Z10.04 Materials, Processes, and Technologies for Advancing In-Space Electric Propulsion Thrusters

Lead Center: GRC

Participating Center(s): JPL

Scope Title
High-Temperature, High-Voltage Electric Propulsion Harness Assembly

Scope Description

Electric propulsion (EP) for space applications has demonstrated tremendous benefit to a variety of NASA, military, and commercial missions. This subtopic seeks to address ongoing challenges with EP performance repeatability, hardware reliability, and total life-cycle cost. Critical NASA EP needs have been identified in the scope area detailed below. Proposals outside the described scope shall not be considered. Proposers are expected to show an understanding of the current state of the art (SOA) and quantitatively (not just qualitatively) describe anticipated improvements over relevant SOA materials, processes, and technologies that substantiate NASA investment.

In EP systems, power, commands, and telemetry are relayed between the power processing unit (PPU) and the thruster via dedicated electrical harness assemblies. These harnesses must support the voltage and current needs of the thruster, survive in-space conditions and the operational thermal environment, and not incur unacceptable line loss, radiated emissions, and mass and volume impacts to the spacecraft. Harnesses must also have sufficient flexibility and abrasion resistance, especially for thrusters that are integrated onto actuated gimbals. Individual EP technologies may have specific needs that must be addressed; for example, low-inductance harnesses are preferred in Hall-effect thrusters to reduce thruster discharge oscillations and to promote system stability.

Thermal management of EP systems is a persistent challenge and can be severe in both high-power (>10 kW) and high-power-density (e.g., compact subkilowatt) thrusters. This solicitation seeks advancements in cable and connector materials and designs to support harness assembly solutions addressing all of the following gridded ion and Hall-effect propulsion system needs:

- Voltages (after derating) up to 600-800 VDC (for Hall-effect thrusters) or up to 1.8-2.1 kVDC (for gridded ion thrusters).
- Operating temperatures of at least 350 °C, survival temperatures down to at least -60 °C, and the ability to survive at least 10,000 on-off thermal cycles.
- Direct currents (after derating) up to 10-15 A (for compact <1-kW systems) or up to 25-200 A (for >10-kW systems).
- Deratings consistent with NASA Technical Standard MSFC-STD-3012A (Appendix A) for connectors and wiring.
- Low outgassing materials consistent with the guideline (i.e., maximum total mass loss (TML) of 1% and maximum collected volatile condensable material (CVCM) deposition of 0.1%) in NASA Technical Standard
MSFC-SPEC-1443B.

- Features (e.g., venting of connectors and backshells) to mitigate Paschen or corona discharges due to materials or trapped volume outgassing at operating temperatures.
- Features to support harness shielding and grounding.
- Available lengths, flexibility (e.g., bend radius), and abrasion resistance comparable to or better than SOA.

**Expected TRL or TRL Range at completion of the Project**

3 to 5

**Primary Technology Taxonomy**

**Level 1**

TX 01 Propulsion Systems

**Level 2**

TX 01.2 Electric Space Propulsion

**Desired Deliverables of Phase I and Phase II**

- Analysis
- Prototype
- Hardware

**Desired Deliverables Description**

**Phase I:**

1. Final report containing test data characterizing key properties that address the critical gaps as well as the design and test plan for an EP harness assembly solution to be implemented in Phase II.
2. Material samples that can be used for independent verification of claimed improvements over SOA.

**Phase II:**

1. Final report containing test data verifying key functional and environmental requirements of the EP harness assembly design, including a functional demonstration in an operating thruster environment (in which partnering with EP developers may be necessary).
2. Prototype harness assembly that can be used for independent verification of claimed improvements over SOA.

**State of the Art and Critical Gaps**

Recent NASA EP harnesses have utilized stranded, plated copper wiring with multilayer, crosslinked fluoropolymer (e.g., polytetrafluoroethylene (PTFE) and ethylene tetrafluoroethylene (ETFE)) insulation consistent with MIL-W-22759/SAE Standard AS22759D. Commercial off-the-shelf (COTS) wiring rated to 600 VDC and 1,000 VDC exists but is limited to temperatures below ~260 °C. Meanwhile, COTS electrical connectors (such as MIL-SPEC circular connectors) typically have even lower temperature limits.

