13kW Advanced Electric Propulsion System Power Processing Unit Development

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Abstract: Solar Electric Propulsion (SEP) can enable missions that are otherwise not feasible, especially missions with a large total impulse requirement [1], [2]. Key to the function of any SEP system is the conversion of power from the spacecraft's unregulated DC bus voltage to the currents and voltages required to reliably start and operate the thruster. NASA has identified the 10 to 15kW class Hall thruster system as necessary for human exploration missions in the coming years [3], [4]. Aerojet Rocketdyne's 13kW Advanced Electric Propulsion System (AEPS) program will provide the propulsion to make deep space missions affordable and sustainable with the first application on the Power Prolusion Element of the Lunar Orbiting Platform Gateway [5]. AEPS provides NASA with a high-power, high-specific impulse, and highly-throttleable Electric Propulsion (EP) string for deep space transport vehicles. The Development of AEPS PPU has addressed many new requirements, which enable Solar Electric Propulsion to provide efficient, reliable propulsion.

Nomenclature

AEPS	=	Advanced Electric Propulsion System
AR	=	Aerojet Rocketdyne
CDR	=	Critical Design Review
DMC	=	Discharge Master Controller
DSU	=	Discharge Supply Unit
EDU	=	Engineering Development Unit

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EP	=	Electric Propulsion
FPGA	=	Field Programmable Gate Array
GRC	=	Glenn Research Center
HERMeS	=	Hall Effect Rocket with Magnetic Shielding
PPE	=	Power and Propulsion Unit
PPU	=	Power Processing Unit
SCB	=	System Control Board
SEP	=	Solar Electric Propulsion
VF	=	Vacuum Facility
XFC	=	Xenon Flow Controller

I. Introduction

The AEPS is an electric propulsion system developed for the PPE [5] and other deep space exploration, which builds upon the NASA research and development efforts of a PPU for the 12.5 kW magnetically-shielded Hall thruster and incorporates Aerojet Rocketdyne's experience with flight PPUs and Electrical Power System experience. The AEPS PPU provides the system control for the AEPS electric propulsion string, the Xenon Flow Controller (XFC), discharge power supply, magnetic power supplies, and the keeper power supply. The PPU software includes the algorithms to control and operate the EP string at the desired thrust and specific impulse levels.

The PPU is capable of processing unregulated input voltage of 95V to 140V and providing 300V to 630V output to the thruster. To address these challenges, the PPU employs a distributed control architecture that consists of a System Control Board (SCB), a Discharge Master Controller (DMC), and four Power Module Controllers at each power module. Each power module contributes up to 3.3kW of power, which combines to deliver output power from 3kW to more than 13kW to the EP string. Each level of control has a discrete set of control functions, fault protections, and telemetry reporting. The resulting architecture allows the electronics to handle a broad range of mission operating requirements such as capability to execute at least four unique HCT startup sequences.

Proper management of the interaction of the PPU's controls ensures stability at the component (SCB, DMC & PMC), subsystem (PPU), and system level. Providing system stability during startup and over the wide range of operating conditions required throughout mission life is a significant challenge. The controls architecture handles instability issues that arise from both mission complexity and electronics complexity, by having configurable parameters and fault ranges expandable while providing synchronization of PPU components.

The effort described in this paper addresses the PPU power and control architectures to accommodate the wide range of input/output operations incorporating controls that have previously resided in higher levels of the spacecraft command and control system. NASA GRC is not only ARs customer, but also a vital independent development partner that greatly contributed to the development of the PPU by providing a collaborative feedback through design review and independent risk reduction test activates.

II. System Architecture

The system produces thrust by ionizing xenon gas and then accelerating the positive xenon ions using electric and magnetic fields. An electric propulsion (EP) string consists of a Hall Current Thruster (HCT), a Power Processing Unit (PPU), a Xenon Flow Controller (XFC), and harnesses that connect these components. Figure 1 illustrates a simplified AEPS functional block diagram

The HCT is a 12.5kW electric thruster using xenon as its propellant. The XFC provides the xenon propellant to the HCT, regulating the xenon pressure, and regulating flow in response to commands from the PPU. The spacecraft can include multiple AEPS strings to provide fault tolerance or to increase total impulse.

