Ablators - From Apollo to Future Missions to Moon, Mars and Beyond

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Abstract

Apollo was designed to carry astronauts safely back from the Moon at return speeds exceeding 11 km/s and required development of a new ablative thermal protection system (TPS) to protect the capsule from entry heating. Mercury and Gemini, that preceded Apollo, were focused on Earth orbiting system demonstration and lessons learned from them were used in Apollo. The ablative material and associated system development for Lunar return conditions required considerable ground and flight testing. Mars Viking Lander missions required a new lighter weight ablator as entry heating was benign compared to Apollo. Pioneer-Venus and Galileo Probe missions required a new and more capable ablator than Apollo. After two decades, Mars Pathfinder followed by Mars Exploration Rover missions, smaller than Viking but more demanding, were able to use Viking ablative TPS. At the same time, advances in man-ufacturing and materials technology led to development of innovative lightweight ablators. These new ablators ena-bled Stardust and Genesis Sample Return Missions. Around the turn of this century, NASA decided on a scaled-up version of the Apollo capsule for human exploration of Moon and Mars and the ablative heat shield to protect the Crew Exploration Vehicle ended up being the Apollo ablative TPS. The Artemis 1 mission is currently fitted with tiled system, different than Orion EFT-1 but with the Apollo ablative material as a result of lessons learned. NASA is currently planning on sample return missions from Mars, and this will require robust ablative TPS that can provide higher reliability than any other past mission. There are still unexplored high scientific value destinations in the solar system. In situ exploration of Uranus, Neptune, Saturn and sample return missions with return speed much higher than Stardust will require ablators capable of withstanding extreme entry that are also efficient. New ablative TPS have been developed in anticipation of these future missions. This paper is intended to tell the story of these ablators, illustrated through examples. We see the use of flight proven ablators was sometimes a risky proposition and new ablators perceived to be higher risk have proved otherwise. The history of ablators illustrates the challenges each mission had to address, either through the use of flight proven or new ablative TPS, to be successful.

1. Introduction

Preparation for the 2019 Paolo Santini Memorial Lecture and writing this companion paper required me to gain much more detailed knowledge than just the anecdotal stories I have heard from my mentors about the Apollo, Viking, and Pioneer-Venus ablative TPS. In 1978, long past the Apollo era, I came to NASA Ames Research Center as a graduate student to do my Ph.D. thesis work in computational fluid dynamics, and my thesis was on predicting the 3-D hypersonic flow around Space Shuttle Orbiter. Since then I have moved from one area to the next, and my first exposure to ablative TPS started in 1992 around the time of Mars Pathfinder (MPF). NASA Ames was involved in MPF and also developing advanced ablative TPS such as Phenolic Impregnated Carbon Ablator (PICA) and Silicone Impregnated Refractory Ceramic Ablator (SIRCA). Between (1997 – 2003), NASA's focus on Mars Sample Return (MSR) Mission and proposed efforts for Venus and Outer-Planet missions required me to learn about Carbon-Phenolic ablative systems. I was fortunate to be mentored by Bernie Laub, Don Curry, Alvin Seiff, James Arnold, Howard Goldstein and others that worked on Apollo, Pioneer-Venus and Galileo heat shields. When

I became the flight system lead for the Crew Exploration Vehicle Thermal Protection System Advanced Development Project (CEV TPS ADP), which was the second coming of Apollo, I was thrown into the thick of ablative TPS development. The CEV TPS ADP ended its role with the selection of Apollo ablative system, around 2009, and I moved on. All of the lessons learned over the past two decades and the challenges of Human Mars missions motivated me to propose and develop new ablative TPS with multiple goals: 1) meet the MSR needs in terms of robustness and 2) withstand extreme entry environments such as Saturn, Uranus and Neptune. This was a challenge and an opportunity that led me to lead the development of a multi-functional ablative material called 3-D MAT, now a part of Artemis 1, and a new ablative heat-shield technology called Heatshield for Extreme Entry Environment Technology (HEEET) that was delivered to NASA recently. HEEET is an option MSR Earth Entry Vehicle (EEV) is currently evaluating. Only time will tell how HEEET is used in future missions. NASA Ames was at the forefront of enabling Apollo with the "blunt body" concept, and it has become the center for both reusable and ablative TPS for the past five decades.

I have been fortunate to work with and learn from the many engineers and material scientists at Ames.

2. Early Years (1949 – 1959)

The first man made object to experience the heat of entry at hypersonic speeds was the second-stage slender rocket, WAC. Sitting on top of V2, the first object of human origin to achieve five times the speed of sound and the only things that survived re-entry were the charred electric switch and part of the tail section [1]. Around the same time, engineers at NACA were beginning to work on practical and theoretical aspects of space flight. The problem that was identified and addressed was "thermal thicket" and the first real breakthrough in the search for a way to surmount the thermal barrier was achieved by Harvey Allen, a senior aeronautical engineer at Ames and chief of the High-Speed Research Division. Allen had invented a technique of firing a gun-launched model upstream through a supersonic wind tunnel to study aerodynamic behavior at high Mach numbers. The Ames supersonic free-flight wind tunnel, opened in 1949, had a test section 18 feet long, one foot wide, and two feet high. By forcing a draft through the tunnel at a speed of about Mach 3 and by firing a model projectile upstream at a velocity of 8000 feet per second, the Ames researchers could simulate a Mach number of about 15. Tests of slender configuration, using metal models in the supersonic wind tunnel at Ames and in rocket launches at Wallops Island, showed that so much heat would be transferred to the vehicle that the warhead would shortly vaporize as it plunged through the atmosphere. No protection system known at that time could prevent its destruction by aerodynamic heating [2].

Harvey Allen took the sharp-nosed Atlas re-entry shape and began making mathematical calculations, using only a pad and pencil. Eventually he reached a conclusion that seemingly contradicted all the years of aeronautical research and streamlined aircraft design. Allen's analysis showed that the best way to cut down re-entry heating was to discard a great deal of one's thinking about orthodox aerodynamics and deliberately design a vehicle that was the opposite of streamlined. *"Half the heat generated by friction was going into the missiles,"* recalled Allen. *"I reasoned we had to deflect the heat into the air and let it dissipate. Therefore, stream lined shapes were the worst possible; they had to be blunt."*

Harvey Allen personally submitted his findings to select persons in the missile industry in September 1952. A secret NACA report memorandum embodying his conclusions on the blunt-nose design, co-authored by Alfred J. Eggers of Ames, went out to industrial firms and the military the next spring. The report bore the date April 28, 1953, but six years passed before the paper was declassified and published in the annual report of NACA [3]. Around 1955, several efforts were underway as to how to tackle the re-entry heating. Ablators were not in

the vocabulary and options considered were heat sinks of high-conductivity materials, liquid internal cooling, ceramics which might ablate and transpiration cooling [4].

George Sutton, in his keynote address [4] describes the early ablator efforts as follows:

"…re-examined every material I had previously considered while at Lockheed. Beryllium oxide, a potential ceramic under investigation by the Chicago Midway Laboratories for G.E., appeared too brittle (because of the wrench criterion). I obtained copies of the Peenemunde documents, and discovered the work of Carl Wagner, who calculated the heating and oxidation of graphite as a nose cone material. I looked carefully at the materials tested for jet vanes, which also must endure a high-temperature, high pressure, and high heat-transfer environment. I read about oak, wet oak, graphite, etc., in reports signed by Werner Von Braun, and later personally discussed these with him. They all worked, but not very satisfactorily… During this time period I studied the properties of fibers, and properties and fabrication of plastics, to more thoroughly understand their subsequent physical properties. The high tensile strength of glass fibers (300,000 psi) was attractive for the application….
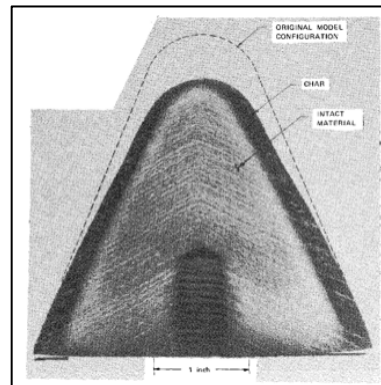

Fig. 1. Silica-phenolic tested in an arc jet showing characteristic that are common to ablators, namely recession and char (1958) [4]. The tape-wound specimen contained radial fibers but at an angle to the surface to alleviate delamination.

"…Of the resins, the phenolic had the highest-temperature capability; and since it was cross linked, it reduced to a char with some structural integrity, especially if there were refractory fibers in it to hold it together. The phenolic resin must be heated and held under high pressure to achieve the cross-linking. During this time the viscosity of the resin is very high, so it is difficult to mold. Finally, it is necessary to release the steam formed by the reactions of the resin. The other resins were generally easier to mold. My plan was to have hemispherical shapes molded and to test them in a rocket exhaust."

Thus, began the concept of how to manufacture and how to test ablators. Fig. 1 is the earliest picture of a

post-test ablative blunt test article , we can find, taken from Ref [4]. Note the year was 1958.

