

WHITE PAPER TO THE NRC DECADAL PRIMITIVE BODIES SUB-PANEL

**Thermal Protection System Technologies for Enabling
Future Sample Return Missions**

by

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INTRODUCTION

This NRC Decadal Survey white paper, provided by the thermal protection technology community, is a general assessment of the current capability of thermal protection systems (TPS) with respect to the return of samples from primitive bodies as well as anticipated TPS requirements needed to support future missions within this aspect of Solar System exploration[3,4]. The paper begins with a brief history of thermal protection systems, including those used for the Stardust and Genesis missions. This is followed by a discussion of current TPS capability and technology issues, and concludes with recommendations for a TPS Technology Program that includes an assessment of the research, development, testing, and manufacturing capabilities needed to support future Sample Return missions.

BACKGROUND: Historical Overview of TPS Development

For vehicles traveling at hypersonic speeds in an atmospheric environment, TPS is a single-point-failure system. TPS is essential to shield the vehicle structure and payload from the high heating loads encountered during (re-)entry. In addition, for the science community, it enables the safe deployment of *in situ* science instruments using probes, landers, balloons, and other instrumented systems. Minimizing the weight and cost of TPS, while insuring the integrity of the vehicle, is the continuing challenge for the TPS community.

The origin of modern thermal protection systems can be traced back to two major technological developments during WW II: the development of the German V-2 rocket and the U.S. atomic bomb. When coupled, the resulting nuclear-tipped ballistic missile capability became the primary strategic objective of the U.S. and U.S.S.R. militaries after WW II and therefore the goal of major development efforts. TPS was a key enabling technology for these missile systems, because without an effective TPS, the nuclear warheads would be unable to survive the heating during the descent phase of their intercontinental trajectory.

Early missile designs had sharp pointed noses and consistently failed during reentry conditions due to the high heat load and lack of a suitable TPS material. Viable reentry vehicles became possible only after two innovations. The first was proposed by H. Julian Allen at the Ames Aeronautical Laboratory and consisted of the counter-intuitive blunt body concept, wherein much of the heat load was deflected away from the vehicle via a strong bow shock wave. The second innovation was ablative TPS, which protects the vehicle via thermochemical phenomena including: 1) an ablation process that lifts the hot shock layer gas away from the vehicle, 2) heat being absorbed by the ablative material and leaving the vehicle as the material ablates away, and 3) the creation of a char layer, which is an effective insulator and also effective at blocking radiated heat from the shock layer. Essentially, the blunt body concept coupled with ablative materials is designed to deflect, reject, and reradiate the heat load - not absorb it. President Kennedy's call for a manned Lunar mission in 1961 resulted in a massive increase in funding for ablative TPS development. Similarly, military reentry vehicles utilized the blunt body concept and ablative TPS approach, even though those flight profiles and heating environment were very different from the Apollo missions.

NASA and military mission requirements led to a rapid development of practical ablative thermal protection materials. Carbon or silica based materials infused with phenolic resin composites proved to be the most suitable candidate materials for many missions. Silica based composites were found to be more efficient at lower heat fluxes (or lower entry speeds) due to lower thermal conductivity and carbon based systems, with a significantly higher temperature capability, were found to be more suitable for higher heat flux entries.

There were two other critical components in the development of viable thermal protection systems in the 1960s: 1) the development of hypersonic ground test facilities including arc

jets, shock tubes, and ballistic ranges, and 2) the development of analytical models and codes that predict the aerothermal environment during entry (both convective and radiative) and the thermal and ablation response of candidate TPS materials.

During the 1960s and into the mid-70s, the ablative TPS community in the U.S. was very active. However, by the late 1970s, the research, development, and testing of ablative TPS materials significantly declined as the nuclear missile programs were completed and the Apollo program was terminated after a dozen Lunar return flights. NASA shifted its focus to the Space Shuttle program that was designed to be a reusable system, including the TPS. While reusable TPS research, development and testing occurred in the late 1970s and through the 1980s, the ablative TPS community saw a serious decline in capability.

