Thermal Protection Systems

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The Space Shuttle design presented many thermal insulation challenges. The system not only had to perform well, it had to integrate with other subsystems. The Orbiter's surfaces were exposed to exceedingly high temperatures and needed reusable, lightweight, low-cost thermal protection. The vehicle also required low vulnerability to orbital debris and minimal thermal conductivity. NASA decided to bond the Orbiter's thermal protection directly to its aluminum skin, which presented an additional challenge.

The External Tank required insulation to maintain the cryogenic fuels, liquid hydrogen, and liquid oxygen as well as to provide additional structural integrity through launch and after release from the Orbiter.

The challenge and solutions that NASA discovered through tests and flight experience represent innovations that will carry into the next generation of space programs.

Orbiter Thermal Protection System

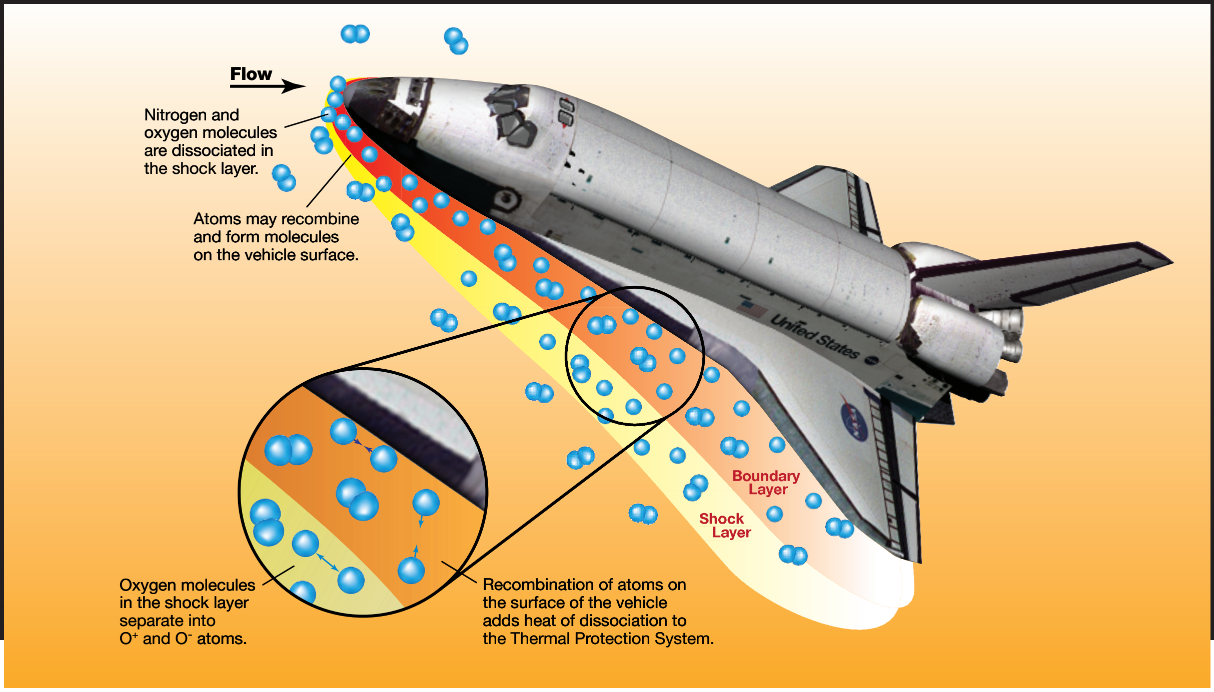
Throughout the design and development of the Space Shuttle Orbiter Thermal Protection System, NASA overcame many technical challenges to attain a reusable system that could withstand the high-temperature environments of re-entry into Earth's atmosphere. Theodore von Karman, the dean of American aerodynamicists, wrote in 1956, "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." He was referring to protecting the intercontinental ballistic missile nose cones. Fifteen years later, the shuttle offered considerably greater difficulties. It was vastly larger. Its thermal protection had to be reusable, and this thermal shield demanded both light weight and low cost. The requirement for a fully reusable system meant that new thermal protection materials would have to be developed, as the technology from the previous Mercury, Gemini, and Apollo flights were only single-mission capable.

Engineers embraced this challenge by developing rigid silica/alumina fibrous materials that could meet the majority of heating environments on windward surfaces of the Orbiter. On the nose cap and wing leading edge, however, the heating was even more extreme. In response, a coated carbon-carbon composite material was developed to

Thermal Protection System Could Take the Heat

Orbiter remained protected during catalytic heating.

While the re-entry surface heating of the Orbiter was predominantly convective, sufficient energy in the shock layer dissociated air molecules and provided the potential for additional heating. As the air molecules broke apart and collided with the surface of the vehicle, they recombined in an exothermic reaction. Since the surface acted as a catalyst, it was important that the interfacing material/coating have a low propensity to augment the reaction. Atomic recombination influenced NASA's selection of glass-type materials, which have low catalycity and allowed the surface of the Orbiter to reject a majority of the chemical energy. Engineers performed precise arc jet measurements to quantify this effect over a range of surface temperatures for both oxygen and nitrogen recombination. This resulted in improved confidence in the Thermal Protection System.



form the contours of these structural components. NASA made an exhaustive effort to ensure these materials would operate over a large spectrum of environments during launch, ascent, on-orbit operations, re-entry, and landing.

Environments

During re-entry, the Orbiter's external surface reached extreme temperatures—up to 1,648°C (3,000°F). The Thermal Protection System was designed to provide a smooth, aerodynamic surface while protecting the underlying metal structure from excessive temperature. The loads endured by the system included launch acoustics, aerodynamic loading and associated structural deflections, and on-orbit temperature variations as well as natural environments such as salt fog, wind, and rain. In addition, the Thermal Protection System had to resist pyrotechnic shock loads as the Orbiter separated from the External Tank (ET).

The Thermal Protection System consisted of various materials applied externally to the outer structural skin of the Orbiter to passively maintain the skin within acceptable temperatures, primarily during the re-entry phase of the mission. During this phase, the Thermal Protection System materials protected the Orbiter's outer skin from exceeding temperatures of 176°C (350°F). In addition, they were reusable for 100 missions with refurbishment and maintenance. These materials performed in temperatures that ranged from -156°C (-250°F) in the cold soak of space to re-entry temperatures that reached nearly 1,648°C (3,000°F). The Thermal Protection System also withstood the forces induced by deflections of the Orbiter airframe as it responded to various external environments.

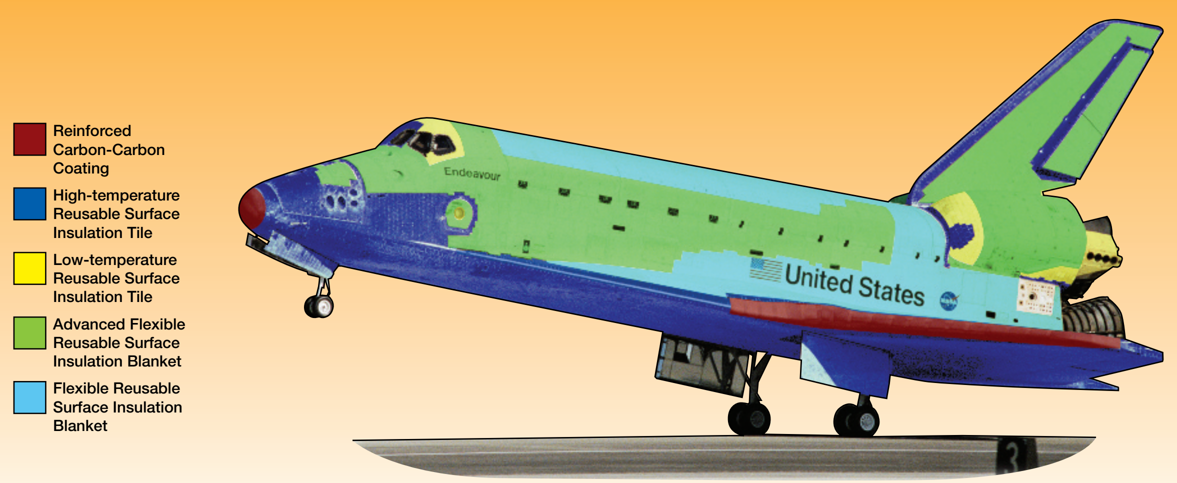
At the vehicle surface, a boundary layer developed and was designed to be laminar—smooth, nonturbulent fluid flow. However, small gaps and discontinuities on the vehicle surface could cause the flow to transition from laminar to turbulent, thus increasing the overall heating. Therefore, tight fabrication and assembly tolerances were required of the Thermal Protection System to prevent a transition to turbulent flow early in the flight when heating was at its highest.