There were many entities working on the ablative TPS for blunt nose missiles during these foundation years that led to the success of Mercury, Gemini and Apollo in quick succession. Some of the most respected experts stated the reentry problem at that time as follows:

"...re-entry...is perhaps one of the most difficult problems one can imagine...It is certainly a problem that constitutes a challenge to the best brains working in these domains of modern aerophysics..."

-    *Theodore von Karman, 1956*

"In looking over the vast amount of recent effort to understand the mechanism of aerodynamic heating, one quickly concludes that, although many advances have been made, there is still a lot to be learned before the engineer be confident in his design"

-    *VAN DRIEST, E.R., 1956*
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3. Mercury and Gemini (1958 – 1966)

Project Mercury ran from 1958 through 1963. The goal was to have an astronaut orbit around Earth and bring him back safely. Newly created NASA conducted twenty un-crewed developmental flights (some using animals), and six successful flights with astronauts. In preparation, McDonnell, the prime contractor, designed the capsule so that it could be fitted alternatively with either a beryllium heat-sink or an ablative heat shield. The long heat-pulse due to shallow entry, combined with safety concerns of the astronaut sitting behind a heat-sink and the progress made by John H. Winter, the heat shield project coordinator at the Cincinnati Testing Laboratory in the development of ablative system, made the ablative material [2] to be the choice for heat shield TPS.
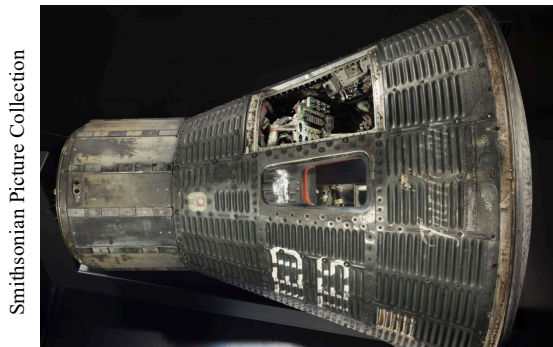
Fig. 2 (a). The Mercury Capsule

The cone shaped spacecraft had a convex base (Fig. 2 (a)), which carried the heat shield consisting of an aluminum-honeycomb covered with multiple layers of fiberglass-phenolic. A small sample taken from a post-flight heat shield is shown in Fig. 2 (b).

Strapped to the heat shield was a retropack consisting of three rockets deployed to brake the spacecraft during re-entry. Between the heat shield and inner wall of the crew compartment was a landing skirt, deployed by letting down the heat shield before landing [2].

In one of the early airdrops a jettisoned shield actually went into "a falling leaf" pattern after detachment. It glided back and collided with the capsule, presenting an obvious potential hazard. This incident prompted the decision that the heat shield would be retained [2].   There were three failures associated with early unmanned Mercury flights but none of them related to entry or heat shield. On February 20, 1962, John H. Glenn Jr., became the first American to orbit Earth. When the post-test inspection was done (~ 1955), it showed the fiberglass phenolic heat shield did not bond to the structure in large areas [8].

Fig. 2 (b). A small core sample from the heat shield (right) showing char and ablated regions, and cracks [5]

Project Gemini's objective was the development of space travel techniques to support the Apollo mission to land astronauts on the Moon. NASA's second human spaceflight program started in 1961 and concluded in 1966.   In 1964 and 1965, two un-crewed Gemini missions were flown to test the heat shield. The capsule was a scaled-up version of Mercury to accommodate two astronauts at 3.0 m (10 ft) diameter (Mercury was 1.89 m (6.21 ft) diameter). Ten crewed Gemini flights, between 1965 and 1966, were conducted successfully.

Ablative TPS selection for Gemini involved three series of plasma-jet tests conducted at different facilities in 1962, starting with 20 materials at a given heating rate and pressure. Four superior materials were selected and tested at reduced pressures and at several heat fluxes. Finally, three best materials were tested in a high-shear stress flow environment as well as for swelling in vacuum [6], which led to the selection of McDonnell's material, which had superior thermal effectiveness and erosion resistance for the range of heating rates of interest. McDonell's material was a silicone elastomer cured in a honeycomb at room temperature (Fig. 3). The silicone elastomer is a paste-like material which hardens after being poured into a honeycomb form. Starting with a load-carrying Fiberglass sandwich structure consisting of two 5-ply faceplates of resin-impregnated *glass* cloth separated by an 0.65 in. thick Fiberglass-honeycomb core, an additional Fiberglas honeycomb is bonded to the convex

side of the sandwich and filled with DC-325 silicone elastomer [6].
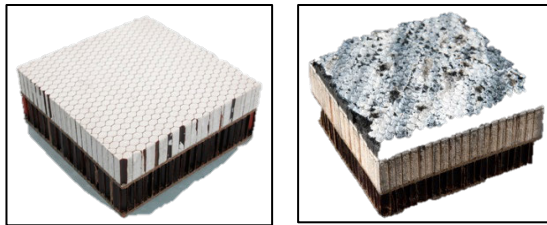
Fig. 3. Gemini ablative system, small samples cores, virgin on the left and post flight test on the right. The white substance on the post-test sample is the glass.

Arc jet test data with instrumented model and recession measurements for a range of conditions were obtained and used in determining flight heat shield thickness with margin to account for uncertainties. Ref. 6 describes aspects of ablative heat shield development from early screening to manufacturing, integration, flight qualification and full-scale flight testing. By 1966, the US had developed a strong system approach to the ablative heat shield as an integral part of the space program. Expertise and approaches developed under Mercury and Gemini were extremely useful and were applied to the development of the Apollo ablative thermal protection system [7]

In addition to the ablative heat shield, metallic shingles (rené 41 and beryllium) were used to protect the rest of the vehicle. The thermal protection was more than a material selection challenge. Defining the operational environment, material and sub-system options, manufacturing, integration, testing, both ground and flight and continued improvements in between flights required new developments across the cross disciplines.

4. Apollo (1966 – 1972)

The Apollo program officially began in 1966 and ended in December 1972. Mission planning, studies and technology development began as early as 1961. An initial set of entry trajectories were developed starting in 1961 and were continually refined. Unlike Mercury and Gemini, Apollo had to fly a lift-guided entry with a blunt body that could accommodate 3 astronauts. Defining the environment and selecting the ablative thermal protection were recognized as key challenges in 1961. Two limiting trajectories, overshoot and undershoot, for a 5000 mile down range were designed specifically to define entry environment for the ablative system. The ablator sizing include the effects of the heating environment during ascent flight (with no specific temperature limit imposed) and the thermal environment in space to a cold temperature of -260^0 F and a hot temperature of 250^0 F at the

beginning of entry, 700^0 F at splash-down for the interface of the ablator and the stainless-steel honeycomb structure, and 200^0 F for the aluminum-honeycomb pressure-vessel structure. In 1963, numerous developments associated with crew module (CM) required weight saving refinements in the design, and new operational logic dictated the need for design trajectory change. The next set of trajectories (Block II) provided an opportunity to resize the ablator as well.
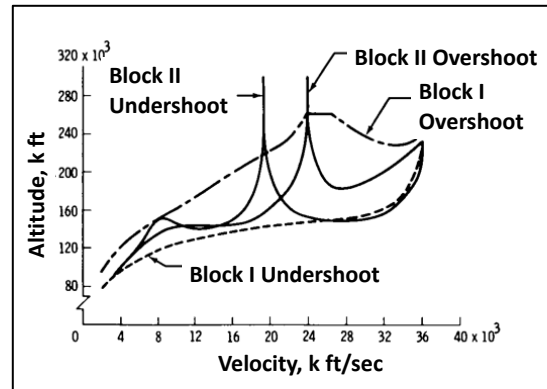

Fig. 4. Design entry trajectories [8]

The external shape of the Apollo CM like the Mercury and Gemini spacecraft consists of a blunt entry face with a conical afterbody that was designed to minimize convective heating during atmospheric entry (Fig. 5). The ablative TPS covered the entire outer shell of the CM and consists of an ablator bonded to a stainless-steel structure in three sub-assemblies (aft heat shield, the primary heat shield, crew compartment and forward heat shield) as shown in Figure 5.
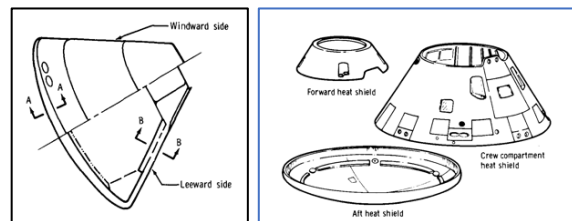

Fig. 5. Apollo outer mold line and the heat-shield sub-assemblies [9].

