Thermal Protection System Technology and Facility Needs for Demanding Future Planetary Missions

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Abstract

NASA has successfully launched numerous science missions to inner and outer planets in our solar system of which the most challenging were to Venus and Jupiter and the knowledge gained from those missions have been invaluable yet incomplete. Future missions will be built on what we have learned from the past missions but they will be more demanding from both the science as well as the mission design and engineering perspectives. The Solar System Exploration Decadal Survey (SSEDS) produced for NASA by the National Research Council identified a broad range of science objectives many of which can only be satisfied with atmospheric entry probes. The SSEDS recommended new probe/lander missions to both Venus and Jupiter.

The Pioneer-Venus probe mission was launched in August 1978 and four probes successfully entered the Venusian atmosphere in December 1978. The Galileo mission was launched in October 1989 and one probe successfully entered the Jovian atmosphere in December 1995. The thermal protection system requirements for these two missions were unlike any other planetary probes and required fully dense carbon phenolic for the forebody heat shield. Developing thermal protection systems to accomplish future missions outlined in the Decadal Survey presents a technology challenge since they will be more demanding than these past missions. Unlike Galileo, carbon phenolic may not be an adequate TPS for a future Jupiter multiprobe mission since non-equatorial probes will enter at significantly higher velocity than the Galileo equatorial probe and the entry heating scales approximately with the cube of the entry velocity. At such heating rates the TPS mass fraction for a carbon phenolic heat shield would be prohibitive. A new, robust and efficient TPS is required for such probes. The Giant Planet Facility (GPF), developed and employed during the development of the TPS for the Galileo probe was dismantled after completion of the program. Furthermore, flight data from the Galileo probe suggested that the complex physics associated with the interaction between massive ablation and a severe shock layer radiation environment is not well understood or modeled. The lack of adequate ground test facilities to support the development and qualification of new TPS materials adds additional complexities.

The requirements for materials development, ground testing and sophisticated modeling to enable these challenging missions are the focus of this paper.

What is TPS?

The Thermal Protection System (TPS) protects (insulates) a body from the severe heating encountered during hypersonic flight through a planetary atmosphere. Since TPS is a single point-of-failure subsystem, it is critical and it's performance needs to be validated through ground test and analysis.

In general, there are two classes of TPS:
Reusable TPS, where after exposure to the entry environment there are no changes in the mass or properties of the TPS materials. Typically, reusable TPS applications are limited to *relatively* mild entry environments (e.g., Shuttle). The characteristics of a reusable TPS are shown in Figure 1 where it is seen that radiative and convective heating results in a significant amount of energy being re-radiated from the heated surface with the remainder conducted into the TPS material. It is advantageous if the (often used) surface coating has high emissivity (to maximize the amount of energy re-radiated) and low surface catalycity (to minimize convective heating by suppressing recombination of dissociated boundary layer species at the heated surface). It is also advantageous if the primary (often inorganic) insulation has low thermal conductivity since that will minimize the mass of material required to insulate the primary structure (backup material).

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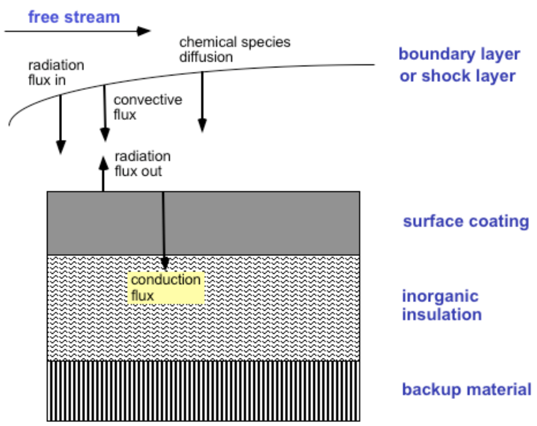