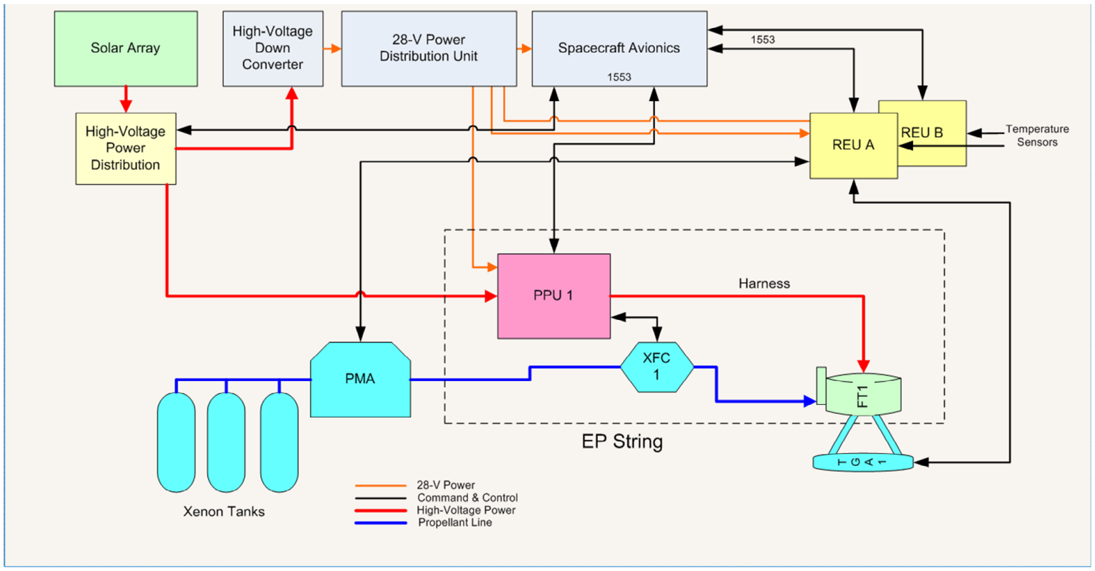

Figure 1: Simplified AEPS Functional Block Diagram

The PPU provides all command, telemetry, and power interfaces between the spacecraft, thruster, and XFC. The PPU operates using two input power buses; 1) The low voltage bus (22V-36V) provide internal housekeeping voltages for command, telemetry and system control. 2) The high voltage bus (95V-140V) provides power for the Auxiliary Power Supply Assembly and the Discharge Supply Unit. Communication between the PPU and the spacecraft for command and data uses a MIL-STD-1553B bus. Figure 2 illustrates the PPU's output capabilities in terms of voltage and current and compared to the power required for the AEPS HCT. This PPU has a wide input and output voltage operating range offering flexibility for a wide range of mission scenarios with a small penalty of decreased power density over other PPUs.

Key metrics for the PPU:

- Input power: 13.3 kW input (14 kW contingency)
- Input voltage range: 95 V to 140 V, unregulated DC bus
- Discharge voltage range: 300 V to 600 V at 20.8 A nominally (up to 630 V contingency)
- Discharge current range: 10 A to 20.8 A nominally (up to 30 A capability)
- Output power: up to 12.5 kW nominally
- Baseplate temperature range for operation: -15°C to 50°C
- Dimensions: 900 mm X 518 mm X 200 mm
- Mass: 62 kg