The entry environment predicted included not just convective heating but also due to shock-layer radiative heating. One unknown concern was nonequilibrium radiation from the shock layer. Theoretical studies and data from shock tubes and light-gas guns showed that the nonequilibrium radiation in comparison with equilibrium radiation decreased with increasing body size. Eventually, basing the Apollo CM configuration upon minimum convective heating was shown to be proper. Early wind-tunnel data on the heating distribution verified that much

of the CM experienced heating well below stagnation-point levels. The uncertainty about the level of heating over parts of the body that is not facing the flow and the problems of surface discontinuity, such as at the Mercury shoulder, led to the choice of an all-ablative system. The predictions were later verified by the FIRE I and II flight experiments (1965), and the comparisons are shown in Fig. 6
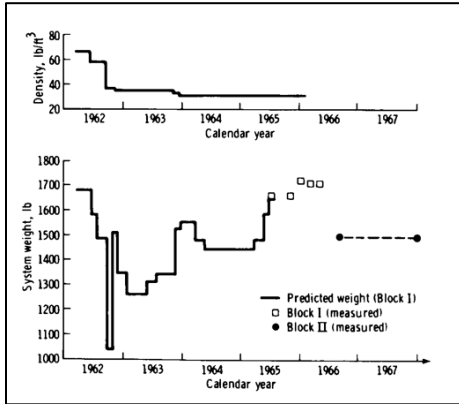

Fig. 6. Measured and predicted heating rates for Project FIRE I during entry [9].

The ablative material initially selected in April 1962 for the Apollo ablative material was Avcoat 5026-22, which had a density of 66 lb/ft^3. The predicted TPS weight in 1961 was 1684 pounds, and it was too heavy. Successive material improvements led to significant reduction in the material density but evolving design details, refined analytical methods replacing crude estimates and rigorous thermal control criteria as the program progressed led to mass increase. The time history of ablator density improvement and the ablative TPS weight changes [8] is shown in Fig. 7.

The thickness of the ablator varied with the local thermal environment. In addition to the basic thermal environment design considerations, the Apollo heat shield also had numerous penetrations and protuberances for the installation of components such as windows, reaction control engines, antennas, and vents. Each of these discontinuities in the TPS required special design considerations such as the recessing of the components and the use of densified ablators in local adjacent areas.

The final ablative material was designated Avcoat 5026-39G and consists of an epoxy-novalac resin reinforced with quartz fibers and phenolic micro-balloons. The density of this material is 31 lb/ft. The ablator is applied in a honeycomb matrix that is bonded to a stainless-steel substructure. The phenolic honeycomb is first bonded to the stainless-steel shell with HT-424 adhesive, and then the ablator is inserted into the individual honeycomb cells with a hypodermic device [9].
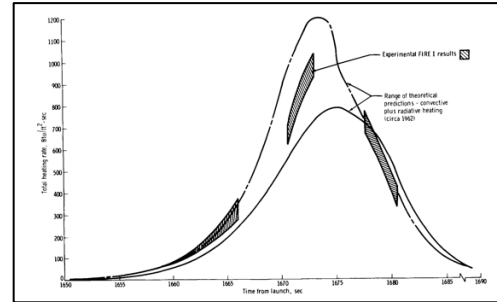

Fig.7. History of ablator density and heat shield system weight [8]

The manufacturing of the ablative material was a significant challenge. Early efforts were to tamper the ablative material into the honeycomb following the Gemini process. But this resulted in significant voids, and a new technique as well as tooling needed to be developed.

The process is described in Ref [9]. The structure is first cleaned, and a primer coating is applied before the bonding of the fiber-glass honeycomb with HT-424 tape adhesive. The fiber-glass honeycomb core sections are then fitted in place and the assembly is vacuum bagged, and oven cured. Inspection of the bonding of the honeycomb to the structure is done using ultrasonic testing and unbonded areas are repaired. The assembly is ready for application of the ablator into the honeycomb. This operation, termed "gunning, " is the injection of the Avcoat 5026-39G into each cell of the honeycomb by means of a special gun developed for that purpose. The cylindrical cartridges containing the ablator are dielectrically heated and are inserted in the gun. When the nozzle is positioned over the honeycomb cell, an air valve injects a blast of air into the cartridge and this entrains the ablator and carries it into the cell, filling it from the bottom to the top. There are approximately 370,000 honeycomb cells and each is filled by hand by the above process. Fig. 8 is a model, given to Smithsonian, made by Avco, to illustrate the end product of each step in the making of Apollo heat shield. Fig. 9 is a screen shot of the gunning process from Ref. [10]. Fig. 10 is a sample virgin Avcoat. Fig. 11 (a) shows larger piece of the post flight-tested Apollo AS-202 heat shield from the Smithsonian collection and also the author inspecting other pieces (Fig. 11 (b)).

In the first 26 months, starting in 1966, there were 7 flight tests, averaging about 4 months a flight test, an amazing record. There was 1 failure at the launch pad (Apollo 1). AS-201, the first flight test qualified the heat shield to orbital reentry speed (February 1966). Apollo 4 qualified the heat shield to lunar reentry speed (November 1967). Apollo 7 was the first crewed Earth orbital demonstrator (October 1968). On July 1969, successful Apollo 11 led to the historic Lunar landing.

Fig. 8. A demonstration article that illustrates the many steps involved in making avcoat heat shield.

Fig. 9. Insertion of the Apollo ablative material Avco 5026-39G into individual honeycomb cells by "gunning" [10]
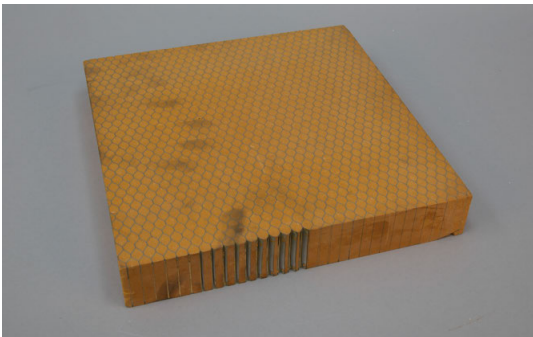

Fig. 10. An Avco 5026-39G panel (virgin)

The expertise and the capabilities in preparing for and executing Apollo program impacted many programs to follow in the future and is worth recalling the key lessons from Apollo [8].

" 1) Thermal performance of the ablation material is one of the lesser criteria in developing a TPS. We need to view its role as part of the integrated system with focus on inspection access, thermal stress, manufacturing, and performance at singularities but never to obtain better thermal performance of the basic ablator.

2) Screening for thermal performance alone is not sufficient, simple analyses do not form a sound basis and flight testing in environments not closely representative of design conditions cause unwarranted concern."

Fig. 11 (a). Post flight-test heat shield part (Apollo AS-202) missions

Fig. 11 (b). The author inspecting the heat shield parts (Apollo AS-202) at the Smithsonian.

5. Mars – Viking Mission (1968 – 1975):

The Viking project was begun by NASA in the winter of 1968, just prior to the Apollo 11 Lunar Landing, and the goal was to make scientific investigation of biological, physical, and related phenomena in the atmosphere and on the surface of Mars [11].

NASA planetary investigators began planning the exploration of Earth's closest neighbors in the 60's. The first Mars lander project called "Voyager" was conducted by the Jet Propulsion Laboratory starting in 1960 with the goal of landing on Mars by 1969. It resulted in 13 proposals by 1963 and notable among them were proposals submitted by GE at Valley Forge, Avco Corp., and Martin Marietta, Denver. The Avco Corp. proposal envisioned a blunt body of the aeroshell to protect the lander during entry and slow the descent with a parachute, deployed when the aeroshell and heat shield were discarded, to slow the craft further.

It is worth noting at this juncture that while ablative TPS developed by Avco Corp. was used on Apollo, Martin Marietta developed ablative TPS that enabled Viking missions, and GE developed the ablative TPS for Pioneer-Venus probes from the DoD materials they had experience with. All three companies though competed during this time and were able to find the right ablative

TPS that met the mission needs. The development of the ablative materials from material performance to integration challenges once addressed, qualified the material-system for the specific application and each one found their way to missions.

By the fall of 1963, the NASA Administrator decided that the resources required, manpower and dollars, precluded a 1969 launch and it was deferred until the first Lunar Landing. NASA LaRC began looking at delivering probes and landers at Mars with their expertise in aerodynamics, starting in 1964. When the Viking project was finally approved, after years of discussion within NASA as to who should lead, NASA Langley Research Center (LaRC) was given the responsibility for managing the Viking I and II missions along with JPL responsible for the Orbiters and the mission operations, and Martin Marietta Corp (which became Lockheed Martin Denver in 1995) became the prime contractor.

By 1967, Martin Marietta had developed light weight ablative systems suitable for Mars entry and published the results [12]. The ablator, SLA 561, is composed of a silicone resin with ground cork, phenolic micro-balloon, silica micro-balloon, and refractory fiber fillers with densities range from 0.19 to 0.25 g/cm3. SLA-561 is one formulation in a family of several successful filled-silicone heat-shield materials developed at Martin Marietta in Baltimore in the 1960's. SLA-561V is the Viking formulation, lower density (14.5 lb/ft3) SLA, reinforced with a fiberglass-phenolic honeycomb (H/C) with a density of 16.0 to 17.0 lb/ft3.
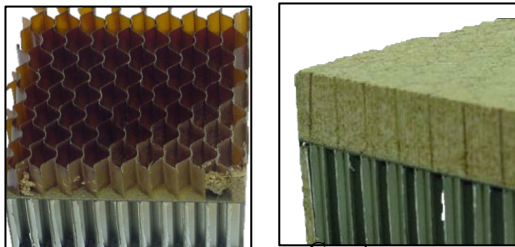
Courtesy:

Fig. 12. Flexcore honeycomb bonded to the structure (left) and the SLA-561V (right).

