Development of a High-Propellant Throughput Small Spacecraft Electric Propulsion System (HT-SSEP) to Enable Lower Cost NASA Science Missions

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Outline

1. Motivation
2. Background
3. Stakeholders, CONOPS, and Requirements
4. Thruster and PPU Development
5. Test Results
6. Concluding Remarks

HT-SSEP: High-Propellant Throughput Small Spacecraft Electric Propulsion

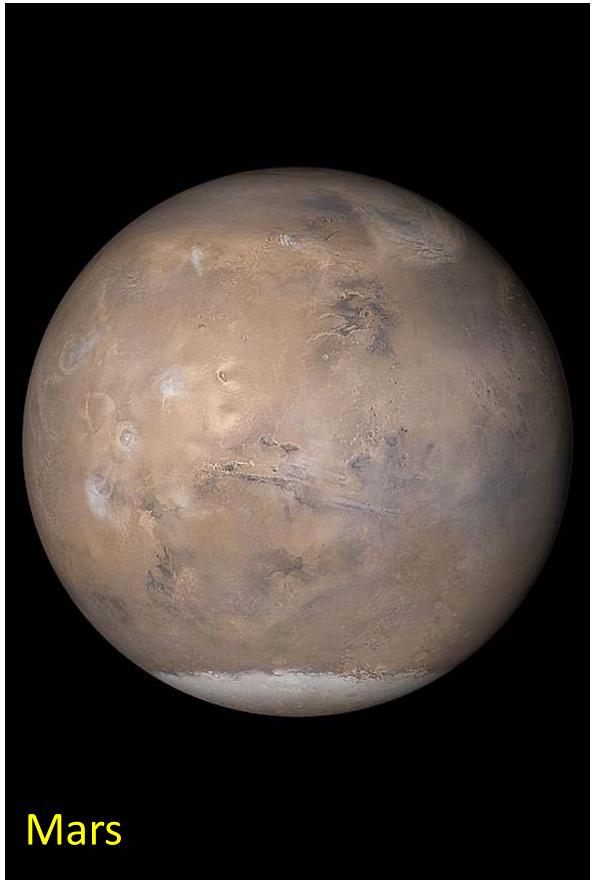
Motivation

"NASA is … expanding the use of lower-cost CubeSats and SmallSats to accomplish our science goals," NASA Strategic Plan 2018

- The lower development cost of smaller spacecraft have the potential to increase the cadence of NASA missions.
- Interesting destinations within reach include asteroids, moons, and Mars.

Martian Moon Deimos

Asteroid Bennu

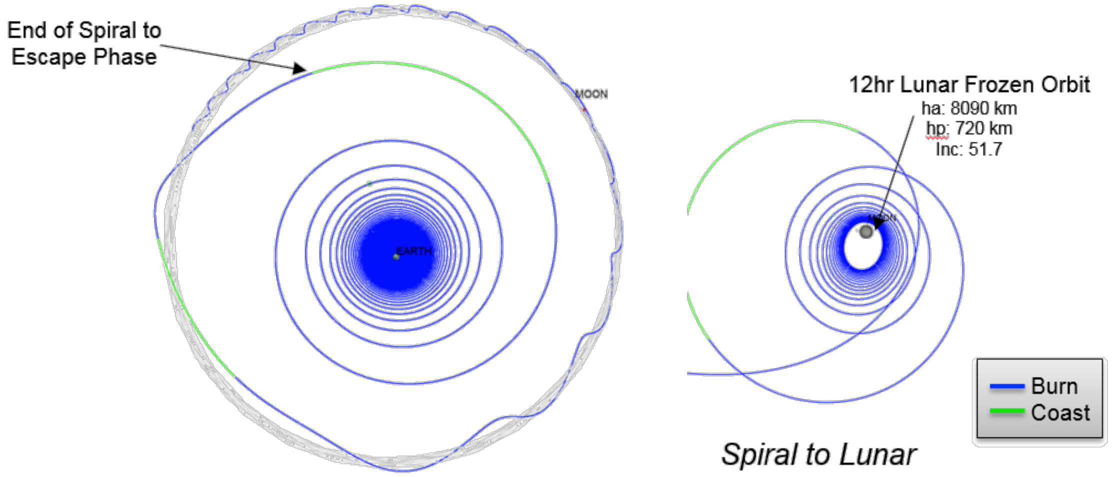


Mars

Example Mission Requiring HT-SSEP

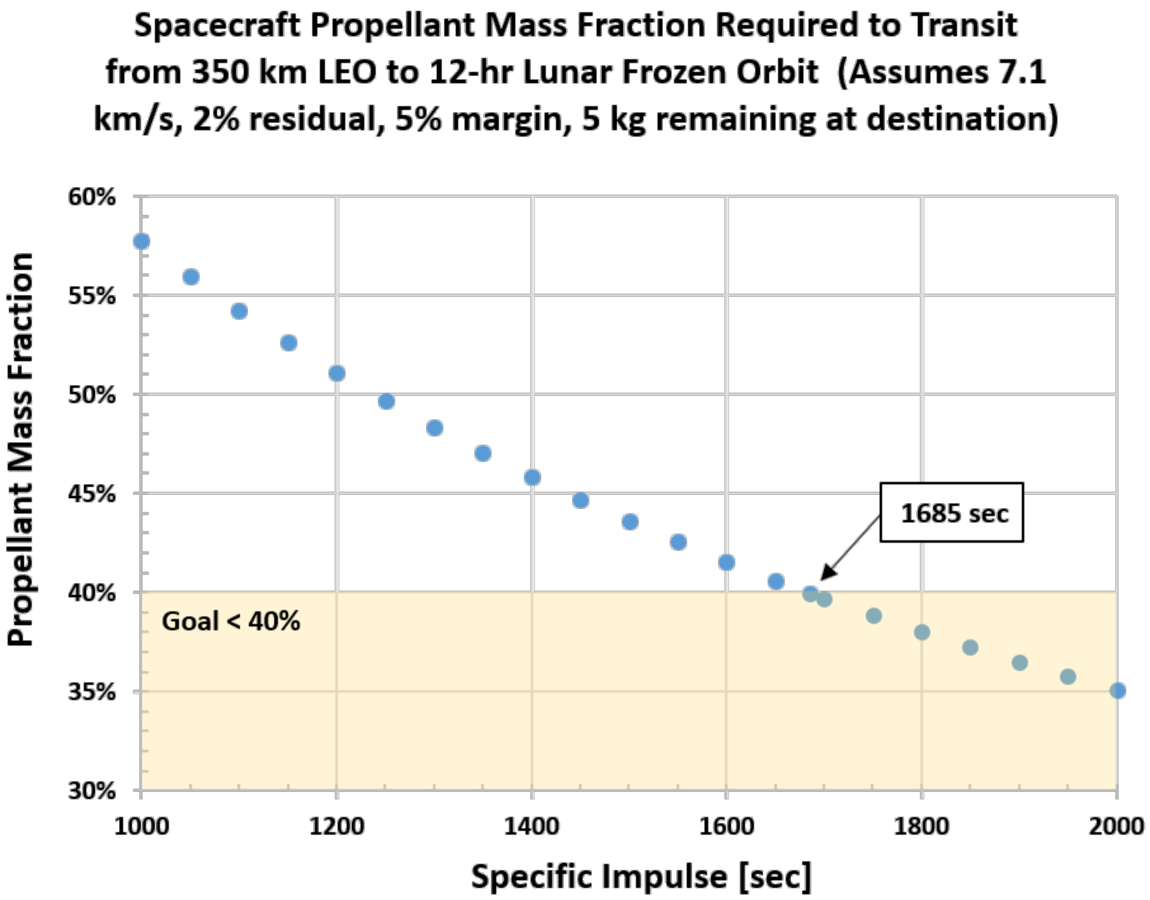
A pair of small spacecraft in a 12-hour lunar frozen orbit could provide near continuous communication between Earth and rovers or other science instruments located near the poles.

- **Delta-v requirement ~ 7.1 km/s**
  - transfer from 350-km LEO to 12-hr lunar frozen orbit.
- **Specific impulse > 1685 s**
  - to achieve a S/C propellant mass fraction less than 0.4.
- **Assume 200-kg S/C, 80-kg propellant**
  - 69.9 kg used for transit.