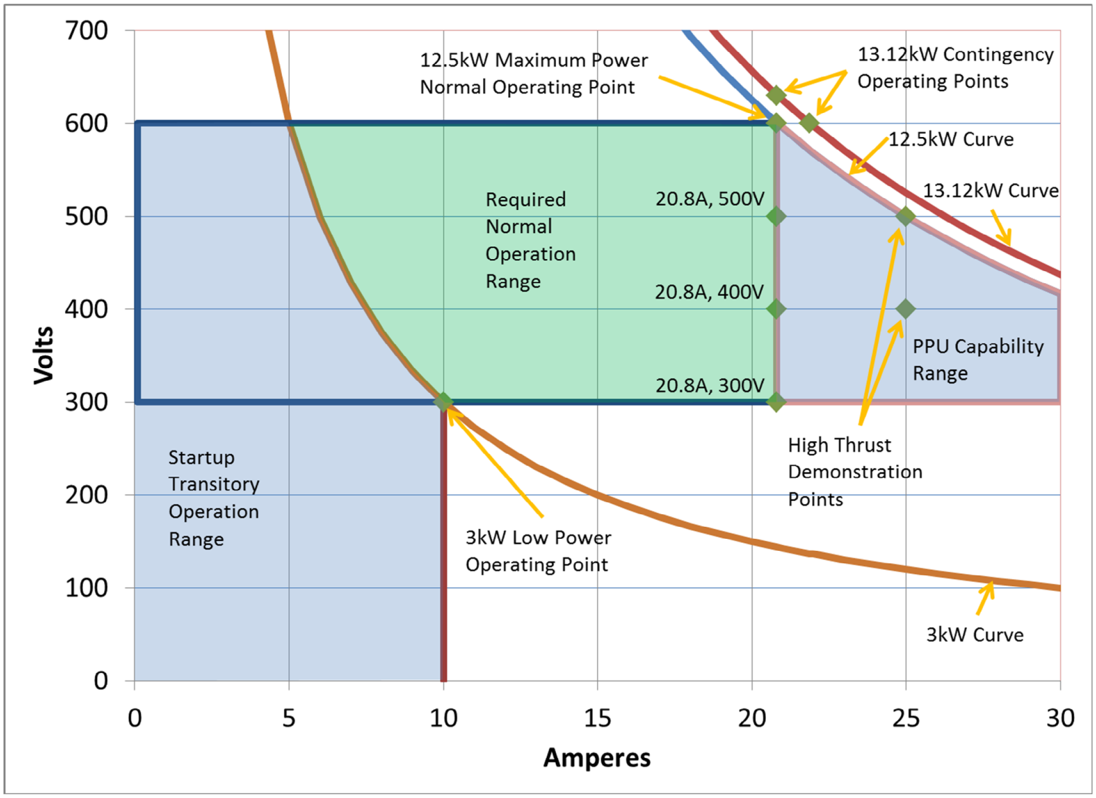

Figure 2: AEPS PPU Discharge Current vs. Discharge Voltage Capabilities

III. Control System

The AEPS PPU contains multiple control systems that provide reliable stable propulsion from the EP string. These control systems include the flow rate control to the anode and cathode Discharge Supply Unit (DSU), discharge current control, xenon flow control of anode and cathode and auxiliary supply control, as shown in Figure 3. The heater, Ignitor/keeper and magnet power supplies provide 516W/16A, 660V/3A and 130W/5.1A respectively. An advancement in the AEPS design over prior Hall thruster auxiliary supplies is the 12-bit set point resolution available. This allows for output current set point resolution down to about 10milliamp level. This results in better than 1 % accuracy measurement of output current over the 15 year design life.

A stability control system is a result of understanding the load characteristics of each PPU output, which have an effect on each converters performance capabilities such as the input impedance. Plasma loads may also have operating points that exhibit a negative impedance that can affect converter stability. The bandwidth of a typical PPU control loop bandwidth is generally well below the breathing mode frequencies, so we developed a method of measuring the low-frequency impedances of Hall thruster discharges. Our test fixture designed such that it prevents interaction and plasma instabilities and is not susceptible to breathing mode currents. The measured magnet coil impedance permitted the control optimization.