The material is tan and has a fine graininess to its surface from ground cork. SLA-561V is a charring ablator system, and the honeycomb stabilizes the silica-based char to reduce mechanical erosion from aerodynamic shear forces. Shown in Fig. 12 are representative of the honeycomb bonded to the structure and finished SLA-561V [14] similar to that used on Viking.

Plasma (arc) jet tests were conducted in N2, Air and CO2 and the material behaved better in all atmospheric compositions compared to other materials available at that time. SLA was tested in two different configurations. One, as part of fiberglass-phenolic honeycomb, and the other without honeycomb as a block ablator. Both variants performed well. The SLA ablative material was flexible even at – 150 F and considerably strong (not brittle) at cold temperature compared to room temperature making it well suited for interplanetary missions. Since the mission goal included search for life, concerns associated with potential contamination of Mars with organisms from Earth needed to be addressed. SLA-561 met this requirement by heat sterilization via prolonged exposure at 1600 F in vacuum. Post sterilization tests showed it had no impact on the mechanical properties [12]. With lower thermal conductivity and higher effective heat capacity compared to other ablative materials available at that time, with good recession characteristics, SLA-561V was an efficient ablator for the Viking missions and was later used on numerous Mars missions. A variant of SLA formulation, consisting of silicone resin and silica fillers without charring carbonaceous components, proved to be RF transparent for use on the backshell, and this also found its use in other missions.

SLA-561V heat shield manufacturing is traceable to that of Gemini and Apollo experiences. The preparation for bonding the flexible, fiber-glass phenolic honeycomb to the structure is similar to that of Apollo. Though SLA-561 formulation allows for its use without honeycomb, the rationale for use of honeycomb is similar to that of Apollo. It makes the heat shield "seamless" and, in addition, the honeycomb provides better char erosion characteristics. During Apollo program hand tamping left voids within each cell, and also, the pressure to push down the ablative compound sometimes damaged the structure underneath and so, Apollo developed the "gunning" process. Due to the difference in consistency / granularity, SLA-561V could not use "gunning" process and had to go back to hand tamping. Similar to Gemini and Apollo, SLA-561V was cured to rigid state prior to machining [13].

NASA's Viking Mission to Mars was composed of two spacecraft, Viking 1 and Viking 2, each consisting of an orbiter and a lander. The primary mission objectives were to obtain high resolution images of the Martian surface, characterize the structure and composition of the atmosphere and surface, and search for evidence of life. The Orbiter Spacecraft with the probe was inserted into orbit first and then the probe was delivered to the surface. Fig. 13 describes the sequence of events associated with landing. It was the first mission to have the heat shield and backshell separate prior to parachute deployment.

Successful landing occurred on July 20, 1976 (Viking 1) and September 3, 1976 (Viking 2). Project specifications called for a return of scientific data from the landers for a minimum of 90 days. Lander-1 collected data until the power shutdown on August 17, 1980 and the Lander-2, functioned until November 13, 1982. The heat shield

was not instrumented . We do not know how conservative the TPS was.
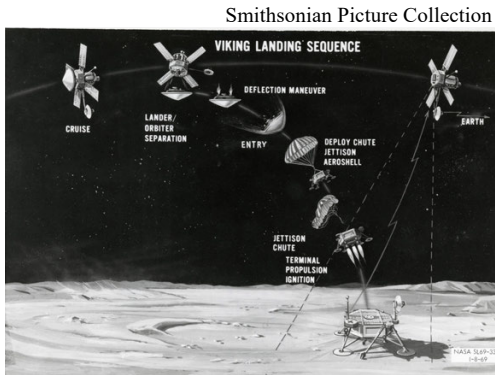
Fig. 13. Viking, the first inter planetary mission entry, descent and landing (EDL) sequence .

6.   Venus and Jupiter:

While Mars Viking entry was relatively benign compared to Apollo, entry conditions for the Pioneer-Venus and Jupiter Probes were radically different and required ablative material and systems that could withstand orders of magnitude more extreme entry conditions (heat-flux, pressure, shear, heat-load) than Apollo.

In January 1969, Goddard Space Flight Center published the results of its studies," A Venus Multiple-Entry Probe Direct-Impact Mission," under the Explorer Program and developed a project plan that scheduled the program to commence during fiscal year 1973 [15]. During this time, Ames Research Center had designed, fabricated, and tested a spacecraft and associated probe along with the instrument systems to demonstrate it in the Earth's atmosphere via the Planetary Atmosphere Experiments Test (PAET) in 1971. PAET demonstrated the capability of selected experiments to determine structure and composition of an unknown planetary atmosphere from a probe entering an atmosphere at very high speed. This was the type of practical data needed for the design of a probe mission into the atmosphere of Venus. In November of 1971, the Planetary Explorer program was discontinued at Goddard and transferred to Ames Research Center primarily due to the wealth of expertise Ames had developed in entry probes. The Pioneer Venus Science Steering Group established in 1972, came up with the proposal for two identical spacecraft, each spacecraft would consist of a bus, a large probe, and three small probes [15].

The probe relative entry velocity was ~11.5 km/s but the probes direct entry could range from entry flight path angle of -20^0 (shallow) to -90^0 (very steep). The probes were much smaller than Viking or Apollo, and the large probe was less than 1.5m diameter. Shock-layer radiation was significant for the Apollo mission with its entry close to 11 km/s at Earth and ablative material selection

needed to take this in to account. For Venus probes, due to Venus atmospheric composition (CO_2) , it was well known that the shock-layer radiation would be even more significant, and theoretical and experimental studies conducted in the 70's by Ames researchers pointed out the shock layer radiation could be equal to the convective heating [17]. As a result, the aeroshell geometry selected was to be a 45^0 sphere-cone for the forebody (shown in Fig. 14), in order to minimize the shock layer radiation on the conical frustum (large area), whereas the heating at the stagnation region could not be minimized. The nose radius could be tailored to balance between aerodynamic stability and heating. The design trade studies led to the final aeroshell shape, shown in Fig. 14, different than the Apollo or Viking heat shield shapes. The design relevant heating conditions from Ref [19] are shown for two bounding cases in Fig. 15.
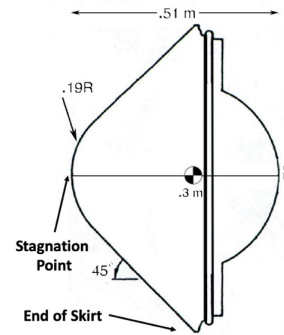

Fig. 14. Pioneer-Venus small probe aeroshell geometry. The large probe aeroshell is photographically scaled (twice as large).
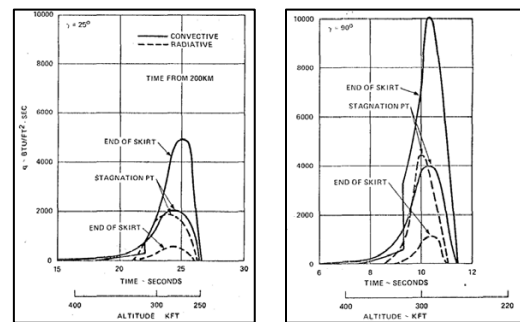

Fig. 15. Convective and Radiative heating for the large and small (right) probe at the stagnation point and at the end of skirt at different entry flight path angles at V – 11.54 km/s [20]

Trade studies were conducted to assess the viability of then available ablative TPS. The ablative TPS material had to withstand extreme total heating at all locations. The need to minimize development cost and to keep reasonable schedule demanded a common aeroshell shape with relatively minimal recession. The concept of

"reflecting" ablative heat shield had been studied earlier [18] in anticipation of Venus missions, and as a result, two concepts, one carbonaceous charring ablator, high density carbon-phenolic and Teflon, a dielectric reflective ablator, were evaluated [19]. The study performed by NASA Ames experts revealed that carbon phenolic loses very little mass, responding to the entry environment as a heat sink with its sizing controlled by the bond-line temperature. Teflon soaks little or no energy but loses significant mass as it was sized by the requirement that sufficient material be present to achieve reflection throughout the radiation pulse. GE conducted their own study with four different ablators and published results of their study showing the robustness of high-density carbon phenolic and relative insensitivity to uncertainties [20]. Orbital cold soak and entry thermal stress analyses provided further confidence in the application of carbon phenolic to this mission.
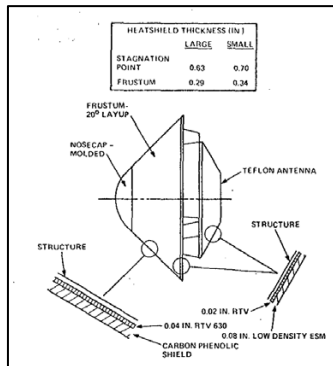

Fig. 16. Pioneer -Venus entry probe ablative TPS.