Requirements for the Thermal Protection System extended beyond the nominal trajectories. For abort scenarios, the systems had to continue to perform in drastically different environments. These scenarios included: Return-to-Launch Site; Abort Once Around; Transatlantic Abort Landing; and others. Many of these abort scenarios increased heat load to the vehicle and pushed the capabilities of the materials to their limits.

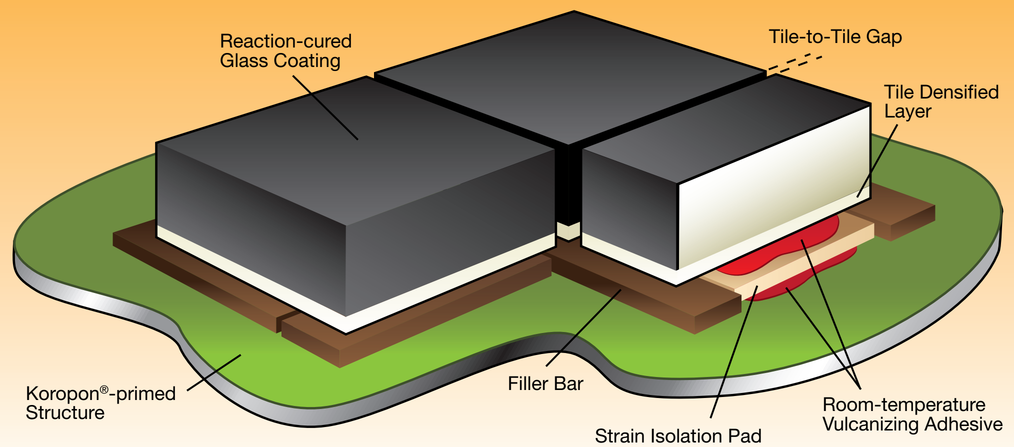
Thermal Protection System Materials

Several types of Thermal Protection System materials were used on the Orbiter. These materials included tiles, advanced flexible reusable surface insulation, reinforced carbon-carbon, and flexible reusable surface insulation. All of these materials used high-emissivity coatings to ensure the maximum rejection of incoming convective heat through radiative heat

Orbiter Tile Placement System Configuration



Orbiter Tile Attachment System High-temperature Reusable Surface Insulation



transfer. Selection was based on the temperature on the vehicle. In areas in which temperatures fell below approximately 1,260°C (2,300°F), NASA used rigid silica tiles or fibrous insulation. At temperatures above that point, the agency used reinforced carbon-carbon.

Tiles

The background to the shuttle's tiles lay in work dating to the early 1960s at Lockheed Missiles & Space Company. A Lockheed patent disclosure provided the first description of a reusable insulation made of ceramic fibers for use as a re-entry vehicle heat shield. In other phased shuttle Thermal Protection System development efforts, ablatives and hot structures were the early competitors. However, tight cost constraints and a strong desire to build the Orbiter with an aluminum airframe pointed toward the innovative, lightweight, and reusable insulation material that could be bonded directly to the airframe skin.

NASA used two categories of Thermal Protection System tiles on the Orbiter—low- and high-temperature reusable surface insulation. Surface coating constituted the primary difference between these two categories. High-temperature reusable surface insulation tiles used a black borosilicate glass coating that had an emittance value greater than 0.8 and covered areas of the vehicle in which temperatures reached up to 1,260°C (2,300°F). Low-temperature reusable surface insulation tiles contained a white coating with the proper optical properties needed to maintain the appropriate on-orbit temperatures for vehicle thermal control purposes. The low-temperature reusable surface insulation tiles covered areas of the vehicle in which temperatures reached up to 649°C (1,200°F).

The Orbiter used several different types of tiles, depending on thermal requirements. Over the years of the program, the tile composition changed with NASA's improved understanding of thermal conditions. The majority of these tiles, manufactured by Lockheed Missiles & Space Company, were LI-900 (bulk density of 144 kg/m^3 [9 pounds/ft^3]) and LI-2200 (bulk density of 352 kg/m^3 [22 pounds/ft^3]). Fibrous Refractory Composite Insulation tiles helped reduce the overall weight and later replaced the LI-2200 tiles used around door penetrations. Alumnia Enhanced Thermal Barrier was used in areas in which small particles would damage fragile tiles. As part of the post-Columbia Return to Flight effort, engineers developed Boeing Rigidized Insulation. Overall, the major improvements included reduced weight, decreased vulnerability to orbital debris, and minimal thermal conductivity.

Orbiter tiles were bonded using strain isolation pads and room-temperature vulcanizing silicone adhesives. The inner mold line of the tile was densified prior to the strain isolation pad bond, which aided in the uniform distribution of the stress concentration loads at the tile-to-strain isolation pad interface. The structure beneath the tile-to-tile gaps was protected by filler bar that prevented gas flow from penetrating into the tile bond line. NASA used gap fillers (prevented hot air intrusion and tile-to-tile contact) in areas of high differential pressures, extreme aero-acoustic excitations and to passivate over-tolerance step and gap conditions. The structure used for the bonding surface was, for the most part, aluminum; however, several other substrates used included graphite epoxy, beryllium, and titanium.

Design Challenges

Determining the strength properties of the tile-to-strain isolation pad interface was no small feat. The allowable strength for the interface was approximately 50% less than the LI-900 tile material used on the Orbiter. This reduction was caused by stress concentrations in the reusable surface insulation because of the formation of "stiff spots" in the strain isolation pad by the needling felting process. Accommodating these stiff spots for the more highly loaded tiles was met by locally densifying the underside of the tile. NASA applied a solution of colloidal silica particles to the non-coated tile underside and baked in an oven at 1,926°C (3,500°F) for 3 hours. The densified layer produced measured about 0.3 cm (0.1 in.) in thickness and increased the weight of a typical 15-by-15-cm (6-by-6-in.) tile by only 27 grams (0.06 pounds). For load distribution, the densified layer served as a structural plate that distributed the concentrated strain isolation pad loads evenly into the weaker, unmodified reusable surface insulation tiles.

NASA faced a greater structural design challenge in the creation of numerous unique tiles. It was necessary to design thousands of these tiles that had compound curves, interfaced with thermal barriers and hatches, and had penetrations for instrumentation and structural access. The overriding challenge was to ensure the strength integrity of the tiles had a probability of tile failure of no greater than $1/10^8$. To accomplish this magnitude of system reliability and still minimize the weight, it was necessary to define the detailed loads and environments on each tile. To verify the integrity of the Thermal Protection System tile design, each tile experienced stresses induced by the following combined sources:

- Substrate or structure out-of-plane displacement
- Aerodynamic loads on the tile
- Tile accelerations due to vibration and acoustics
- Mismatch between tile and structure at installation
- Thermal gradients in the tile
- Residual stress due to tile manufacture
- Substrate in-plane displacement

Other Thermal Protection System Materials? NASA had it Covered.

Flexible Reusable Surface Insulation

White blankets made of coated Nomex® Felt Reusable Surface Insulation protected areas where surface temperatures fell below 371°C (700°F). The blankets were used on the upper payload bay doors, portions of the mid-fuselage, and on the aft fuselage sides.

