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13kW Electric Propulsion System Architecture Development

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Abstract

Aerojet Rocketdyne's Advanced Electric Propulsion System (AEPS) program is completing development, qualification and delivery of five flight 13.3kW EP systems to NASA. The flight AEPS includes a magnetically-shielded, long-life Hall thruster, power processing unit (PPU), xenon flow controller (XFC), and intra-system harnesses. The Hall thruster operates at input powers up to 13.3kW while providing a specific impulse over 2600s at an input voltage of 600V. The power processor is designed to accommodate an input voltage range of 95 to 140V, consistent with operation beyond the orbit of Mars. The integrated system is continuously throttleable between 3 and 13.3kW. The program has completed the System Requirement Review (SRR); the system, thruster, PPU and XFC Preliminary Design Reviews (PDRs); development of a system performance model, and Early Integrated System Testing (EIST). In preparation for the critical design reviews, PPU, thruster and XFC flight-like engineering development units (EDU) will be tested to qualification levels, and then integrated into the EP string configuration and hot-fire tested in a vacuum chamber. This paper summarizes the status of the AEPS program, including the design updates that are being incorporated prior to EDU testing and Critical Design Reviews (CDRs) in 2019. These improvements were identified during the EIST that verified successful operation of the EP string. In order to successfully execute this contract, there is a close collaboration between the team members at Aerojet Rocketdyne (AR), ZIN Technologies, VACCO, the NASA Jet Propulsion Laboratory (JPL), and the NASA Glenn Research Center (GRC). This paper will present the high power AEPS flight system architecture in development for a range of mission applications.

I. Nomenclature

AEPS	=	Advanced Electric Propulsion System
AR	=	Aerojet Rocketdyne
CDR	=	Critical Design Review
DMC	=	Discharge Master Controller
DSU	=	Discharge Supply Module
EDU	=	Engineering Development Unit
EIST	=	Early Integrated System Test
EP	=	Electric Propulsion
GRC	=	Glenn Research Center
HEOMD	=	Human Exploration and Operations Mission Directorate
HERMeS	=	Hall Effect Rocket with Magnetic Shielding
ICD	=	Interface Control Document
JPL	=	Jet Propulsion Laboratory
LOP-G	=	Lunar Orbiting Platform - Gateway
NASA	=	National Aeronautics and Space Administration
PDR	=	Preliminary Design Review
PMA	=	Propellant Management Assembly
PPE	=	Power Propulsion Element
PPU	=	Power Processing Unit
SEP	=	Solar Electric Propulsion

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SMD	=	Science Mission Directorate
SRR	=	System Requirements Review
STMD	=	Space Technology Mission Directorate
SRR	=	System Requirements Review
TDU	=	Technology Development Unit
VF	=	Vacuum Facility
XFC	=	Xenon Flow Controller
XFCM	=	Xenon Flow Control Module

II. 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]

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.

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 maturation of a high-power Hall thruster and power processing unit developed by NASA.[3,4] This technology development work has since transitioned to Aerojet Rocketdyne via a competitive procurement selection for the Advanced Electric Propulsion System (AEPS) contract. Aerojet Rocketdyne has worked with NASA to develop the 13 kW Advanced Electric Propulsion System (AEPS) thruster strings that will be the primary propulsion for the PPE. It is NASA's intention to perform a thorough in-space test of the entire SEP system including AEPS and the advanced solar arrays.[4] Among the capabilities to be demonstrated is refueling of the xenon tanks to allow for additional thrusting and extending 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 vehicle 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.[5]