Fig. 1 Energy accommodation of reusable TPS materials

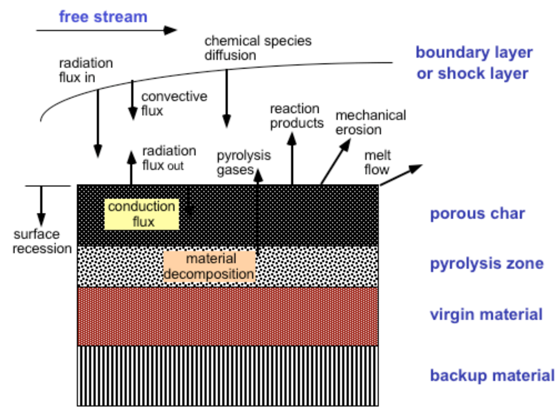

Fig. 2 Energy accommodation mechanisms of ablative TPS materials

Ablative TPS materials, in contrast, accommodate high heating rates and heat loads through phase change and mass loss. Ablative materials have been the classical approach to TPS used for over 40 years in a broad range of applications. All NASA planetary entry probes (to date) have used ablative TPS. The characteristics of ablative TPS materials are illustrated in Figure 2. Most ablative TPS materials are reinforced composites employing organic resins as binders. When heated, the resin pyrolyzes producing gaseous products (mostly hydrocarbons) that percolate toward the heated surface and are injected into the boundary layer. Resin pyrolysis also produces a carbonaceous residue that deposits on the reinforcement. The resulting surface material is termed "char." The pyrolysis process is typically endothermic and the pyrolysis gases are heated as they percolate toward the surface thus transferring some energy from the solid to the gas. The injection of the pyrolysis gases into the boundary layer alters the boundary layer properties, typically resulting in a reduction in convective heating. However, the gases may undergo chemical reactions with the boundary layer gases that will have an effect on the net heating 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 environmental gases is much more complex as there are many more mechanisms to accommodate the entry heating.

Ablative TPS – a short history

Early NASA missions (Gemini, Apollo, Mars Viking) employed new ablative TPS materials that were tailored for the 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. As an example, the Pioneer Venus and Galileo missions employed fully dense carbon phenolic that was developed by the United States Air Force for ballistic missile applications. Over the past 30 years NASA adopted a "risk averse" philosophy relative to TPS, i.e., use what was used before, even if it isn't optimal, since it has been flight-qualified. An unintended consequence was that the ablative TPS community in the United States slowly disappeared.

The Stardust and Genesis missions were exceptions in that employed new ablative TPS simply because those missions could not be accomplished with existing, flight-proven TPS materials.

To illustrate, Figure 3 shows a chronology of NASA entry missions that have employed ablative TPS. As seen, in over 40 years, NASA entry probes have only employed a few ablative TPS materials. The red symbols indicate materials still available. The black symbols indicate materials no longer manufactured, and the blue symbols indicate materials that may have to be re-qualified due to the unavailability of heritage precursor materials. It should be apparent that half of these materials are (or are about to be) no longer available.

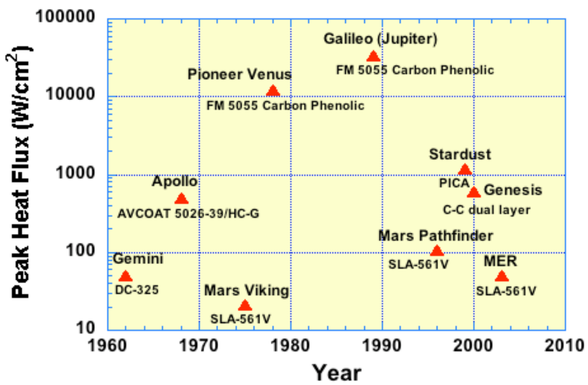

Fig. 3 Chronology of ablative TPS for NASA entry missions

Figure 3 also indicates the broad range of peak heat fluxes that these various missions encountered. Note the logarithmic scale of the ordinate. But Figure 4 provides a better representation as it illustrates both peak heat flux and stagnation pressure for these missions. In addition it includes values for the TPS mass fraction[1] for each mission. It should be apparent that NASA entry probes have successfully survived entry environments ranging from the very mild (Mars Viking ~25 W/cm^2 and 0.05 atm. to the extreme (Galileo ~30,000W/cm^2 and 7 atm.)