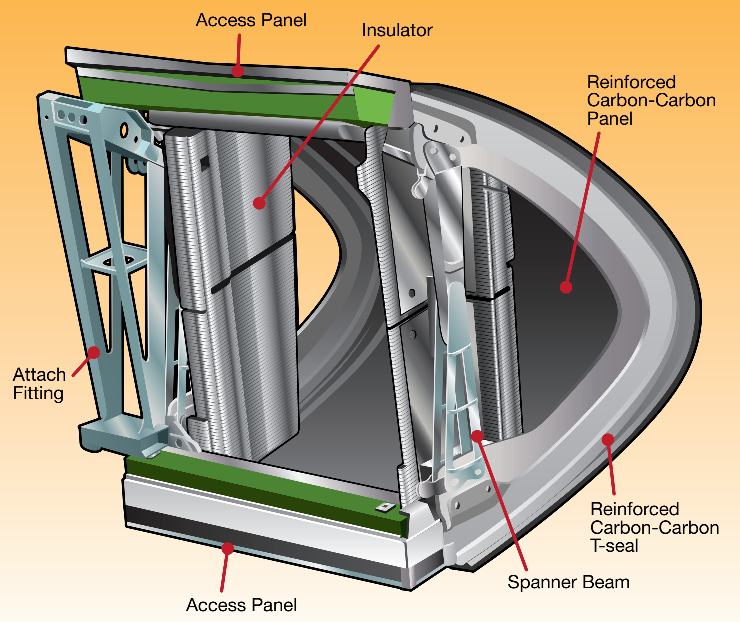
Advanced Flexible Reusable Surface Insulation

After initial delivery of Columbia to the assembly facility, NASA developed an advanced flexible reusable surface insulation consisting of composite quilted fabric insulation batting sewn between two layers of white fabric. The insulation blankets provided improved producibility and durability, reduced fabrication and installation time and costs, and reduced weight. This insulation replaced the majority of low-temperature reusable surface insulation tiles on two of the shuttles: Discovery and Atlantis. Following Columbia's seventh flight, the shuttle was modified to replace most of the low-temperature reusable surface insulation tiles on portions of the upper wing. For Endeavour, the advanced flexible reusable surface insulation was directly built into the shuttle.

Additional Materials

NASA used additional materials in other areas of the Orbiter, such as in thermal glass for the windows, Inconel® for the forward Reaction Control System fairings, and elevon seal panels on the upper wing. Engineers employed a combination of white and black pigmented silica cloth for thermal barriers and gap fillers around operable penetrations such as main and nose landing gear doors, egress and ingress flight crew side hatch, umbilical doors, elevon cove, forward Reaction Control System, Reaction Control System thrusters, mid-fuselage vent doors, payload bay doors, rudder/speed brake, and gaps between Thermal Protection System tiles in high differential pressure areas.

Orbiter Wing Panel Assembly



Reinforced Carbon-Carbon

The temperature extremes on the nose cap and wing leading edge of the Orbiter required a more sophisticated material that would operate over a large spectrum of environments during launch, ascent, on-orbit operations, re-entry, and landing. Developed by the Vought Corporation, Dallas, Texas, in collaboration with NASA, reinforced carbon-carbon formed the contours of the nose cap and wing leading edge structural components.

Reinforced carbon-carbon is a composite made by curing graphite fabric that has been pre-impregnated with phenolic resin laid up in complex shaped molds. After the parts are rough trimmed, the resin polymer is converted to carbon by pyrolysis—a chemical change brought about by the action of heat. The part is then impregnated with furfuryl alcohol and pyrolyzed multiple times to increase its density with a resultant improvement in its mechanical properties.

Since carbon oxidizes at elevated temperatures, a silicon carbide coating is used to protect the carbon substrate. Any oxidation of the substrate directly affects the strength of the material and, therefore—in the case of the Orbiter—had to be limited as much as possible to ensure high performance over multiple missions. Silicon carbide is formed by converting the outer two plies of the carbon-carbon material through a diffusion coating process, resulting in a stronger coating-to-substrate interlaminar strength.

As a result of the silicon carbide formation, which occurs at temperatures of 1,648°C (3,000°F), craze cracks develop in the coating on cool-down as the carbon substrate and coating have a different coefficient of thermal expansion. Impregnating the carbon part with tetraethyl orthosilicate and applying a brush-on sealant provides additional protection against oxygen paths to the carbon from the craze cracks.

The tetraethyl orthosilicate is applied via a vacuum impregnation with the intent of filling any remaining porosity within the part. Once the tetraethyl orthosilicate has cured, a silicon dioxide residue coats the pore walls throughout the part, thus inhibiting oxidation. After the tetraethyl orthosilicate process is complete, a sodium silicate sealant is brushed onto the surface of the reinforced carbon-carbon. The sealant fills in the craze cracks and, once cured, forms a glass. The craze cracks close at high temperatures and the sealant will flow onto the surface; however, since there is sufficient viscosity, the sealant remains on the part. When the reinforced carbon-carbon cools down, the glass fills back into the craze crack.

Why Reinforced Carbon-Carbon?

The functionality of the reinforced carbon-carbon is largely due to its ability to reject heat by external radiation (i.e., giving off heat from surface to the surroundings) and cross-radiation, which is the internal reinforced carbon-carbon heat transfer between the lower and upper structures. Reinforced carbon-carbon has an excellent surface emissivity and can reject heat by radiating to space similar to the other Thermal Protection Systems. It is designed as a shell section with an open interior cavity that promotes cross-radiation.

Since the highest heating is biased toward the lower surface, heat can be cross-radiated to the cooler upper surfaces, thus reducing temperatures of the lower windward surface. Another benefit is that the thermal gradients across the part are minimized.

While reinforced carbon-carbon is designed to withstand high temperatures and maintain its structural shape, the material has a relatively high thermal conductivity so it did not significantly inhibit the heat flow to reach the internal Orbiter wing structure. The metallic attachments that mated the reinforced carbon-carbon to the wing structure were crucial for accommodating the thermal expansion of reinforced carbon-carbon and maintaining a smooth outer mold line of the vehicle. Protecting these attachments and the spar structure itself required internal insulation. Incoflex®, an insulative batting encased by a thin Inconel® foil, protected the metal structural components from the internal cavity radiation environment.

Certification

Prior to the Orbiter's first flight, NASA performed extensive test and analysis to satisfy all requirements related to the natural and induced environments. The space agency accomplished certification of the wing leading edge subsystem for flight by analyses verified with development and qualification tests conducted on full-scale hardware. Engineers performed subscale testing to establish thermal and mechanical properties, while full-scale testing ensured the system performance and provided the necessary data to correlate analytical models. This included a full-scale nose cap test article and twin wing leading edge panel configuration tested through multiple environments (i.e., acoustic/vibration, static loads, and radiant testing). Full-scale testing ensured that the metallic mechanisms worked in concert with the hot structure as a complete system in addition to meeting the multi-mission requirements.

Reinforced Carbon-Carbon Flight Experience Lessons Learned

While NASA confirmed the fundamental concepts and design sufficiency through the wing leading edge subsystem certification work and early flight test phase of the Space Shuttle Program, the agency also identified design deficiencies. In most cases, modifications rectified those deficiencies. These modifications included addressing the gap heating between the reinforced carbon-carbon and reusable surface insulation to inhibit hot gas flow-through and retrofitting hardware to the wing leading edge subsystem design to account for a substantial increase in the predicted airloads. With increasing design environment maturity, temperature predictions on the attach fittings were significantly lowered, which allowed a design change from steel to titanium and a weight reduction of 136 kg (300 pounds).

Over the 30 years of flight, the shuttle encountered many anomalies that required investigative testing and analysis. Inspections revealed several cracks in the T-seals—i.e., components made of reinforced carbon-carbon that fit between reinforced carbon-carbon panels that allowed for thermal expansion of those components while keeping a smooth outer mold line. The cracks were later found to be caused by convoluted plies from the original layup of the T-seals. NASA corrected the cracking by modifying the manufacturing techniques and implementing additional inspections. In 1993, the agency identified small pinholes that went down to the carbon substrate and were subsequently traced to a change in maintenance of the launch pad structure. Engineers altered the silica/cement topcoat over the zinc primer such that zinc particles were able to come into contact with the wing leading edge and react with the silicon carbide coating during re-entry, thereby forming pinholes. NASA developed criteria for the pinholes as well as vacuum heat clean and repair methods.

Improved Damage Assessment and Repair With Return to Flight After Columbia Accident

NASA performed rigorous testing and analysis on the Thermal Protection System materials to adequately identify risks and to mitigate failure as much as practical. Engineers developed impact testing, damage-tolerance assessments, and inspection and repair capabilities as part of the Return to Flight effort.

Impact Testing

The greatest lesson learned was that failure of the reinforced carbon-carbon and the catastrophic loss of the vehicle was caused by a large piece of foam debris that was liberated from the ET.

