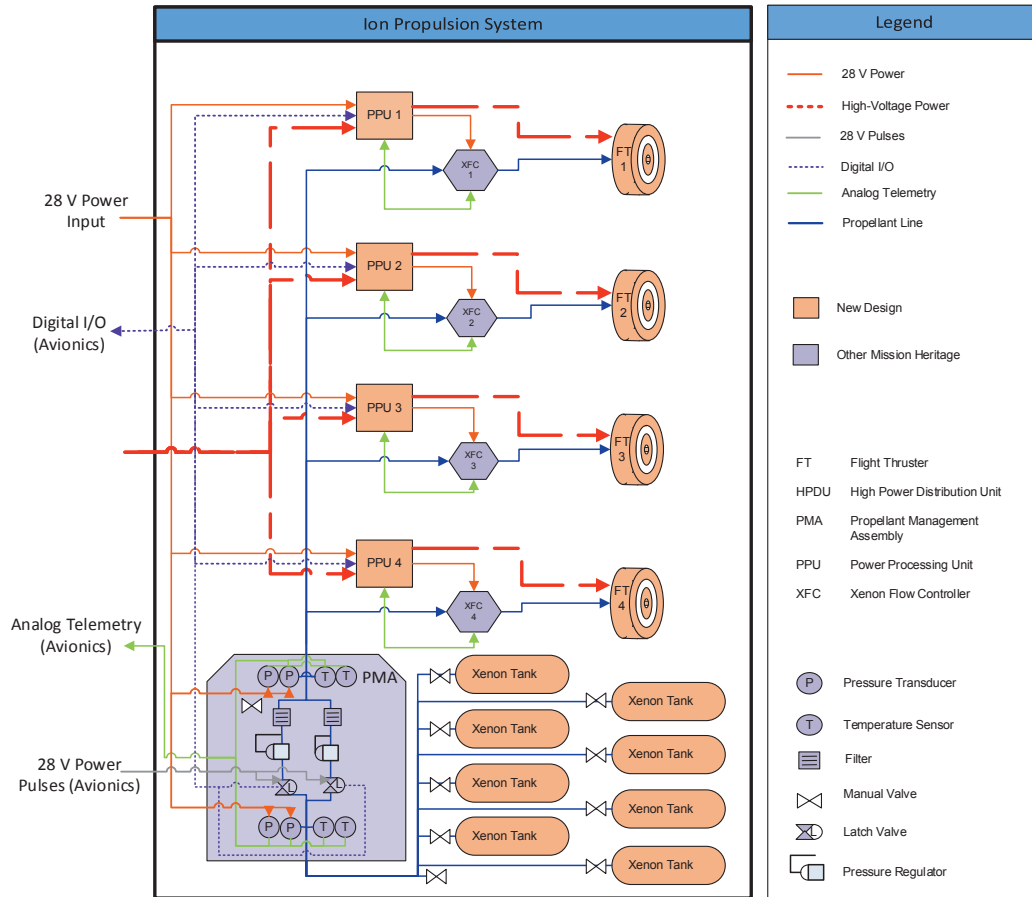


Figure 3. Top-Level ARRM Conceptual Ion Propulsion System (IPS) Block Diagram.

III. Subsystem Descriptions

In 2010 NASA's Space Technology Mission Directorate (STMD) began developing large, deployable photovoltaic solar array structures for high-power electrical power generation and high-power electric propulsion technologies.²⁻⁷ The maturation of the critical technologies required for the high-power SEP vehicle has made mission concepts utilizing high-power SEP viable.¹⁵ The high-power electric propulsion investments were in areas assessed as having high technical risks and/or long-lead times for the IPS. NASA in-house development of the 12.5 kW Hall thruster and power processing unit have resulted in two high-fidelity development model thrusters and a brassboard power processing unit that have been extensively tested and characterized separately as well as demonstrated as an integrated system. The NASA work validated subsystem design methodologies, developed critical diagnostics, demonstrated performance that meets current mission requirements, made significant strides in life qualification, developed and validated an array of models, and potentially provided a departure point for the technology maturation and flight system development within industry for the mission. The conceptual design of the IPS components for the current ARV concept is described below.

Desired EP string performance for ARRM at beginning-of-life is shown in Table 2. The table utilizes the current best estimates from the in-house 12.5 kW thruster and PPU based on demonstrated performance including experimental uncertainty. The desired flight electric propulsion string should meet or exceed the performance in Table 2 while simultaneously meeting the ARRM lifetime requirement over all environmental conditions and input voltages.