It should also be apparent that TPS mass fraction does not correlate with peak heat flux and/or pressure. As seen in Figure 5, The TPS mass fraction for an entry probe is a strong function of the total integrated heat load (e.g., ≈ 50% for Galileo) and the TPS material

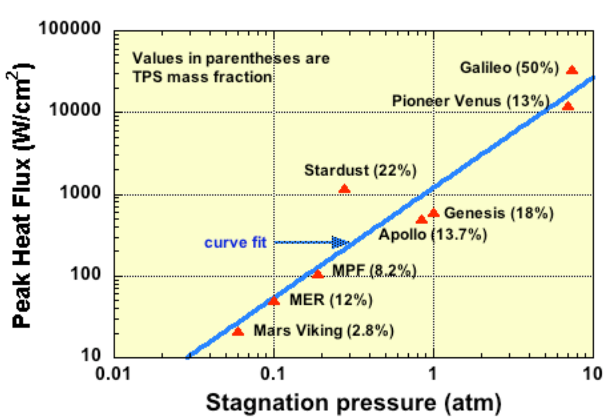

Fig. 4 Mission environments for ablative TPS applications

optimal performance characteristics. TPS material selection requires an assessment of the entry environment and a trade between ablation and insulation performance. Pioneer-Venus with 13% TPS mass fraction is an excellent example of TPS optimization for a very demanding mission, i.e., high heat fluxes, high pressures, and a relatively modest total heat load. Carbon phenolic, which is not a very good insulator but an excellent ablator, was a good choice.

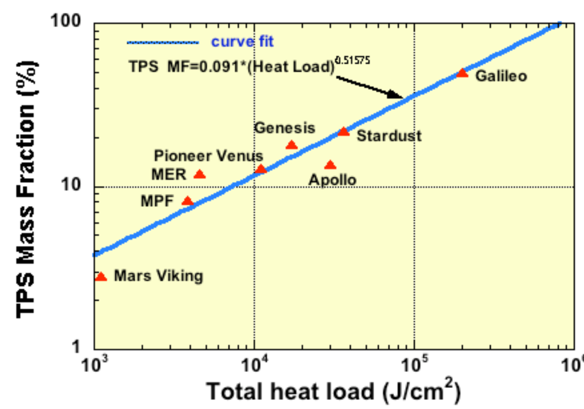

Fig. 5 TPS mass fraction for prior ablative TPS missions

It is also important to recognize that there are several classes of ablative materials and each class has its performance limitations. Typically, we categorize ablative TPS materials by density, i.e., low density, mid density and high density. Material strength increases with density, but so does the thermal conductivity. Consequently, materials selection for a given mission entry environment requires a balance between ablative and insulation efficiency while recognizing the optimal performance regime for each class of materials. When a material is used outside of its optimal zone, its performance is inefficient which leads to a non-minimal TPS mass fraction. This is illustrated in Figure 6, which suggests, notionally, that as density increases the threshold for char spallation moves to higher pressures and heat fluxes. Char spallation is an undesirable phenomenon as it consumes mass (periodically) with minimal loss of thermal energy and, importantly, is difficult to characterize and predict.

[1] TPS mass fraction is that fraction of the entry probe mass devoted to TPS.

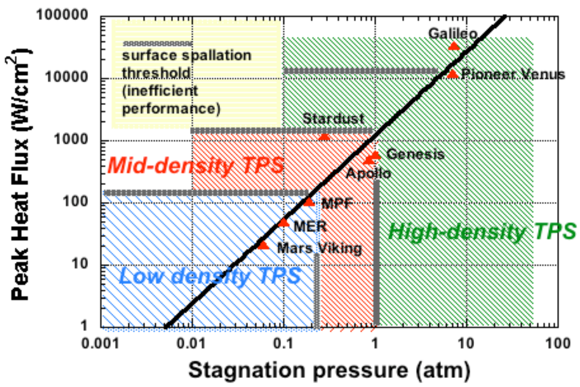

Fig. 6 Limitations of ablative TPS classes

Jupiter Missions

Lessons Learned from Galileo

The Galileo probe to Jupiter was the most challenging entry mission ever undertaken by NASA. The probe employed a 45 deg blunt cone aeroshell and it entered the Jovian atmosphere at a velocity of ≈ 47.4 km/s. The forebody TPS employed fully dense carbon phenolic (ρ = 1450 kg/m^3) that, at the time, was the best ablator available. The entry environment was very severe and estimates of the peak heating (combined convective and radiative) were on the order of 35 kW/cm^2 with a total integrated heat load of ≈ 200 kJ/cm^2. It's important to note that the above numbers include the effects of blockage due to ablation products.