High density carbon phenolic manufacturing technique and material specifications were adopted from ballistic missile use for the conical frustum. The tape-wrapping technique, where a long strip of carbon fabric impregnated with phenolic resin is wrapped around at a wrap angle of 20 deg to the normal and then the mold is cured to result in a high density conical piece part that can be machined and bonded on to the structure. This technique could not be extended to the nose section. GE adopted another technique where small, pre-impregnated phenolic carbon-fabric squares were placed in a mold and cured into a cylindrical part at high pressure and temperature. This part can then be machined to form the spherical nose segment. The nose and the frustrum together are bonded to the structure.

The Pioneer-Venus mission was launched on May 1978 and had a single spacecraft bus with three small probes and one large probe. The small probes were identical and were 50% of the size of the large probe. On December 9, 1978, all the probes were released from the orbiter spacecraft as it approached Venus and all the probes entered at a relative velocity of 11.54 km/s, but the entry flight path angle varied from -25.4^0 to -65.7^0

[16]. All probes were successful in collecting data, and one small probe survived all the way to the surface and transmitted data from the surface for over an hour. The heat shields on all probes were instrumented with thermocouple plugs. The post-flight analysis came to the conclusion that the heat shield performed as expected, and the heat shield thickness was designed conservatively [21].

Galileo Probe mission: Jupiter was rated as the number one priority in the Planetary Science Decadal Survey published in the summer of 1968. In the 1970s *Pioneer 10* and *Pioneer 11*, and *Voyager 1* and *Voyager 2* spacecraft went past Jupiter. Around the same time, scientists were beginning to contemplate a deep space orbiter along with a probe mission to Jupiter. Alvin Seiff of NASA Ames had proposed back in 1969 that a Jupiter entry vehicle be designed and constructed (he later became the PI for the atmospheric structure experiment on the Galileo Mission). Work on the Galileo mission began around 1973 [22]. While there are similarities between the successful Pioneer Venus large probe and the Galileo Probe mission in terms of concept of operation, size and shape of the probe, the choice of TPS, etc., entry at Venus was benign compared to entry at Jupiter at nearly 47.4 km/s of relative velocity (and the inertial velocity was 60 km/s) by taking advantage of the rotation of the Jupiter rotation with a pro-grade trajectory. The gas composition of Jupiter required development of physics-based models for H_2/He. During the 70's, theoretical and experimental data and prediction of the of the shock layer radiation were undertaken by teams at NASA Langley and Ames, and these studies pointed to extreme convective and radiative environment.

Work on the spacecraft began at Jet Propulsion Laboratory in 1977. JPL designed and built the *Galileo* spacecraft and managed the *Galileo* mission. NASA Ames managed the descent probe, which was built by Hughes Aircraft Co. GE was responsible for the ablative TPS for the heat shield and back shell.

The definition of the entry heating environment and the design of heat shields was a major program effort. The work was carried out as a joint NASA-industry development with significant evolution of the basic inputs during the Probe design phase from 1978 to 1981 [24]. For both the forebody and afterbody, radiative transfer was the prime energy-transport mechanism. Because of the complexity of detailed heat shield performance calculations in the radiative regime for bodies with high ablation rates, NASA provided benchmark calculations for the design and the contractors at GE used it to develop and verify the results of engineering-type estimates [23,24]. An additional consideration included in the final heat-shield design was mechanical erosion (spallation) of heat-shield material. The predicted peak stagnation heat-flux was ~ 40 kW/cm^2 at 7 atm of stag. Pressure and the

integrated heat-load at stagnation was 200 kJ/cm^2. The estimated surface peak heat-flux [24] is shown in Fig. 17.

Fig. 18 from [23] makes the case that the entry conditions were extreme (note: the plot is in log-log scale) compared to Pioneer Venus or Apollo and were beyond existing facility capabilities.
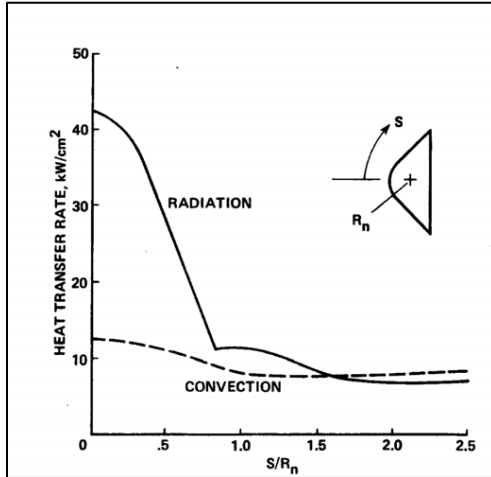

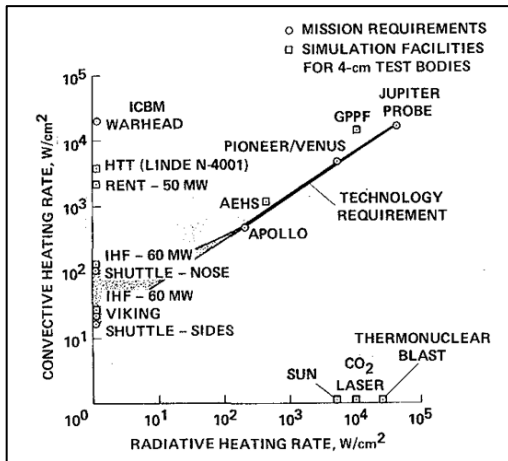Fig. 17. Predicted peak heat-flux on the surface of the Galileo probe during entry [24]

Fig. 18. Galileo, Pioneer Venus and Apollo heating environment in comparison to test facility capabilities [23]. Note the entry was more severe than an. ICBM warhead flying through a thermonuclear blast combined.

Though not adequate, small-scale (about 1-in.) samples were tested in the Giant Planet Facility at Ames Research Center as it could reach about 25 kW/cm^2 combined heating [23]. The Gas-dynamic Laser Facility at Ames was also used in heat-shield development tests, particularly in evaluating spallation effects [23]. Testing provided data with limited value, and it was difficult to develop confidence due to extrapolation. Testing at relevant conditions and relevant scale was impossible given the facility limitations. In addition, complex interaction between radiation and ablation products, turbulence coupled with spallation particulates in the flow, and massive surface blowing made flight predictions highly uncertain. The saving grace was the use of carbon phenolic, the only material that could withstand the anticipated entry environment however uncertain it was. Use of heritage carbon phenolic along with the expertise developed by GE in manufacturing the Pioneer Venus heat shield were leveraged [25]. The predicted peak heat-flux of a ~ 1kW/cm^2 on the backshell was sufficiently high to warrant phenolic nylon as the afterbody ablative material [24]. The final flight sizing of the carbon phenolic and phenolic nylon ablators is shown in Fig. 19.
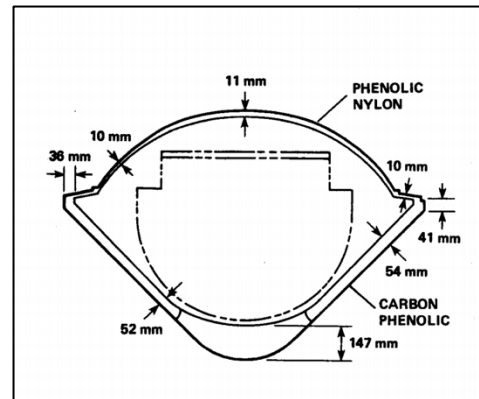

Fig. 19. The ablative TPS on Galileo Probe and the thicknesses [24].

The heat shield was made in two segments (chop-molded and tape-wrapped) and bonded together and then bonded to the forebody structure. The back shell TPS was also constructed in two segments (spherical dome and a ring) and then bonded to the structure [24]. Unlike Apollo, Mars Viking or Pioneer Venus, the Galileo heat shield was relatively massive (~ 50% of the mass of the entry mass).

Galileo, after multiple delays, finally launched on October 18, 1989. On July 13, 1995, Galileo's descent probe, which had been carried aboard the parent spacecraft, was released and began a five-month freefall toward Jupiter [22]. The Galileo aeroshell was instrumented with recession sensors as part of the atmospheric structure experiment. The flight measurement showed the recession to be very significant (50% mass loss in less than 15 sec). The comparison of the recession data with pre-flight prediction, shown in Fig. 20 makes it obvious: a) Design was overly conservative in the spherical nose region, b) TPS thickness was nonconservative near the shoulder, and c) mission came close to failure.
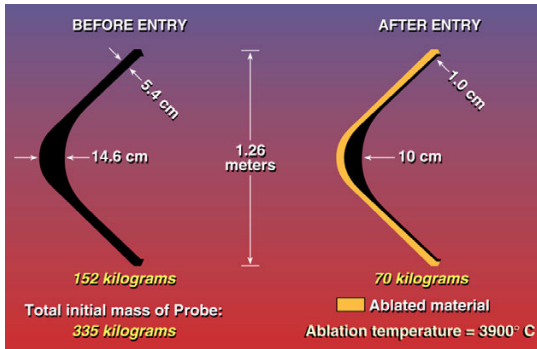
Fig. 20. Pre- and Post-flight comparison of the heat shield carbon phenolic thickness from flight measurements [25]

It is worth noting here that since the mid 1980's heritage carbon yarn has not been made due to pre-cursor (rayon) manufacturing plant was shut down and also, the rayon processing into carbon was banned, since then, due to toxic byproducts. A thirty-year gap in lack of deep space or Venus missions since Galileo and Pioneer Venus missions, also played into carbon phenolic atrophy in manufacturing capabilities as well as experience base.