However, NASA continued to require ablative TPS for an occasional robotic entry probe mission (e.g., Mars Viking, Pioneer-Venus, Galileo). Fortunately, TPS requirements for these missions were satisfied with existing ablative materials. In particular, NASA leveraged the significant investment made by the U.S. military in the 1970s in developing FM5055 carbon phenolic for use as heat shields on ICBM reentry vehicles. Since then, NASA and Industry have made modest investments in ablative TPS in support of specific missions.

In the mid-90s NASA invested in the development of two new lightweight ablators, Phenolic Impregnated Carbon Ablator (PICA) and Silicone Impregnated Reusable Ceramic Ablator (SIRCA). Owing to its performance in the $\sim 1 \text{ kW/cm}^2$ heating environment and low heat shield mass, PICA enabled the Stardust Sample Return Mission. SIRCA has been used on the backshell of Mars Pathfinder and the Mars Exploration Rover missions. Furthermore, as an evolution of the Shuttle wing leading edge Reinforced Carbon-Carbon Composite (RCC), industry developed Advanced Carbon-Carbon (ACC). The Genesis Sample Return Mission heat shield was comprised of ACC atop a low density, rigid carbon fiber insulation.

During the last 4 years, NASA has been developing the Orion Crew Exploration Vehicle (CEV) to replace the Space Shuttle. Orion is designed to be capable of lunar return, therefore the CEV requires an ablative TPS. In response to this need, NASA formed an Advanced Development Project (ADP) to ensure that industry could build a large, 5-meter diameter ablative heat shield. The ADP evaluated five candidate ablators from industry, including Avcoat (the Apollo TPS material) and PICA, during 3 years of intensive effort. In April 2009, Avcoat was down-selected for the heat shield. **One very important lesson learned by the ADP was that even with a detailed specification in place, getting an “off-the-shelf” material, such as the Apollo Avcoat, back to its prior “heritage” level requires several years of intense and expensive effort due to the mothballed industrial capability and the lack of key personnel.**

Another important event was the MSL post-CDR (Critical Design Review) discovery of material failures in the thermal performance of SLA-561V at moderate heat flux and pressure, high shear and lower enthalpy conditions, resulting in the selection of an alternate TPS – PICA - for the MSL heat shield. Because the mechanisms that caused SLA-561V to fail are still not fully understood, the combination of parameters that may lead to failure in flight could not be established. Fortunately, the ADP was developing a tiled PICA concept for Orion and had begun to bring PICA production back on line. Without the CEV ADP investment of early PICA development and manufacturing efforts, MSL would not have met its original 2009 launch opportunity. This is a prime example of how NASA’s TPS community can work together and co-develop materials and technology that can benefit multiple projects and programs. **The very important lesson learned here is that it is wise to have at least two viable candidate TPS materials in place for mission projects** because although the selected TPS sufficed for a previous project, it may not be adequate for the next, even if the entry environments are only slightly more severe.

What is often overlooked is that since the late 70s, the expertise in ablative TPS, both at NASA and in industry, evaporated. The recent MSL and CEV TPS projects provided an opportunity for the few remaining senior personnel in the community to pass their knowledge on to a new generation of TPS scientists and engineers. The situation is much improved today, but without an effort to maintain this expertise, it will erode, as it did previously.