Temperature derating requirements for electrical connectors mating to SOA EP thrusters have been challenging for recent NASA missions and have complicated mechanical retention and strain relief at the interface. Custom connector solutions or extensive component testing to relax derating requirements are possible approaches, but they are unattractive as increased development costs would be incurred for each mission. Harness material and design improvements that increase the maximum allowable harness temperature would improve the thermal margin for derating purposes on SOA thrusters and facilitate the development of thrusters with higher powers or...
power densities relative to SOA.

SOA EP harnesses frequently employ custom insulation wraps on COTS wiring in order to support high thruster operating voltages. Such wraps can be mechanically fragile and complicate harness handling and installation. Harness material and design improvements that increase the voltage rating are desirable to improve system reliability and to reduce life-cycle costs.

**Relevance / Science Traceability**

Both NASA's Science Mission Directorate (SMD) and Human Exploration and Operations Mission Directorate (HEOMD) need spacecraft with demanding propulsive performance and greater flexibility for more ambitious missions requiring high duty cycles and extended operations under challenging environmental conditions. SMD spacecraft need the ability to rendezvous with, orbit, and conduct in situ exploration of planets, moons, and other small bodies (i.e., comets, asteroids, near-Earth objects, etc.) in the solar system; mission priorities are outlined in the decadal surveys for each of the SMD divisions ([https://science.nasa.gov/about-us/science-strategy/decadal-surveys](https://science.nasa.gov/about-us/science-strategy/decadal-surveys)). For HEOMD, higher power EP is a key element in supporting sustained crewed exploration of cislunar space and Mars.

This subtopic seeks innovations to meet future SMD and HEOMD propulsion requirements in EP systems related to such missions. The roadmap for such in-space propulsion technologies is covered under the 2020 NASA Technology Taxonomy ([https://www.nasa.gov/offices/oct/taxonomy/index.html](https://www.nasa.gov/offices/oct/taxonomy/index.html)), with supporting information archived in the 2015 NASA Technology Roadmap TA-2 ([https://www.nasa.gov/offices/oct/home/roadmaps/index.html](https://www.nasa.gov/offices/oct/home/roadmaps/index.html)).

**References**


**Scope Title**

Advanced Thermal Management for Hall-Effect Thrusters

**Scope Description**

Electric propulsion (EP) for space applications has demonstrated tremendous benefit to a variety of NASA, military, and commercial missions. This subtopic seeks to address ongoing challenges with EP performance repeatability, hardware reliability, and total life-cycle cost. Critical NASA EP needs have been identified in the scope area detailed below. Proposals outside the described scope shall not be considered. Proposers are expected to show an understanding of the current state of the art (SOA) and quantitatively (not just qualitatively) describe anticipated improvements over relevant SOA materials, processes, and technologies that substantiate NASA investment.

As Hall-effect thrusters are scaled up in power for next-generation missions with large payloads (including human
crews), thermal management poses a major design challenge. Compact subkilowatt thrusters for small spacecraft
also typically operate with high power density and face similar challenges. To protect critical components such as
electromagnets, technological advances are needed to improve the efficiency with which heat can be radiated or
conducted away from temperature-sensitive areas of the thruster.

NASA is soliciting proposals for high-emissivity coatings that are compatible with high thruster operating
temperatures (300 to 400 °C) and remain compliant with the material outgassing guideline (i.e., maximum total
mass loss (TML) of 1% and maximum collected volatile condensable material (CVCM) deposition of 0.1%) in NASA
Technical Standard MSFC-SPEC-1443B. Development of discharge channels and anodes made from intrinsically
high-emittance materials is also encouraged. Plasma-facing materials and coatings must be able to survive for
>20,000 hr of thruster operation while maintaining their thermal performance.

Other approaches of interest include novel radiator geometries that can either be easily attached to existing
thrusters or integrated into the design of existing thruster components. Heat pipes integrated into a standard Hall-
effect thruster design are also of interest. The solutions must be compatible with expected maximum local
temperature in a high-power thruster at the implementation location (e.g., 400 to 600 °C in the vicinity of the inner
magnet coil) as well as elevated saturation temperatures that do not produce excessive vessel pressure.