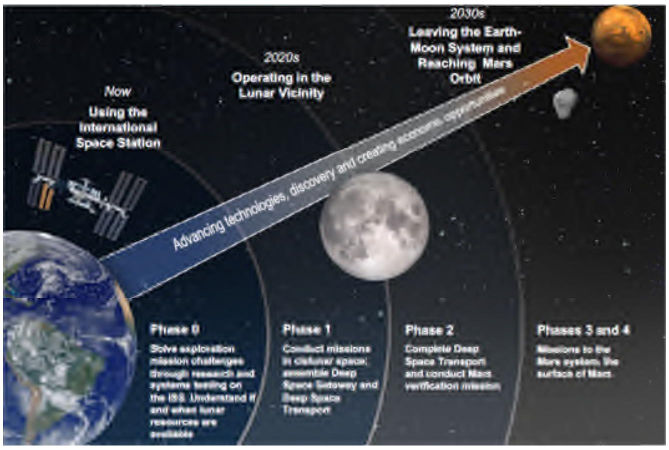

Figure 1. NASA Exploration Approach

III. Program Overview

AEPS is a NASA contract that was competitively-selected and consists of the development of an Engineering Development Unit (EDU) EP string, qualification of a flight system and an option to deliver four 13.3kW EP flight systems to NASA. The AEPS program was awarded to Aerojet Rocketdyne on April 28 of 2016. In execution of this program, there is close collaboration between Aerojet Rocketdyne, NASA GRC, and JPL. The industry AEPS team includes two Aerojet Rocketdyne sites, Redmond and Los Angeles, as well as ZIN Technologies, who is providing elements of the PPU, and VACCO, who is providing the Xenon Flow Controller (XFC). The management of the contract is being led by the NASA Glenn Research Center.

The AEPS Hall thruster is based upon the 12.5 kW Hall Effect Rocket with Magnetic Shielding (HERMeS) that was originally developed and demonstrated by NASA GRC and JPL. The thruster operates at input powers up to 12.5 kW while providing a specific impulse of over 2600 s at a discharge voltage of 600 V. This thruster design approach resulted in an estimated life of 50,000 hours[6] enabled by magnetic shielding. Aerojet Rocketdyne first demonstrated long-life Hall thruster technologies on the XR-5 (previously named BPT-4000).[7] NASA continues to perform further development testing of the HERMeS Technology Development Units (TDUs) including wear testing, environmental testing and cathode development in order to better understand implications for spacecraft accommodations and mitigate risk for the AEPS program.

The AEPS Power Processing Unit (PPU) leverages the work performed by NASA GRC on a brassboard power processor that was utilized in the integration testing of the HERMeS thruster. 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.[8] 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.

Unlike the HCT or PPU, the fidelity of the XFC for AEPS is already at a high Technology Readiness Level (TRL). The AEPS XFC is a derivative of the Xenon Flow Control Module (XFCM), which was previously developed under a NASA contract by VACCO. The XFCM is a highly integrated feed system that accepts unregulated xenon directly from storage tanks and outputs precision, throttleable flow through two independent channels. The XFCM completed qualification testing and was delivered to NASA GRC on 7 June 2012.[7]

IV. System Engineering

Development of the AEPS system architecture has been accomplished through a rigorous system engineering process. The system specification effort was developed through the integration of requirements from NASA-GRC to consolidate all the supplied and derived requirements into a single set of requirements for the EP string. This required close coordination with NASA to select the best representative set of requirements to meet the program's risk mitigation needs. The initial requirements and the derivation methodology were reviewed at SRR and completed after PDR.