7. Smaller Missions (1990 - 2005):

JPL in partnership with Lockheed Martin (LM), Denver led the following three missions that were low cost, used smaller aeroshell and driven by schedule to meet then NASA adage " Faster, Better and Cheaper." JPL/LM were willing to take risks. Ablators that have not been used for over two decades or new ablators under development were considered acceptable so as to maintain mission cadence.

Pathfinder: In 1991, NASA Ames published the Mars environmental survey (MESUR) feasibility study, which envisioned a series of 16 lightweight hard landers with Viking-based entry aeroshells to produce a Mars meteorological network. In mid-1992 and 1993 JPL asked Martin Marietta to provide cost and technical proposals based on MESUR, for a direct entry concept to place a small rover on the surface of Mars and Martin Marietta recommended a scaled version of the Viking aeroshell with SLA-561V as the heat shield ablator. This became the Mars Pathfinder Mission and was launched on December 4, 1996. SLA-561V manufacturing processes were 20 years out of date and required significant efforts to recover, find replacements for solvents and other components no longer available and Martin Marietta had to requalify SLA-561V [26]. In addition to manufacturing challenges, the peak stagnation point heat-flux was estimated to be 100 W/cm^2, and this also dictated the newly reconstituted SLA-561V be arc jet tested at these high conditions for the first time. The 1990 version of SLA-561V was tested in the NASA Ames arc jet

successfully up to 188 W/cm^2. The flight heat shield was instrumented with thermocouples and the successful Pathfinder landing on Mars provided limited temperature data at the stagnation point and at the shoulder. The comparison between flight data and prediction using the best estimated trajectory was in good agreement, and it established SLA-561V to be a more capable ablator.

Stardust: This was the first sample return mission. In the early 1990's NASA Ames, recognizing the need for light weight ablators, developed SIRCA and PICA. PICA, with a density of ~0.25 g/cc, performed well at conditions exceeding 1000 W/cm^2 and 0.5 atm pressure (higher than Apollo). In addition, NASA Ames was working closely with Fiber Materials Inc. and transferred the technology. In the mid 90's, Lockheed Martin (LM), Denver (Martin Marietta was merged into Lockheed) successfully proposed a low cost, sample return mission to bring back samples from comet Wiley [27]. The mission needed a light weight, efficient ablator capable of around 1000 W/cm^2. Though PICA was in early development stage, the risk tolerance was acceptable given NASA's philosophy of "Faster, Better, Cheaper" at that time, and PICA was selected.
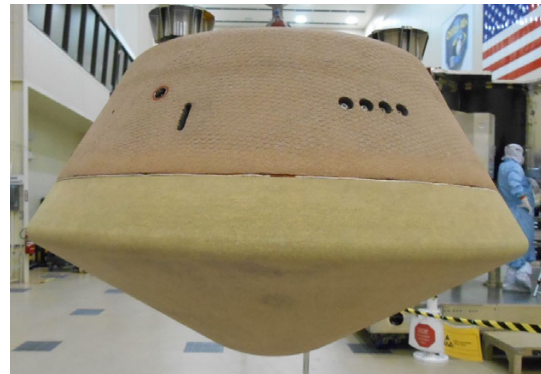

Fig. 21. OSIRIS-REx aeroshell with PICA heat shield and SLA-561V backshell (build-to-print to Stardust).

FMI Inc then developed the process of molding carbon FiberForm™ into a net casted shape and used the NASA Ames developed phenolic infusion process to make a single piece PICA heat shield. FMI, Inc. provided a machined heat shield that LM then bonded on to the heat shield structure. The afterbody ablative TPS was SLA-561V, which LM had resurrected and used it on the Pathfinder mission. The OSIRIS-REx sample return capsule (SRC) with PICA and SLA-561V [27], build-to-print aeroshell based on Stardust design, is shown in Fig. 21. The Stardust mission was launched on the 7th of February 1999 and returned samples on the 15th of January 2006. The PICA heat shield performed flawlessly.

Genesis: It was the second sample return mission that followed Stardust in mid 90's and was launched on August 8, 2001 and returned on September 8, 2004. The

objective was to collect samples of solar wind particles and return them to Earth for analysis. The aeroshell was 1.5 m diameter, twice as large as Stardust. Genesis considered PICA, but PICA required development to scale up from 0.8m to 1.5 m. LM did not have the time or money to develop a system from scratch, and an alternative was needed. Carbon-carbon TPS has been in development and use for DoD for some time. LM, leveraging the expertise of CCAT Inc., developed the heat shield for Genesis. In order to minimize the mass, the high-density carbon-carbon, with its high conductivity, has to be bonded to low density carbon tile (similar to Fiber-Form[TM]) and then on to the structure to be used as a seamless heat shield. SLA-561V was used on the backshell similar to its use on Stardust. One unique aspect is that Genesis, similar to Apollo, had to deal with heat shield penetration for integration with the spacecraft. A metallic heat-sink was used at the local attachment.

The mission was a partial success, as upon reentry, the parachute did not open and as a result, the aeroshell impacted the ground and broke apart. Post flight recovery and inspection showed the heat shield and the backshell performed flawlessly.

8. Mars Science Laboratory (MSL):

MSL, a JPL led mission, began in 2001 as the Mars Smart Lander project, a concept to land very large rover (compared to Pathfinder or MER), with a launch in 2007. However, budget cuts pushed the launch to 2009, and the mission was renamed Mars Science Laboratory. Work started in 2003, and it finally launched in 2011. MSL's goal was to land the largest ever rover, Curiosity, to investigate Mars' habitability, study its climate and geology, and collect data that can aid in future human exploration. The MSL design required the largest aeroshell ever built at the time (4.5m dia.), and LM proposed to build it with SLA-561V as the heat shield ablative TPS. MSL was heavier than all previous Mars missions, and it had to use lift in order to gain altitude for parachute deployment and subsequent precision landing. Generating lift meant flying at angle-of-attack and using GN&C to guide the aeroshell along a lift guided trajectory. Successive design iterations resulted in mass growth along with increased entry velocity, and increased heating due to turbulent augmentation on the leeward side of the heat shield. Trajectory design studies, including 6-DOF simulations, resulted in peak heat fluxes which started out around 140 W/cm^2 and exceeded 250 W/cm^2 at the time of Preliminary Design Review. SLA-561V had never flown at these conditions. Bernie Laub at NASA Ames led the effort to test SLA-561V in the arc jet at higher conditions (2004 – 2005), with instrumented coupons, up to 300 W/cm^2 (see Fig. 22) and the material performed well [28]. The Ames team developed a higher fidelity thermal response model for SLA-561V [28]. Hence there

was confidence that SLA-561V would be adequate for this demanding mission.

Fig. 22. SLA-561V arc jet tests with stagnation coupons. Left at 150 W/cm^2 and right at 300 W/cm^2. The appearance differs due to glass vaporization vs glass melt that occurs depending on the local conditions.

Around the time of Critical Design Review (2007), with the launch scheduled for 2009, testing of SLA-561V, to demonstrate adequate performance at a combination of heat-flux, pressure and shear conditions resulted in un-anticipated failure behavior [29]. Fig. 23 from Ref. [29] shows two images that show the anticipated and un-anticipated (failure) behaviors. Tests were repeated and yet SLA failed at lower than peak conditions.
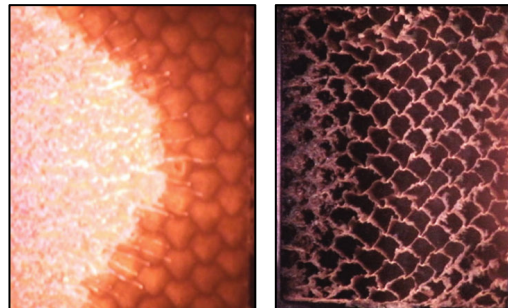

Fig. 23. Results from SLA-561V arc jet tests - wedge configuration. Picture on the left shows the anticipated behavior, after 45s, including melt flow at higher heat-flux and lower pressure. Picture on the right shows the failure in 4 s at lower heat-flux and higher pressure [29].

The arc jet does not create conditions that match all of the flight parameters, and "test as you fly" could not be achieved due to facility limitations. On the other hand, no testable hypothesis was obvious as to why SLA-561V failed. The MSL project had to find an alternate quickly. The Crew Exploration Vehicle (now known as Orion) TPS Advanced Development Project (CEV TPS ADP) had performed extensive testing of PICA as an option for Lunar return conditions and had substantial data. It had also established, at FMI, an expanded manufacturing

capability to produce PICA blocks, and had undertaken characterization of it. MSL quickly assessed and adopted PICA in 2007 and began addressing the challenges of integration of PICA tiles on to the heat shield.