Novel radiator geometries, integral heat pipes, and/or channels for pumped fluid loops also open the design space
to additively manufactured implementation of these features. Hiperco® is a typical material that these features
could be additively manufactured from, but other magnetic materials may also be considered. Whatever solutions
are presented, a reduction of at least 50 to 100 °C in peak inner coil temperatures is desired.

Expected TRL or TRL Range at completion of the Project

3 to 5

Primary Technology Taxonomy

Level 1

TX 01 Propulsion Systems

Level 2

TX 01.2 Electric Space Propulsion

Desired Deliverables of Phase I and Phase II

- Analysis
- Prototype
- Hardware

Desired Deliverables Description

Phase I:

1. Final report containing data from small-scale or coupon testing of the proposed heat rejection technology
   and a design and test plan for scaling up the technology to a Hall-effect thruster in Phase II.
2. If applicable, material samples that can be used for independent verification of claimed improvements over
   the SOA (e.g., this would apply to surface coatings).

Phase II:
1. Final report containing test data verifying thermal performance of the novel or improved heat rejection technology, demonstrated in an operating Hall-effect thruster environment (in which partnering with EP developers may be necessary).
2. If applicable, hardware prototypes delivered to NASA in order to enable testing of the new technology on additional laboratory thrusters (e.g., this would apply to bolt-on radiators, coatings, etc.).

State of the Art and Critical Gaps

High-emissivity coatings (such as black oxide) have been tested on high-power Hall-effect thruster components, but adhesion over >1,000 thermal cycles and during extended thruster operation remains challenging. Coatings exist that can radiate away heat efficiently while still having low absorptivity of radiated power from the background environment (Conversano et al. 2019). Exterior-facing thruster surfaces may be constructed from carbon to facilitate radiative heat loss (Reilly et al. 2016). The dominant heat load in the thruster arises from plasma impacting the discharge channel and anode (Reilly et al. 2016), so improving the ability of these surfaces to radiate could have significant benefits. A smaller heat load is generated within the magnetic coils, but the thermal conductivities of the coil bobbins, ferromagnetic cores, and potting material are usually low, making the coils a problem area thermally. SOA thrusters are designed to maintain good thermal contact between internal components to maximize heat conduction from the interior to the exterior (Myers et al. 2016), but novel solutions such as heat pipes could dramatically improve heat transport efficiency. Radiators extending from the thruster body have been used for heat rejection in recent NASA Hall thruster designs (Myers et al. 2016, Conversano et al. 2019).

Relevance / Science Traceability

Both NASA's Science Mission Directorate (SMD) and Human Exploration and Operations Mission Directorate (HEOMD) need spacecraft with demanding propulsive performance and greater flexibility for more ambitious missions requiring high duty cycles and extended operations under challenging environmental conditions. SMD spacecraft need the ability to rendezvous with, orbit, and conduct in situ exploration of planets, moons, and other small bodies (i.e., comets, asteroids, near-Earth objects, etc.) in the solar system; mission priorities are outlined in the decadal surveys for each of the SMD divisions (https://science.nasa.gov/about-us/science-strategy/decadal-surveys). For HEOMD, higher-power EP is a key element in supporting sustained crewed exploration of cislunar space and Mars.

This subtopic seeks innovations to meet future SMD and HEOMD propulsion requirements in EP systems related to such missions. The roadmap for such in-space propulsion technologies is covered under the 2020 NASA Technology Taxonomy (https://www.nasa.gov/offices/oct/taxonomy/index.html), with supporting information archived in the 2015 NASA Technology Roadmap TA-2 (https://www.nasa.gov/offices/oct/home/roadmaps/index.html).

References

Cost-Effective Carbon-Based Electrodes for High-Power, High-Performance Gridded Ion Thrusters

Scope Description

Electric propulsion (EP) for space applications has demonstrated tremendous benefit to a variety of NASA, military, and commercial missions. This subtopic seeks to address ongoing challenges with EP performance repeatability, hardware reliability, and total life-cycle cost. Critical NASA EP needs have been identified in the scope area detailed below. Proposals outside the described scope shall not be considered. Proposers are expected to show an understanding of the current state of the art (SOA) and quantitatively (not just qualitatively) describe anticipated improvements over relevant SOA materials, processes, and technologies that substantiate NASA investment.