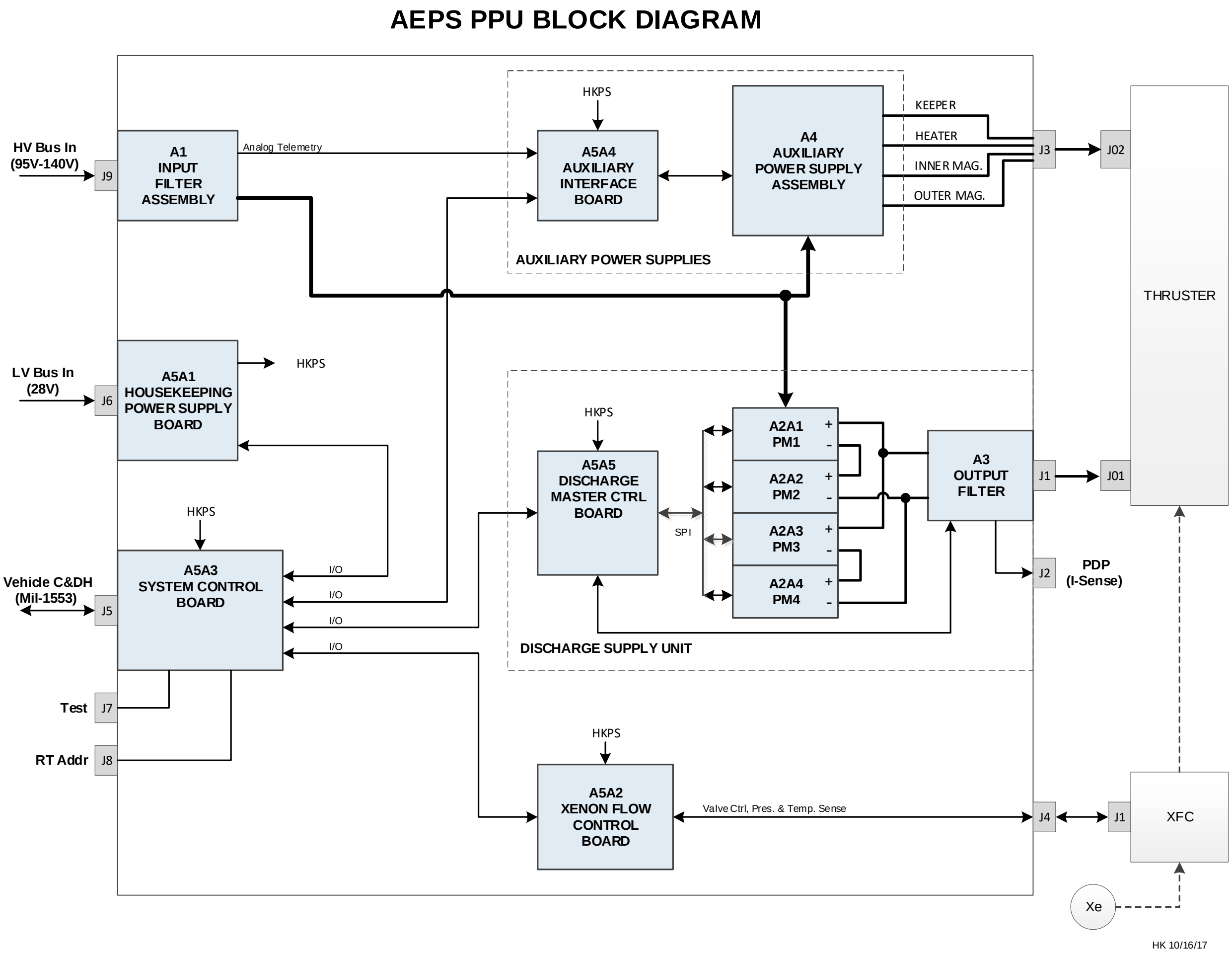

Figure 3: AEPS PPU Block Diagram

IV. Discharge Supply Unit

The DMC receives feedback from the analog telemetry circuit and provides commands to enable and set the power level for each PM. The control system closes the loop output voltage by comparing the feedback to the voltage commanded by the spacecraft. Simultaneously, the control system distributes the same power command to Power Module and thus equal power dissipation. This results in a tightly regulating the output voltage while assuring components run at optimal efficiency.

The DSU control system provides many benefits to the overall design. The discharge voltage and discharge current directly affect thruster ISP and thrust. Therefore, it is imperative to accurately control and measure discharge voltages and current, which are the fundamental parameters to calculate thrust for vehicle navigation. The enhanced digital and analog filter of the control system enable the minimization of noise sources, which cause telemetry error.

The digital control loop has configurable parameters, which are adjustable to tailor voltage and current limits as well as gain coefficients. These parameters are defaulted to tuned values or values specific to the current AEPS mission. However if mission parameters change or the design needed to be applied to a PPU which could provide different power levels the design could be dropped in place without having to redesign the control system. The programmable gain allows the DSU to be tuned to have different output impedance levels and gain/phase margins. During the first integrated test with the AEPS Thruster, the gains were adjusted during the test campaign on the fly to improve thruster ripple levels.

Original gains. 24.8Vp-p New gains. 12Vp-p

Figure 4: a) Initial Gain Settings, b) Optimized Gain Settings

Figure 4a shows the discharge ripple voltage with the initial gain settings, which yielded significantly high voltage ripple than desired with additional low frequency oscillations not shown. Figure 4b shows over 50% voltage ripple reduction after the gains were adjusted to provide a higher system bandwidth. Also eliminating the low frequency oscillations. This will provide a capability to adjust parameters throughout the life of the system to accommodate planned and unplanned changes in spacecraft conditions.

A unique control system exists between the PPU, HCT and XFC. The PPU orchestrates the control of this complex system. Thruster and XFC valve telemetry are used by the PPU to control thrust and keep the thruster in ideal operating conditions.