To enable qualification testing of the TPS, NASA Ames developed and built test facilities that included a new arc jet test facility called the Giant Planet Facility (GPF) and a laser test facility to understand the spallation characteristics. The GPF arc jet operated on an H_2-He gas mixture and was capable of producing very high heat fluxes (convective and radiation) on test samples. Figure 7 presents the convective and the radiative heating environment for many of the missions including the Galileo and the Pioneer-Venus probes and also shows the operational environment for the GPF facility. The arc jet testing was augmented by testing with continuous wave (CW) carbon dioxide (CO_2) lasers which were capable of even higher heat fluxes, albeit with small spot sizes on target.

The Galileo Probe TPS design employed engineering tools developed in the 1970s and was very sophisticated for the time. A handful of teams independently developed and applied their methods for the analysis and evaluation of the design. These models addressed the coupled chemically reacting boundary layer and shock layer in the presence of thermochemical ablation and some spall. But it was also apparent that some of the models could not be validated (e.g., shock layer radiation) due to limitations in existing ground test facilities.

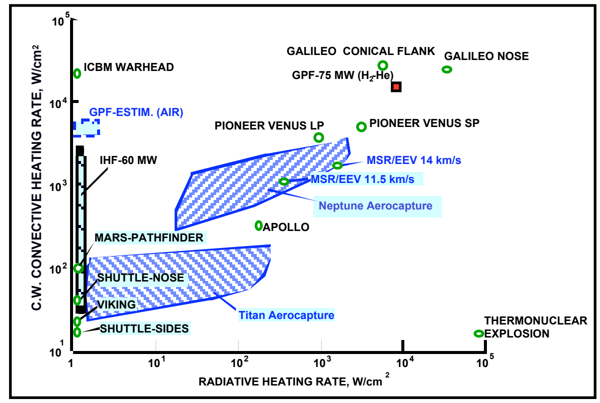

Fig 7. The stagnation region convective heat-flux and the radiative heat-flux for various missions along with the Giant Planet Facility (GPF) operational conditions are compared. Note: Galileo stagnation region heating is equivalent to the combined heating of an ICBM warhead flying through a thermonuclear explosion

The final TPS design, specifically the thickness distribution, was determined by adding a margin to the conservative side of the many predictions and the margin thickness was determined by consensus between many teams. Fortunately, ablation sensors were installed in the forebody TPS and the flight data provides us the basis to assess the accuracy of the design method. These data were extremely valuable in defining the actual performance of the Galileo probe TPS during entry. Figure 8 illustrates the ablation profile of the forebody TPS as deduced from the ARAD ablation sensors.

The Galileo ablation data demonstrated that stagnation point recession was less than predicted but ablation at the shoulder was significantly greater than predicted. In fact, the data suggest that there was almost a burn-through at the shoulder. Current physical models cannot explain the Galileo flight recession data. There remains significant uncertainty in the coupled environment /ablation physics.

TPS Challenges for Future Jupiter Missions

Based on the Galileo mission, fully dense carbon phenolic is the only TPS material with *heritage*. However, from the Galileo recession data it should be

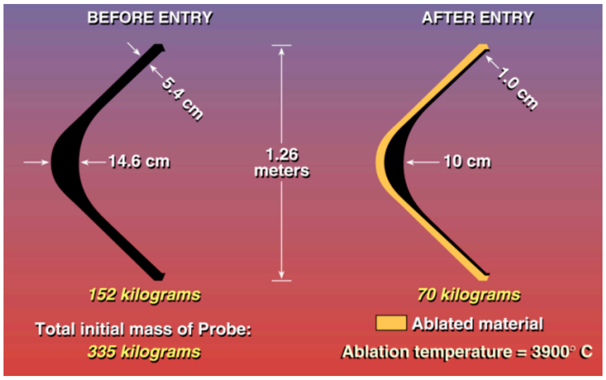

Fig. 8 Galileo probe heat shield ablation

apparent that for a similar Jovian equatorial entry probe, the TPS mass fraction would probably be greater than the 50% employed on Galileo. However, the science community sees the value of a multiprobe mission to Jupiter with some of the probes going to higher latitudes. But the entry velocity for entry probes to higher latitudes is even greater (~ 55km/sec at 30 deg latitude) and, since the heating increases with the cube of entry velocity (approximately), the heating rates will be too severe for even fully dense carbon phenolic, i.e., mass loss by char spallation will become the dominant ablation mechanism. A guestimate of the TPS mass fraction for such a mission using carbon phenolic would exceed 70%, which leaves little mass for science.