While modifications to the thermal protection foam on the tank reduced the risk of shedding large debris during launch, NASA still expected smaller-sized debris shedding. It was critical that engineers understand the impact of foam shedding on the Orbiter's wing leading edge and tiles. The Southwest Research Institute, San Antonio, Texas, conducted many of these impact tests to understand the important parameters that governed structural failure of reinforced carbon-carbon and tile materials. Additionally, NASA developed finite element modeling capabilities to derive critical-damage thresholds.

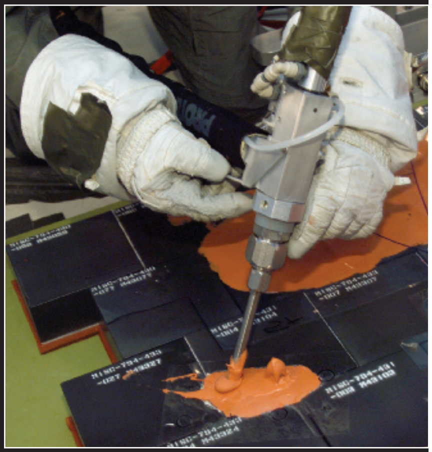
Tile Repair—A Critical Capability Was Developed

Prior to the first shuttle launch, NASA recognized the need for a capability to repair tiles on orbit. The loss of a tile during launch due to an improper bond posed the greatest threat. In response, NASA prioritized the development of an ablative material, MA-25S, for repairs of missing or damaged tiles. The biggest obstacle, however, was finding a stable work platform. Thus, NASA cancelled the early repair effort in 1979.

After the Columbia accident in 2003, NASA prioritized tile repair capability. Prior to the Columbia accident, the inspections after every flight revealed damage greater than 2.5 cm (1 in.) in approximately 50 to 100 locations. The original ablative material formed the basis for the repair material developed in the Return to Flight effort.

Some reformulation of MA-25S began in 2003. At that time, NASA changed the name of the material to Shuttle Tile Ablator, 865 kg/m^3 (54 pounds/ft^3) (STA-54). This material decreased the amount of swell during re-entry while maintaining a low enough viscosity to dispense with the extravehicular activity hardware. The material did not harden and would remain workable for approximately 1 hour but still cured within 24 hours in the on-orbit environments.

Simulating a damaged shuttle tile created dust that prevented the STA-54 from penetrating the surface of the tiles. This led to the development of additional materials: a gel cleaning brush that was coated with a sticky silicone substance used to clean tile dust from the repair cavity prior to filling; and primer material that provided a contact surface to which the STA-54 could adhere. Once the primer was cured, the bond strength was stronger than the shuttle tile.



Ground test of Orbiter tile repair.

Finally, NASA performed an on-orbit experiment during STS-123 (2008). Crew member Michael Foreman dispensed STA-54 into several damaged tile specimens. The on-orbit experiment was a success, showing that the material behaved exactly as it had during vacuum dispenses on the ground.

Damage Tolerance Criteria

To make use of the inspection data, NASA developed criteria for critical damage. Damage on reinforced carbon-carbon ranged from spallation (i.e., breaking up or reducing) of the silicon carbide coating to complete penetration of the substrate. Tiles could be gouged by ascent debris to varying depths with a wide variety of cavity shapes. The seriousness of any given damage was highly dependent on local temperature and pressure environments. NASA initiated an extensive Arc Jet test program during Return to Flight activities to characterize the survivability of multiple damage configurations in different environments. Testing in an Arc Jet facility provided the closest ground simulation for the temperature and chemical constituents of re-entry. Engineers performed numerous tests for both reinforced carbon-carbon and tile to establish damage criteria and verify newly developed thermal math models used for real-time mission support.

Inspection Capability

NASA developed an inspection capability to survey the reinforced carbon-carbon and tile surfaces. This capability provided images to assess any potential impact damages from ascent and orbital debris. A boom with an imagery sensor package attached to the Shuttle Robotic Arm was used to perform the inspection. The sensor package contained two laser imaging systems and a high-resolution digital camera. Additionally, astronauts residing on the International Space Station (ISS) photographed the entire Orbiter as it executed an aerial maneuver, similar to a backflip, 182 m (600 ft) from the ISS. The crew transmitted photographs to Houston, Texas, where engineers on the ground evaluated the images for any potential damage.

NASA employed an additional detection system to gauge threats from ascent and on-orbit impacts to the wing leading edge. As part of preparing the

Reinforced Carbon-Carbon Repair—Damage Control in the Vacuum of Space

Following the Space Shuttle Columbia accident in 2003, a group of engineers and scientists gathered at Johnson Space Center to discuss concepts for the repair of damaged reinforced carbon-carbon in the weightless vacuum environment of space. Few potential repair materials could withstand the temperatures and pressures on the surface. Of those materials, few were compatible with the space environment and none had been tested in this type of application. Thus, the team developed two repair systems that were made available for contingency use on the next flight.

Astronaut Andrew Thomas (left) watches as Charles Camarda tests the reinforced carbon-carbon plug repair (STS-114 [2005]).

The first system—Non-Oxide Adhesive Experimental—was designed to repair coating damage or small cracks in reinforced carbon-carbon panels. This pre-ceramic polymer had the consistency of a thick paste. COI Ceramics, Inc., headquartered in San Diego, California, developed this system and the NASA repair team slightly modified it to optimize its material properties for use in space. Technicians used a modified commercial caulk gun to apply the material to the damaged wing. The material was spread out over the damage using spatulas similar to commercial trowels. Once dried and cured by the sun, Non-Oxide Adhesive Experimental used the heat of re-entry to convert the material into a ceramic, which protected exposed damage from extreme temperatures and pressures.

For larger damages, a plug repair system protected the reinforced carbon-carbon using a series of thin, flexible composite discs designed to fit securely against the curvature of the surface. Engineers developed 19 geometric shapes, which were flown to provide contingency repair capability. An attach mechanism held the plugs in place. The anchor was made up of a refractory alloy called titanium zirconium molybdenum that was capable of withstanding the 1,648°C (3,000°F) re-entry temperature.

Orbiter for launch, technicians placed accelerometers on the spar aluminum structure behind the reinforced carbon-carbon panels at the attachment locations. Forty-four sensors across both wings detected accelerations from potential impacts and relayed the data to on-board laptops, which could be transmitted to ground engineers. Using test-correlated dynamic models, engineers assessed suspected impacts for their level of risk based on accelerometer output.

Conclusion

The Orbiter Thermal Protection Systems on the shuttle proved to be effective, with the exception of STS-107 (2003). On that flight, the catastrophic loss was caused by a large piece of foam debris that was liberated from the ET. Advanced materials and coatings were key in enabling the success of the shuttle in high-temperature environments. Experience gathered over many shuttle missions led the Thermal Protection Systems team to modify and upgrade both design and materials, thus increasing the robustness and safety of these critical systems during the life of the program. Through the tragedy of the Columbia accident, NASA developed new inspection and repair techniques as protective measures to ensure the success and safety of subsequent shuttle missions.

External Tank Thermal Protection System

The amount of Thermal Protection System material on the shuttle's External Tank (ET) could cover an acre. NASA faced major challenges in developing and improving tank-insulating materials and processes for this critical feature. Yet, the space agency's solutions were varied and innovative. These solutions represented a significant advance in understanding the use of Thermal Protection System materials as well as the structures, aerodynamics, and manufacturing processes involved.

The tanks played two major roles during launch: containing and delivering cryogenic propellants to the Space Shuttle Main Engines, and serving as the structural backbone for the attachment of the Orbiter and Solid Rocket Boosters. The Thermal Protection System, composed of spray-on foam and hand-applied insulation and ablator, was applied primarily to the outer surfaces of the tank. It was designed to maintain the quality of the cryogenic propellants, protect the tank structure from ascent heating, prevent the formation of ice (a potential impact debris source), and stabilize tank internal temperature during re-entry into Earth's atmosphere, thus helping to maintain tank structural integrity prior to its breakup within a predicted landing zone.

Basic Configuration

NASA applied two basic types of Thermal Protection System materials to the ET. One type was a low-density, rigid, closed-cell foam. This foam was sprayed on the majority of the tank's "acreage"—larger areas such as the liquid hydrogen and liquid oxygen tanks as well as the intertank—also referred to as the tank "sidewalls." The other major component was a composite ablator material (a heat shield material designed to burn away) made of silicone resins and cork.