Table 2. Electric propulsion string performance including experimental uncertainty.

EP String Total Input Power (kW)	Discharge Voltage (V)	Thrust (mN) ¹	Mass Flow Rate (mg/s) ²	System Efficiency ³
13.33	400	686	30.66	0.58
13.33	500	633	25.97	0.58
13.33	600	589	22.91	0.57
13.33	700	553	20.36	0.56
13.33	800	526	18.52	0.56
10.00	300	562	29.69	0.53
10.00	400	521	24.30	0.56
10.00	500	482	20.89	0.56
10.00	600	446	18.31	0.54
10.00	700	413	16.17	0.53
10.00	800	342	12.88	0.45
6.67	300	386	21.65	0.52
6.67	400	342	17.55	0.50
6.67	500	311	14.83	0.49
6.67	600	267	11.43	0.47
6.67	700	242	9.99	0.44

¹ Thrust shown here is Current Best Estimate minus experimental uncertainty.

² Mass flow rate shown here is Current Best Estimate plus experimental uncertainty.

³ System Efficiency is shown here for information only (it is calculated directly from the other parameters).

B. Flight Thruster (FT)

The conceptual ARV IPS design utilizes the 12.5 kW Hall Effect Rocket with Magnetic Shielding (HERMeS). The HERMeS thruster design attributes from the ARRM mission requirements are shown in Table 3. HERMeS is unique because it is the first Hall thruster designed for magnetic shielding over its entire service life. The design of HERMeS incorporates technologies developed

by NASA over nearly two decades, and is enabled through the use of magnetic shielding to effectively eliminate discharge chamber erosion.^{3,16-18} The result is a significant increase in the operational lifetime of state-of-the-art for Hall thrusters, with HERMeS being designed to operate at 3000 s specific impulse with a lifetime exceeding 50 kh. These capabilities are expected to meet or exceed the requirements for ARRM. The methodology used to design HERMeS emphasized the use of thermal, structural, and physics-based plasma modeling that were used in nearly every aspect of the thruster design to achieve its design goals.^{17,19,20} The design of HERMeS resulting from this approach uses an integrated magnetic and thermal design, pole piece covers, a graphite cathode keeper, an internally mounted cathode, and a downstream-plenum gas distributor.

Table 3. 12.5kW Hall-Effect Rocket with Magnetic Shielding (HERMeS) attributes (assuming 94% PPU efficiency).

Attribute	Value
Maximum input power to PPU	13.3 kW
Maximum discharge power	12.5 kW
Discharge voltage at 3000 s I_{sp}	800 V
Discharge current at 3000 s I_{sp}	15.6 A
Service life (unmargined)	50,000 h
Minimum discharge power	6.25 kW (50% maximum power)

The HERMeS development plan was formulated from a set of technical risks that could impact mission success. Each element of the development plan is traceable to these risks. The development consists of three major hardware build phases: the Technology Development Unit (TDU), Engineering Development Unit (EDU), and the combined Qualification Model (QM) and Flight Model (FM) phases. HERMeS is now in the TDU phase, which is scheduled to complete by mid-2016, and the project is on track to complete all of its planned tasks prior to the EDU phase starting.

HERMeS Technology Demonstration Unit 1 (TDU-1) has been fabricated and has recently completed the first test campaign of its performance, stability, thermal, and wear characteristics. TDU-1 thruster is shown operating in VF5 at NASA GRC in Fig. 4. Figure 5 shows the operating conditions that have been investigated to date in the laboratory as discrete points overlaid on the operating envelope developed for ARRM.

The completion of the first HERMeS test campaign has demonstrated thruster performance, verified magnetically shielded operation at high specific impulse, and affirms that the internally mounted cathode minimizes the effects of facility pressure on performance. Details regarding TDU thruster design, mission-required operating envelope, and test results that are being presented this summer at various conferences and in future publications are detailed in Refs. 3, 16, 17, and 19-35. A second thruster, TDU-2, is currently being fabricated and will undergo environmental testing by the end of 2015.

