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. For example, if existing materials were to be used for the proposed Mars Sample Return mission, the TPS mass fraction would be on the order of 40%. The potential savings that could be achieved with some investment in TPS technology development is sizeable.
to protect the crew and cargo from entry heating at entry velocities of approximately 8 km/s for Space Station missions, 11 km/s for lunar return missions, up to 8 km/s for Martian aerocapture and entry, and between 12 - 15 km/s for Martian return missions. Ablative TPS is an enabling technology for all CEV superorbital reentry missions.

**Ablation Modeling**

The heat shield for CEV will employ a TPS material that pyrolyzes and ablates at high temperature for mass-efficient rejection of the aerothermal heat load. Pyrolysis is an internal decomposition of the solid that releases gaseous species, whereas ablation is a combination of processes that consume heat shield surface material (including chemical reactions, melting, and vaporization). For the design and sizing of TPS materials, it is imperative to have reliable simulation tools that can compute surface recession rate, in-depth pyrolysis, and internal temperature histories under general heating conditions. In addition, lunar and Martian reentry environment heating will consist of significant radiation from the shock layer. The models need to include the effect of not only convective but radiative heating as well.

Therefore, advances are sought in modeling of radiation, gas surface interactions, ablation mechanisms, pyrolysis, and other processes such as coking and charring. Specifically for charring, advances are sought in the development of a low density charring ablator model to give insight into how conductivity changes as function of temperature and pressure for the virgin material and for the material as it pyrolyzes.

**Instrumentation**

TPS sensors and experimental diagnostic tools are required to provide traceability of TPS sizing tools, design, and material performance. Traceability will lead to higher fidelity design tools, which in turn will lead to risk reduction and decreased heat shield mass on missions requiring atmospheric aerocapture or entry/reentry. Decreasing heat shield mass will enable certain missions that are not otherwise feasible and directly increase payload. Heat flux sensors and surface recession diagnostic tools are essential to advancing the state of TPS traceability for material modeling and aerothermal simulation.

Advances in the understanding of how heat flux sensor performance changes upon integration of the sensors into TPS materials in ablative environments through simulation or experimental investigation are sought. Specifically, the following list of sensor materials is of primary interest:

- Type K, C, R, and S thermocouples
- Sapphire windows
- Inconel superalloys
- Pure platinum
- Teflon

For surface recession, advances in optical methods (photometrics/tomography) are sought.
Non-destructive Testing Techniques and Novel Techniques for Material Characterization

The CEV heat shield will be the largest ever built. During manufacturing and integration, it will be necessary to understand the variability in material properties, to determine voids and inclusions, to assess bondline integrity, and to ensure that the established flight heat shield requirements are met.

For this purpose, advances in NDE and proposals of novel techniques for material characterization applicable for ablative TPS are sought.

Ablation Materials Development

Early NASA missions employed new ablative TPS materials that were tailored to each specific entry environment. However, after Mars Viking, NASA-sponsored ablative TPS development essentially ceased as the research focus shifted to reusable TPS in support of the Space Shuttle. For example, the Pioneer Venus (1978) and Galileo (1995) missions employed carbon phenolic TPS material that had previously been developed by the United States Air Force for ballistic missile applications. Over the past 40 years, NASA has adopted a risk averse philosophy relative to TPS, i.e. use what was used before since it has been flight-qualified. For Mars Direct Return, the entry velocities will be in the range of 12-15 km/s. Heritage carbon phenolic can satisfy Mars Return requirements however the TPS mass fraction would be less than optimal. Thus, advances toward new reliable and efficient TPS materials are desired. Similarly, development of adhesives, joints, penetrations, and seals are of equal importance and advances are sought.