Prior to MSL, all Mars missions had flown only with seamless heat shields. NASA's experience with tiles with seam was limited to Space Shuttle Orbiter. The CEV TPS ADP had acquired data that showed inadequate performance of the gap filler materials and did not have a gap design for Lunar conditions. Heat shields undergo deflections from launch through entry, and the flexing of the heat shield could crack the PICA tiles or the seam can open up to allow hot gas burrowing. The integrated system with PICA tiles had to be robust. MSL had to quickly solve the seam problem to avoid runaway failure. RTV proved to be a good gap filler based on the arc jet tests. By choosing a stiff, PICA compatible structure and by selecting the size of the PICA tile from the known mechanical properties, including variability, and by performing structural design, analysis and testing, a robust design was achieved [30].
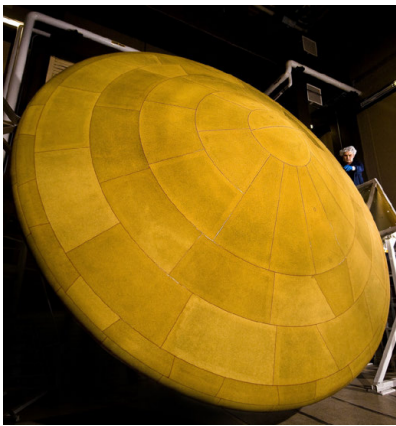

Fig. 24. MSL Tiled PICA heat shield

The tiled PICA heat shield, shown in Fig. 24 protected MSL during entry and performed its function flawlessly. Fortunately, NASA decided to instrument the heat shield with thermal plugs and pressure sensors [32]. Post-flight analysis showed PICA receded very little as the entry conditions at the end were benign (~ 100 W/cm^2) even though the system was designed for a heat-flux of ~ 250 W/cm^2. As a result of the MSL success, missions that need large aeroshell with design conditions close to MSL, such as Mars 2020, Dragonfly to Titan, and MSR Sample Retrieval Lander (SRL) are all planning to utilize tiled PICA.

9. Orion:

In January 2004, President George W. Bush announced a new Vision for Space Exploration that would return humans to the Moon by 2020 in preparation for human exploration of Mars. Dr. Mike Griffin, within a

month of becoming the NASA Administrator, initiated a 90-day Exploration Systems Architecture Study (ESAS) in May of 2005. The goal was to develop a reference architecture along with top-level requirements and configurations for crew and cargo launch systems in order to reduce the capability gap between Shuttle's retirement by 2010 and CEV's Initial Operations by 2014. In addition, the study was to identify key technologies required to enable and significantly enhance these reference exploration systems. The results of the study led to a CEV that is a scaled version of Apollo. The study recommended immediate investment to be made in the development of TPS, since the Apollo ablative TPS no longer existed and qualification of a new or replacement material would require extensive analysis and testing.

The CEV TPS Advanced Development Project (ADP) was established with the primary objective to develop a single heat shield design that met both lunar direct return and LEO return Earth entry requirements. NASA Ames was the lead center supported by 6 other NASA centers. The ADP screened eight different candidate ablative materials, from 5 different commercial vendors and this included SLA-561V, PICA and Avcoat. At the end of phase 1, two final candidates, Avcoat and PICA were selected for maturation. The maturation program involved fabrication and characterization, establishing thermal performance capability for Lunar and LEO, material thermo-mechanical performance capability, integrated heat shield and component design including system performance capability, and manufacturing of a 5m demonstration unit for both PICA and Avcoat. Since Avcoat was not proprietary to Textron, Textron provided all of the materials by recovering the Avcoat process and also manufactured the 5 m heat shield. For PICA, Boeing was competitively selected to manufacture the PICA heat shield with FMI Inc. supplying machined PICA.

There were many challenges. PICA proved to be a very capable thermal performance material. The primary challenge was the integrated system performance of a tiled arrangement, defining the gap width and finding suitable seam material for Lunar return conditions. Though there were promising solutions, the problem was not satisfactorily solved before the down selection. In addition, PICA bonding required strain isolation from the structure if the structure were incompatible and bond verification was an added complexity.

Recovering Apollo era Avcoat was problematic as well as some of the components that constituted the ablative material were no longer made or met the specification. The honeycomb was different. So, the reconstituted Textron Avcoat had to be certified through extensive testing. Manufacturing required re-creating the tools and training and certifying personnel involved in every step. The Apollo heat shield and backshell together required 270,000 cells to be filled and on the average, there were

30,000 cells defective (nearly 10%) that needed to be repaired. The Orion heat shield at 5m dia. had 300,000 cells and the defects were fewer but still required repair.

Fig. 25. PICA gap and seam testing (2008) with different seam materials [33]

In March of 2009, after a three-year effort in assembling and analysis of the data, a review was held that included LM, the prime contractor for Orion. The CEV TPS ADP recommended Textron's Avcoat as the better of the two options. LM took over the continued development and engineering of the Avcoat heat shield and Textron manufactured the heat shield.
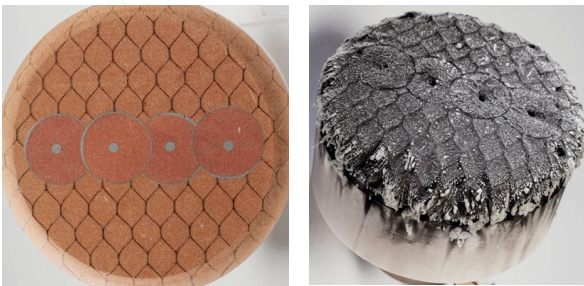

Fig. 26. Crack repaired pre-test coupon on the left and the post-test coupon on the right [34]

Similar to the Apollo experience, the Avcoat heat shield developed 28 cracks, primarily in the seam between honeycomb gores during curing due to stress concentration [34]. The crack repair procedure developed was tested in the arc jet tests and showed acceptable performance. The procedure was used to repair cracks prior to flight.

Cracks were predicted to occur during flight based on the material strength measured using witness coupons due to negative stress margin. Despite the lack of positive structural margins, based on experimental evidence from arc jet testing that the crack causes a very modest increase in the temperature at the bondline, the probability of catastrophic consequence of cracking was acceptably small and based on this the heat shield was qualified for the EFT-1 flight. On December 5, 2014, the EFT-1 flight

was executed successfully. The recovered heat shield was extensively cored, and the cores were studied.

Options for making changes to the Orion capsule were being studied even before the EFT-1 flight. After the successful first flight, Orion did make significant changes including to the heat shield [35]. The manufacturing was changed from a monolithic, individual cell injection process to a "Block Avcoat" process. In the block process, the Avcoat is cast and cured in blocks by eliminating the honeycomb from the system. The block process will allow production automation and reduce time and cost. Also, defects in the heat shield could be minimized, and repairs avoided. In addition, removal of the honeycomb increased the material properties performance and improved structural margin. The one disadvantage with the block approach is the gaps and seam, and the challenges were addressed by developing a seam and demonstrating through testing that the integrated system can withstand the Lunar return loads.

During early heat-shield development there was an area of concern that was addressed specifically for EFT-1 but not for Artemis 1, and it is connected with a local feature, the compression pads. During launch and space operations, the crew (CM) and service (SM) modules remain attached through the use of tension ties. Explosive bolts, (6 on EFT-1) are used to sever the tension ties prior to entry to allow the CM to separate from SM. The compression pad is a circular pad that allows the tension tie to pass through and is designed to function both as a structure prior to entry and an ablator during entry. Apollo used a fiberglass phenolic shingles system; the heritage material could not be found, and a carbon phenolic material system was developed and integrated with Avcoat. Similar to Apollo, Orion had to demonstrate that the differential recession between Avcoat and the compression pad would not adversely impact the system thermally as well as its thermo-structural performance. It was known, and post-flight test provided evidence, that the EFT-1 compression pad could not be extended to Artemis 1. A new material called 3-D MAT, a three-dimensionally woven quartz/cyanate-ester, was invented, tested, qualified and certified. This is now integrated on to Artemis 1 [36].

On a side note, SLA-561V arc jet tests performed at LEO return condition, which were similar to that of MSL entry conditions, led to the very first evidence that SLA-561V has failure mode at relatively benign conditions and were reported to MSL project. MSL switching from SLA-561V to PICA was described earlier.

The successful flight test of Artemis 1 will lead to crewed mission Artemis 2 in the near future. Orion is also intended to bring back astronauts returning from Mars by the use of Gateway under the Artemis program.

10. Looking to the Future – Beyond Moon and Mars

The Ice Giants and Saturn are high priority destinations for in-situ investigation [37]. NASA held two workshops in 2010 and 2012 at NASA Ames and reported out that the capability to make heritage carbon phenolic heat shield from the 1980's that enabled the Galileo mission had atrophied and also that the Galileo post flight analysis pointed to a need for a more efficient system for outer planet exploration. In 2014, NASA funded a technology development project called Heatshield for Extreme Entry Environment Technology (HEEET) based on exploratory studies done by NASA Ames personnel using 3-D weaving and resin infusion to construct a dual layer ablative TPS [38]. After five years, the HEEET project delivered a mature heat shield system ready for mission use to enable missions to Ice Giants, Saturn, Venus and sample return missions from Mars [39]. In addition, HEEET is being considered for the Mars Sample Return (MSR) mission Earth Entry Vehicle (EEV) as it offers robust thermal protection but also protection against micrometeoroid and orbital debris impact [40].