Spiral to Escape

Spiral to Lunar Frozen Orbit



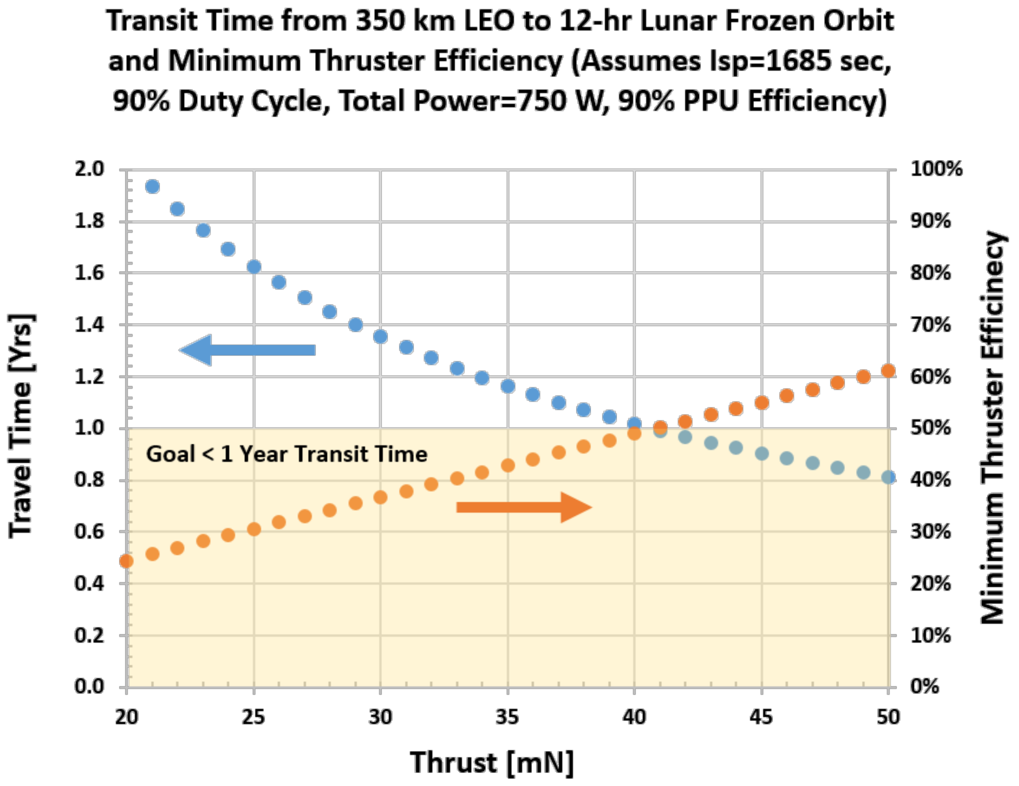
Example Mission Requiring HT-SSEP

A pair of small spacecraft in a 12-hour lunar frozen orbit could provide near continuous communication between Earth and rovers or other science instruments located near the poles.

- Transit time from 350-km LEO to 12-hr lunar frozen orbit assuming I_{sp} of 1685 seconds and 90% duty cycle.
- Minimum thruster efficiency assuming 750-W input to PPU and 90% PPU total efficiency.
- Transit can be accomplished in 1 year if thruster discharge input to beam power conversion efficiency exceeds 49%.

Example HT-SSEP Specification	
Specific Impulse	1685 s
Thrust	40 mN
Thruster Efficiency	49 %
PPU Total Efficiency	90 %
Propellant Throughput Capability	80 kg
Lifetime	10,200 hrs

Example Mission: 350 km LEO to 12-Hr Lunar Frozen Orbit	
Delta-V	7.1 km/s
Power to Propulsion System	750 W
Transit Time	1 Year
Duty Cycle	90 %
Spacecraft Wet Mass	200 kg
Initial Propellant Mass	80 kg
Propellant Consumed in Transit	69.9 kg



Motivation for HT-SSEP

- Most Hall thruster technologies under development are known to have capabilities optimized for LEO missions, rather than deep space.
- Significant focus on servicing growth in the LEO constellation market.
- NASA requirements typically drive product cost beyond viable for commercial use in LEO/GEO applications.
- Lack of a clear source for flight-ready, low-power HT-SSEP results in considerable technical risk for NASA to undertake small spacecraft missions beyond LEO using EP.
- NASA GRC is well positioned to advance the state-of-the-art in sub-kW electric propulsion to support future NASA mission needs by leveraging recent advances in Hall thruster technology.
- **Goal:** Develop HT-SSEP technology and provide a no-cost, non-exclusive license U.S. industry.

GRC has identified no domestic flight-ready sub-kW high-propellant throughput electric propulsion systems proven capable of meeting NASA high delta-v mission needs beyond LEO.

Background – SKEP Project

Sub-Kilowatt Electric Propulsion (SKEP) Project

- In FY18, NASA STMD funded the Sub-Kilowatt Electric Propulsion (SKEP).
- The project pursued development of a high-propellant throughput sub-kW thruster and power processing unit (PPU). (Patent Pending: LEW-20035-1, LEW-20041-1, LEW-19799-1)

Commercialization path from project inception:

- Project sought industry input from project inception in definition of system requirements, identification of commercial applications benefiting from high delta-v propulsive capability, and further optimization of the commercialization (i.e., technology transfer) strategy.
- Project sought to compile a comprehensive package of design / process documents licensable at no cost non-exclusively to U.S. industry.
- Project benefited from the existing, competitively-selected Space Flight Systems Development and Operations Contract (NNC13ZMA015R) for engineering support toward delivery of a qualification unit.

The project sought to limit recurring unit cost and improve commercial viability by systematically optimizing the propulsion system (*thruster, power processing unit, and feed system*), rather than exclusively optimizing the thruster.

Background - Stakeholders

- The HT-SSEP development is a technology push.
  - The stakeholders were not well defined at project inception.
  - The project sought interest from NASA mission directorates and U.S. industry (specifically potential end-users).
- A key NASA stakeholder identified was the SMD Planetary Sciences Division (PSD).
- A handful of U.S. industry stakeholders were identified.
  - Interest in the technology was first identified at a JANNAF conference in May 2018.
- A Request for Information (RFI) was drafted to further engage U.S. industry
  - Release of the RFI was blocked by NASA HQ (concerned industry was receiving too many RFIs).
- Feedback from PSD and U.S. industry was sufficient to develop a useful requirements set to focus the SKEP project effort and provide a high likelihood of meeting future mission needs.

Concept of Operations (subset)

Through discussions with the identified stakeholders, a concept of operations was formulated.

The HT-SSEP system should:

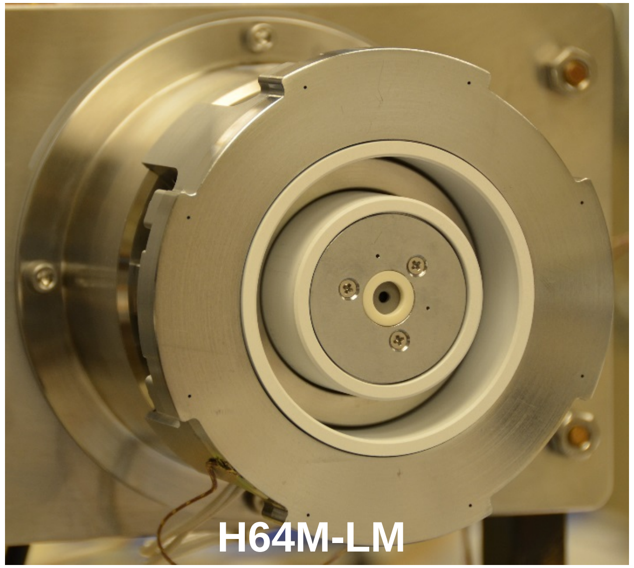
- impart ~2 MN-s on a small spacecraft, while consuming less than 1-kW input power,
- be capable of more than 8,000 cycles to support station keeping,
- have a reversible magnetic field (to switch direction of swirl torque),
- be electrically isolated and thermally insulated from the spacecraft bus,
- protect the spacecraft against all conceivable propulsion system electrical faults,
- accept an unregulated low-voltage power bus consistent with most small spacecraft,
- accept commands and return telemetry via a common small spacecraft communication protocol,
- operate at two qualified firing conditions, including nominal power and a reduced power state,
- operate at additional higher and lower than nominal power firing conditions,
    - although they may not be fully qualified to minimize development cost
- employ a scalable, low-cost PPU architecture able to accommodate more than one domestic small spacecraft Hall-effect thruster, and
- require no on-orbit maintenance by the spacecraft other than survival heaters.