Investment Strategies and Benefits

To enable such a mission would require advanced TPS materials capable of reducing TPS mass fraction in comparison to that projected for carbon phenolic.

Qualification of such advanced materials would require a capability to demonstrate performance in ground test. Unfortunately, the Giant Planet Facility was dismantled after the Galileo program. To pursue the TPS development and design for another Jupiter entry probe mission, re-establishment of the Giant Planet Facility or something similar would be required.

The Galileo flight data demonstrated that the physical models employed for that design were not adequately validated and improvements are required. This would necessitate resurrecting, updating, and improving the 70s vintage tools by adapting computational techniques developed over past 15 years to these new applications.

The physical models would have to be updated using ground-test data. The development of such data, in itself, would be a challenge due to limitations in existing ground test facilities.

Venus Missions

Lessons Learned from Pioneer-Venus

In 1978 NASA launched the Pioneer-Venus mission that included one large entry probe (Sounder) and three smaller entry probes (Day, Night and North). All probes employed a common geometry, an aeroshell with a blunt 45 deg half-cone angle shape. Entry velocity was ≈ 11.54 km/s. The predicted entry heating environments for these probes was severe with peak convective heating rates in the range from 3900-7200 W/cm^2 and peak radiative heating rates in the range from 1300-3400 W/cm^2. Total integrated heat load (convective + radiative) was in the range from 12-14 kJ/cm^2. The forebody TPS for all probes employed fully-dense carbon phenolic that, at the time, was the only well-characterized robust ablator capable of handling such high heating rates.

TPS Challenges for Future Venus Missions

Currently, NASA is planning a future mission to put a Lander on the surface of Venus. If such a mission retains the same aeroshell shape as Pioneer-Venus, it would be logical to employ the same forebody TPS. However, the heritage material employed for Pioneer-Venus may no longer be available since it used a carbon cloth derived from a specific rayon fabric produced in the 1970s. Similar, carbon phenolic composites are currently being evaluated using carbon cloth derived from alternate rayon fabrics or other precursors. Characterization and qualification of such composites is straightforward but will require time and resources.

NASA is also evaluating the use of aerocapture to place an orbiter around Venus. The aerothermal environment for Venus aerocapture will experience lower peak heat fluxes but significantly larger total heat loads. While fully dense carbon phenolic would be a logical candidate for such a mission, it would not be the best choice as, given the large heat load, it would impose a significant TPS mass penalty on such a payload. A mid-density TPS with better insulation properties would be a better choice. Alternatively, a multi-layer system employing a robust ablator backed by a high temperature, low-density insulator would also be attractive for a Venus aerocapture mission.

During the period when the Pioneer-Venus probes were designed, the Giant Planet Facility did not exist. Testing TPS materials for a Venus entry mission was a challenge then and remains so today. No existing arc jet facilities operate on CO_2. Peak heating rates and pressures projected for Venus entry are attainable in existing arc jet facilities, albeit in air and on small samples. Radiative heating rates can be simulated with existing high-energy laser facilities, although the radiative spectrum would not be representative. Fortunately, that is not a major issue for high-density carbonaceous materials as such materials are surface absorbers over a broad range of wavelengths. It must also be noted that the TPS community understands the performance of high-density carbonaceous TPS materials and analytical scaling of their performance in air to the Venus atmosphere is straightforward.

Investment Strategies and Benefits

For direct entry missions to Venus, fully dense carbon phenolic remains a viable choice for the forebody TPS. However, as discussed previously, the heritage material may no longer be available and some investment is required to develop a modern carbon phenolic using a carbon fabric derived from alternate rayon or other precursors.