CURRENT CAPABILITIES: TPS for Sample Return Missions

Density	Forebody Heat Shield	Supplier	Flight Qual or TRL	Potential Limit		Entry velocity, km/s		Other Potential Missions
				Heat flux, W/cm ²	Pressure, atm	<13	>13	
FOREBODY HEAT SHIELD								
Low-Mid	PICA	FMI	Stardust	~1200	< 1	●	◐	SR, CEV, Mars
	Avcoat	Textron	Apollo	~ 1000	<1	●	✗	Venus (aerocapture)
	ACC	LMA/C-Cat	Genesis	> 2000	> 1	◐	◐	SR, CEV, Mars
	BPA	Boeing	TRL 3-4	~ 1000	~ 1	◐	✗	Venus (aerocapture)
	PhenCarb Family	ARA	TRL 5-6	(1000 – 4000)	>1	◐	◐	MSR, CEV, Venus, Earth
High	3DQP	Textron	DOD (TRL4)	~ 5000	> 1	■	■	SR, Venus
	Heritage Carbon phenolic	Several capable, none active	Venus, Jupiter	(10,000 – 30,000)	>> 1	■	■	MSR, Venus, Jupiter, Saturn, Neptune
● Fully capable ◐ Potentially capable, qual needed ■ Capable but heavy ✗ Not capable								

Table 1. Candidate ablative TPS materials for forebody heat shield for Sample Return applications

Materials

Given the superorbital re-entry velocities for sample return vehicles, their forebody thermal protection systems will almost certainly consist of ablative materials[2]. The heating environment over the afterbody of a blunt cone aeroshell, like that used for Stardust, is an order of magnitude less severe than the forebody environment. For these regions, several lower density materials are available that have been demonstrated to provide reliable performance at such conditions. There are also regions on the backshell where Radio Frequency (RF) transparent TPS materials may be required to allow communications, and a number of candidate materials are available to meet this requirement as well.

Table 1 illustrates the capabilities of currently available ablative TPS materials, potential performance limits, and potential regions of applicability for future Sample Return missions. Table 1 also shows two classes of missions; those with an entry velocity less than 13 km/s (which includes Stardust with its return speed of 12.9 km/s) and those greater than 13 km/s (where the environments are more severe). For the latter regime, the capability of current CFD codes to predict the aerothermal environment is limited, due to increased levels of ionization in the shock layer.

In the low- to mid-density range, for entry velocities <13 km/s, two candidate materials are fully capable and three candidate materials are potentially capable. The lower density materials are typically better choices, being lower mass solutions. While the single-piece PICA heat shield performed flawlessly for the Stardust Sample Return Mission, future missions may re-

quire larger heat shields, which will then require a tiled configuration for PICA. A significant qualification program would be required for the tile and gap filler system, due to the fact that the MSL environments are much less severe than those encountered for these Sample Return missions. Avcoat could then be a viable solution for missions with larger aeroshells.

For these materials to be available when required for future missions, manufacturing capability must be sustained by industry on a continuing basis. Recent experiences with resurrecting the PICA and Avcoat manufacturing processes have proven to be very costly and time intensive. Without a sustained TPS technology program, these specialty materials tend to languish, and eventually the capability for reliable production disappears. As a result, the manufacturing capability and expertise may not exist in the future for materials that have demonstrated successful performance on past missions.

For higher entry velocities, three low- to mid-density candidate materials are potentially capable with a significant qualification program. Unfortunately, the timeline to develop and qualify new TPS materials is generally on the order of 10 years. Heritage fully dense carbon phenolic is the only material with demonstrated capability to handle the aerothermal environment resulting from these high entry velocities, however there is a significant mass impact. The advantage of using heritage material lies in the extent and maturity of the performance database and design models. There are several vendors that routinely manufacture Tape Wrapped Carbon Phenolic (TWCP) to the heritage specifications, but Chopped Molded Carbon Phenolic (CMCP) has rarely been manufactured since the Galileo program. In addition, today's supply of the specific rayon precursor employed to fabricate the heritage carbon phenolic is extremely limited. Moreover, the U.S. companies with the capability to fabricate rayon suitable for heat shield development have gone out of business. There is currently no domestic supplier of aerospace-grade rayon in the U.S. The term "aerospace-grade" implies a quality control system that produces a uniform, consistent product, not required for most other rayon applications. NASA Ames Research Center acquired a modest supply of 1970s-vintage rayon from the limited stockpile held by the Navy's Strategic Systems Program Office, which will allow NASA to fabricate only a few heritage carbon phenolic heat shields, of modest size, for upcoming missions requiring a high performance heat shield. Recovering heritage carbon phenolic manufacturing capabilities will allow for a quicker and more cost effective design solution for higher speed Sample Return missions. In addition, work should be performed to develop carbon phenolic from other available aerospace grade rayon precursors as an alternate to the heritage carbon phenolic material. These investments not only benefit Sample Return missions, but they directly benefit Venus, Outer Planet, and Mars Sample Return missions as well.