NASA oversaw the development of the closed-cell foam to keep propellants at optimum temperature—liquid hydrogen fuel at -253°C (-423°F) and liquid oxygen oxidizer at -182°C (-296°F)—while preventing a buildup of ice on the outside of the tank, even as the tank remained on the launch pad under the hot Florida sun.

The foam insulation had to be durable enough to endure a 180-day stay at the launch pad, withstand temperatures up to 46°C (115°F) and humidity as high as 100%, and resist sand, salt fog, rain, solar radiation, and even fungus. During launch, the foam had to tolerate temperatures as high as 649°C (1,200°F) generated by aerodynamic friction and rocket exhaust. As the tank reentered the atmosphere approximately 30 minutes after launch, the foam helped hold the tank together as temperatures and internal pressurization worked to break it up, allowing the tank to disintegrate safely over a remote ocean location.

Though the foam insulation on the majority of the tank was only about 2.5 cm (1 in.) thick, it added approximately 1,700 kg (3,800 pounds) to the tank's weight. Insulation on the liquid hydrogen tank was somewhat thicker—between 3.8 and 5 cm (1.5 to 2 in.). The foam's density varied with the type, but an average density was 38.4 kg/m^3 (2.4 pounds/ft^3).

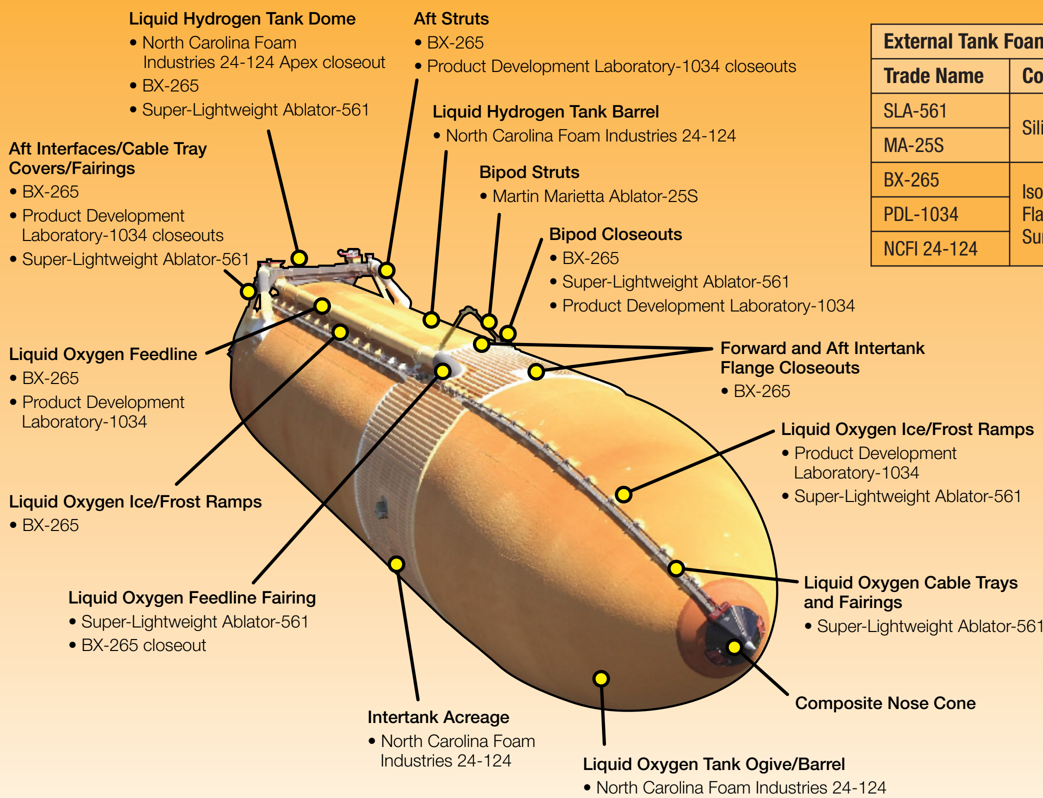
The tank's spray-on foam was a polyurethane material composed of five primary ingredients: an isocyanate and a polyol (both components of the polymeric backbone); a flame retardant; a surfactant (which controls surface tension and bubble or cell formation); and a catalyst (to enhance the efficiency and speed of the polymeric reaction). The blowing agent—originally chlorofluorocarbon (CFC)-11, then hydrochlorofluorocarbon (HCFC)-141b—created the foam's cellular structure, making millions of tiny bubble-like foam cells.

NASA altered the Thermal Protection System configuration over the course of the Space Shuttle Program; however, by 1995, ET performance requirements led the program to baseline four specially engineered closed-cell foams. The larger sections were covered in polyisocyanurate (an improved version of polyurethane) foam (NCFI 24-124) provided by North Carolina Foam Industries. NCFI 24-124 accounted for 77% of the total foam used on the tank and was sprayed robotically. A similar foam, NCFI 24-57, was sprayed robotically on the aft dome of the liquid hydrogen tank. Stepanfoam® BX-265 was sprayed manually on closeout areas, exterior tank feedlines, and internal tank domes. The tank's ablator, Super-Lightweight Ablator (SLA)-561, was sprayed onto areas subjected to extreme heat, such as brackets and other protuberances, and the exposed, exterior lines that fed the liquid oxygen and liquid hydrogen to the shuttle's main engines. NASA used Product Development Laboratory-1034, a hand-poured foam, for filling odd-shaped cavities.

Application Requirements

Application of the foam, whether automated or hand-sprayed, was designed to meet NASA's requirements for finish, thickness, roughness, density, strength, adhesion, and size and frequency of voids within the foam. The foam was applied in

External Tank Thermal Protection Systems Materials



External Tank Foam Material	
Trade Name	**Composition**
SLA-561	Silicone Resin, Cork
MA-25S	Silicone Resin, Cork
BX-265	Isocyanate Polyol, Flame-Retardant, Surfactant Catalyst
PDL-1034	Isocyanate Polyol, Flame-Retardant, Surfactant Catalyst
NCFI 24-124	Isocyanate Polyol, Flame-Retardant, Surfactant Catalyst

The External Tank's Thermal Protection System consisted of a number of different foam formulations displayed here. NASA selected materials for their insulating properties, and for their ability to withstand ascent aerodynamic forces.

specially designed, environmentally controlled spray cells and sprayed in several phases, often over a period of several weeks. Prior to spraying, engineers tested the foam's raw material and mechanical properties to ensure the materials met NASA specifications. After the spraying was complete, NASA performed multiple visual inspections of all foam surfaces as well as tests of "witness" specimens in some cases.

More than 90% of the foam was sprayed onto the tank robotically, leaving 10% to be applied by manual spraying or by hand. Most foam was applied at Lockheed Martin's Michoud Assembly Facility in New Orleans, Louisiana, where the tank was manufactured. Some closeout Thermal Protection System was applied either by hand or manual spraying at the Kennedy Space Center (KSC) in Florida.

Design and Testing

In the early 1970s, NASA developed a spiral "barber pole" Thermal Protection System application technique that was used through the end of the program. This was an early success for the ET Program, but many challenges soon followed.

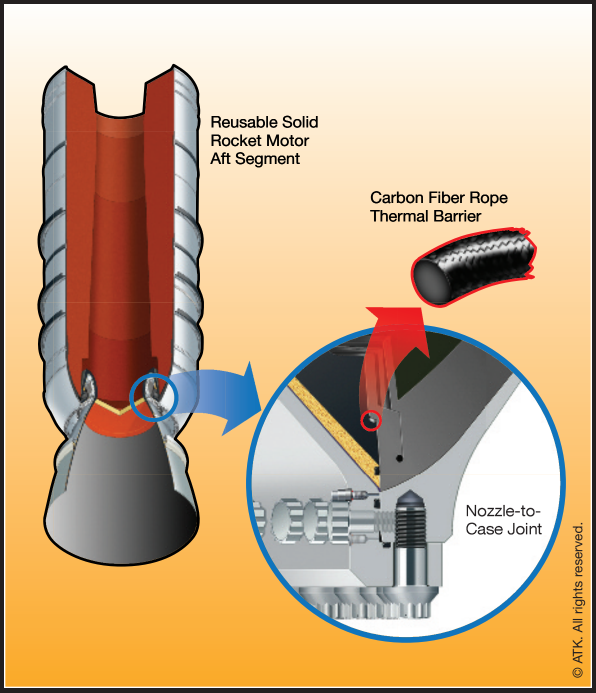
As the ET was the only expendable part of the shuttle, NASA placed particular emphasis on keeping tank manufacturing costs at a minimum. To achieve this objective, the agency based its original design and manufacturing plans on the use of existing, well-proven materials and processes with a planned evolution to newer products as they became available.