System Requirements (subset)

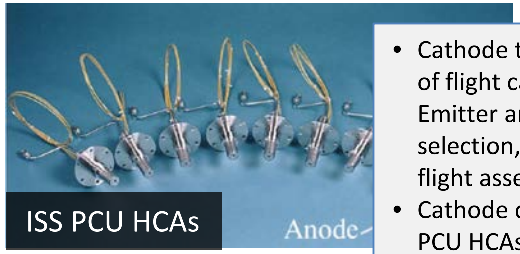
Name	Requirement
Nominal Thrust	The propulsion system shall generate >= 40 mN of thrust at the nominal operating condition over its full operational lifetime.
Nominal Total Specific Impulse	The propulsion system total (anode + cathode flow) specific impulse shall be >= 1600 seconds at the nominal operating condition over its full operational lifetime.
Propellant Throughput	The propulsion system shall have a total propellant throughput capability of >= 100 kg at the nominal operating condition.
Propellant	The propulsion system shall meet all performance requirements when fed xenon propellant.
Cyclic Lifetime	The propulsion system shall be capable of 8,000 on/off cycles over its full operational lifetime.
Reversible Swirl Torque	The propulsion system shall have the ability to set the direction of the swirl torque induced by thruster operation.
Input Voltage	The propulsion system shall operate over an input voltage range of 24 to 34 VDC.
Enable Time	The propulsion system shall reach 95% of steady-state thrust in less than 10 seconds following receipt of the enable command.
PMA Outlet Pressure	The propulsion system shall meet all performance requirements when exposed to or operating at a pressure management assembly (PMA) outlet pressures of 40 +/- 3 psia.
System Mass	The propulsion system shall have a mass <= 10 kg (including thruster, PPU, flow controller, and interconnects, but excluding tank, gimbal, and PMA).

HT-SSEP Development Approach

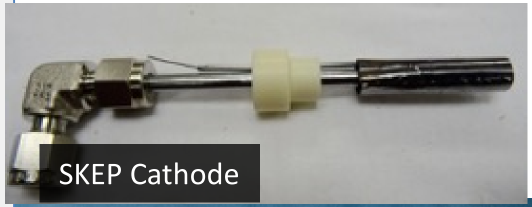
- Project goals were achieved by leveraging prior NASA investments.
- **Development approach:**
  - Minimize parts count
  - Minimize “touch labor” in assembly
  - Maximize use of heritage components and/or processes (high TRL)
  - Leverage recent Hall-thruster technology advancements
  - Performance optimization was **not** included in the approach
    - Goal was to provide an initial robust capability
    - Reserve optimization for future projects
- **Key technologies leveraged:**
  - Advanced Hall-effect thruster magnetic circuit design
  - Centrally-mounted hollow cathode
  - GRC-pioneered hollow cathode assemblies (derived from the International Space Station plasma contactor units)
  - GRC-patented robust propellant manifold assembly design
  - Extensive in-house power processing unit development expertise



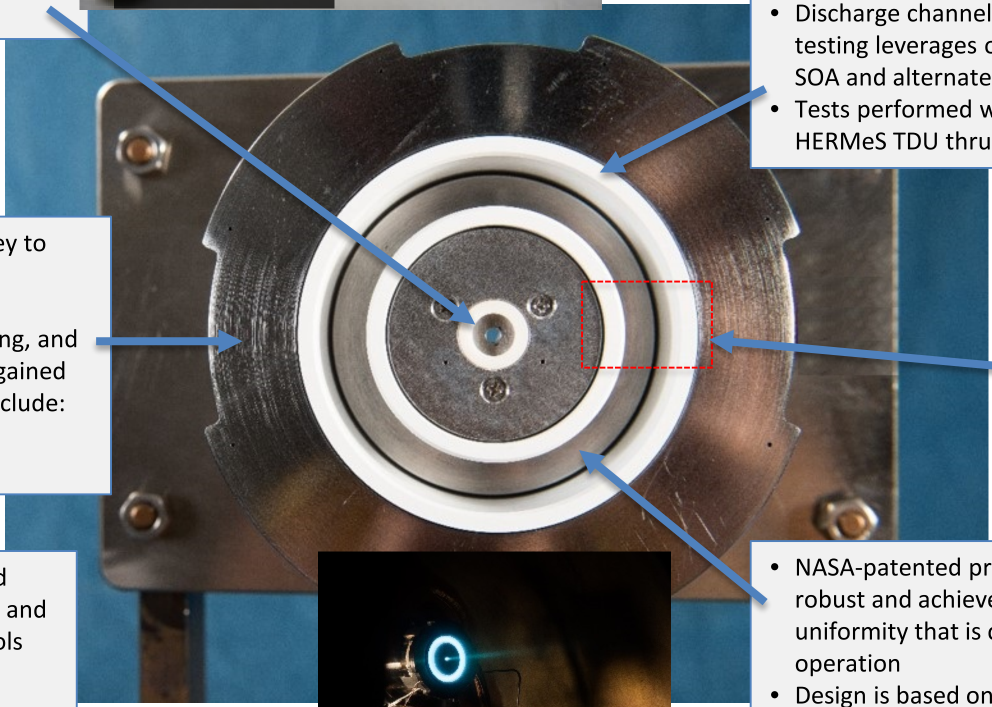
HT-SSEP Thruster Design Heritage



- Cathode technology leverages over 35 years of flight cathode development that includes: Emitter and cathode assembly materials selection, swaged heater manufacturing, and flight assembly manufacturing process
- Cathode developments include: NASA ISS PCU HCAs, NSTAR, DAWN, NEXT-C, AR XR-5 (NASA heritage), and AEPS



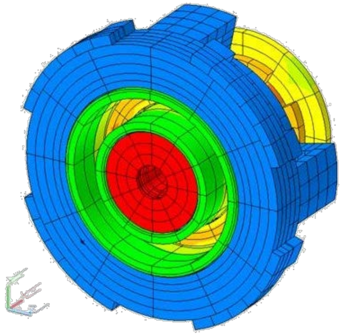
- Functional test procedures (electrical, magnetic, and flow) that assure thruster design will meet performance design objectives
- Developed detailed pre-test checkouts to confirm hardware test readiness



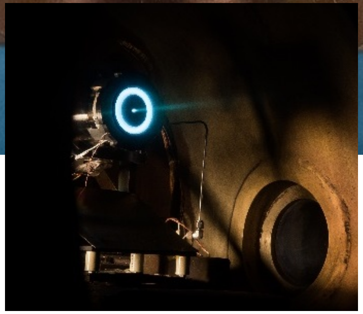
- Discharge channel material selection and testing leverages over 25 years of tests with SOA and alternate materials
- Tests performed with NASA HiVHAc and HERMeS TDU thrusters

- Magnetic thruster components are key to attaining the designed magnetic field topology
- Magnetic material selection, processing, and manufacturing leverages knowledge gained from previous NASA programs that include: NASA 300M, HiVHAc, and HERMeS

- Magnetic field topology design leverages over 25 years of NASA GRC & JPL research and development efforts to optimize the magnetic field configuration to maximize thruster performance and lifetime
- Technology implemented and demonstrated on: NASA 120, 173, 300M, 300MS, 400M, 457M, HiVHAc, H6MS, and HERMeS thrusters



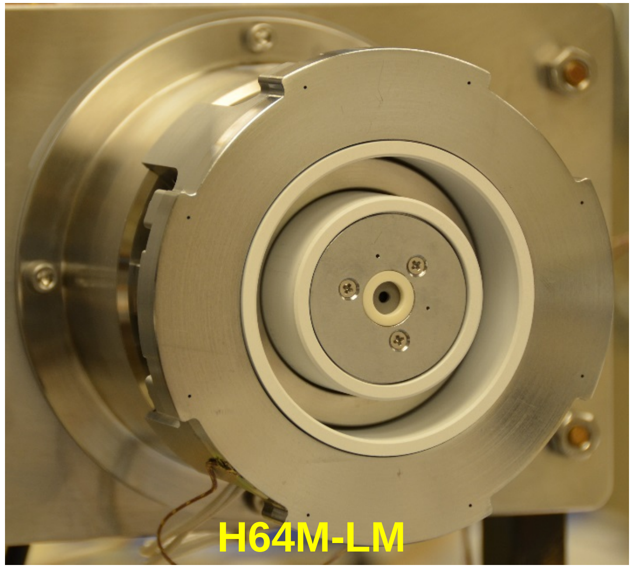
- Thruster magnetic, plasma, thermal, and structural modeling leverages NASA GRC and JPL extensive and validated modeling tools that can predict thruster performance, temperatures, structural viability, and lifetime
- Tools include: Magnet7, JPL Hall2De, JPL OrCad2, & Thermal Desktop