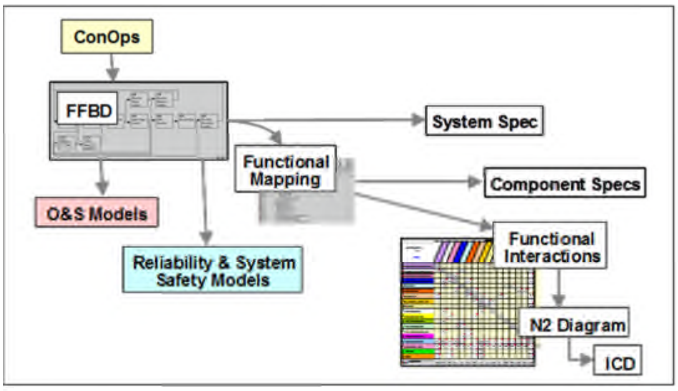

Figure 2. Functional Analysis and Requirements Allocation Process

Requirements validation ensured that each requirement statement is accurate, concise and specific to what it requires the design to accomplish. Requirements were allocated down to the next level of detail and the requirement validity is dependent on traceability of these allocated requirements, including allocation analyses and rationale (see Figure 2). Functional analysis was performed to establish traceability of system requirements to functions, with subsequent allocation to AEPS components. CONOPS functions were then mapped to the EP String. System functions and interfaces were defined early in the design process which was used to help guide the design activities. A functional flow block diagram (FFBD) was developed to support allocation / flow down of requirements to subsystems. Interface control is a vital activity performed in support of system integration in identifying, defining, negotiating and controlling all interface types (e.g., physical, functional, electronic, data, software, etc.). System integration is accomplished by controlling interfaces among system and subsystem elements. External interfaces are those between AEPS and the mission customers. Internal interfaces include all subsystems and component interfaces within the EP string. The interface control document (ICD) is a collaborative effort between AR and NASA to define and control these interfaces. This process is facilitated with an N^2 diagram which identifies existence and aspects of interfaces between the subsystems and components for control and verification.

Each requirement was also assessed against AR standards for effective requirements, specifically to ensure that it is verifiable, attainable, and that it effectively and objectively communicates the parameter and tolerance that the design is expected to meet. Requirements validation was performed in reviews including all stakeholders – customer, designers, users and support personnel. The requirements, verification and preliminary design concept maturation has been an iterative process beginning prior to SRR and completing in the 3rd quarter of 2018 to a CDR maturity level. (see Figure 3).

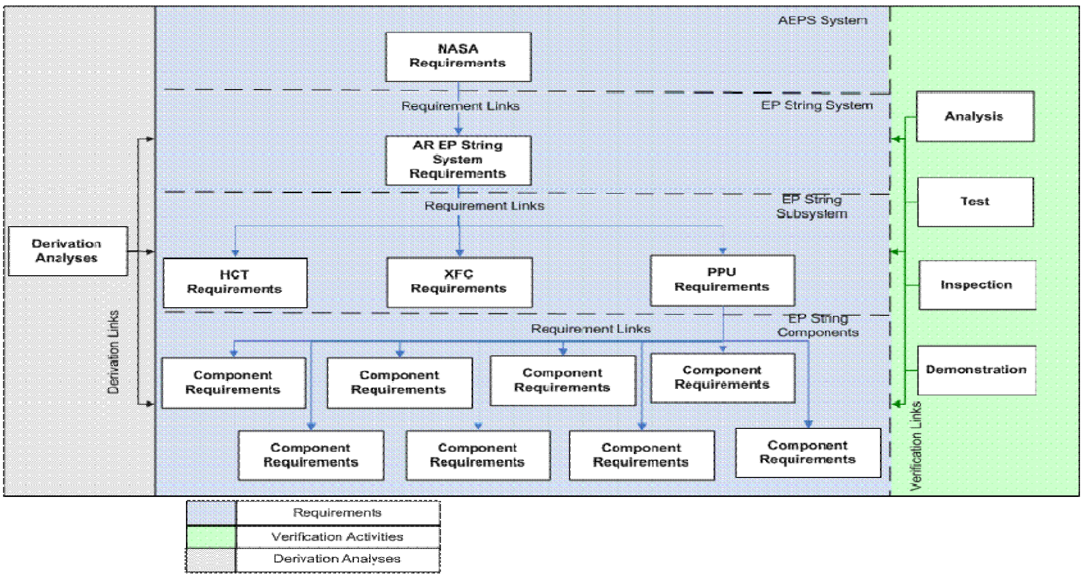

Figure 3. Requirements Traceability

The requirements contained in the program specifications and derived requirements are maintained in a technical requirements database using DOORS™. The DOORS™ database also contains trace links between the requirements provide show traceability from the customer requirements through system-level requirements and allocations to component and subsystem-level requirements. All levels of requirements are integrated in the database.