Gridded ion thruster technology offers high efficiency, high specific-impulse capabilities, and has been used successfully to support NASA science missions as well as commercial Earth-orbiting applications. The primary life limiter for these devices is typically erosion of the accelerator electrode due to bombardment by charge-exchange ions. While NASA gridded ion thrusters have achieved the necessary lifetimes in the past by operating at derated current densities, there is interest in operation at higher thrust and power densities that would increase mission capture and allow for more compact thruster designs. Higher power and current densities result in increased erosion rates of the accelerator electrode, such that the refractory metals used on previous designs may no longer be sufficient to meet demanding lifetime requirements.

Carbon-based electrodes have shown promise by offering significantly higher erosion resistance compared to refractory metals. Innovative solutions are desired that would result in manufacturing processes for carbon-based electrodes that are cost-effective relative to prior efforts, making them competitive with SOA electrode manufacturing using refractory metals. These solutions must be capable of producing carbon-based electrodes with the following geometries, operating voltages, and thermal properties:

- Screen and accelerator electrode thicknesses of ~0.33 mm and ~0.50 to 0.75 mm, respectively.
- Screen and accelerator electrode open area fractions of ~70% and ~25%, respectively.
- Screen and accelerator aperture diameters of ~2 mm and ~1.25 mm, respectively.
- Gap between the screen and accelerator electrode of ~0.50 to 0.75 mm.
- A shallow spherical dome (i.e., dished) geometry for both screen and accelerator electrodes.

Note: Dome and flat geometries are both of interest to NASA. However, a dome geometry ensures sufficient electrode stiffness and first-mode natural frequency to withstand expected structural loading during launch as well as maintaining required electrode gaps and avoiding buckling due to compressive stresses caused by nonuniform temperature distributions along electrodes. Manufacturing solutions capable of producing only flat electrodes will also be considered but must demonstrate that structural loading during launch and potential buckling during operation will not be issues.

- Extensibility to beam extraction (i.e., perforated) diameters of 40 cm or larger.
- Tight tolerances on apertures' locations (<0.1 mm) to facilitate proper alignment of apertures between screen and accelerator electrodes.
- Minimum voltage standoff capability between screen and accelerator electrodes of 2 kV.
- Peak operating temperatures of 450 °C.
- Coefficients of thermal expansion less than or equal to that of molybdenum (4.8x10-6 K⁻¹).

Proposals are desired that offer solutions which are applicable for manufacturing of carbon-based screen and accelerator electrodes. However, proposals that focus only on carbon-based accelerator electrodes will be considered if such solutions are shown to be compatible with screen electrodes made with heritage refractory metals.
Expected TRL or TRL Range at completion of the Project

3 to 5

Primary Technology Taxonomy

Level 1

TX 01 Propulsion Systems

Level 2

TX 01.2 Electric Space Propulsion

Desired Deliverables of Phase I and Phase II

- Analysis
- Prototype
- Hardware

Desired Deliverables Description

Phase I:

1. A final report detailing the material properties and the manufacturing processes for the carbon-based electrodes, as well as an evaluation of the extensibility of the processes to sizes of interest (i.e., 40-cm perforated diameter or larger).
2. A scaled-down sample of each carbon-based electrode (either screen and accelerator or accelerator only, depending on the approach) representative of typical electrode thickness and open area fraction to be delivered to NASA for independent assessment and tests.

Phase II:

1. A final report detailing final manufacturing processes and an updated evaluation of the extensibility of these processes to sizes of interest (i.e., 40-cm perforated diameter or larger).
2. Carbon-based screen and accelerator electrodes (or accelerator electrode only, depending on the approach) at least 30 cm in diameter that can be hot-fire tested with a gridded ion thruster (in which partnering with EP developers may be necessary).