The original baseline Thermal Protection System configuration called for the sprayable Stepanfoam® BX-250 foam (used on the Saturn S-II stage) on the liquid hydrogen sidewalls (acreage)

Solid Rocket Motor Joint—An Innovative Solution

Alliant Techsystems (ATK) Aerospace Systems, in partnership with NASA Glenn Research Center, developed a solution for protecting the temperature-sensitive O-rings used to seal the shuttle reusable solid rocket motor nozzle segments. The use of a carbon fiber material promoted safety and enabled joint assembly in a fraction of the time required by previous processes, with enhanced reproducibility.

The reusable solid rocket motors were fabricated in segments and pinned together incorporating O-ring seals. Similarly, nozzles consisted of multiple components joined and sealed at six joint locations using O-rings. A layer of rubber insulation, referred to as "joint fill" compound, kept the 3,038°C (5,500°F) combustion gases a safe distance away from these seals. In a few instances, however, hot gases breached the compound, leaving soot within the joint. NASA modified the compound installation process and instituted reviews of postflight conditions. Although the modifications proved effective, damage was still possible in the unlikely event that gases breached the compound.



Using carbon fiber rope instead of rubber insulation in solid rocket motor nozzle joints simplified the joint assembly process and improved shuttle safety margins.

ATK chose an innovative approach through emerging technologies. Rather than attempt to prevent gas intrusion with manually applied rubber fill compound, the heat energy from internal gases would be extracted with a special joint filler and the O-ring seals would be pressurized with the cooled gas.

ATK's solution was based on a pliable, braided form of high-performance carbon material able to withstand harsh temperature environments. The braided design removed most of the thermal energy from the gas and inhibited flow induced by pressure fluctuations. The carbon fiber thermal barrier was easier to install and significantly reduced motor assembly time.

In a rocket environment, carbon fibers withstood temperatures up to 3,816°C (6,900°F). The braided structure and high surface area-to-mass ratio made the barrier an excellent heat exchanger while allowing a restricted yet uniform gas flow. The weave structure allowed it to conform to tolerance assembly conditions. The thermal barrier provided flexibility and resiliency to accommodate joint opening or closing during operation. Upon pressurization, the thermal barrier seated itself in the groove to obstruct hot gas flow from bypassing the barrier.

The carbon fiber solution increased Space Shuttle safety margins. Carbon fibers are suited to a nonoxidizing environment, withstanding high temperatures without experiencing degradation. The barrier provided a temperature drop across a single diameter, reducing gas temperature to O-rings well below acceptable levels. The thermal barrier also kept molten alumina slag—generated during solid fuel burn—from contacting and affecting O-rings.

and forward dome, and SLA-561 (used on the Viking Mars Lander) on the aft dome, intertank, and liquid oxygen tank in the areas of high heating.

In the late 1970s, however, design of the Orbiter tiles advanced to the point where it became apparent that they were susceptible to damage from ice detaching from the ET. This caused a reassessment of the Thermal Protection System design to prevent the formation of ice anywhere on the tank forward of the liquid hydrogen tank aft-end structural ring frame. The Orbiter/ice issue drove the requirement to cover the entire tank with Stepanfoam® BX-250, except for the high-heating aft dome, which remained SLA-561. Ice was to be prevented on tank pressurization lines through the use of a heated purge. Certain liquid oxygen feedline brackets, subject to extensive thermal contraction, could not be fully insulated without motion breaking the insulation. Therefore, NASA accepted ice formation on these brackets as unavoidable.

While attempting to prevent ice buildup on the tank, NASA also worked to characterize both the ablator material and the foams for expected heating rates. NASA worked with Arnold Engineering Development Center in Tennessee to modify its wind tunnel to provide the capability to test foam materials under realistic flight conditions. SLA-561 was tested in the plasma arc facility at NASA's Ames Research Center in California, which could deliver the required high heating rates. Better understanding of ablation rates and the flow fields around ET protuberances permitted refinement of the Thermal Protection System configuration.

A secondary function of the Thermal Protection System was to stabilize tank internal temperature during re-entry into Earth's atmosphere, thus helping to maintain tank structural integrity prior to its breakup over a remote ocean location.

The key to the External Tank's foam Thermal Protection System insulating properties was its cellular structure, creating millions of tiny bubble-like foam cells. The sprayed foam (NCFI 24-124) can be seen here after application to an area of the tank's aluminum "acreage," consisting of the liquid oxygen tank, liquid hydrogen tank, and intertank.

Another unique project was the testing of spray-on foam insulation on a subscale tank, measuring 3 m (10 ft) in diameter, in the environmental hanger at Eglin Air Force Base, Florida. The insulated tank was filled with liquid nitrogen and subjected to various rain, wind, humidity, and temperature conditions to determine the rate of ice growth. These data were then converted to a computer program known as Surfice, which was used at KSC to predict whether unacceptable ice would form prior to launch.

To provide information on application techniques, the agency ran cryogenic flexure tests that verified substrate adhesion and strength as well as crush tests on the Thermal Protection System materials.

In a continuous search for optimum Thermal Protection System performance, NASA—still in the Thermal Protection System design and testing phase—decided to use Chemical Products Research (CPR)-421, a commercial foam insulation with good high-heating capability. Lockheed Martin developed a sprayable Thermal Protection System to apply to tank sidewalls and aft dome. Application needed a relative humidity of less than 30%, which resulted in the addition of a chemical dryer at Michoud. Also, the tank wall had to be heated to 60°C (140°F). This required passing hot gas through the tank while it was being rotated for the "barber pole" foam application mode.

Ice Detection Prevents Catastrophic Problems

NASA had a potentially catastrophic problem with ice that formed on the cryogenic-filled Space Shuttle External Tank. Falling ice could have struck and damaged the crew compartment windows, reinforced carbon-carbon panels on the wing leading edge of the Orbiter, or its thermal protection tiles, thus placing the crew and vehicle at risk.

Kennedy Space Center and the US Army Tank Automotive and Armaments Research, Development and Engineering Center confirmed that a proof-of-concept system, tested by MacDonald, Dettwiler and Associates Ltd. of Canada, offered potential to support cryogenic tanking tests and ice debris team inspections on the launch pad. NASA and its partners initiated a program to develop a system capable of detecting ice on the External Tank spray-on foam insulation surfaces. This system was calibrated for those surfaces and used an infrared strobe, a focal plane sensor array, and a filter wheel to collect successive images over a number of sub-bands. The camera processed the images to determine whether ice was present, and it also computed ice thickness. The system was housed in nitrogen-purged enclosures that were mounted on a two-wheeled portable cart. It was successfully applied to the inspection of the External Tank on STS-116 (2006), where the camera detected thin ice/frost layers on two umbilical connections.

Robert Speece, NASA engineer, is shown operating the ice detection system at the pad, prior to shuttle launch.

The system can be used to detect ice on any surface. It can also be used to detect the presence of water.

First Flight Approaches

As the Space Shuttle Program moved toward the first shuttle flight in 1981, NASA faced another challenge. Approximately 37 m^2 (400 ft^2) of ablator became debonded from the tank's aluminum surface the first time a tank was loaded with liquid hydrogen. While the failure analysis was inconclusive, it appeared that the production team had tried to bond too large an area and did not get the ablator panels under the required vacuum before the adhesive pot life ran out. Technicians at Michoud Assembly Facility reworked the application process for the ET at their facility and the first tank at KSC.

Following the ablator bonding problem, NASA intensified its analysis of the ablator/aluminum bond line. This analysis showed that the higher coefficient of thermal expansion of the ablator binder, as compared to the aluminum, would cause the ablator to shrink. This would introduce biaxial tension in the ablator and corresponding shear forces at the bond line near any edges, discontinuities, or cracks. Then, when the tank was pressurized, tank expansion from pressure would compound this shear force, possibly causing the bond line to fail. NASA decided to pre-pressurize the liquid hydrogen tank with helium gas prior to filling the tank for launch—and to pressures higher than flight pressures—to stretch the ablator when it was warm and elastic.