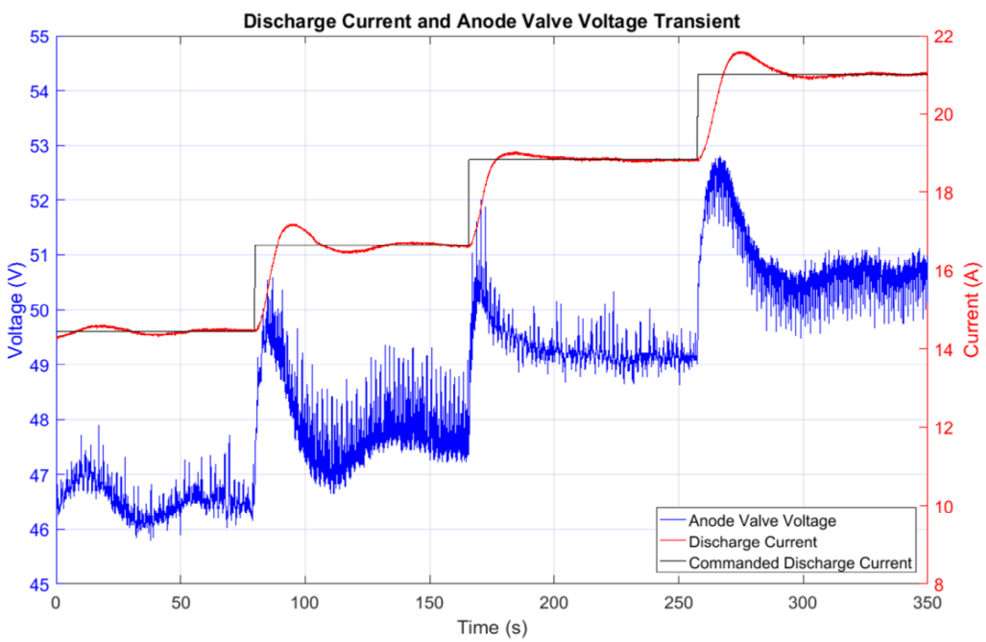

Figure 5: Discharge Current Control with flow control

The operation of closed loop discharge current mode is shown above in the Figure 5. The discharge current settles on the discharge current command. This is a result of the control system detecting difference between command current and telemetry and then commanding the anode valve voltage to adjust which assures the correct flow rate occurs which in turn adjusts the discharge current. Much like the DSU voltage loop, these control system parameters are adjustable. Different piping lengths of the final EP-string will not be an issue for the control system because the control loop can be retuned for the final length without having to update the hardware design.

The initial PPU design included six power modules with two parallel-connected strings of output rectifier stages. Each string had the output rectifiers from three power modules connected in series in order to reduce the voltage stresses across the output rectifier diodes [6]. Environmental conditions can increase the diode voltages beyond the derated voltage. Various clamping or snubber circuits limit the rectifier diode voltages, and some are able to recirculate energy instead of dissipating it [7]. The clamping circuits that recirculate energy instead of dissipating it are often called lossless snubbers [8]. We developed a highly effective lossless snubber circuit that recirculates the energy while limiting voltage spikes to an acceptable level. Consequently, we were able to reduce the number of series-connected output stages from three to two while still meeting the diode voltage derating.

Figure 6 shows the test results of the breadboard DSU and Engineering Development Unit. The results indicate that the EDU will DSU will have a full power efficiency of approximately 95%.