The verification activity descriptions for each requirement are maturing post-PDR in support of the system and subsystem verification activities. The verification planning process is an iterative process that involves input from team members to develop verification activities. The verification activities identify the requirements verified, the objective of the activity, and the verification method: test, analysis, inspection, or demonstration. The verification activities also identify what the team members and systems engineers determine as the final output and success criteria. Analysis and test planning is driven by the requirements verification definition. The verification descriptions will be linked to the requirements in DOORS along with the artifacts to document requirement closure.

V. System Architecture

The AEPS 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. The Hall thruster uses xenon propellant and up to 12.5 kW 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 3000 psi in the event of a spacecraft xenon pressure regulator failure scenario. The AEPS program has completed the System Requirement Review (SRR), the system and component Preliminary Design Reviews (PDRs); system model development, and early system integration testing.

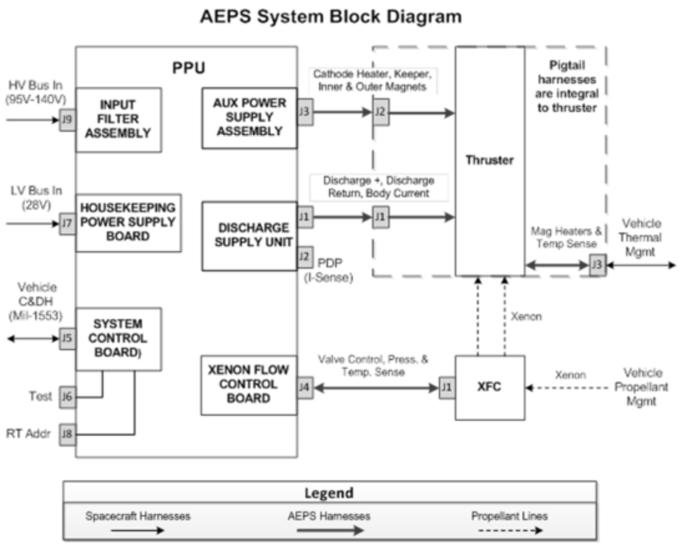

Figure 4. AEPS Block Diagram with electrical and propellant interfaces

The primary design objective for the system architecture is to provide a high performance propulsion system that efficiently utilizes both electrical power and propellant. The system provides the capability to throttle between 3 kW and 13.3 kW of input power providing a range in discharge voltage between 300 V and 600 V. 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 PPU continuously monitors the health of the entire EP string, allowing continuous diagnostics for changing conditions over the mission life. This monitoring includes reporting warning or fault level status and discrete flags for the thruster to the spacecraft. The system returns telemetry and can accept updates to the software and data tables during mission operations.

The performance requirements for AEPS are summarized in Table 1 below. The required system input power and propellant flow rates will be determined by the throttle set points commanded by the spacecraft. Significant effort has been focused on maximizing the electrical efficiency of each component of the propulsion string and ensuring repeatable performance throughout the life of the mission.

A block diagram of the AEPS string (See Figure 4) shows external interfaces to the spacecraft. The system receives high voltage power for thruster operation and low voltage power for housekeeping and XFC operation from the spacecraft power buses. When low voltage (housekeeping) power is applied, the PPU performs initialization checks and waits for spacecraft commands. Following verification that the high voltage bus is active, the system enters startup, conditions the system for operation, and can then start the thruster. The spacecraft command and data handling bus provides commands to the system and receives telemetry from the system. The system receives pressure-regulated xenon propellant from the spacecraft xenon feed system.

Table 1. Performance Requirements of Advanced Electric Propulsion System.

EP String Total Input Power, max (kW)	Discharge Voltage @ Thruster (V)	Thrust, min (mN)	Specific Impulse, (sec)	Total System Efficiency
13.3	600	589	2800	57%
11.1	500	519	2600	55%
8.9	400	462	2300	54%
6.7	300	386	1900	52%

The AEPS harnessing between the components is designed to allow easier spacecraft integration and gimbaling of the thruster by dividing the PPU-to-thruster power between two harnesses, thereby reducing the thickness and stiffness as compared to using a single harness. One harness is dedicated to the primary discharge power. The other is dedicated to thruster auxiliary power, which includes power required by the cathode heater, cathode keeper, and electromagnets.