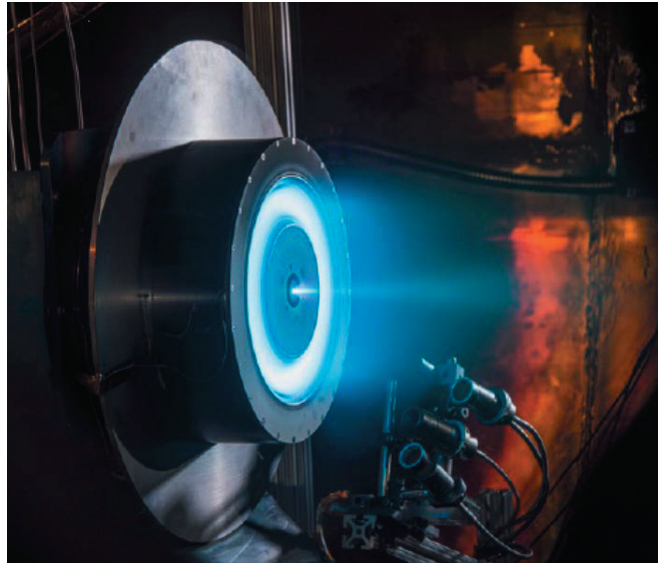


Figure 4. 12.5kW Hall-Effect Rocket with Magnetic Shielding (HERMeS) operating in VF5 at NASA GRC.

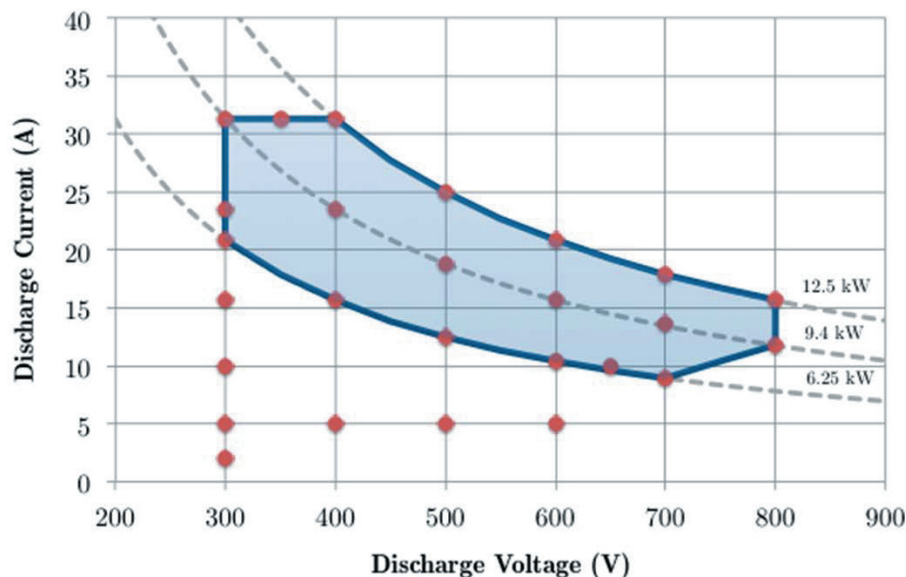


Figure 5. HERMeS operating envelope for the ARV IPS (shaded region) and the discrete operating conditions demonstrated to date in the laboratory (red markers).

C. Power Processor Unit (PPU)

The HP-120V PPU converts power from the 95 – 140 V spacecraft solar arrays into the currents and voltages needed to start and operate the thruster. The PPU also communicates with the spacecraft and controls the xenon flow controller. The overall strategy for the NASA in-house power converter electronic design utilizes a full-bridge topology. This topology is good for high power applications because it has low electrical stress on the power transistors with efficient transformer utilization. Metal-oxide-semiconductor field-effect transistors (MOSFETs) are paralleled as needed on each leg of the bridge to reduce conduction losses on the transistors. A full bridge topology is also used for the auxiliary supplies to allow for ample room for growth and to be able to reuse control and gate drive printed wiring boards.

For the NASA-developed HP-120V PPU, breadboard circuits of the power converter for the discharge and auxiliary supplies were built and tested. These were used to evaluate the main power transformer, output inductor, and gate drive circuit design. The design was then used to develop a brassboard PPU capable of operating in vacuum, but without the final form and fit of a flight PPU. The PPU brassboard unit was designed to operate in simulated operational environments in order to assess critical performance functions. The overall strategy for thermal design was to locate components with high dissipation like transformers, inductors, MOSFETs and diodes with a direct thermal path to the baseplate. Printed circuit boards have thermal planes and are mounted vertically to aluminum brackets that removed heat from the perimeter of the PCB into the baseplate.