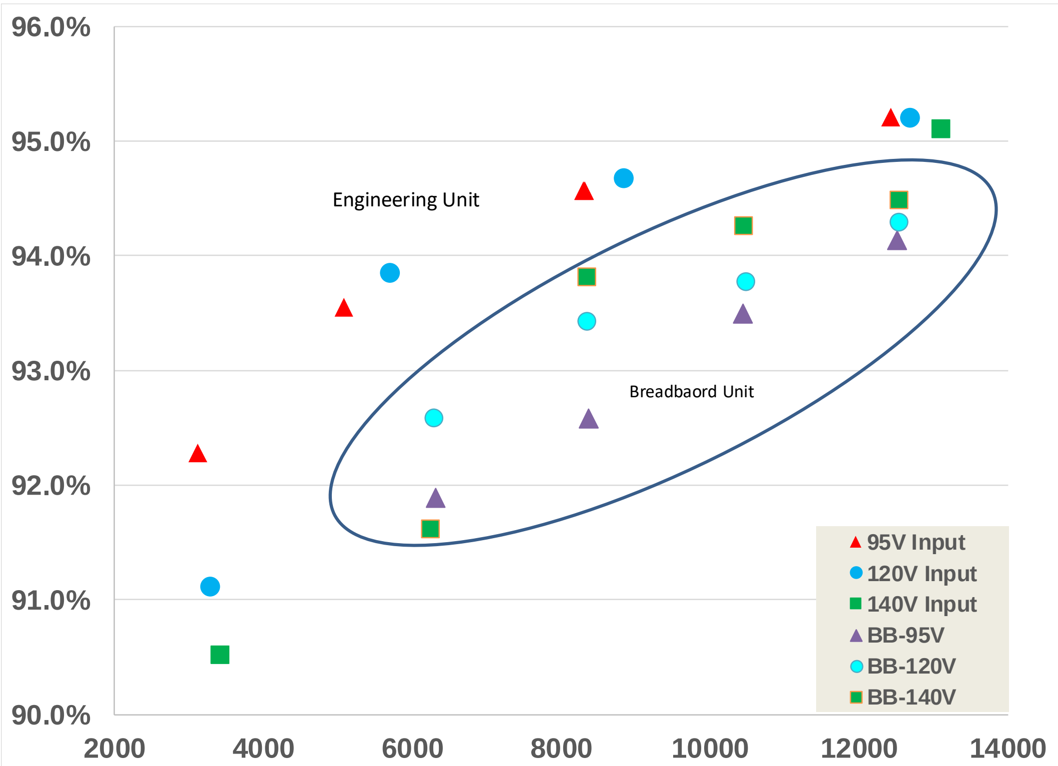

Figure 6: DSU Efficiency

A key safety feature for protecting the PPU against H-bridge failures is the Primary Over-Current (POC) protection [9]. The POC protection circuit provides cycle-to-cycle over-current protection with less than one µs response time preventing the PPU from sustaining an over-current event that could damage the H-bridge or other components.

One particular safety feature for the thruster is the programmable output over-current protection, which limits current in the event of a spark event. During a spark event, the impedance of the thruster load can act as a short circuit, so current limiting during this situation is critical to limit component stress and not draw excess power from the spacecraft power bus. The programmable nature of the overcurrent protection also facilitates alternate thruster starting techniques like a glow mode start where the PPU initially operates in a current limited state. Similarly, the PPU has a programmable output over-voltage protection that protects both the PPU's output components from exceeding derating nominally, and rating in the extreme cases, but can also protect the thruster.

For design robustness, the PPU's outputs to the thruster are designed to survive a short between any two output nodes. The PPU has built-in mechanisms to verify continuity and isolation during Assembly, Test, and Launch Operations (ATLO) allowing spacecraft-level verification without de-mating the flight harnesses.

V. Analog Telemetry

Control Loop Telemetry

In order to provide the required data to perform digital control of the Discharge Supply Unit, voltage and current telemetry needs to be measured for each of the power modules and provided to the FPGA for use in the control loop logic.

Since the telemetry is referenced to the cathode, which is isolated from the primary power supplies, the telemetry circuitry architecture needed to include isolation capable of withstanding the full range of voltages that can be developed between the thruster cathode and PPU housekeeping voltage returns. Conventionally, voltage and current telemetry are isolated using magnetic or optical isolation to provide a primary return referenced signal for measurement by a primary return referenced analog-to-digital converter, however this can result in a loss of accuracy when transmitting analog signals across the isolation barrier. In this implementation, rather than provide isolation on the signals being measured, an isolated DC-DC converter and a digital isolator IC were used to eliminate the need to pass the analog signals across the isolation barrier and instead perform the analog-to-digital conversion referenced to the cathode, and isolate the digital signals.

The analog telemetry that is used as an input to the digital control loop is required to operate at a sample rate of at least 100 ksps in order to ensure that the DSU operates with sufficient gain and phase margin. Since additional telemetry, such as cathode potential and individual PM voltages are necessary for error handling and fault protection but not digital control, a scheduling scheme was implemented to sample certain parameters at the full 100 ksps and others at a lower data rate, removing the need for additional data buses or needlessly fast components.

Ripple Telemetry

In order to assess whether or not the thruster is at a stable operating point, during ground testing an assessment of thruster ripple is frequently used. By comparing the RMS and peak-to-peak values of the ripple current or voltage, it can be determined whether or not the thruster is in a stable operating point or fluctuating. During laboratory testing, oscilloscope data was used to determine thruster stability (see Figure 7) and equivalent data was required to be available to the control system. This is accomplished by taking the thruster voltage and current telemetry and using a bandpass filter to extract the ripple portion of the signal for measurement. While this does result in large peak-to-peak waveforms during commanded changes in power level, software can be used to omit data gathered during and immediately after any commanded transitions. This ripple signal is sampled by a positive and negative peak detector and differentiated to generate a DC representation of the peak-to-peak ripple. An analog multiplier IC with an integrator in the feedback loop is used to implement an RMS to DC converter. Both of these measurements on both the ripple voltage and current are included in the output telemetry stream to the same FPGA being used for control.