Propellant flow rate is controlled and regulated by AEPS via the PPU. Power from the PPU is provided to the XFC on a single harness. The PPU provides the necessary current to open and close the XFC latch valve. The PPU also provides the necessary voltages to control the size of the orifice in the piezoelectric anode and cathode valves, which regulate the propellant flow rate to the anode and cathode propellant lines on the thruster. The architecture facilitates system operation in the event of a spacecraft propellant regulator failure. In the event of such failure, the PPU will monitor the temperature on the XFC and provide sufficient current to an integrated XFC heater, ensuring that the high pressure propellant remains in a gaseous phase as it passes through the anode and cathode flow control valves. 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. Figure 5 includes the CAD images of the three AEPS components.

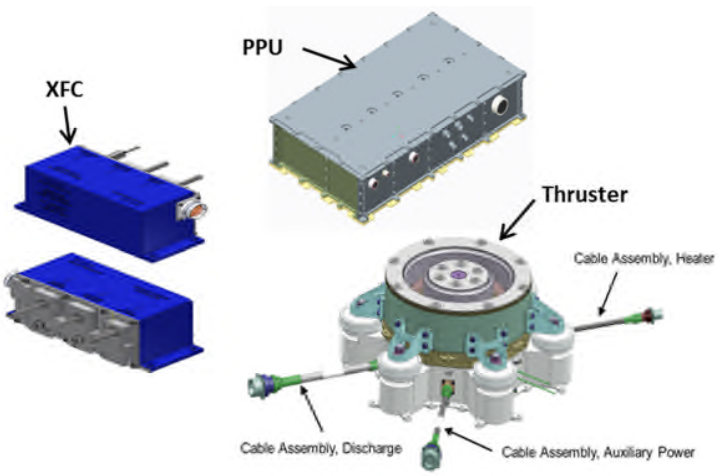

Figure 5. AEPS components – Thruster, XFC & PPU

The XFC and PPU are maintained within their required thermal environments via thermal conduction through temperature-controlled mounting surfaces on the spacecraft. The AEPS architecture allows installation of the XFC inside or outside of the spacecraft. The thruster is designed to be thermally isolated from the spacecraft. The thruster is equipped with integrated heaters and temperature sensors that may be operated by the spacecraft thermal management system to maintain the thruster above its minimum qualified temperature limits. This allows the thruster to be located on a gimbal or boom far away from temperature-controlled surfaces and minimizes plasma impingement on spacecraft surfaces. 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 system survives deep space solar radiation for up to 15 years. The thruster radiates only 22 W to the spacecraft when operating at discharge power levels of up to 12.5 kW.

The PPU is designed to be radiation tolerant for a deep space environment. This covers missions with durations up to 15 years and 23,000 hours of operating time and 1,700 on/off cycles. The modular design of the PPU provides the capability for operation at lower power output when one or more of the power modules are disabled.

The architecture provides the spacecraft with accurate measurements of anode and cathode propellant flow rates and all relevant voltages, currents and temperatures. (See the two columns of telemetry in Table 2).