- NASA-patented propellant manifold that is robust and achieves excellent propellant flow uniformity that is critical for efficient thruster operation
- Design is based on the HERMes TDUs and AEPS thruster design

NASA-H64M and NASA-H71M

- While system requirements were being developed, a proof-of-concept thruster called the H64M was manufactured and evaluated.
  - H64M is a nominally a 600-W thruster at 350 V that produces around 37 mN of thrust with an Isp of approximately 1600 seconds.
  - Demonstration of the H64M indicated that all project goals were achievable.
- Following demonstration of the H64M and collection of NASA / industry inputs, the thruster design was iterated accordingly to a nominal power of 750 W at 350 V. This thruster iteration is known as the H71M.
  - The anticipated thrust is ~ 50 mN with an Isp of ~ 1700 seconds.
  - The H71M design is still in progress.
  - The H71M is a lower-cost implementation of the technology demonstrated with the H64M, while maintaining the key features that give the H64M its enhanced performance relative to the SOA low-power Hall thrusters.



H64M-LM Development Activities

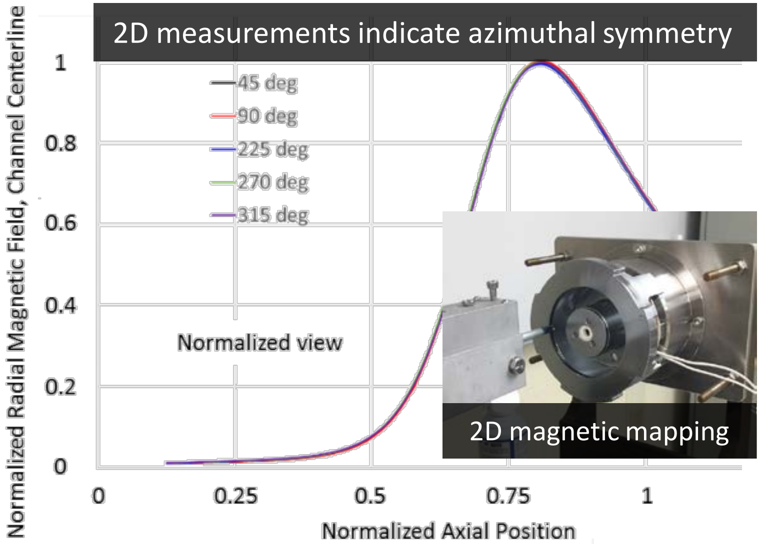
Magnetic Circuit

H64M-LM magnetic circuit design leverages NASA GRC's experience with magnetic topology design of several NASA Hall thrusters that include HiVHAc, 300M, and HERMeS.

The advanced magnetic circuit results in:

- High thruster performance by implementing a magnetic lens; and
- High-propellant throughput capability, which translates to long thruster life

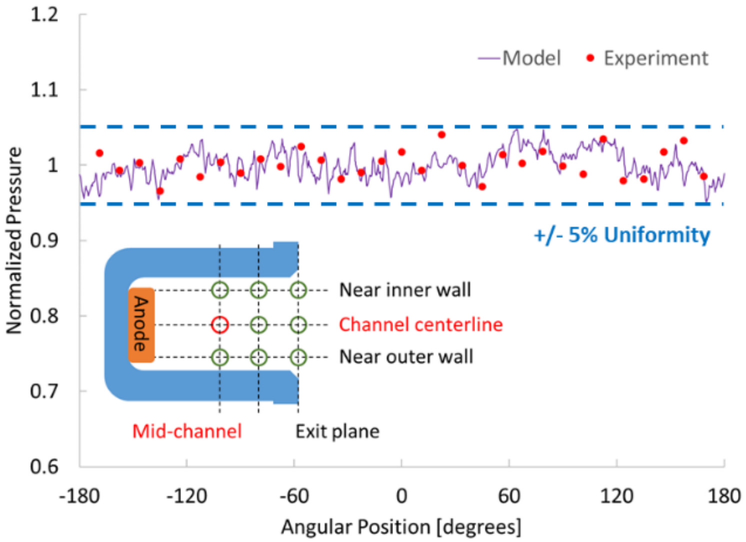
The H64M magnetic circuit has been assembled, tested, and mapped (matches model prediction).

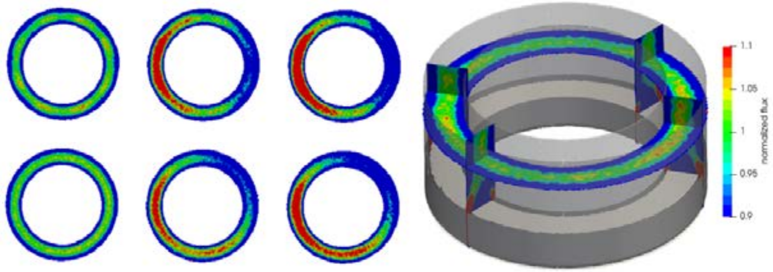


Propellant Distributor / Anode

Must promote neutral flow uniformity and tolerate electron backstream heating.

Modeling suggests and testing agree azimuthal flow uniformity within ±5%



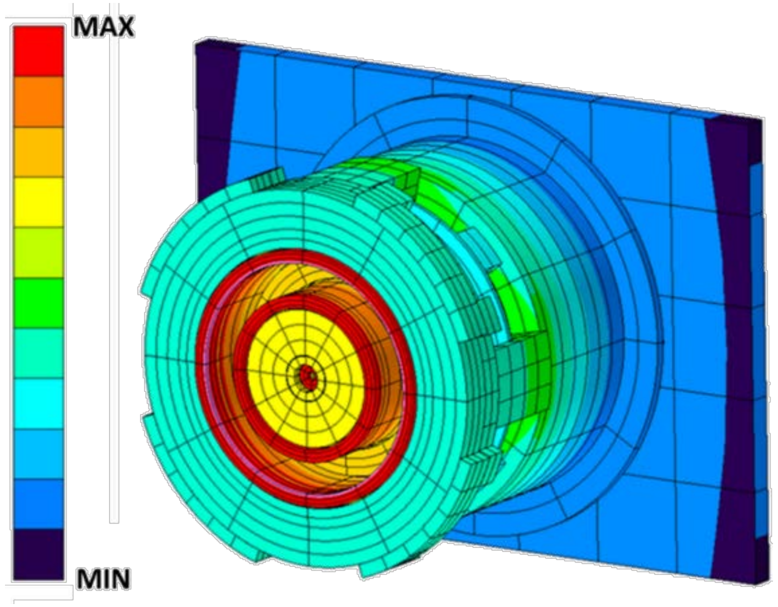


Thermal Design

Develop a thermal model of H64M-LM to aid the design process and assess thermal management options as the design evolves.

