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ORBITER THERMAL PROTECTION SYSTEM

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ABSTRACT

The major material and design challenges associated with the Orbiter thermal protection system (TPS), the various TPS materials that are used, the different design approaches associated with each of the materials, and the performance experienced during the flight test program are described. The first five flights of the Orbiter Columbia and the initial flight of the Orbiter Challenger have provided the necessary data to verify the TPS thermal performance, structural integrity, and reusability. The flight performance characteristics of each TPS material are discussed. This discussion is based on postflight inspections and postflight interpretation of the flight instrumentation data. The flights to date indicate that the thermal and structural design requirements for the Orbiter TPS have been met and that the overall performance has been outstanding.

INTRODUCTION

One major technical accomplishment of the Space Shuttle program involved the development of a reusable thermal protection system (TPS). To meet the challenge of providing a reusable TPS for the Orbiter, new concepts of thermal protection materials and design approaches were necessary. Before the Shuttle Program, all manned space vehicles had used ablator materials having a one-mission capability. In contrast, the TPS for the Shuttle Orbiter had to be reusable for 100 missions to minimize operational costs. It also had to be extremely weight-efficient to meet vehicle performance requirements.

Four principal thermal protection concepts appeared to have potential application for the Shuttle Orbiter program. These concepts included (1) replaceable ablator panels, (2) nonablating, nonmetallic insulative-reradiative materials, (3) metallic reradiative heat shields, and (4) carbon-carbon reradiative hot structures. Extensive technology development activities were undertaken at the various National Aeronautics and Space Administration (NASA) centers and by the major aerospace firms from 1969 to 1972. Although the ablator technology was well developed at the time, it was not economically feasible to develop a low density replaceable system. The metallic reradiative heat shields had significant disadvantages as far as temperature limitation, expensive manufacturing, and difficult inspection techniques. The nonreceding, nonmetallic ceramic heat shield possessed two unique advantages from the onset; that is, design simplicity and reuse capability. In contrast, the metallic heat shields were quite complex due to the design features needed to minimize thermal distortion, panel-to-panel joints, as well as the insulation with its packaging that would be required to protect the vehicle primary structure. The carbon-carbon material was the only known material that showed potential for providing reuse capability for the high temperature areas of the Orbiter (>2300° F) such as the wing leading edge and nose cap regions.

It was recognized that major technological developments would have to be undertaken to bring the nonmetallic ceramic materials from the laboratory state to actual vehicle application. However, the significant weight savings and design simplicity inherent with the ceramic materials led to their selection as the primary Orbiter TPS. The carbon-carbon was the clear choice for leading-edge applications where a durable reusable material was required. Its temperature capability also added to its preference. However, significant developments in coatings preventing oxidation would have to be made to make carbon-carbon a multi-mission material.

This paper will cover the development activities that were undertaken for the TPS materials selected for the Orbiter, the major pacing material and design issues that evolved, and finally the performance based on the recent flight test program.

DESIGN REQUIREMENTS - GENERAL

The thermal protection for the Orbiter is designed to operate successfully over a spectrum of environments typical of both aircraft and spacecraft as shown in figure 1. During the ascent and entry phases of the mission, the Orbiter structure must be maintained at temperatures less than 350° F. In addition to withstanding the thermal environments, the TPS must also perform satisfactorily in other induced environments, such as launch acoustics, structural deflections induced by aerodynamic loads, and onorbit cold soak as well as the natural environments such as salt fog, wind, and rain.