Table 2. Analog Telemetry

Analog Telemetry	
Cathode heater current	I/F Assy 2 Temp
Cathode heater voltage	Discharge Module 1 Temp
Cathode keeper current	Discharge Module 2 Temp
Cathode keeper voltage	Discharge Module 3 Temp
Inner Magnet Current	Discharge Module 4 Temp
Inner Magnet Voltage	High voltage bus input voltage
Outer Magnet Current	High voltage bus input current
Outer Magnet Voltage	Low voltage bus input voltage
Discharge Current	Low voltage bus input current
Discharge Voltage	Discharge current ripple (RMS and Peak to Peak)
Cathode-to-PPU chassis DC voltage	Discharge voltage ripple (RMS and Peak to Peak)
Cathode-to-Thruster-Body Current	PPU Housekeeping 1 - HK Pwr Supply +3.3V_{DC} Voltage
Aux Pwr Supply 1 Temp	PPU Housekeeping 2 - HK Pwr Supply +5V_{DC} Voltage
Aux Pwr Supply 2 Temp	PPU Housekeeping Voltage 3 - HK Pwr Supply +15V_{DC} Voltage
Aux Pwr Supply 3 Temp	PPU Housekeeping Voltage 4 - HK Pwr Supply -15V_{DC} Voltage
Aux Pwr Supply 4 Temp	XFC Anode-Leg flow control device temperature
HK Pwr Supply Temp	XFC Anode-Leg Pressure
Aux I/F Board Temp	XFC Cathode-Leg control device Temperature
XFC Board Temp	XFC Cathode-Leg pressure
I/F Assy 1 Temp	All XFC drive voltages

To ensure successful development of the AEPS flight system, Aerojet Rocketdyne has developed a time dependent system performance model to address the interactions between components, design adjustments, production tolerances as well as the major interactions with the spacecraft. The AEPS system performance model has been designed to account for these interactions in a way that allows evaluation of the sensitivity of the system to expected changes over the planned mission as well as to assess the impacts of normal component and assembly variability during the production phase of the program. The model also allows for the assessment of the system stability when exposed to oscillatory perturbations and assessment of the system response to transients The results will ensure the component requirements do not unnecessarily drive the system cost or overly constrain the development program. Finally, the model is available to assist in troubleshooting any future unforeseen development challenges.[8]

VI. EP String Operation Architecture

The EP string has nine modes of operation which are defined in Table 3. Boot mode is performed only by the Boot Loader software. All other modes are performed with the operational OFP software. For nominal thrusting operations, when power is applied, the system performs initialization checks and waits for spacecraft commands to perform requested operation.

Table 3. EP String Modes of Operation

Mode	Description
Unpowered	All PPU outputs off and XFC valves closed. No communication.
Boot	Only mode that can accept software updates.
SwInit	Initialization of software internal data structures and external interfaces
Standby	Bus inputs are in range, communications available, all outputs off.
Initialization	Magnet bakeout, propellant line bakeout, cathode conditioning, cathode purge, anode purge and xenon venting.
Operation	Startup, steady-state thrusting, throttling, shutdown.
Safe	Places string in a safe state. All outputs off, communications available.
Manual	Allows for manual operation of the PPU for troubleshooting purposes.
Maintenance	Allows for data table updates, memory writes and verification.

The system initiates startup following receipt of the spacecraft command. The SwInit and Initialization sequences are performed only once during the initial operation on-orbit. The Operation mode consists of four states shown in Figure 6. The system can transition between any two operational states when commanded by the spacecraft. The Heater Only and Keeper Only states maintain requisite cathode temperatures for efficient electron emission and keep the EP string in a state of readiness for rapid thruster start. While in the Thruster Control state, the system can maintain a constant throttle level, perform a thruster start, throttle to a new operating point or shut down.