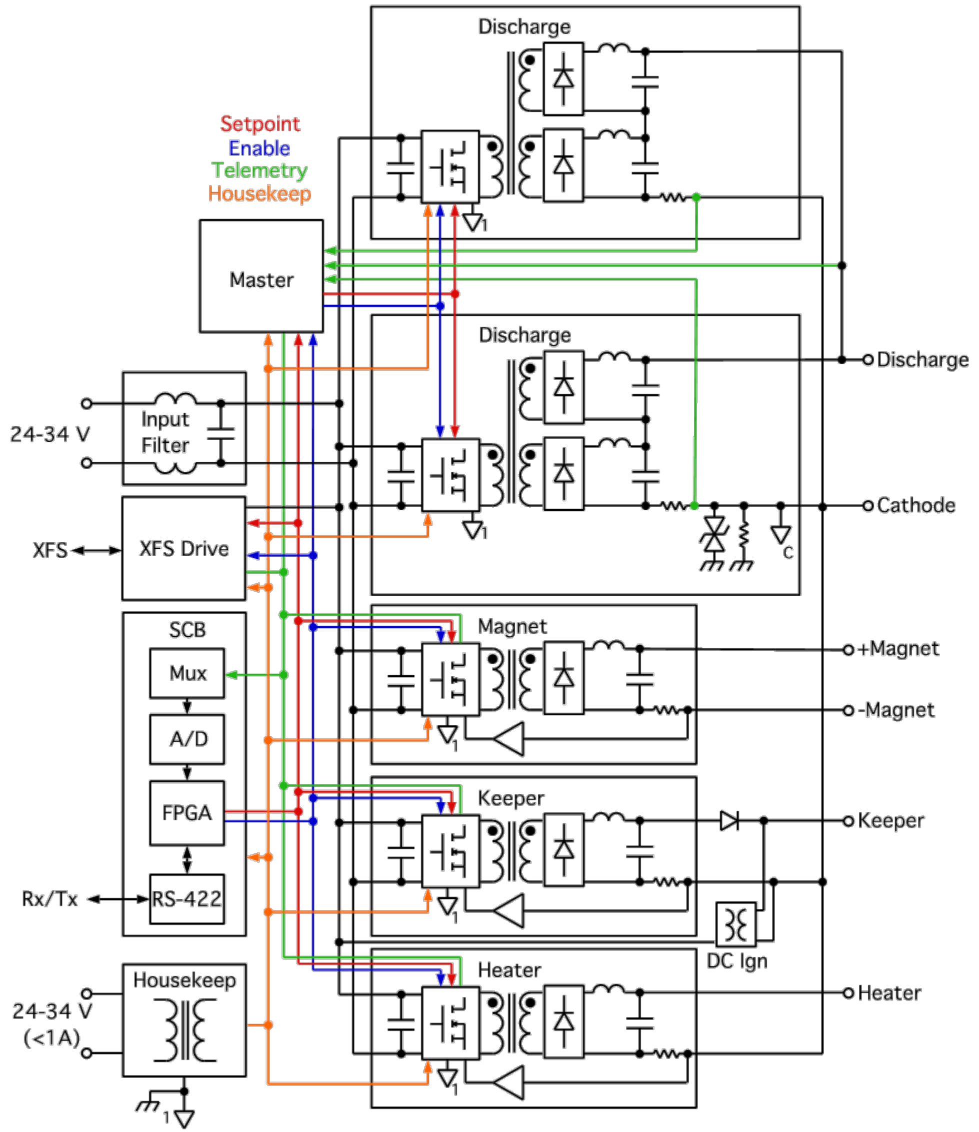
During the thermal characterization test, the thruster was equipped with ~20 thermocouples.

Comparison of correlated thermal model and experimental test results show good agreement. The results show moderate temps for all components with large thermal margins on traditional limits.



PPU Architecture

- Discharge module based on GRC heritage full-bridge designs
  - Paralleling modules enable scaling power for different thrusters, minimizing future PPU development costs
- Parallel module approach mitigates efficiency losses due to the low-voltage/high-current input
- Use of 2 series-stacked full wave rectifiers on the output allows the use of lower-voltage components
- An additional rectifier stage could be added for output voltages greater than 350 V, though electrical efficiency would be reduced
- A flyback converter is the baseline approach for each auxiliary.
  - The same architecture for each auxiliary supply reduces overall system costs.
  - Keeper differs only slightly with the addition of DC ignitor circuit.



Parallel Discharge Summary & Estimates

Pros: (compared to single module approach)
- Lower power per module
- Lower MOSFET and diode stress
- Higher efficiency
- Staggered switching reduces input current ripple/filter size
- Easy to scale power by adding/removing modules
- Lower development cost than a line of different power PPUs

Cons: (compared to single module approach)
- More complex, more parts, greater unit cost
- Greater mass
- Master controller required for current sharing

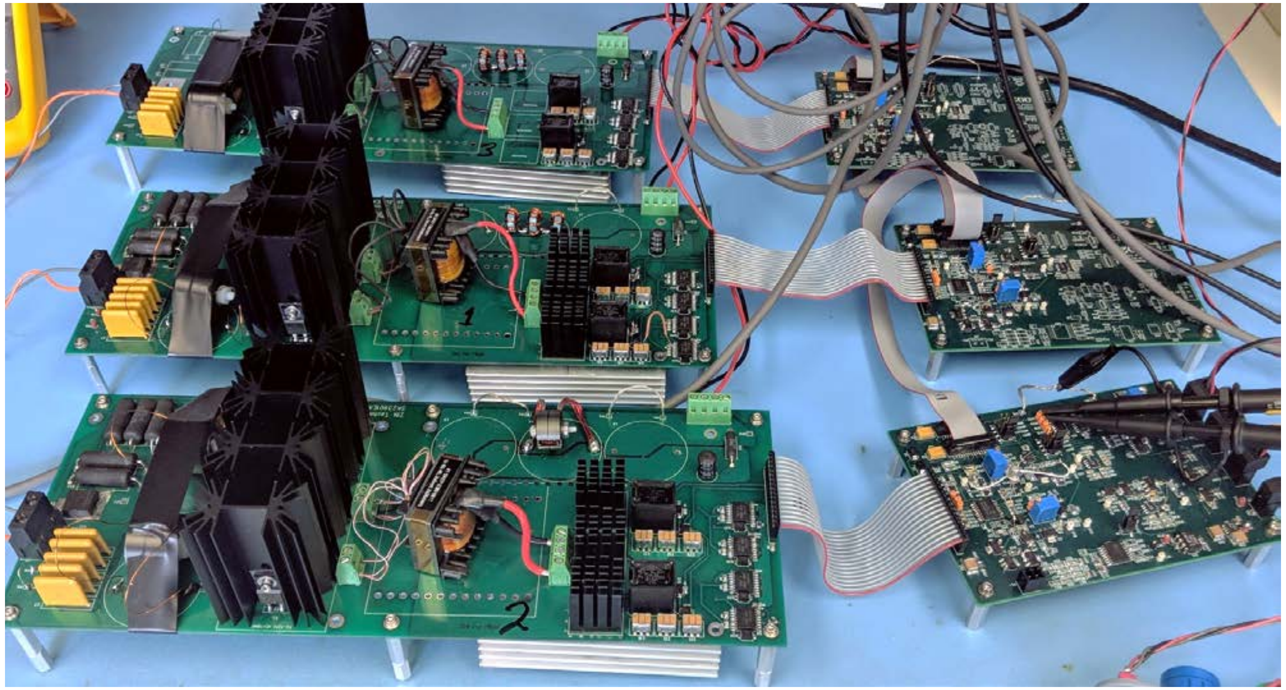
Preliminary Efficiency Est.:
- 93.0 % at 900 W

Preliminary Mass Est.:
- 4.9 kg for 900 W

Preliminary Volume Est.:
- ~ 5 L for 900 W

Assumptions:
- 50-kHz PWM full-bridge discharge converter
- 100-kHz PWM flyback auxiliary converters
- FPGA-based SCB
- EMI/EMC low-pass input filter
- COTS housekeeping converters

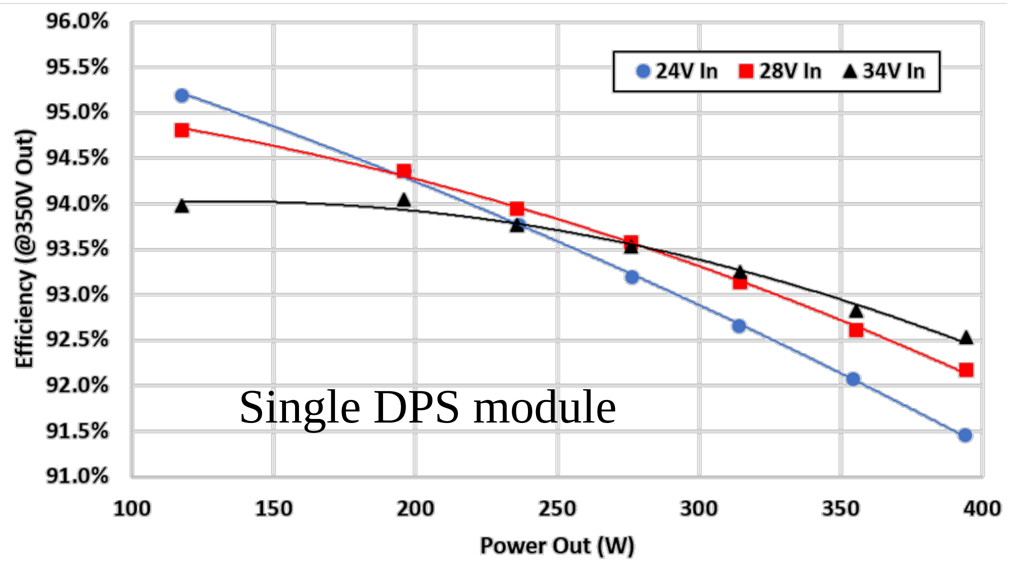
Three Module Breadboard Discharge Supply