State of the Art and Critical Gaps

While extensive research and development of carbon-based electrodes have resulted in solutions that were technically adequate, the complexity and associated costs of manufacturing have been prohibitive toward widespread adoption into ion thruster technology. The material used for electrodes has historically been refractory metals, whose thermal and mechanical properties allow the electrodes to withstand the temperatures and launch loads they will experience while offering adequate erosion resistance. Fabrication using refractory metals such as molybdenum typically involves chemical etching to produce the apertures within the electrodes. Carbon-based solutions have been developed previously by several organizations and include carbon-carbon, amorphous graphite, and pyrolytic graphite (PG). Fabrication techniques for carbon-based electrodes have been rather varied and complex and have included methods such as chemical vapor deposition and carbonization. Apertures in carbon-based electrodes have been created using laser drilling, electric discharge machining (EDM), or machining. As such, innovative solutions are desired that would result in manufacturing processes for carbon-based electrodes that are less complex and/or more cost-effective than prior efforts.
Relevance / Science Traceability

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References


Scope Title

High-Power Electric Propulsion Thrusters for Mars-Class Missions

Scope Description

Electric propulsion (EP) for space applications has demonstrated tremendous benefit to a variety of NASA, military, and commercial missions. This subtopic seeks to address ongoing challenges with EP performance repeatability, hardware reliability, total life-cycle cost, and future needs.

Critical NASA EP needs have been identified in the scope area detailed below. Proposals outside the described...
scope shall not be considered. Proposers are expected to show an understanding of the current state of the art (SOA) and quantitatively (not just qualitatively) describe anticipated improvements over relevant SOA materials, processes, and technologies that substantiate NASA investment.

Megawatt-class EP has been identified as a key NASA need for enabling sustained human Mars exploration missions. This solicitation seeks solutions that would advance the technical maturation of high-power EP thrusters. NASA is interested in thruster technologies that meet both of the following requirements:

1. Expected operability at greater than or equal to 100 kW of electrical power, with scalability or clustering approaches capable of supporting greater than or equal to 1 MW of electrical power.
2. Present technology readiness level (TRL) of greater than or equal to 4 at the thruster level per NASA NPR 7123.1C Appendix E, in which TRL 4 is defined in this solicitation’s context as a low-fidelity laboratory thruster with test performance demonstrating agreement with analytical predictions for a relevant environment.

To remain within the scope of SBIR awards, proposals addressing component-level or subcomponent-level innovations are desired with a clearly defined path toward thruster-level integration and ground demonstration. Proposals shall address the following:

- Justified compliance with the thruster-level power and TRL requirements listed above.
- Key performance parameters, both SOA and anticipated, relative to the baseline metrics in Table 1.3 of the National Academies’ "Space Nuclear Propulsion for Human Mars Exploration" 2021 report.
- Critical technical challenges identified to date associated with maturing the thruster technology (including interfacing with other elements of a complete EP subsystem) and how the proposed solution addresses one or more of the critical challenges.
- Anticipated compliance with the desired SBIR deliverables (Technological Details section, below).
- Anticipated compliance with the expected TRL range at completion of the project (Technological Details section, below).

Note: The expected TRL range at completion of the project addresses the TRL of the proposed component-level or subcomponent-level innovations. When integrated and demonstrated with a thruster during Phase II, the proposed innovations must support a thruster-level TRL greater than or equal to 4.

**Expected TRL or TRL Range at completion of the Project**

3 to 4

**Primary Technology Taxonomy**

**Level 1**

TX 01 Propulsion Systems

**Level 2**

TX 01.2 Electric Space Propulsion

Desired Deliverables of Phase I and Phase II

- Analysis
- Prototype
- Hardware

**Desired Deliverables Description**
Phase I

1. Final report containing:
   - Design and test plan (to be implemented in Phase II) for thruster-level integration and demonstration of the proposed innovation.
   - Data from proof-of-concept or breadboard testing of the proposed innovation, along with comparisons to SOA and predicted performance.

Phase II

1. Final report containing test data verifying the performance of the proposed innovation, including a functional demonstration in an operating thruster environment.

State of the Art and Critical Gaps

Chapter 3 of the National Academies’ “Space Nuclear Propulsion for Human Mars Exploration” 2021 report provides an overview of SOA technologies and critical gaps for high-power EP thrusters.

Relevance / Science Traceability

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References