The ARV performance estimates for the PPU are based upon measured values in the NASA-developed HP-120V PPU. The PPU processes electrical power from high voltage solar arrays and a 28 V unregulated, battery-backed bus. The brassboard unit powers a high power Hall thruster from 2 to 14 kW of discharge power and provides auxiliary outputs for inner and outer magnets, cathode keeper and heater. The discharge supply in this PPU is capable of supplying up to 800 V for specific impulse of 3,000 s and up to 14 kW of discharge power. The discharge supply consists of four discharge modules that may operate at a maximum of 3.5 kW each. The output current of each discharge module is limited to 20 A. Each discharge module can provide an output voltage of up to 200 V and the outputs are stacked in series to generate the required 800V output as shown in Fig. 6. When the required output voltage is lower than 400 V, two of the four discharge modules can be disabled by command to increase loading and improve performance. Bypass diodes were included to further improve performance when operating in this mode.

The desired PPU specifications changed after the HP-120V PPU, shown in Fig. 7, was already being fabricated. It was desired to increase the maximum PPU output current from 20 A up to 33 A to match the HERMeS thruster capability. While this capability does not currently exist within the HP-120V PPU, with the addition of switches the four discharge modules could be reconfigured to provide 400V and 40A discharge output maximums. For the conceptual ARV design, this switching capability was assumed in order to estimate the system performance curves during the maturation of the ARRM mission concept.

Also in this design, power for the magnet supplies comes from the 120 V input. Magnet power is only needed while discharge power is applied. This reduces the impact of the PPU on the 28V power management and distribution system in the spacecraft. Keeper and heater power is still processed from the 28V bus so that the thruster cathode can be operated during an eclipse while spiraling around the Earth. A high-level HP-120V PPU electrical schematic is shown in Fig. 8.

This project fabricated and tested a high power, high performance PPU design. Modeling with physics-based thermal and circuit models was performed to support the development and testing. The testing included electrical performance and

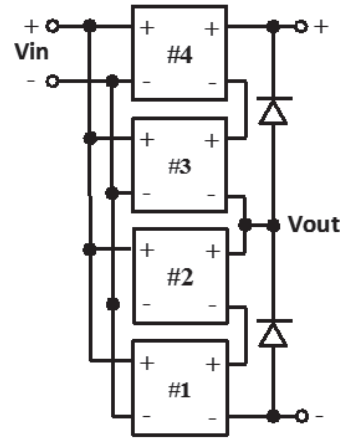


Figure 6. HP-120V PPU Discharge Supply configuration consisting of 4 discharge modules.

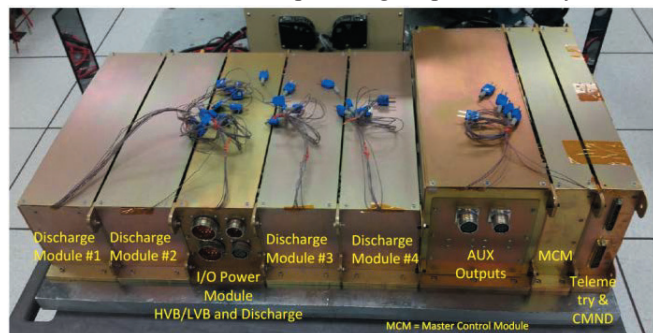


Figure 7. 14kW HP-120/800V Brassboard PPU.

thermal characterization using resistive and thruster loads, in ambient and vacuum conditions. The 120V PPU discharge supply demonstrated a peak efficiency of 95.5% and higher than 94% efficiency down to 30% of the maximum output power. With the magnet and keeper supplies operating at levels representative of the thruster, the PPU demonstrated total efficiencies as high as 95% at 100 V input, 12.5 kW discharge power, and 700V discharge voltage. After completion of benchtop testing, the HP-120V PPU was tested in ambient with the HERMeS thruster in VF5 at NASA GRC demonstrating the capability to operate a dynamic thruster load over the full range of HP-120V specifications including the 12.5kW, 3000s specific impulse thruster design point. Following successful completion of the ambient and thruster integration tests, the HP-120V PPU was characterized in vacuum. The results from the HP-120V PPU tests can be found in Ref. 36.