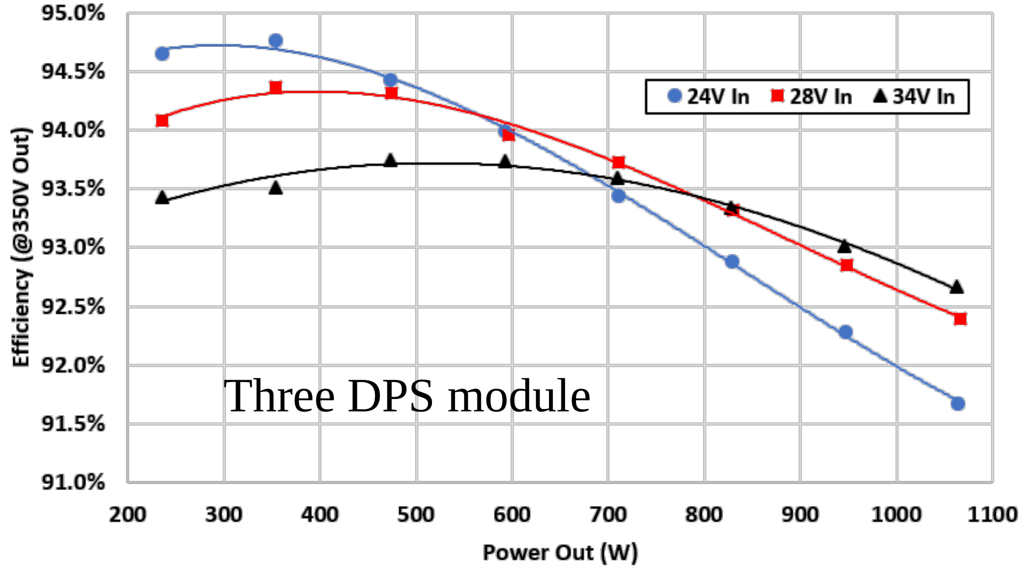


DPS Steady State Performance: Efficiency

- **Single Module Efficiency**
  - Typical single module with all improvements to date performed with 92-95% efficiency across full power and input bus voltage.

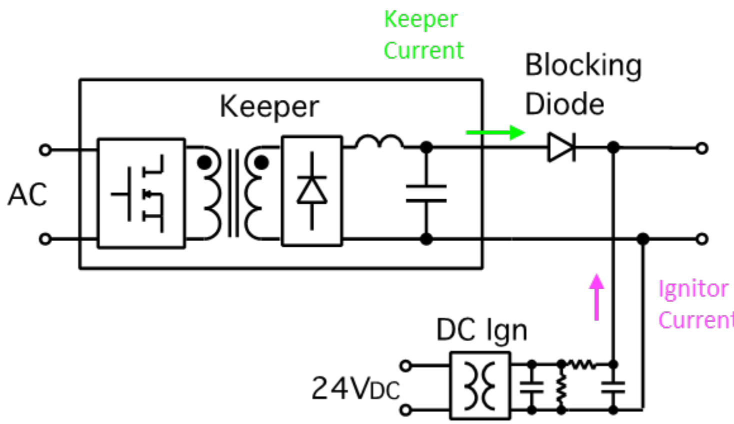
- **Three Module Efficiency**
  - Three modules together at higher power levels had similar performance to a single module.
  - Further improvement is anticipated after two remaining input filter inductors replaced with custom design.



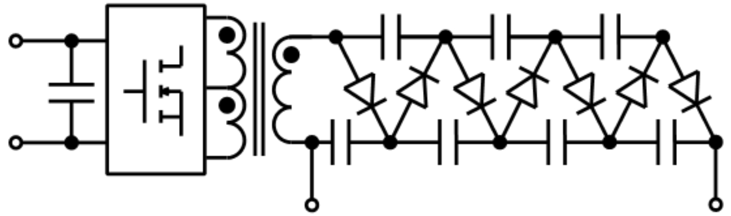


Auxiliary Supplies

- **Auxiliary supplies include the keeper, heater, and magnets**
  - The auxiliary supplies are anticipated to be nearly identical, although the cathode keeper will have modest differences.
- **A breadboard keeper supply has been designed and fabricated.**
  - Operating range has been demonstrated from ~300 mA to 1 Amp
  - Shown below is data for 1 Amp output, at both 15 and 30 ohm
  - Voltage on FET is acceptable, does not exceed derating
  - Circuit was tested with a cathode in GRC VF-8.
  - Test objectives were cathode ignition and steady-state operation.
  - Successfully demonstrated ignition and steady-state operation of an appropriately sized cathode.



Schematic of cathode keeper supply


Schematic of keeper voltage multiplier ignitor

Load [Ohms]	V_{in} [V]	Power [W]	I_{out} [A]	Efficiency
15.2	24	19.3	1.0	78.8%
15.2	29	19.1	1.0	79.6%
15.2	34	19.1	1.0	79.6%
30.1	24	37.5	1.0	80.3%
30.1	29	37.2	1.0	80.9%
30.1	34	37.0	1.0	81.4%

H64M-LM Thruster Performance Testing

Objectives

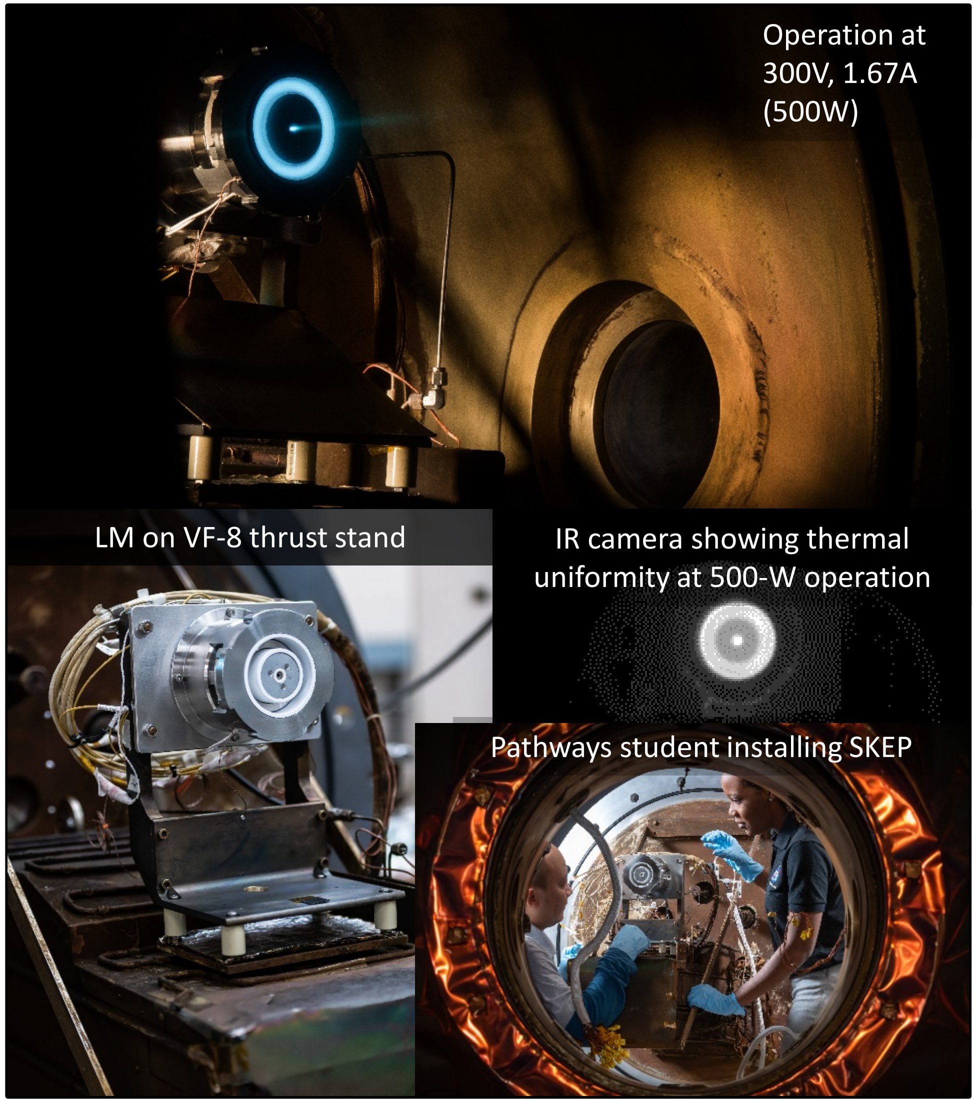
1. Investigate the performance of the H64M-LM thruster across its entire design throttle table 200-600W, I_d of 0.67 to 2A, V_d of 200-350 V.
2. Measure select thruster component temperatures and use measurements to fine-tune thermal model.
3. Perform a preliminary assessment of the thruster stability across the design magnetic field range.