For potential aerocapture missions to Venus, newer TPS materials, at lower TRL, offer potential TPS mass savings. For example, materials and/or concepts that are robust ablators but better insulators could potentially reduce the mass fraction requirements by an estimated 50% in comparison to fully dense carbon phenolic.

Aerocapture Mission at Titan

A study to develop a conceptual design for an aerocapture mission at Titan was conducted during 2002 by a NASA systems analysis team comprised of technical experts from several of the NASA centers. Multidisciplinary analyses demonstrated that aerocapture could be accomplished at Titan with a blunt 70° (half angle) rigid aeroshell entering the Titan atmosphere at an inertial entry velocity of $\approx$ 6.5 km/s. Aerothermal analyses demonstrated that the peak convective heating rates are relatively mild but the radiative heating rates, due to shock layer radiation from CN, are significantly larger and lie totally in the narrow UV band from 3500 to 4200 Å. TPS sizing analyses were conducted for a broad range of candidate TPS materials and, as expected, low density materials are the most attractive from a TPS mass standpoint. However, there is significant uncertainty associated with the interaction of low-density TPS materials with UV radiation (i.e., the potential for in-depth absorption).

Of some concern is the interaction of CN radiation with low density, porous TPS materials. Figure 9 illustrates the spectral distribution of the predicted CN radiation where it is seen that almost all the radiation lies in a relatively narrow band in the ultraviolet (UV) with the peak at $\approx$ 3800Å (0.38μm). Studies conducted during the 1980s, evaluated the performance of dozens of ablative materials exposed to high-energy lasers. The types of materials evaluated spanned the range from low-density organic resin composites to fully dense carbon-carbon composites. Materials were tested with both continuous wave (CW) and repetitively pulsed (RP) lasers at wavelengths from the visible (0.53μm) to the infrared (10.6μm). While material performance was strongly dependent on the type of material and the irradiance (heat flux) it was exposed to, the data also suggested a general trend where material performance degraded at the shorter wavelengths. Further studies demonstrated that the materials did not become semi-transparent at the shorter wavelengths, but rather the absorption length became larger as the wavelength got shorter. The potential for in-depth radiant absorption is of concern since it could lead to char spallation that would significantly degrade material performance. It should be noted that none of the materials that NASA is evaluating as TPS candidates for Titan aerocapture were evaluated under these laser studies.

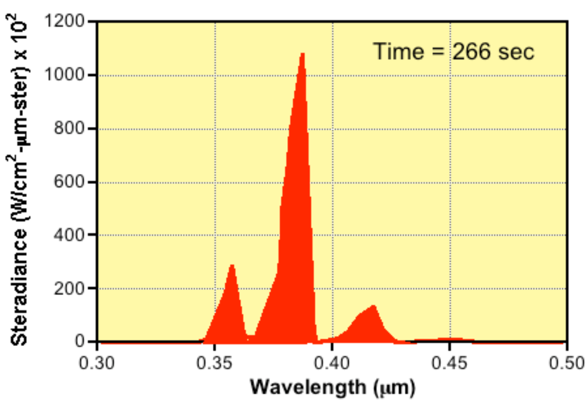

Fig. 9 Spectral distribution of CN radiation predicted for Titan aerocapture

It is also worth noting that the Huygens probe is scheduled to enter the Titan atmosphere in late 2004. The forebody TPS on the Huygens probe is AQ60, is a low density ($\approx$ 300 kg/m^3), porous silica fiber felt with phenolic resin reinforcement. From what we have been able to learn, during design of the Huygens probe TPS AQ60 was tested with a radiant IR source but not tested at UV wavelengths. Doing ground tests with a UV radiant source capable of producing heat fluxes of interest is a challenge. However, NASA is currently assembling a facility to do just such tests, which are currently scheduled for the spring of 2004. The results of those tests will either demonstrate that the performance of porous, low-density TPS materials is not degraded when exposed to UV radiation or, alternatively, they are not good candidates for a Titan mission and more robust (i.e., higher density) materials are better choices despite the weight penalty they would impose. Unfortunately, the Huygens probe TPS is not instrumented so flight performance will remain unknown.

