NASA SBIR 2009 Phase I Solicitation

X9 Thermal Protection Systems

The Thermal Protection System (TPS) protects a spacecraft from the severe heating encountered during hypersonic flight through a planetary atmosphere. In general, there are two classes of TPS: reusable and ablative. Typically, reusable TPS applications are limited to relatively mild entry environments like that of Space Shuttle. No change in the mass or properties of the TPS material results from entry with a significant amount of energy being re-radiated from the heated surface and the remainder conducted into the TPS material. Typically, a surface coating with high emissivity (to maximize the amount of energy re-radiated) and with low surface catalycity (to minimize convective heating by suppressing surface recombination of dissociated boundary layer species) is employed. The primary insulation has low thermal conductivity to minimize the mass of material required to insulate the primary structure. Ablative TPS materials, in contrast, accommodate high heating rates and heat loads through phase change and mass loss. All NASA planetary entry probes to date have used ablative TPS. Most ablative TPS materials are reinforced composites employing organic resins as binders. When heated, the resin pyrolyzes producing gaseous products that are heated as they percolate toward the surface thus transferring some energy from the solid to the gas. Additionally, the injection of the pyrolysis gases into the boundary layer alters the boundary layer properties resulting in reduced convective heating. However, the gases may undergo chemical reactions with the boundary layer gases that could return heat to the surface. Furthermore, chemical reactions between the surface material and boundary layer species can result in consumption of the surface material leading to surface recession. Those reactions can be endothermic (vaporization, sublimation) or exothermic (oxidation) and will have an important impact on net energy to the surface. Clearly, in comparison to reusable TPS materials, the interaction of ablative TPS materials with the surrounding gas environment is much more complex as there are many more mechanisms to accommodate the entry heating.

NASA has successfully tackled the complexity of thermal protection systems for numerous missions to inner and outer planets in our solar system in the past; the knowledge gained has been invaluable but incomplete. Future missions will be more demanding. Better performing ablative TPS than currently available is needed to satisfy requirements of the most severe CEV missions, e.g., Mars Landing with 8 km/s entry and Mars Sample Return with 12-15 km/s Earth entry. Beyond the improvement needed in ablative TPS materials, more demanding future missions such as large payload missions to Mars will require novel entry system designs that consider different vehicle shapes, deployable or inflatable configurations and integrated approaches of TPS materials with the entry system sub-structure.

Subtopics

X9.01 Ablative Thermal Protection Systems

Lead Center: ARC

Participating Center(s): GRC, JPL, JSC, LaRC

The technologies described below support the goal of developing higher performance ablative TPS materials for higher performance CEV as well as future Exploration missions.

- Developments are sought for ablative TPS materials and heat shield systems that exhibit maximum robustness, reliability and survivability while maintaining minimum mass requirements, and capable of enduring severe combined convective and radiative heating, including: development of acreage materials, adhesives, joints, penetrations, and seals. Two classes of materials will be required.
    - One class of materials, for Mars aerocapture and entry, will need to survive heat fluxes of 200-400 W/cm^2 (primarily convective) and integrated heat loads of up to 25 kJ/cm^2. These materials or material systems must improve on the current state-of-the-art recession rates of 0.25 mm/s at heating rates of 200 W/cm^2 and pressures of 0.3 atm and improve on the state-of-the-art areal mass of 1.0 g/cm^2 required to maintain a bondline temperature below 250Â°C.
    - The second class of materials, for Mars return, will need to survive heat fluxes of 1500-2500 W/cm^2, with radiation contributing up to 75% of that flux, and integrated heat loads from 75-150 kJ/cm^2. These materials or material systems must improve on the current state-of-the-art recession rates of 1.00 mm/s at heating rates of 2000 W/cm^2 and pressures of 0.3 atm and improve on the state-of-the-art areal mass of 4.0 g/cm^2 required to maintain a bondline temperature below 250Â°C.
- In-situ heat flux sensors and surface recession diagnostics tools are needed for flight systems to provide better traceability from the modeling and design tools to actual performance. The resultant data will lead to higher fidelity design tools, risk reduction, decreased heat shield mass and increases in direct payload. The heat flux sensors should be accurate within 20%, surface recession diagnostic sensors should be accurate within 10%, and any temperature sensors should be accurate within 5% of actual values.
- Non Destructive Evaluation (NDE) tools are sought to verify design requirements are met during manufacturing and assembly of the heat shield, e.g. verifying that anisotropic materials have been installed in their proper orientation, that the bondline as well as the TPS materials have the proper integrity and are free of voids or defects. Void and/or defect detection requirements will depend upon the materials being inspected. Typical internal void detection requirements are on the order of 6-mm, and bondline defect detection requirements are on the order of 25.4-mm by 25.4-mm times the thickness of the adhesive.
- Advances are sought in ablation modeling, including radiation, convection, gas surface interactions, pyrolysis, coking, and charring. There is a specific need for improved models for low and mid density as well as multi-layered charring ablators (with different chemical composition in each layer). Consideration of the non-equilibrium states of the pyrolysis gases and the surface thermochemistry, as well as the potential to couple the resulting models to a computational fluid dynamics solver, should be included in the modeling efforts.