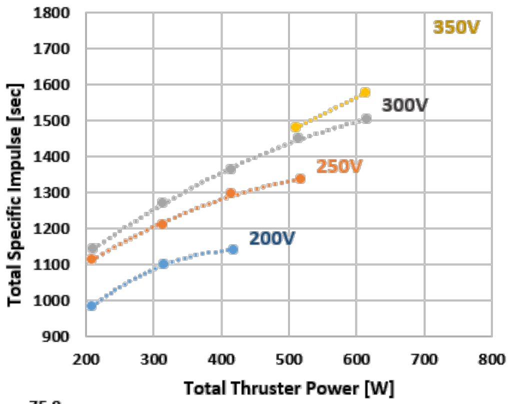
Summary conclusions

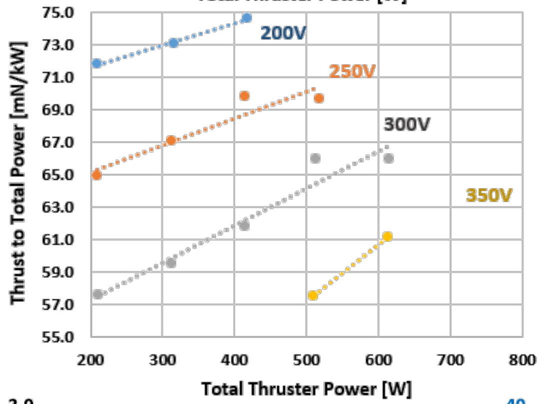
- ***Thruster performance exceeds SOA.***
- ***Thruster demonstrated ~ 450 hrs, including 110 hrs of uninterrupted operation @ 500W.***
- ***Integrated demo with breadboard DPS***

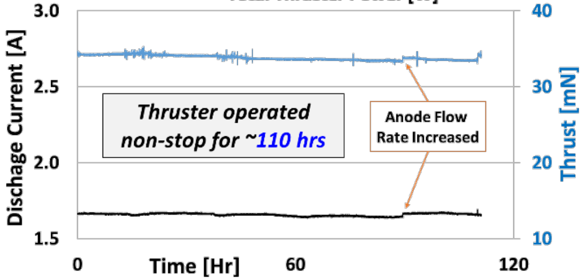
- Facility background pressure was 7 µTorr-Xe during thruster operation @ full power.
- Facility pressure conditions appropriate for thruster functional and performance assessment.
- Test setup enabled unattended duration tests of LM.

NASA GRC VF-8
Low-Power EP Test Chamber







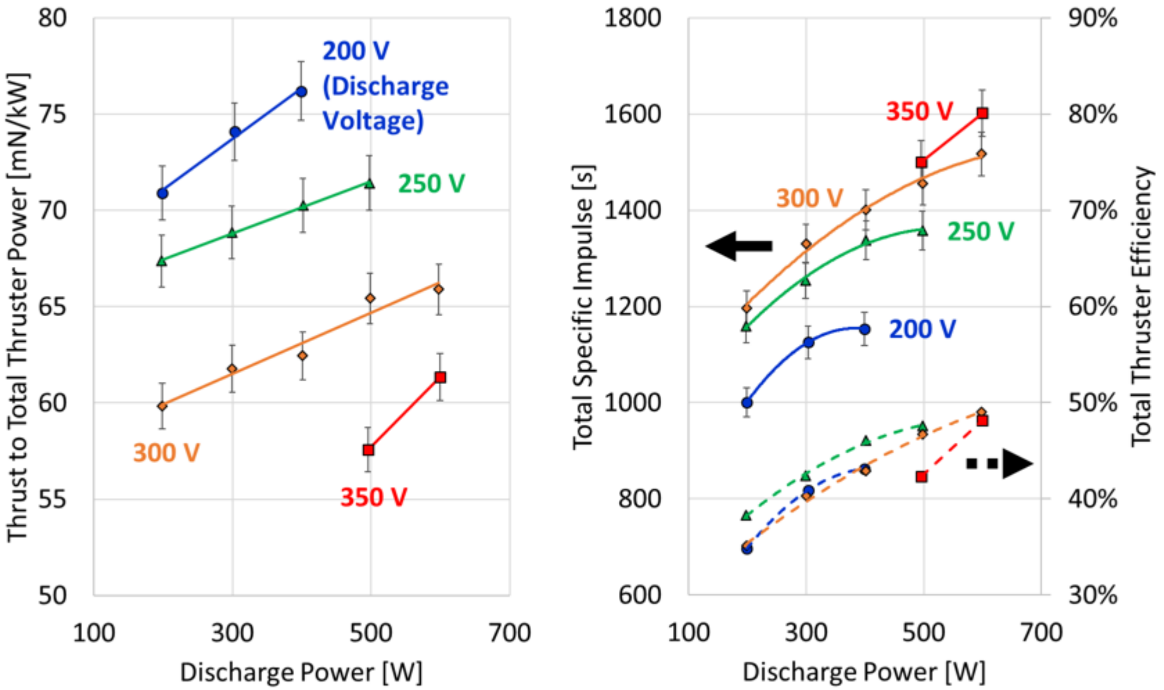


H64M-LM Test Phases

Phase	Thruster	Description	Duration [hr]
A	LM	Performance and stability characterization: 200 to 600-W discharge power, 200 to 350-V discharge voltage, magnetic and cathode flow fraction mapping	27
B	LM	Short-duration wear: 500-W, 300-V discharge condition	275
C	LM	Performance and stability characterization: 200 to 600-W discharge power, 200 to 350-V discharge voltage, magnetic and cathode flow fraction mapping	5
D	LM	Thermal characterization: 300 to 600-W discharge power, 300 and 350-V discharge voltage	18
E	LM	Accelerated wear characterization: 500-W, 300-V discharge condition	108
F	LMW	Performance and stability characterization: 300 to 700-W discharge power, 300 and 350-V discharge voltage, magnetic and cathode flow fraction mapping, integrated demonstration with breadboard discharge module	10
		Cumulative Thruster Operation	**443 hrs**