Mars Missions

To date, NASA has launched several lander missions to Mars, i.e., Mars Viking (1976), Mars Pathfinder (1997) and Mars Exploration Rover (launched in 2003 with expected arrival in early 2004). All of these entry missions employed a blunt, 70 deg half-cone angle aeroshell and the same forebody TPS material, Lockheed-Martin's SLA-561. Currently, NASA is developing the Mars Science Laboratory (MSL) mission with scheduled launch in 2009.

There is also considerable interest in utilizing aerocapture at Mars. Given the low density of the Martian atmosphere and estimates of the aerothermal environment for direct entry or aerocapture, there are no significant TPS challenges for Mars entry missions since existing materials, including some newer low density materials are adequate.

However, a Mars Sample Return (MSR) mission is also of significant interest and the TPS issues such a mission introduces are not trivial. Direct entry from Mars into the Earth's atmosphere would be at velocities in the range from 12-14 km/s. The Planetary Protection requirements would, necessarily, be very stringent. This would dictate the use of a highly reliable TPS. However, from the NASA perspective, only (heritage) carbon phenolic can satisfy that requirement although TPS mass fraction would not be optimum. As discussed previously, the availability of heritage carbon fabric is, at best, uncertain.

Other Sample Return Missions

There are several other sample return missions in process or in the planning stages. For example, the Stardust mission is collecting comet samples from Wild 2 and is scheduled to return to Earth in January 2006. The aeroshell is a 60 deg half-angle blunt cone with a diameter of 0.827 m. An entry velocity of ≈ 12.6 km/s is anticipated. Peak stagnation point heat fluxes of ≈ 1200 W/cm^2 convective and ≈ 130 W/cm^2 are predicted with a total integrated heat load of ≈ 36 kJ/cm^2. A new TPS (PICA-15) developed at NASA Ames was selected for this mission since it was a lightweight solution that could reliably handle the anticipated heating. In reality, PICA-15 *enabled* this mission since the TPS mass fractions for other existing candidate materials were prohibitive.

The Genesis spacecraft is gathering solar wind particles and is scheduled to return to Earth in September 2004. The aeroshell is a 60 deg half-angle blunt cone with a diameter of 1.51 m. An entry velocity of ≈ 10.8 km/s is anticipated Peak stagnation point heat fluxes of ≈ 700 W/cm^2 convective and ≈ 30 W/cm^2 are predicted with a total integrated heat load of ≈ 16.6 kJ/cm^2. The forebody TPS for this mission is a new carbon multilayer (carbon-carbon facesheet insulated with carbon fiberform) developed by Lockheed-Martin. Similar to Stardust, this new TPS was enabling as it offered a lightweight solution in comparison to other TPS candidates.
NASA has several other lunar and comet sample return missions in the planning stages. Typically, these missions would return samples to Earth with entry velocities in the range from 10-16 km/s. Dependent upon aeroshell shape and dimensions, peak heat fluxes would be in the range from 500-2500 W/cm^2. Shallow entry angles would limit peak fluxes to the lower end of this range but would result in large total integrated heat loads. In contrast, steep entry angles would result in very high peak heat fluxes but would reduce total integrated heat load. The optimum TPS for such missions would minimize TPS mass fraction with modest ablation. Analytical studies have demonstrated that this is best accomplished with medium density TPS materials (ρ ≈ 500-700 kg/cm^3). Unfortunately, there are no well-characterized, qualified TPS materials of that class currently available (Apollo era materials no longer made) which implies a need for mid-density TPS development.

Neptune Missions

There is significant interest in missions to Neptune. Some potential missions involve probes for direct entry while others are looking at aerocapture to place an orbiter around Neptune.

Direct Entry Probes

Mission studies have determined that entry velocities at Neptune lie in the range from ≈ 28-32 km/s. Predicted stagnation point heating rates are very high, i.e., higher than Venus entry but lower than Jupiter entry. It should not be surprising that fully dense carbon phenolic would be the primary forebody TPS candidate for a Neptune probe mission. However, as stated previously, the availability of the *heritage* carbon phenolic used in Pioneer-Venus and Galileo *may* no longer be available. Similar composites are under development, but need to be characterized and qualified. Similar to Jupiter, improvements in physics-based models are needed, specifically in the areas of radiative heating, turbulent convective heating, and the coupling in the presence of ablation products. Improvements in these models can reduce uncertainty and TPS mass fraction significantly. However, similar to Jupiter entry, limitations in existing ground test facilities present significant challenges to validating any of these models.