The mass of this brassboard PPU is 62 kg. It is projected that after optimization of the converter design and packaging improvements, mass could be reduced by at least 33 percent. This would result in a Hall thruster PPU with a variable input with a specific mass of less than 3 kg/kW which compares favorably to state-of-art (SOA) PPUs.

In parallel to the HP-120V PPU development at NASA GRC, a System Control Board (SCB) is under development at JPL to handle the communication with the spacecraft, operate the XFC, control the PPU outputs, and provide fault protection. The system control board is the PPU (and EP string) interface with the spacecraft computer, taking commands and sending status. A prototype SCB has been fabricated and is undergoing functional testing before integration with the HP-120V PPU. A schematic of the system control board interfaces is shown in Fig. 9.

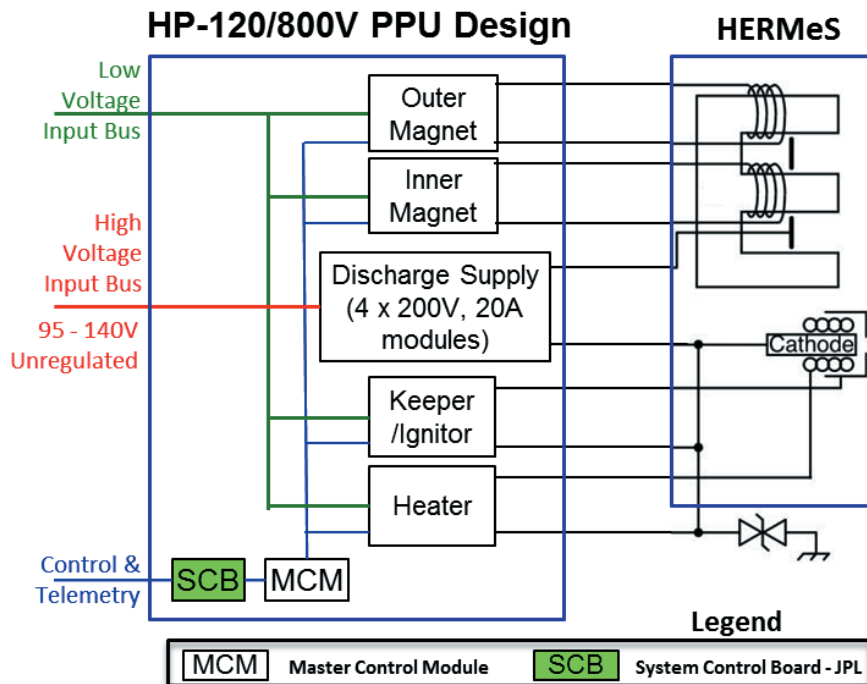


Figure 8. 14kW HP-120/800V Brassboard PPU top-level design including thruster connections.

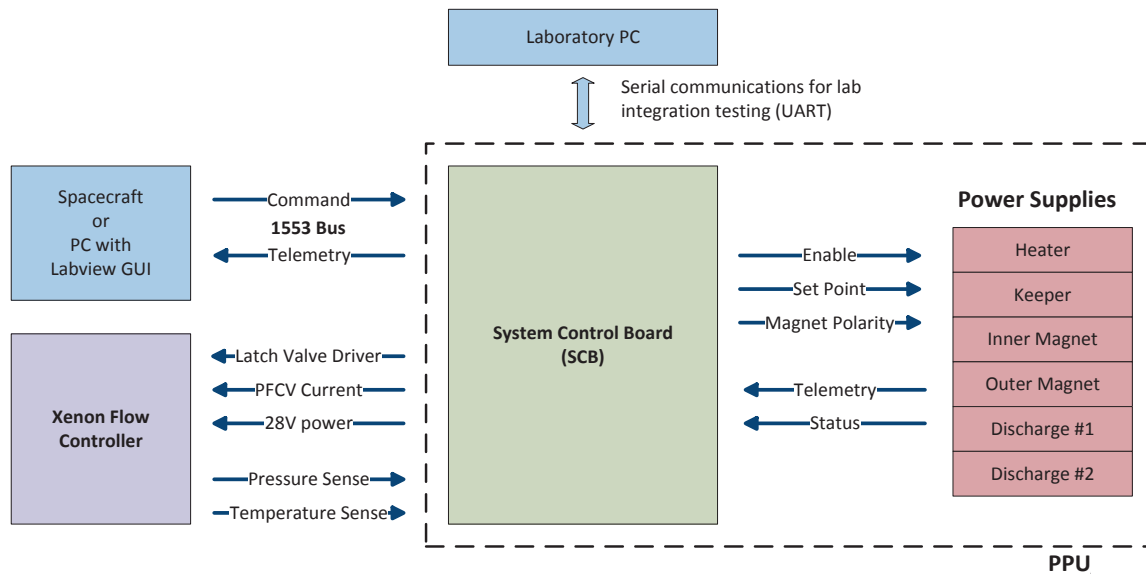


Figure 9. System Control Board Interfaces.