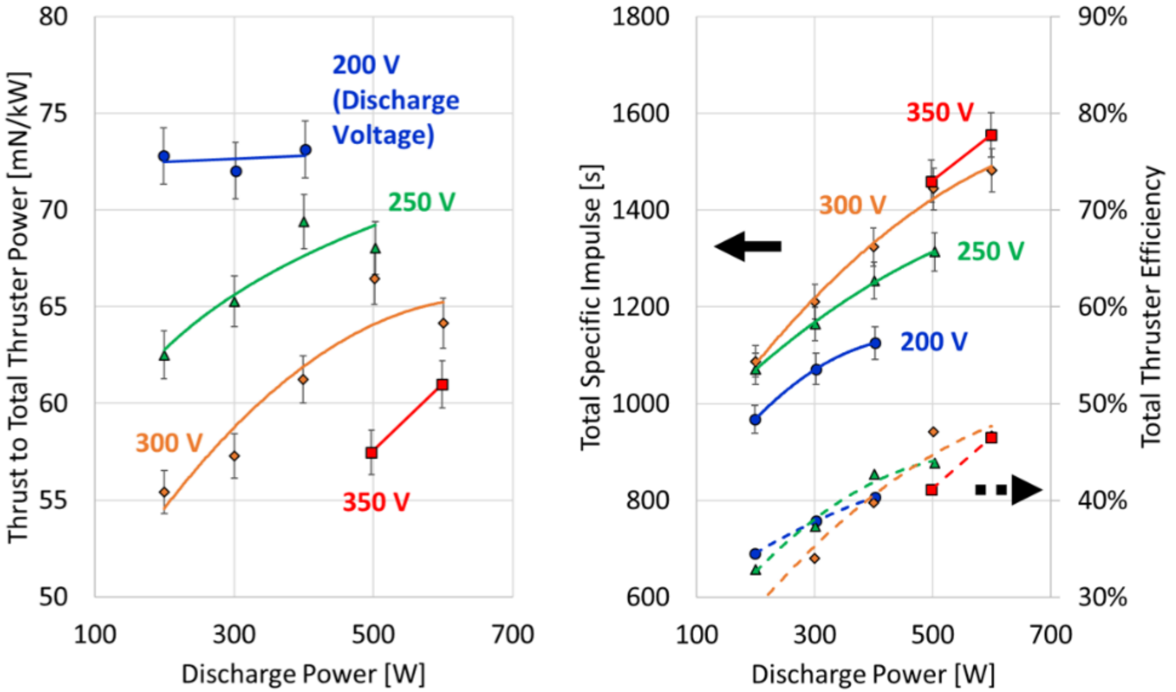
H64M-LM (BOL vs 300 Hours)

BOL



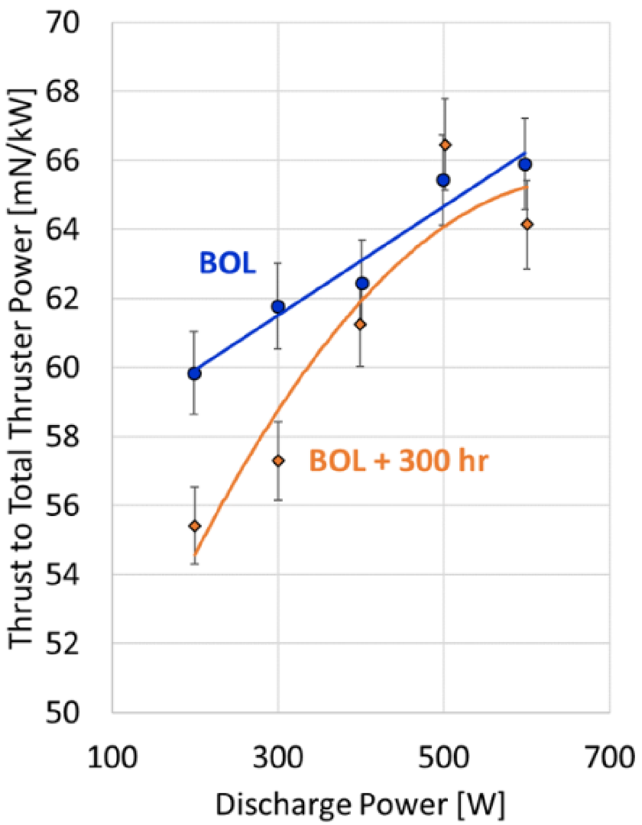
NASA-H64M-LM performance at beginning-of-life (BOL) with performance-optimized magnetic field strengths, 7% cathode flow fraction, and electrically floating configuration. The total thruster efficiency uncertainty bars (up to a maximum of ±5%) are not shown for clarity.

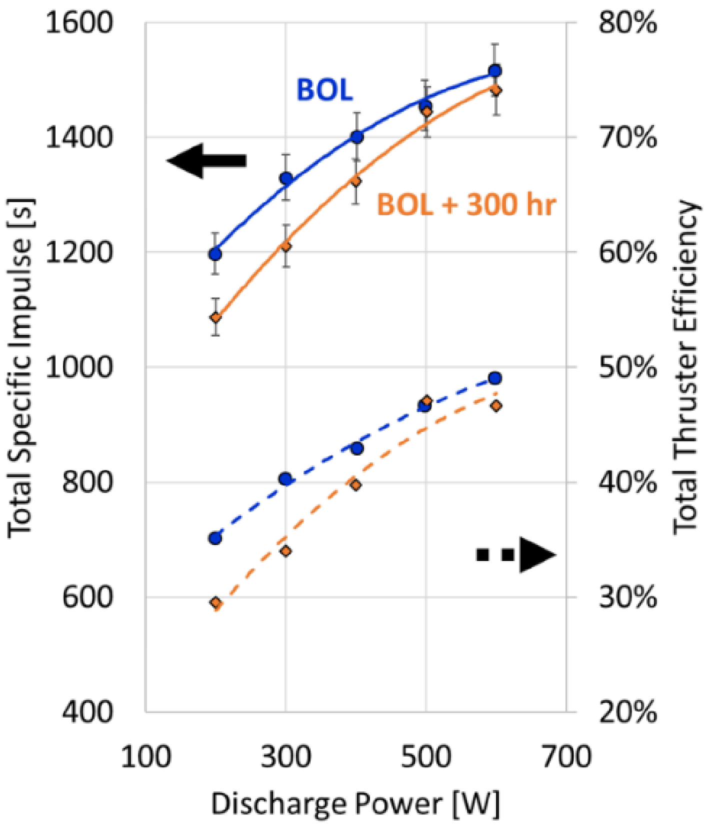
300 Hrs



NASA-H64M-LM performance at BOL + 300 hr with the same magnetic field settings as the results in Figure Left, 7% cathode flow fraction, and electrically floating configuration. The total thruster efficiency uncertainty bars (up to a maximum of ±5%) are not shown for clarity.

H64M-LM (BOL vs 300 Hours)





Comparison of NASA-H64M-LM performance at BOL versus BOL + 300 hr for 300-V discharge voltage, identical magnetic field settings, 7% cathode flow fraction, and electrically floating configuration. The total thruster efficiency uncertainty bars (up to a maximum of ±5%) are not shown for clarity.

Wider Channel Variant (H64M-LMW)

Thruster Configuration / Condition	Discharge Power [W]	Discharge Voltage [V]	Thrust [mN]	I_{sp} [s]	η_T
LM BOL	500	300	33	1455	47%
LM BOL	600	300	41	1517	49%
LM BOL	500	350	29	1500	42%
LM BOL	600	350	38	1601	48%
LM BOL + 300 hr	500	300	34	1444	47%
LM BOL + 300 hr	600	300	40	1483	47%
LM BOL + 300 hr	500	350	29	1460	41%
LM BOL + 300 hr	600	350	37	1555	47%
LMW BOL	600	300	42	1602	53%
LMW BOL	700	300	48	1608	53%
LMW BOL	600	350	38	1671	51%
LMW BOL	700	350	44	1702	52%

Discharge Voltage [V]	Thrust to Total Thruster Power	Total Specific Impulse	Total Thruster Efficiency
300	1.03X	1.06X	+4%
350	1.02X	1.04X	+3%

Concluding Remarks

- NASA is expanding the use of lower-cost SmallSats to accomplish science goals.
- Small spacecraft launched as secondary payloads can reduce the total mission cost of NASA robotic science missions (especially for constellations, swarms, or multiple spacecraft formation flying).
- No flight-ready sub-kW EP with sufficient propellant throughput to support small spacecraft beyond GEO.
- SKEP project was a low-risk, high-payoff technology development effort to rapidly fielded HT-SSEP.
- SKEP aimed to make available a comprehensive package of design/process documents to U.S. industry.
- SKEP project team enlisted the cooperation of U.S. industry to refine system requirements.
- NASA GRC engineering expertise, fabrication capabilities, and world-class test facilities in the field of electric propulsion facilitated the aforementioned work to be completed in one year in one design iteration.
- The thruster and PPU work leveraged numerous prior NASA investments and produced many innovations to solve the unique challenges of a miniaturized, long-life Hall-effect electric propulsion system.
- Pending continued funding, GRC aims to deliver an initial high total impulse capability for NASA small spacecraft science missions through collaborations with industry.
- The ultimate goal is to make the described thruster and PPU designs available domestically to credible electric propulsion developers on a non-exclusive no-cost basis.