For the aft body requirements, Table 2 shows that there are several suitable TPS materials. Most materials, however, will require a qualification program to verify performance capabilities relevant to the specific mission. Several of the candidate materials are also RF transparent, to allow for communication throughout all mission phases.

In addition to the need for sustaining current material manufacturing capabilities, as well as recovering heritage material manufacturing capabilities, a significant gap remains in heat shield materials development for Sample Return entry technology in the upper entry velocity range. The emerging mid-density TPS materials that could fill this gap, such as PhenCarb, lack the extensive ground testing and flight heritage necessary for TRL ratings higher than 5. These materials could lead to return of higher-mass samples relative to missions using fully dense carbon phenolic.

Ground Test Facilities

Arc jet facilities continue to provide the best simulation of the TPS flight environment, with certain limitations [1]. Existing arc jet facilities are not capable of simulating combined convective and radiative heating, and Sample Return missions to date have not required this capability due to relatively lower entry velocities coupled with smaller aeroshells (diameter < 1m).

Table 2. Candidate TPS materials for back shell heat shield for Sample Return applications

Density	TPS	Supplier	Flight Qual or TRL	Potential Limit		Entry velocity, km/s		Other Potential Missions
				Heat flux, W/cm ²	Pressure, atm	<13	>13	
BACKSHELL TPS								
Low	SLA 561V*	LMA	Mars	~ 300*	< 1	●	●	Mars, Earth
	SRAM Family	ARA	TRL 5-6	< 300	~ 1	◐	◐	Mars, Venus (backshell)
	SIRCA†	Ames	Mars	< 150	> 1	◐	✖	Mars, Venus
	Acusil II†	ITT	DOD	~100	< 1	◐	✖	Mars
High	Teflon (PTFE)†	Several	Varies	>500	>1	■	■	Mars, Venus, Earth, Saturn
	AD3DQ†	Textron	DOD	< 2500	> 1	◐	◐	DoD

● Fully capable ◐ Potentially capable (qual. needed) ■ Capable but heavy ✖ Not capable

*For low shear environments

†RF transparent

Table 3 specifies the heating rates for a Stardust-type entry body, and a -8 degree entry flight path angle for entry velocities ranging from 11.5 to 15 km/s. As previously noted, ionization becomes important at speeds greater than about 13 km/s, so the conditions are approximate. For small capsules and entry velocities less than 13 km/s, like Stardust, shock layer radiative heating is about 10% of the total heat flux. For larger capsules or higher entry velocities, radiative heating becomes increasingly more important. As the radiative heating component becomes more significant, Sample Return missions could benefit from future facility upgrades, like those planned for the Orion Program, which will incorporate the radiative heating component.

Table 3. Aerothermal environment conditions for a Stardust-sized entry capsule (-8° entry flight path angle)

Entry velocity, km/s	Convective heat-flux	Radiative heat-flux	Total heat-flux	Stagnation pressure
11.5	480*	30*	510*	0.33*
12.9	940†	90†	1030†	0.38†
15.0	1250†	360†	1610†	0.42†

*Engineering estimate

†estimates from CFD simulations

Note: Heat load will be relatively low (8 - 15) kJ/cm², as these are all ballistic entries.