D. Xenon Feed System (XFS)

The conceptual ARV XFS, shown in Fig. 10, includes eight xenon propellant tanks, four Xenon Flow Controllers (XFC), and a Propellant Management Assembly (PMA) that consists of the isolation valves, service valves, and mechanical regulators. There has been no development work for the xenon feed system to date because the system is low-risk and does not require a long-lead procurement as a result of multiple options available utilizing flight qualified components. The exception to this is the xenon tanks. Qualification and testing issues during carbon-overwrapped pressure vessel (COPV) tank development for the Dawn mission highlighted the potential for risk associated with a new tank development.¹⁴ The conceptual ARV xenon tank attempts to mitigate this risk by utilizing a seamless, aluminum liner with a diameter that is currently being fabricated by multiple vendors. This leverages existing expertise and tooling within industry for the new (longer) tank development. To further mitigate this risk, a competitive contract is being initiated for industry to develop a pathfinder xenon tank. The tank specifications for the conceptual ARV design are as follows:

- 1) 60 cm diameter x 305 cm long
- 2) Storage of 1300 kg of xenon at maximum design pressure of 1750 psia at 40 C
- 3) Seamless, aluminum-lined, carbon-overwrapped pressure vessel

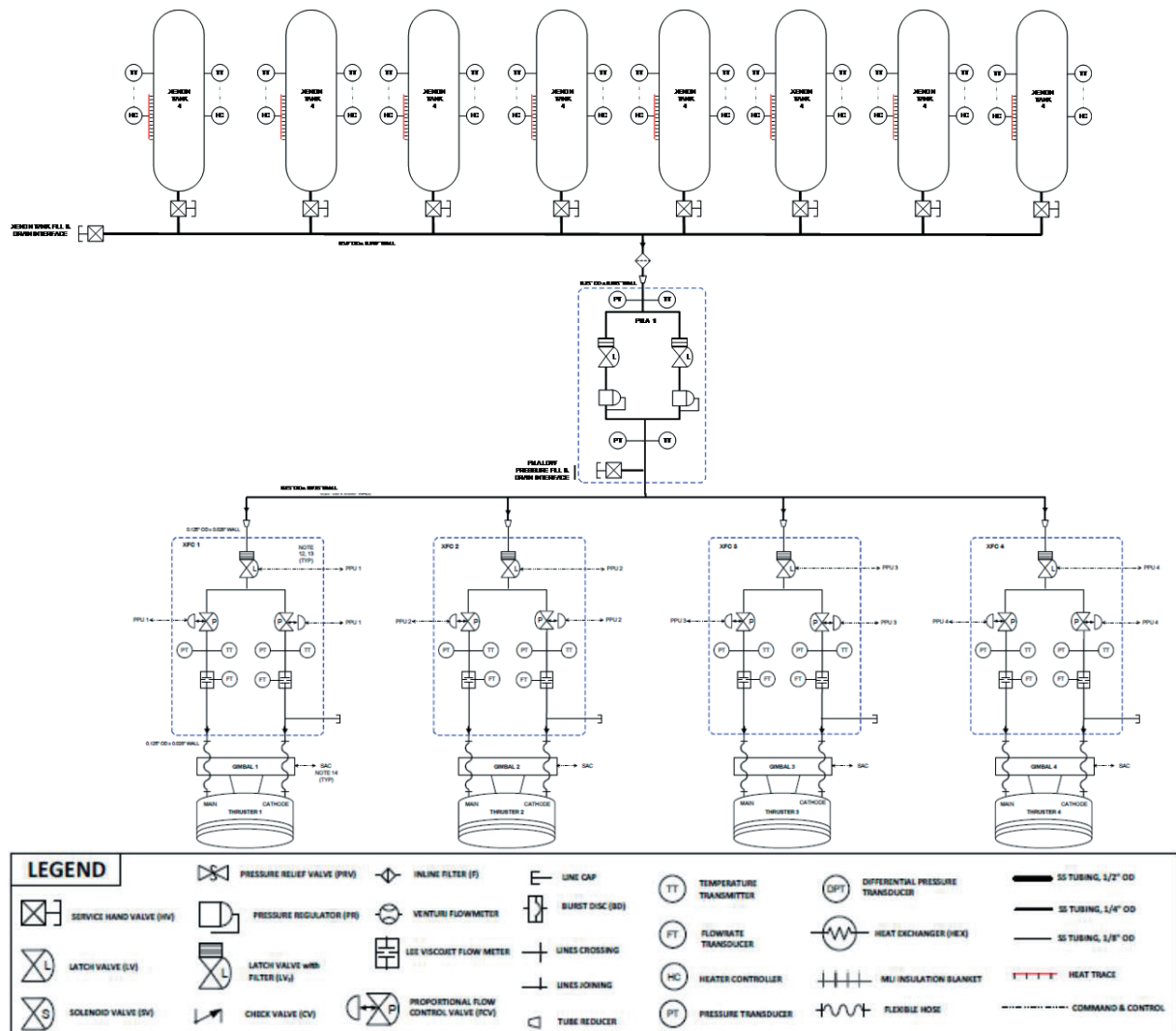


Figure 10. Conceptual Asteroid Redirect Vehicle (ARV) Xenon Feed System (XFS) Design.

E. Mechanical Integration Hardware (MIH)

The MIH includes structural and thermal hardware integral with IPS elements and cabling between IPS subsystems and components including structural elements for installing the IPS and heat rejection elements.

IV. System and Acquisition Status

Given the lead times required for the IPS components, acquisition for most of the major components has been initiated. A draft RFP for an Engineering Development Unit (EDU) EP string and optional Qualification Model (QM) and Flight Model (FM) hardware delivery was released on May 5, 2015.³⁷ This potential procurement includes the thrusters, power processing units, xenon flow controllers, and electric harnesses between the subsystems. A draft of the procurement schedule is shown in Table 3.

Table 3. Draft Electric Propulsion String Procurement Dates.

Event	Date (Notional)
Draft RFP Release	May 21, 2015
Industry Conference	June 10, 2015
Comments on Draft Due	June 22, 2015
Final RFP Release	July 14, 2015
Proposal Due Date	August 28, 2015