11. Concluding Remarks

To be successful, exploring our solar system with humans and robots has required vision and innovation, commitment to the development of technologies that form the backbone of complex integrated systems, and learning from both ground and flight tests. We have gained significant knowledge in the past 6 decades from the Apollo mission to the Mars Science Laboratory, from Stardust to OSIRIS-REx. Even with limitations of ground test facilities and challenges of interpreting flight-test results, we have developed an adequate understanding of how ablative materials behave and have engineered integrated systems that have led to these mission successes. Preparing for every mission, from Mercury onward, has required challenges to be overcome and the history of ablative TPS has taught us the following:

1. TPS cannot be allowed to fail.
2. System performance and system integration are far more important than optimized thermal performance or ablative TPS mass (Apollo, Orion, MSL).
3. Heritage ablative system could arguably be attractive at the beginning from cost and schedule perspective, can become problematic over time, due to limited understanding of the capability and/or evolving requirements (MSL and Orion).
4. Atrophy is a fact of life. Recovery can take substantial resources and time, and the recovered system will require re-certification.
5. Development of new ablative systems can be equally or more challenging.

References

[1] Anderson Jr, J. D. , "Hypersonic and high-temperature gas dynamics," American Institute of Aeronautics and Astronautics, 2006.

[2] Swenson, Loyd S., James M. Grimwood, and Charles C. Alexander. This new ocean: a history of Project Mercury. Vol. 4201. National Aeronautics and Space Administration, NASA History Office, Office of Policy and Plans, 1998.

[3] H. Julian Allen and Alfred J. Eggers, Jr., "A Study of the Motion and Aerodynamic Heating of Ballistic Missiles Entering the Earth's Atmosphere at High Supersonic Speeds," NACA Tech. Report 1381, Forty-Fourth Annual Report of the NACA-1958 (Washington, 1959}, 1125-1140.

[4] G.W. Sutton, " The Initial Development of Ablation Heat Protection, An Historical Perspective," Invited talk on History of Key Technologies paper as part of AIAA's 50th Anniversary celebration, AIAA 82-4038, Journal of Spacecraft, Vol. 19, No. 1, Jan.-Feb. 1982.

[5] https://en.wikipedia.org/wiki/Project_Mercury

[6] Malik, P. W., and G. A. Souris. "Project Gemini-A technical summary," NASA CR-1106 Mcdonnell Douglas Corporation, 1968.

[7] Grimwood, J.M., Hacker, B.C., and Vorzimmer, P. "Project Gemini – Technology and Operations – A Chronology," NASA SP-4002, 1969.

[8] Erb, R. B., Greenshields, D. H., Chauvin, L. T., Pavlosky, J. E., & Statham, C. L., "Apollo thermal-protection system development. Journal of Spacecraft and Rockets," 1970, 7(6), 727-734.

[9] Pavlosky, J. E., und Leger, L. G.. "Apollo Experience Report - Thermal Protection Subsystem," NASA TN D-7564, National Aeronautics and Space Administration Washington, D.C., January 1974.

[10] Textron, Inc. " 1000 Seconds to Home: Apollo Heat Shield," July 2013.
https://www.youtube.com/watch?v=64UM3CUqSfg

[11] Ezell, E.C., and Ezell, L.N., " On Mars: Exploring the Red Planet, 1958-1978", NASA SP-4212, NASA History Series, 1984.

[12] Strauss, Eric L. "Superlight ablative systems for Mars lander thermal protection." Journal of Spacecraft and Rockets 4, no. 10 (1967): 1304-1309.

[13]https://www.lockheedmartin.com/content/dam/lockheed-martin/space/documents/thermal/SLA-561%20Product%20Information.pdf

[14] Willcockson, W., "Reentry Thermal Protection Systems – NASA Roadmap Feedback," presented at the

National Academy of Sciences meeting, March, 2011, http://sites.nationalacademies.org/cs/groups/depssite/documents/webpage/deps_063892.pdf

[15] Fimmel, R. O., Colin, L. and Burgess, E., "Pioneer Venus", NASA SP-461, 1983.

[16] Planetary Mission Entry Vehicles Quick Reference Guide, NASA/SP-2006-3401, 2006.

[17] Page, W. A. and Woodward, H. T., "Radiative and Convective Heating During Venus Entry", AIAA Journal, Vol. 10, No. 10, Oct. 1972, pp.1379-1381.

[18] Nachtsheim, P. R., Peterson, D. L., and Howe, J. T., "Reflecting Ablative Heat Shields for Radiative Environments," AAS Paper 71-47, Seattle, Wash., 1971

[19] Peterson, D.L, and Nicolett, W.E., " Heat Shielding for Venus Entry Probes," AIAA Paper 73-712, AIAA 8th Thermophysics Conference, Palm Springs, Calif., July 16-18, 1973.

[20] Brewer, R.A.," Selection Of A Heat Protection System For Venusian Entry," AIAA Paper 75-731, AIAA 10th Thermophysics Conference, Denver, Colorado, May 1975.

[21] W. C. PITTS AND R. M. WAKEFIELD, "Performance of Entry Heat Shields on Pioneer Venus Probes," Journal of Geophysical Research, Vol. 85, No. A13, Pages 8333-8337, DECEMBER 30, 1980

[22] Meltzer, M.," MISSION TO JUPITER: A History of the Galileo Project," NASA SP-2007-4231, NASA History Division, NASA, Washington, DC 2007.

[23] Howe, J. T., Pitts, W. C., and Lundell, J. H., "Survey of the Supporting Research and Technology for the Thermal Protection of the Galileo Probe," AIAA Paper 81-1068, Palo Alto, Calif., 1981.

[24] Givens, J., L. Nolte, and L. Pochettino. "Galileo Atmospheric Entry Probe System-Design, development, and test." AIAA-83-0098, presented at the 21st Aerospace Sciences Meeting, Reno, NV, 1983.

[25] Milos, Frank S. "Galileo probe heat shield ablation experiment." *Journal of Spacecraft and Rockets* 34, no. 6 (1997): 705-713.

[26] Willcockson, W. H., "Mars pathfinder heatshield design and flight experience." *Journal of Spacecraft and Rockets* 36, no. 3 (1999): 374-379.

[27] Willcockson, William H. "Stardust sample return capsule design experience." *Journal of Spacecraft and Rockets* 36, no. 3 (1999): 470-474

[28] Laub, B., Chen, Y. K., and Dec, J., "Development of a High-Fidelity Thermal/Ablation Response Model for SLA-561V," AIAA Thermophysics Conference, AIAA Paper 2009-4232, June 2009.

[29] Driver, D., et al., "Arc Jet Testing in a Shear Environment for Mars Science Laboratory Thermal Protection System," AIAA Thermophysics Conference, AIAA Paper 2009-4230, June 2009.

[30] Slimko, E., Szalai, C., and Hoffman, P., "MSL heatshield development: From Failure to Success," IEEE Aerospace Conference, 2011.

[31] Szalai, C., Slimko, E., & Hoffman, P., "Mars science laboratory heatshield development, implementation, and lessons learned," Journal of Spacecraft and Rockets, 51(4), 1167-1173.

[32] Munk, M., Little, A., Kuhl, C., Bose, D., and Santos, J., "Mars Science Laboratory Entry, Descent, and Landing (MEDLI) Hardware," 23rd AAS/AIAA Space Flight Mechanics Meeting, American Astronautical Soc. Paper AAS-2013-310, Feb. 2013.

[33] Kowal, T. J., "Thermal protection system (heat shield) development-advanced development project," JSC Commercial Human Space Flight Symposium October 13-14, 2010. https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20100036525.pdf

[34] Vander Kam, J., "Burn to Shine: Experiences and Lessons from the Orion Heat Shield," NASA Summer Series, June 2015.

https://www.youtube.com/watch?v=ccJ6LpnSK20

[35] Norris, S. D., Marshall, P., & Cichan, T., "Orion: lessons Learned from EFT-1 and EM-1, AA-2, and EM-2 status," AIAA SPACE Forum, 13 - 16 September 2016, Long Beach, California 2016.

[36] Feldman, J. D., Venkatapathy, E., Wilkinson, C., and Mercer, K. J., "Development of an Ablative 3D Quartz/Cyanate Ester Composite for the Orion Spacecraft Compression Pad," SAMPE TP15-0195, Composites and Advanced Materials Expo, Dallas Convention Center, Dallas, TX, Oct. 2015.

[37] Committee on the Planetary Science Decadal Survey, "*Vision and Voyages: For Planetary Science in the Decade 2013-2022*," National Academies Press, 2011.

[38] Venkatapathy, E., Ellerby, D., Stackpoole, M., Peterson, K., Gage, P., Beerman, A., Blosser, M., Chinnapongse, R., Dillman, R., Feldman, J. and Gasch, M., "Heatshield for Extreme Entry Environment Technology (HEEET)," presented at the VEXAG meeting, 2013.

[39] Venkatapathy, Ethiraj, D. Ellerby, P. Gage, M. Gasch, H. Hwang, D. Prabhu, M. Stackpoole, and Paul Wercinski. "TPS for Outer Planets," presented at the OPAG meeting, 2018.