Because early test data showed the tank insulation could be adversely affected by ultraviolet light, NASA painted the first several tanks white, using a fire-retardant latex paint. Exposure testing of foam samples on the roof of the Michoud Assembly Facility, however, showed the damage to be so shallow that it was insignificant. NASA decided not to paint the tanks, resulting in a weight savings of about 260 kg (580 pounds), lowered labor costs, and the introduction of the "orange" tank.

Environmental Challenges

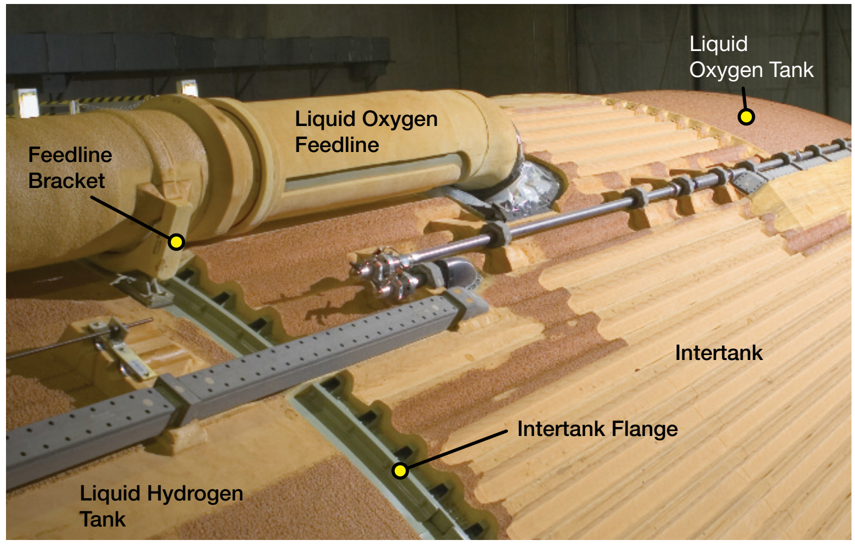
Knowledge of toxic properties and environmental contaminations increased over the 30 years of the Space Shuttle Program. Federal laws reflected these changes. For instance, ozone-depleting substances, including some Freon® compounds, reduced the protecting atmospheric ozone layer. NASA worked with its contractors to reduce both toxicity and environmental consequences for the cooling agents and the foam compounds.

During the 1990s, the University of Utah published data showing that CPR-421 was potentially toxic. Based on this analysis, Chemical Products Research withdrew CPR-421 from the market. NASA's ET office had Chemical Products Research reformulate this foam, with the new product identified as CPR-488.

New challenges arose related to emerging environmental policies that necessitated changes to Thermal Protection System foam formulations. In 1987, the United States adopted the Montreal Protocol on Substances that Deplete the Ozone Layer, which provided for the eventual international elimination of ozone-depleting substances. The United States implemented the protocol by regulations under the Clean Air Act. Ozone-depleting substances, including CFC-11—the Freon® blowing agent used in the production of the Thermal Protection System sprayable foams for the tanks—were scheduled to be phased out of production. After the phaseout, CFC-11 would only be available for such uses through a rigorous exemption process.

To prepare for the upcoming obsolescence of the foam blowing agent, Marshall Space Flight Center (MSFC) along with Lockheed Martin tracked and mitigated the effect of emerging environmental regulations. After extensive research and testing of potential substitutes, NASA proposed that HCFC-141b replace the CFC-11 blowing agent. NASA continued to use stockpiled supplies of CFC-11-blown foam until the HCFC-141b foam was certified for tank use and phased in beginning in 1996.

NASA undertook the development and qualification of a foam to be phased in as a replacement for the tank



The foam's approximately 2.5-cm (1-in.) thickness borders the circumferential flange that joins the intertank with the liquid hydrogen tank. The ribbed area is the intertank, that, like the liquid oxygen tank in the background and the liquid hydrogen tank in the foreground, was robotically sprayed with NCFI 24-124 foam. The flange would later be hand-sprayed with Stepanfoam® BX-265. The liquid oxygen feedline at the top of the tank and a feedline bracket have been hand-sprayed with BX-265 foam.

A technician at NASA's Michoud Assembly Facility sprays the flange that connects the intertank and liquid hydrogen tank. Stepanfoam® BX-265 was sprayed manually on closeout areas, exterior tank feedlines, internal tank domes, closeout areas of mating External Tank subcomponent surfaces, and small subcomponents.

sidewall foam, CPR-488. North Carolina Foam Industries reformulated CPR-488 and developed a new product.

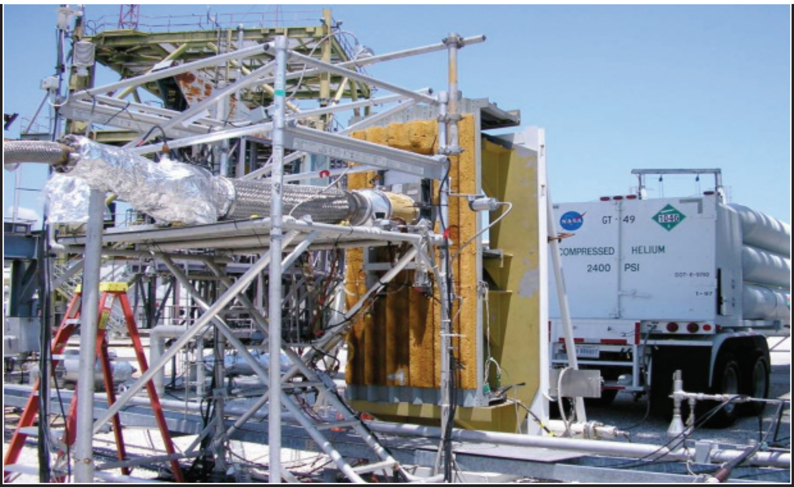
As part of qualifying this new product, Lockheed Martin, Wyle Laboratories, and MSFC developed an environmental test. This test used a flat aluminum plate machined to match aft dome stress levels. The plate was attached to a cryostat filled with liquid helium and then strained with hydraulic jacks to the flight biaxial stress levels. Radiant heat lamps were installed to match the radiant heating from the solid rocket motor plumes, and an acoustic horn blasted the test. This simulated the aft dome ascent environment as well as possible. The test results indicated the need to spray ablator on the aft dome. To provide the capability to spray the ablator, personnel at Michoud Assembly Facility built two spray cells, with an additional cell to clean and prime the liquid hydrogen tank before ablator application.

To save the weight of this ablator and its associated cost, NASA had North Carolina Foam Industries develop a foam adequate for the aft dome environment without ablator. The foam was phased in on the aft dome, flying first on Space Transportation System (STS)-79 in 1996. The first usage of the new foam on the tank sidewalls was phased in over three tanks starting with STS-85 in August 1997.

Environmental Protection Agency regulations also required NASA to replace Stepanfoam® BX-250, which was sprayed manually—with a CFC-11 blowing agent—on the tank's "closeout" areas. During STS-108 (2001), Stepanfoam® BX-265—with HCFC-141b as its blowing agent—first flew as a replacement for BX-250. BX-250 continued to be flown in certain applications as BX-265 was phased into the manufacturing process.

The use of HCFC-141b as a foam blowing agent, however, was also problematic. It was classified as a Class II ozone-depleting substance and was subject to phaseout under the

Aerogel-based Insulation System Precluded Hazardous Ice Formation



Testing of gaseous hydrogen vent arm umbilical disconnect equipment at Kennedy Space Center.

During the STS-114 (2005) tanking test, the External Tank Gaseous Hydrogen Vent Arm Umbilical Quick Disconnect formed ice and produced liquid nitrogen/air. The phenomenon was repeated during subsequent testing and launch. For the shuttle, ice presented a debris hazard to the Orbiter Thermal Protection System and was unacceptable at this umbilical location. The production of uncontrolled liquid nitrogen/air presented a hazard to the shuttle, launch pad, and ground support equipment.

NASA incorporated a fix into the existing design to preclude ice formation and the uncontrolled production of liquid nitrogen/air. The resolution was accomplished with two changes to the umbilical purge shroud. First, the space agency improved the shroud purge gas flow to obtain the desired purge cavity gas concentrations. Second, technicians wrapped multiple layers of aerogel blanket material directly onto the quick disconnect metal surfaces within the purged shroud cavity.