Contract Award	March 29, 2016

Preparations are underway for the potential prototype tank procurement. The thruster gimbal assembly is low-risk as the required ARRM capability exists within flight-qualified designs that require only minor modifications to accommodate the thruster mass and size. The conceptual ARV thruster gimbal assembly design is a modified version of a flight qualified thruster gimbal. The estimated lead time for the gimbal development does not currently require acquisition initiation.

The conceptual ARV design includes the capability of xenon storage capability up to 10,000 kg. A xenon propellant load of 10 metric tons is approximately 19% of the global annual production rate of xenon (currently estimated at 53 tons).³⁸ A single procurement of this size with short-term delivery can disrupt the xenon market and drive up cost, which could make the propellant costs for the mission prohibitive. Additional considerations regarding xenon acquisition given xenon pricing volatility and market considerations are important to the xenon acquisition for ARRM. A multi-year strategy will be required for the acquisition of up to 10,000 kg of xenon without causing a spike in the xenon market price. One such strategy would be a single long-term acquisition over 3.5 – 4.5 years allowing the market to smoothly adjust to this new demand.³⁸

V. Conclusion

The Ion Propulsion System for Solar Electric Propulsion Technology Demonstration Missions including the Asteroid Redirect Robotic Mission, is currently being developed focusing on the higher-risk and long-lead items. This work is a continuation of efforts initiated by NASA's Space Technology Mission Directorate several years ago. Collaborative HERMeS thruster and HP-120V PPU in-house developments conducted by NASA GRC and JPL have resulted in characterized high-fidelity laboratory hardware that proves the viability of the conceptual IPS for ARRM and provides demonstrated performance for mission design. Ongoing advanced technology work is focused on reducing the mission technical risks in a systematic, traceable approach. With some additional development the thruster and PPU designs could become the basis of future flight NASA missions. An electric propulsion string procurement has been initiated that will develop engineering-model hardware with an option for qualification and flight strings that will meet the ARRM requirements and support the target launch date.

Acknowledgments

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