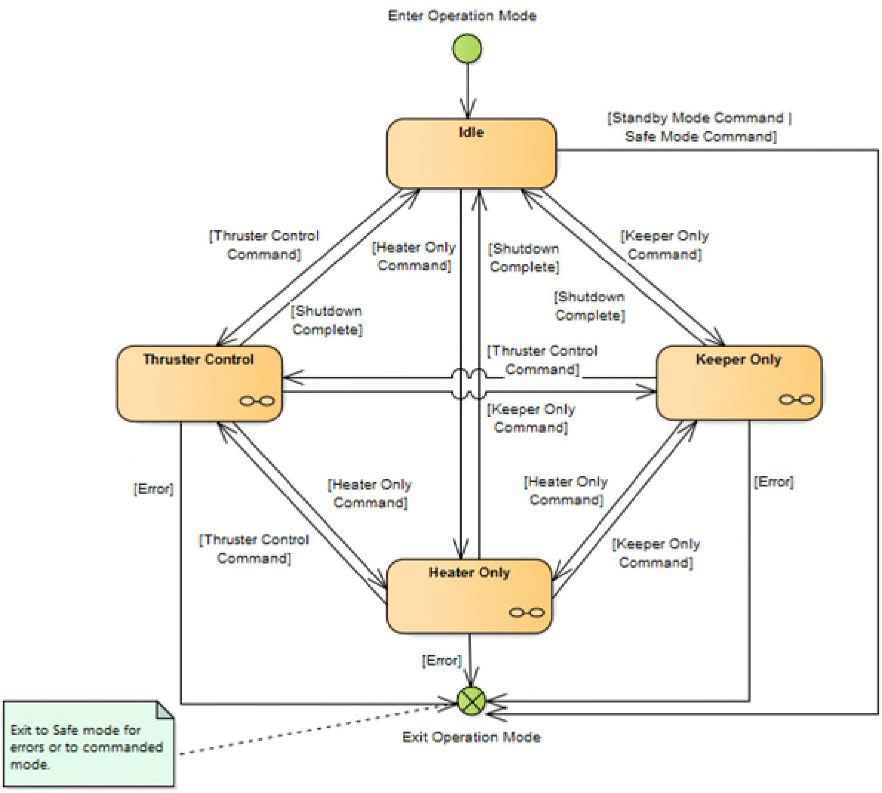

Figure 6. EP String Operation Sequences

In the event of anomalous operation, AEPS will self-protect through an integrated fault monitoring system. If telemetry strays outside of the expected range, the PPU will notify the spacecraft of the anomaly. The spacecraft has the option to maintain the current operating point or change operating points in an effort to resolve the anomaly. If telemetry strays too far from its expected range, the PPU will notify the spacecraft of a fault and then perform an automated shutdown of the system to avoid potential damage. The expected operating range for the fault protection system will be established during the Engineering Development test phase. The design of the fault protection system allows for reconfiguration throughout the mission.

The major control loop within AEPS is focused on achieving the desired thruster discharge current commanded by the spacecraft. The PPU does not control discharge current directly. Instead, it regulates propellant flow rate to the thruster. While the electrical interactions between the PPU and thruster are very fast, the propellant flow interactions between the XFC and thruster are relatively slow. Flow rate changes are on the order of seconds, while the electrical changes may occur over periods smaller than a millisecond. Transients associated with these interactions may occur over several seconds or less than a millisecond. Designing a control loop that provides the prompt command response desired by the mission and stability against a wide spectrum of perturbations is challenging, especially when the system is designed to allow integration on a variety of spacecraft configurations.

VII. EP String Testing

The program completed an early integrated system test (EIST) which demonstrated operation of the laboratory propulsion system for the first time at NASA Glenn Research Center's Vacuum Facility 6. Laboratory versions of the PPU, HCT, and XFC were used for this test. The breadboard PPU consisted of an AR Discharge Supply Unit (DSU) which contained input and output power modules and a ZIN Technologies built System Flow Controller (SFC), which was used to control XFC valves. A NASA developed Test Demonstration Unit One (TDU-1) thruster and a VACCO's Xenon Flow Control Module (XFCM) completed the rest of the test EP string. While the NASA TDU-1 thruster and VACCO XFCM have been tested as a system before, this test was the first time that they were operated together with an AR's DSU and SFC in closed-loop operation. An overview of the EP string used during the EIST is shown in Figure 7.

The system under test included Aerojet Rocketdyne's breadboard power processing unit (PPU), NASA's Hall Effect Rocket with Magnetic Shielding (HERMeS) Technology Demonstration Unit-1 (TDU-1), and VACCO's xenon flow control module (XFCM).