Aerocapture

A NASA Systems Analysis Team conducted a detailed conceptual design study of a Neptune aerocapture mission in 2003. It was demonstrated that an aeroshell with higher lift-to-drag (L/D) ratio than provided by a blunt aeroshell would be required, i.e., an aeroshell shape capable of L/D ≈ 0.8 would be required. Aerothermal studies for an entry velocity of ≈ 29 km/s demonstrated that very high convective and radiative heating rates would be anticipated in the stagnation region, i.e., in the range 10-15 kW/cm^2. Furthermore, due to the long flight time in the atmosphere to effect aerocapture, stagnation region heat loads would be enormous, i.e., in the range from 1000-1500 kJ/cm^2. This combination of high heat flux and heat load presents a significant TPS challenge. At such high heat fluxes only fully dense carbonaceous materials are viable candidates. But the very large heat loads dictate TPS thicknesses that may be beyond the capabilities of materials suppliers to produce uniform, reliable composites.

Of course in areas away from the stagnation region heat fluxes are lower but heat loads, although proportionately lower, are still very large. The TPS design for such mid L/D configurations will require a suite of TPS solutions.

Investment Strategies and Potential Benefits

TPS design for Neptune aerocapture will require utilization of several TPS materials to minimize TPS mass fraction. New TPS materials (e.g., mid-density ablators) would be attractive choices on the windward side, but away from the stagnation region. Such materials, at lower TRL, offer potential TPS mass savings, but require development. Ground test facilities to evaluate such materials do not currently exist. Re-establishment of the Giant Planet Facility or similar would be required. Models for evaluating the aerothermal environment and coupling with ablation products need to be updated and improved.

Summary

There's been little ablative TPS development work in the USA over the past 20+ years. NASA has already done the "easy" missions with materials (for the most part) developed over 30 years ago. However, NASA's ambitious exploration vision requires TPS *innovations.*

Many future missions require TPS materials and/or concepts not currently available or, in some cases, new versions of old materials. New TPS materials, ground test facilities, and improved analysis models are required and will take some time to develop, Advances and improved TPS capabilities will benefit an array of missions (and *enable* some).

Figure 10 repeats the data shown previously in Figure 5, but includes TPS mass fraction estimates for some future missions currently in the planning stages.

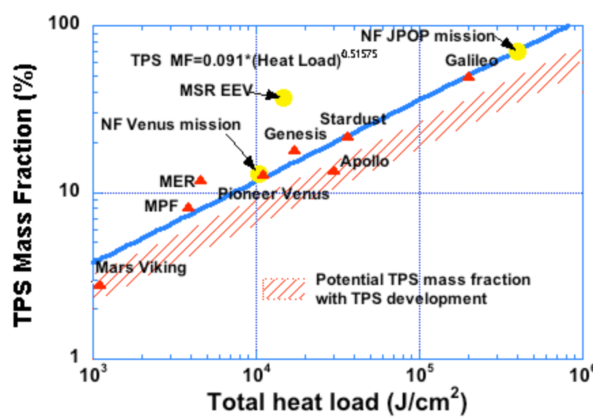

Fig. 10 TPS mass fraction for prior and future planetary missions employing ablative TPS

As seen, TPS mass fraction requirements for proposed New Frontiers missions (e.g., Jupiter Polar Orbiter with Probes - 70%) and Sample Return Missions (Mars Sample Return especially) become prohibitive and/or demanding with use of existing materials. The crosshatched region is an estimate of the potential savings in TPS mass fraction that could be achieved (≈ 20%-50% savings in TPS mass fraction) with some investment in TPS technology development. In this sense, *TPS technology* includes ground test facilities and improvements in models to predict the heating environment, as well as TPS materials.

Nomenclature

ARAD	Analog Resistance Ablation Detector
CN	Cyanogen radical
GPF	Giant Planet Facility
MSL	Mars Science Laboratory
MSR	Mars Sample Return
NASA	National Aeronautics & Space Administration
PICA	Phenolic Impregnated Carbon Ablator
SSEDS	Solar System Exploration Decadel Survey
TPS	Thermal Protection System
TRL	Technology Readiness Level
UV	Ultraviolet

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