NASA tested the design modifications at the Kennedy Space Center Cryo Test Lab. Tests showed that the outer surface of the shroud was maintained above freezing with no ice formation and that no nitrogen penetrated into the shroud purge cavity. NASA used the modified design on STS-121 (2006) and all subsequent flights.

Aerogel insulation is a viable alternative to the current technology for quick disconnect shrouds purged with helium or nitrogen to preclude the formation of ice and liquid nitrogen/air. In most cases, aerogel insulation eliminates the need for active purge systems.

Clean Air Act effective January 2003. NASA was granted exemptions permitting the use of HCFC-141b in foams for specific shuttle applications. These exemptions applied until the end of the program.

Post-Columbia Accident Advances in Thermal Protection

Following the loss of Space Shuttle Columbia in 2003, NASA undertook the redesign of some tank components to reduce the risk of ice and foam debris coming off the tank. These hardware changes drove the need to improve the application of Thermal Protection System foam that served as an integral part of the components' function. The major hardware addressed included the ET/Orbiter attach bipod closeout, protuberance air load ramps, ice frost ramps, and the liquid hydrogen tank-to-intertank flange area.

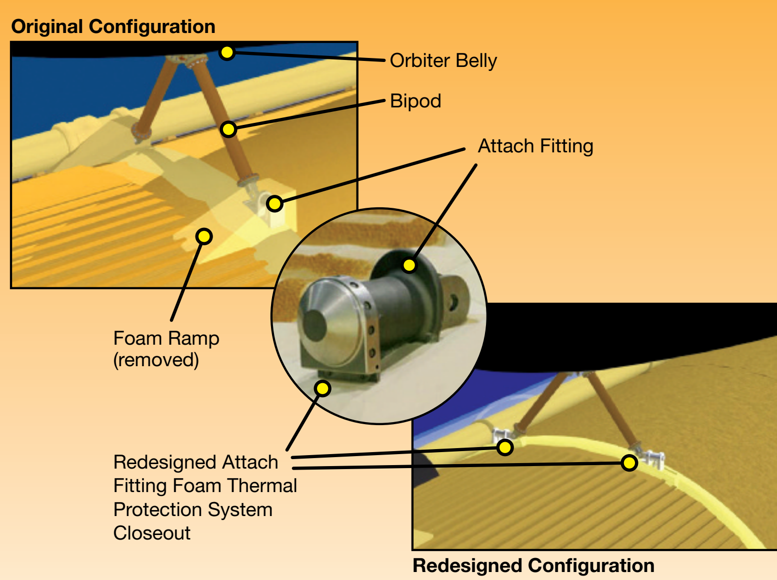
The ET bipod attached the Orbiter to the tank. The redesign removed the foam ramps that had covered the bipod attach fittings, and which had been designed to prevent the formation of ice when the ET was filled with cold liquid hydrogen and liquid oxygen on the launch pad. This left the majority of each fitting exposed. NASA installed heaters as part of the bipod configuration to prevent ice formation on the exposed fittings.

NASA developed a multistep process to improve the manual bipod Thermal Protection System spray technique. Validation of this process was accomplished on a combination of high-fidelity mock-ups and a full-scale ET test article in a production environment. Wind tunnel tests demonstrated Thermal Protection System closeout capability to withstand maximum aerodynamic loads without generating debris.

The ET protuberance air load ramps were manually sprayed wedge-shaped layers of insulating foam insulation along the pressurization lines and cable tray on the side of the tank. They were designed as a safety precaution to protect the tank's cable trays and pressurization lines from airflow that could potentially cause instability in these attached components. Foam loss from the ramps during ascent, however, drove NASA to remove them from the tank. This required extensive engineering. NASA created enhanced structural dynamics math models to better define the characteristics of this area of the tank and performed numerous wind tunnels tests.

The ET fuel tank Main Propulsion System pressurization lines and cable trays were attached along the length of the tank at multiple locations by metal support brackets. These were protected from forming ice and frost during tanking operations by foam protuberances called ice frost ramps. The feedline bracket configuration had the potential for foam and ice debris loss. Redesign changes were

External Components Redesign



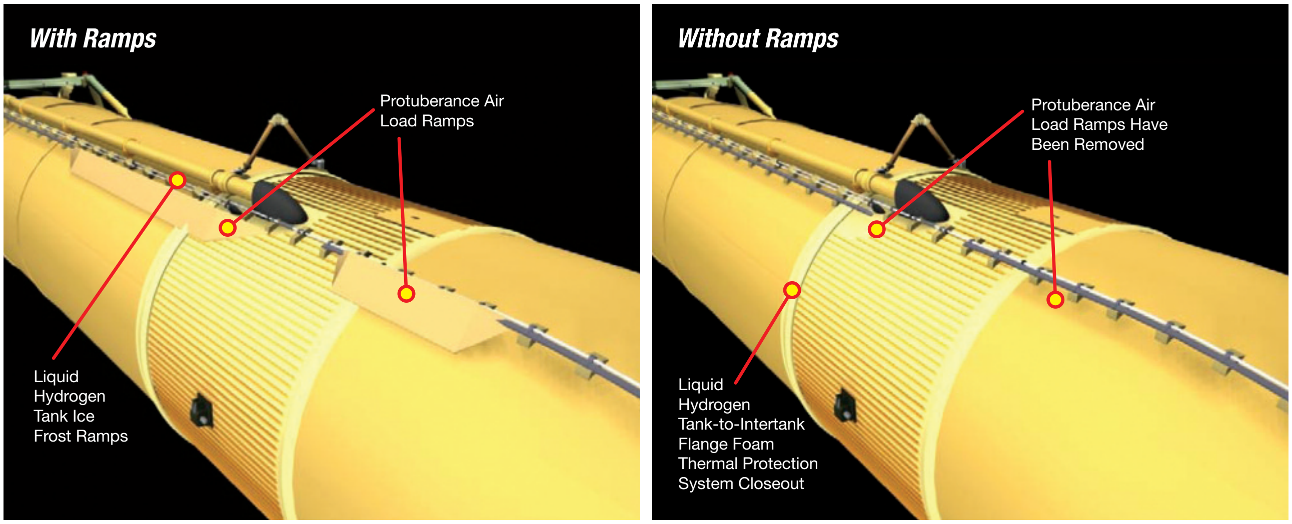
After the Columbia accident, NASA implemented a number of improvements to External Tank components and related Thermal Protection System elements. One such measure was the redesign of the Orbiter/External Tank attach bipod fitting mechanism, which included a meticulous reworking of the attach fitting Thermal Protection System configuration.

In what used to be a one-person operation, a team of technicians at NASA's Michoud Assembly Facility prepares to hand-spray BX-250 foam on the bipod attach fittings. The videographer (standing) records the process for later review and verification. A quality control specialist (left) witnesses the operation, while two spray technicians make preparations.

incorporated into the 17 ice frost ramps on the liquid hydrogen tank to reduce foam loss. BX-265 manual spray foam replaced foam in the ramps' closeout areas to reduce debonding and cracking.

The NASA/Lockheed Martin team also developed an enhanced three-part procedure to improve the Thermal Protection System closeout process on the liquid hydrogen tank-to-intertank flange area.

In all post-Columbia Thermal Protection System enhancement efforts, NASA modified process controls to ensure that defects were more tightly kept within the design envelope. The space agency simplified application techniques and spelled out instructions in more detail, and technicians had the opportunity to practice their application skills on high-fidelity component models. MSFC and Lockheed Martin also developed an electronic database to store information for each spray. New application certification requirements were added. Improvements included the forward bellows heater, the liquid oxygen feedlines, and titanium brackets. Improved imagery analysis and probabilistic risk assessments also allowed NASA to better track and predict foam loss. Thermal protection debris could never be completely eliminated, but NASA had addressed a complex and unprecedented set of problems with determination and innovation.



NASA decided to delete the tank's protuberance air load ramps and implement design changes to the 17 ice frost ramps on the liquid hydrogen tank. Both these measures required adjustments in the components' Thermal Protection System configuration and application processes. Materials and techniques were also altered to improve the Thermal Protection System closeout of the flange joining the liquid hydrogen tank with the intertank.