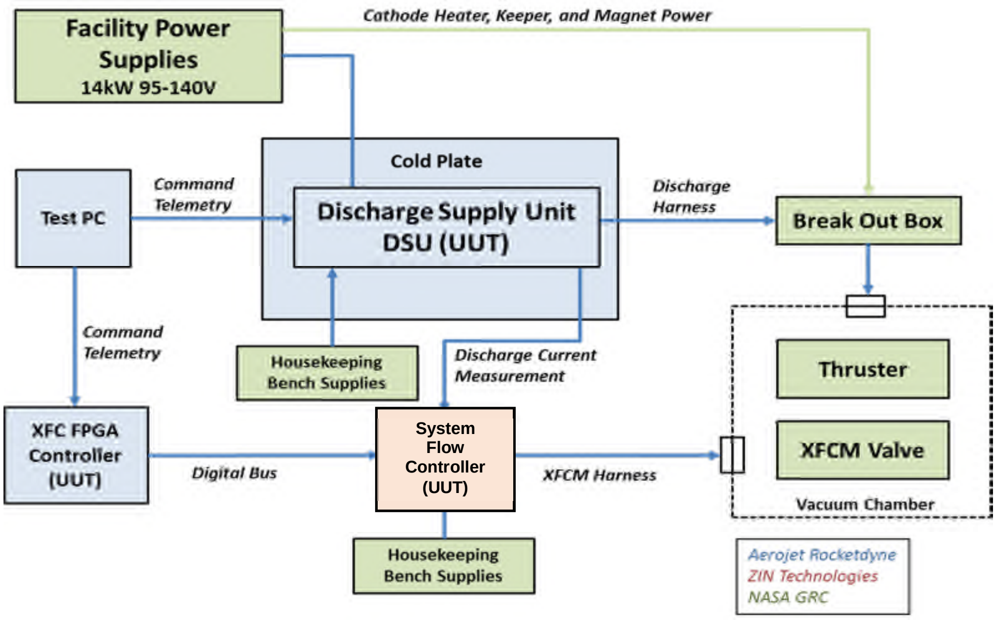

Figure 7. Early Integrated System Test Diagram

The PPU architecture demonstrated operation over input voltages between 95 V and 140 V. The PPU provided discharge powers between 2.4 kW and 13.1 kW to the TDU-1 thruster. Discharge voltages varied between 300V and 630 V and discharge currents varied between 8 A and 25 A. The PPU series/parallel architecture demonstrated proof of concept by sharing power between discharge modules in a tightly controlled manner. The PPU successfully demonstrated closed-loop control of both the discharge current and the XFCM flow rate.

Thruster performance measurements were made at reference firing conditions and were characterized over a 2x range of magnetic field strengths. Test results and thruster plume measurements were also found to be comparable to previous NASA measurements. This testing will form the baseline for the system to compare against, as enhancements are implemented during the engineering development phase.

The PPU and XFCM integrated together to provide stable steady-state operation, consistent transient operation, and precise flow rates. The system demonstrated the capability to meet power contingency objectives and finely tune throttle points. Conducted emissions, thruster impedance, and thruster discharge oscillations were also characterized.[9]

A development integrated string test that is currently slated for spring 2019. It will use an Engineering Development PPU, HCT, XFC and AEPS automated special test equipment. This test will encompass the baseline performance, characterization, extended, and margin testing that was conducted as part of the EIST. Environmental testing will be completed at the component level. In addition, the EDU system will undergo a 200 hour test prior to Critical Design Review (CDR) in mid-2019 and then an extended 4300 hour wear test afterwards. A system level radiated emissions characterization is also planned as part of the development effort.

Qualification testing will be performed on the first string of flight production units in 2020. Upon completion of component level testing, the HCT, PPU, XFC and test harnesses will be integrated and system level testing performed. The system testing includes electrical checks and a full system performance test to ensure the integrated system operates as expected. All string testing is planned to take place at the NASA GRC Space Simulation Test Facilities.

VIII. Conclusion

Development of the AEPS system architecture has been accomplished through a rigorous system engineering process. Early system testing of the prototype hardware demonstrated the capability of system performance and characterized system interactions between elements. Opportunities to optimize the design of the individual elements were also identified during EIST which have been incorporated into the EDU design. A system performance model has been developed and anchored by testing and supports additional optimization as the team continues of the path to CDR in 2019.

IX. Acknowledgements

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