Technology Readiness Levels (TRL) of 4 or higher are sought.

Research should be conducted to demonstrate technical feasibility during Phase 1 and show a path toward a Phase 2 hardware demonstration, and when possible, deliver a demonstration unit for functional and environmental testing at the completion of the Phase 2 contract.

This subtopic is also a subtopic for the "Low-Cost and Reliable Access to Space (LCRATS)" topic. Â Proposals to this subtopic may gain additional consideration to the extent that they effectively address the LCRATS topic (See topic O5 under the Space Operations Mission Directorate).

X9.02 Advanced Integrated Hypersonic Entry Systems

Lead Center: ARC

Participating Center(s): GRC, JPL, JSC, LaRC

The technologies below support the goal of developing advanced integrated hypersonic entry systems that meet the longer term goals of realizing larger payload masses for future Exploration missions.

- Advanced integrated thermal protection systems are sought that address: (1) thermal performance efficiency (i.e., ablation vs. conduction), (2) in-depth thermal insulation performance (i.e., material thermal conductivity and heat capacity vs. areal density), (3) systems thermal-structural performance, and (4) system integration and integrity. Such integrated systems would not necessarily separate the ablative TPS material system from the underlying sub-structure, as is the case for most current NASA heat shield solutions. Instead, such integrated solutions may show benefits of technologies such as hot structures and/or multi-layer systems to improve the overall robustness of the integrated heat shield while reducing its overall mass. The primary performance metrics for concepts in this class are increased reliability, reduced areal mass, and/or reduced life cycle costs over the current state of the art.

- Advanced multi-purpose TPS solutions are sought that not only serve to protect the entry vehicle during primary planetary entry, but also show significant added benefits to protect from other natural or induced environments including: MMOD, solar radiation, cosmic radiation, passive thermal insulation, dual pulse heating (e.g., aero capture followed by entry). Such multi-purpose materials or systems must show significant additional secondary benefits relative to current TPS materials and systems while maintaining the primary thermal protection efficiencies of current materials/systems. The primary performance metrics for concepts in this class are reduced areal mass for the combined functions over the current state of the art.

- Integrated entry vehicle conceptual development is sought that allow for very high mass (> 20 mT) payloads for Earth and Mars entry applications. Such concepts will require an integrated solution approach that considers: TPS, structures, aerodynamic performance (e.g., L/D), controllability, deployment, packaging efficiency, system robustness / reliability, and practical constraints (e.g. launch shroud limits, ballistic coefficients, EDL sequence requirements, mass efficiency). Such novel system designs may include slender or winged bodies, deployable or inflatable entry systems as well as dual use strategies (e.g., combined launch shroud and entry vehicle). New concepts are enabling for this class of vehicle. Key performance metrics for the overall design are system mass, reliability, complexity, and life cycle cost.

- Advances in Multidisciplinary Design Optimization (MDO) are sought specifically in application to address combined aerothermal environments, material response, vehicle thermal-structural performance, vehicle shape, vehicle size, aerodynamic stability, mass, vehicle entry trajectory / GN&C, and cross-range, characterizing the entry vehicle design problem.

Technology Readiness Levels (TRL) of 4 or higher are sought.

Research should be conducted to demonstrate technical feasibility during Phase 1 and show a path toward a Phase 2 hardware demonstration, and when possible, deliver a demonstration unit for functional and environmental testing at the completion of the Phase 2 contract.

This subtopic is also a subtopic for the "Low-Cost and Reliable Access to Space (LCRATS)" topic. Â Proposals to this subtopic may gain additional consideration to the extent that they effectively address the LCRATS topic (See topic O5 under the Space Operations Mission Directorate).