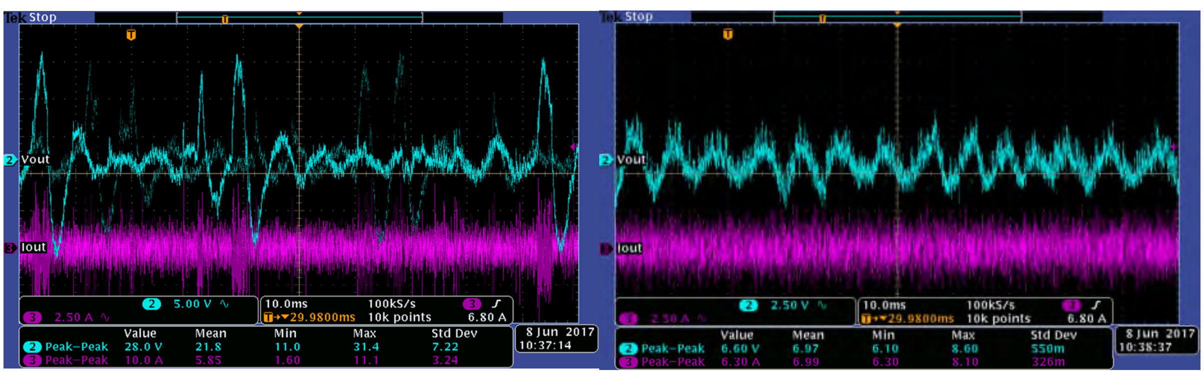

Figure 7: a) Unstable Breathing Mode b) Stable breathing mode

Since software only collects this data infrequently, the FPGA reports the minimum, maximum, and average values of both the peak-to-peak and RMS voltages and currents over the cycle period. This allows for a comparison of long-term average ripple to short-term spikes by using the average RMS and the min and max peak-to-peak measurements.

VI. *AEPS PPU: Risk Reduction Activities at NASA GRC*

The development of the AEPS PPU has been a collaborative effort by Aerojet-Rocketdyne, ZIN Technologies, and NASA Glenn Research Center (GRC). NASA GRC has developed a number of PPU technologies in-house during the past decade, some of which include: a 12.5 kW PPU that operates from a nominal input of 120 Vdc and a 15 kW PPU designed to operate with a high-input voltage of 300 Vdc to 500 Vdc [10]. Leveraging this in-house expertise in power electronics and PPU technology development, NASA GRC's contribution to AEPS has been focused on improving the robustness of the AEPS PPU technology by performing hardware testing of one Power Module (PM) breadboard assembly and the following Auxiliary power supply sub-assemblies: Inner Magnet, Outer Magnet, Heater, and Keeper/Igniter. The knowledge gained during these in-house risk reduction test campaigns was shared with the two industry partners.

The PM breadboard tested at NASA GRC was an early stage hardware prototype built by AR (see Figure 8). Since it did not include the Discharge Supply Unit's digital controller for closed-loop operation, an analog controller was designed by GRC to command the PM into an open-loop fixed output state. One of the first tasks of this in-house risk reduction effort was to assess the component rating requirements and their respective margins by measuring the peak MOSFET voltages, transformer currents, and voltage excursions across the rectifier diodes. Test results provided useful information for updating the design and improving component margins.

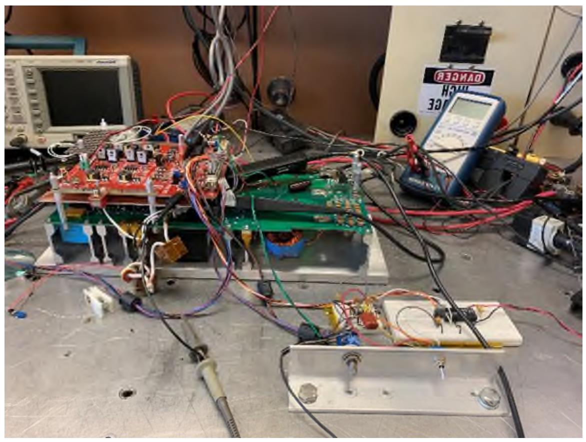

Figure 8: Power Module Breadboard Hardware at NASA GRC

NASA GRC also investigated the implementation of the PM's output rectifier lossless snubber circuit and its effect on the control circuitry. Overall, the snubber showed an impressive performance in reducing rectifier voltage excursions and improving efficiency compared to other design solutions. However, because of the snubber implementation, the complexity of the control circuitry increased, and it was noted that the controller was more sensitive to noise and introduced a non-linear high gain. Modifications were made that improve the noise sensitivity and linearize the gain. The plots of output current vs current command before (Figure 9) and after (Figure 10) circuit modifications are shown below.