Technical Engineering Development

The last two decades have seen the emergence of very sophisticated CFD real-gas codes - DPLR and LAURA. They are considered premier design tools and have been heavily used for the design of the Orion and MSL heat shields. However, these codes are not designed to reliably predict flow environments wherein significant ionization occurs, such as in missions with higher Earth return velocities (>13 km/s). Improvements in these codes are needed to verify material response and qualification test conditions. Integral in this process is also the development of improved testing techniques that support traceability from ground to flight aerothermal environment conditions. Additionally, data from tests that investigate material and system failure modes are needed to develop improved system reliability analysis methodologies.

Flight Instrumentation

Uncertainties in CFD and material thermal response models are significant and limit the ability to design and validate a more robust entry, descent, and landing architecture for future entry missions. Reducing uncertainties requires validation with relevant flight data. There has been very limited TPS flight data obtained, with the bulk being acquired in the 1960s primarily in support of the Apollo program. While many early planetary missions contained at least a minimal set of instrumentation, attempts to instrument recent science mission vehicles, including Stardust, have been largely unsuccessful. The exception to this is TPS instrumentation on the MSL heat shield, which will provide more than an order of magnitude more data than all previous Mars entry missions combined. The instrumentation will provide thermal, recession, and pressure data that will capture important measurements of the aerothermal environments, material response, vehicle orientation, and atmospheric density of the Martian atmosphere. While this data will prove invaluable in validating current CFD and material thermal response models for PICA during Martian entries, there is still a critical need to obtain relevant earth re-entry TPS flight data to benefit Sample Return missions.

RECOMMENDATIONS

Sample Return missions can be accomplished in the near term with existing TPS and design methodologies. However, to support future Sample Return missions, the national capabilities that ensure continued availability of existing TPS materials must be maintained. Recent experiences with resurrecting the PICA and Avcoat manufacturing processes have proven to be costly in time and dollars. Without a sustained TPS technology program, it cannot be guaranteed that the capability and expertise will exist for supplying materials that have demonstrated successful performance on past missions. In addition, development of new materials in the mid-density ablator class, as well as improved design analysis tools, are enabling for future higher speed Sample Return missions (>13 km/s).

Specifically it is recommended that NASA establish a cross cutting TPS Technology program with elements focused on sustaining current technologies and elements focused on enabling future higher speed Sample Return missions. The program will need to focus on the following:

Materials:

1. Sustaining current material manufacturing capabilities and expertise to ensure that at least two proven TPS materials, for both the heat shield and back shell, are available for future Sample Return missions. Recertification or qualification of these materials needs to occur every few years.
2. Fully recovering the heritage manufacturing process for carbon phenolic TPS.
3. Developing alternate carbon phenolic materials using currently available precursors.

4. Developing and qualifying mid-density ablative materials as an alternative to carbon phenolic TPS.

Arc jet Facilities:

Leveraging arc jet facility upgrades already underway in support of the Orion Program.

Technical Engineering Development:

1. Improving design and analysis tools, such as CFD and material response models, to verify material response and qualification test conditions.
2. Developing improved testing techniques to support traceability from ground to flight aerothermal environment conditions.
3. Designing tests that investigate material and system failure modes and use such data to develop improved system reliability analysis methodologies.

Flight Instrumentation:

Including TPS instrumentation in all Sample Return missions to generate a database of relevant flight data that will aid in the design of future entry missions.

In conclusion, TPS technologies for Sample Return missions are NASA unique, challenging, and cross cutting. It requires specialized resources in terms of expertise, facilities and capabilities across NASA and industry. These can be deployed to support different NASA missions and not just sample return. The Decadal committee needs to consider not only specific recommendations made above for destinations of interest to the Primitive Bodies sub-panel, but also the TPS needs for other Science destinations addressed in companion TPS white papers and the needs of other NASA stakeholders to ensure that taxpayer dollars provide the maximum return on investment.

Finally, it is requested that during the course of the development of recommendations by the new Decadal planning team, the TPS community be given feedback on those missions that appear to be emerging as high priority and that involve atmospheric flight. With this information, these recommendations could be focused towards the prioritized missions with supporting analysis. If helpful, estimates for cost and schedules could be provided upon request.

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