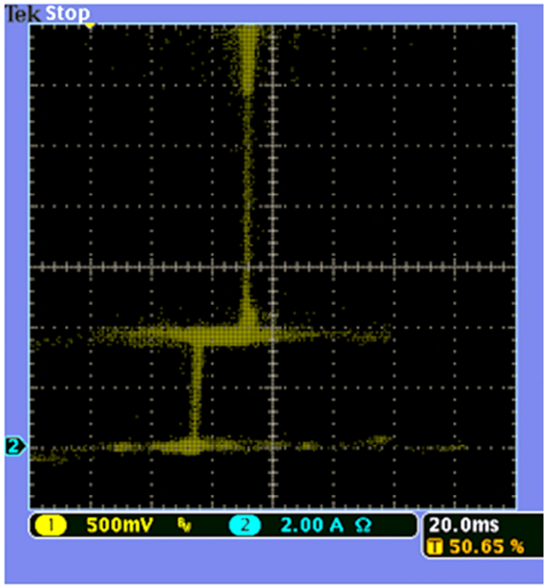

Figure 9: Transfer Function of Output Current (vertical axis) vs Commanded Current (horizontal axis) before Control Circuit Modifications

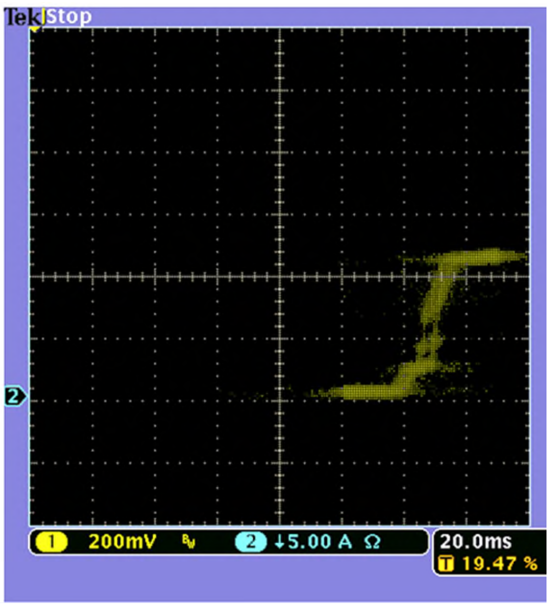

Figure 10: Transfer Function of Output Current (vertical axis) vs Commanded Current (horizontal axis) after Control Circuit Modifications

Additionally, NASA GRC has been characterizing the performance of the Auxiliary power supply sub-assemblies: Inner Magnet, Outer Magnet, Heater, and Keeper/Igniter. These power supplies have been designed by ZIN Technologies in collaboration with AR and NASA GRC, and leverage the heritage designs from the NEXT-C PPU [11]. As with the PM testing, the hardware tested at NASA GRC is of breadboard fidelity, consisting of a common housekeeping supply board, a unique controller, and a power stage board for each supply. Although these breadboards are somewhat different from the finalized hardware, NASA GRC has still gained valuable insight and understanding into the performance of the supplies by supplementing the data collected by ZIN with additional data to characterize the supplies. ZIN performed testing to verify requirements, such as output current range, load resistance, output ripple, loop stability, and over-current limits. NASA GRC expanded on this testing to include evaluation of component de-rating, transient response, regulation, open and short circuit operation, loop stability, and thermal performance. This testing will complement analyses done by the project and provide a means for evaluating the robustness of the design, and look for avenues where the flight design could be improved. Thus far, NASA GRC has found that the auxiliary supplies are performing well, meeting AEPS requirements and otherwise operating as expected.

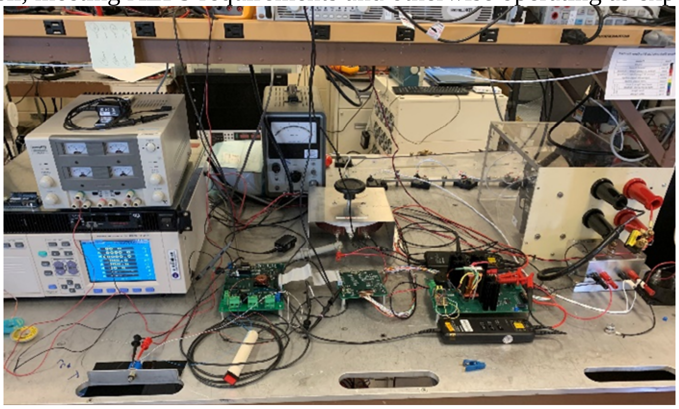

Figure 11: Auxiliary Magnet Power Supply Hardware Testing at NASA GRC

VII. Conclusion

The AEPS PPU has needs beyond the typical PPU and provides broader range of capabilities that will enable mission operators and scientist new insight into the flight performance of an EP system. AR and NASA have developed a PPU that meets the needs AEPS with new health monitoring capability to evaluate thruster stability as well as employ programmable limits for numerous fault protection parameters. This resulted in a PPU capable of providing over 12.5kW of discharge power with a 95% efficiency.

Acknowledgments

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