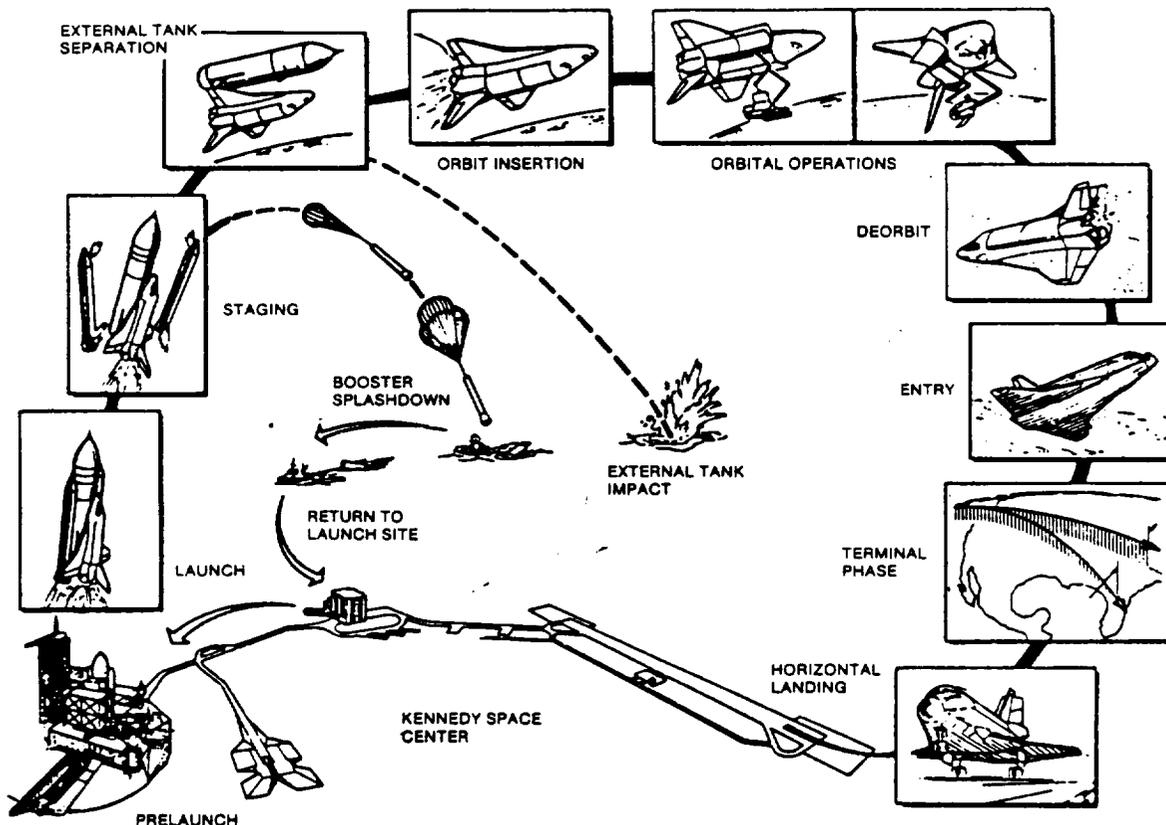


FIGURE 1.- ORBITER MISSION SEQUENCES.

The exterior surfaces of the TPS must also provide an acceptable aerodynamic surface to avoid early tripping of the high-temperature boundary layer (from laminar to turbulent flow). This would significantly increase the thermal heat load to the structure. This requirement resulted in maintenance of rigid fabrication tolerances during the manufacture of the TPS.

The key driver to the design of the TPS has been the requirement for the TPS to function for 100 missions with minimal weight, maintenance, and refurbishment.

THERMAL PROTECTION MATERIALS

The location of the various thermal protection materials that are applied to the Orbiter structure is shown in figure 2. The allocation was based primarily on the inherent temperature capability of the materials. The following sections cover the individual material characteristics.

RSI TPS

Three material systems applied to the Orbiter are broadly characterized as reusable surface insulation (RSI). Low-density silica ceramic insulation comprises two of these material systems. The third material consists of a coated nylon felt system.

The ceramic tiles are classified in two categories: the high-temperature reusable surface insulation (HRSI) and the low-temperature reusable surface insulation (LRSI). The primary difference in these material systems is in the surface coating. The HRSI tiles (predominantly on the lower Orbiter surfaces) are coated with a black borosilicate glass. Whereas, the LRSI tiles contain a white coating. Both the HRSI and the LRSI TPS tiles are manufactured by the Lockheed Missiles and Space Company (LMSC). The actual tile installation on the Orbiter structure is performed by Rockwell International.

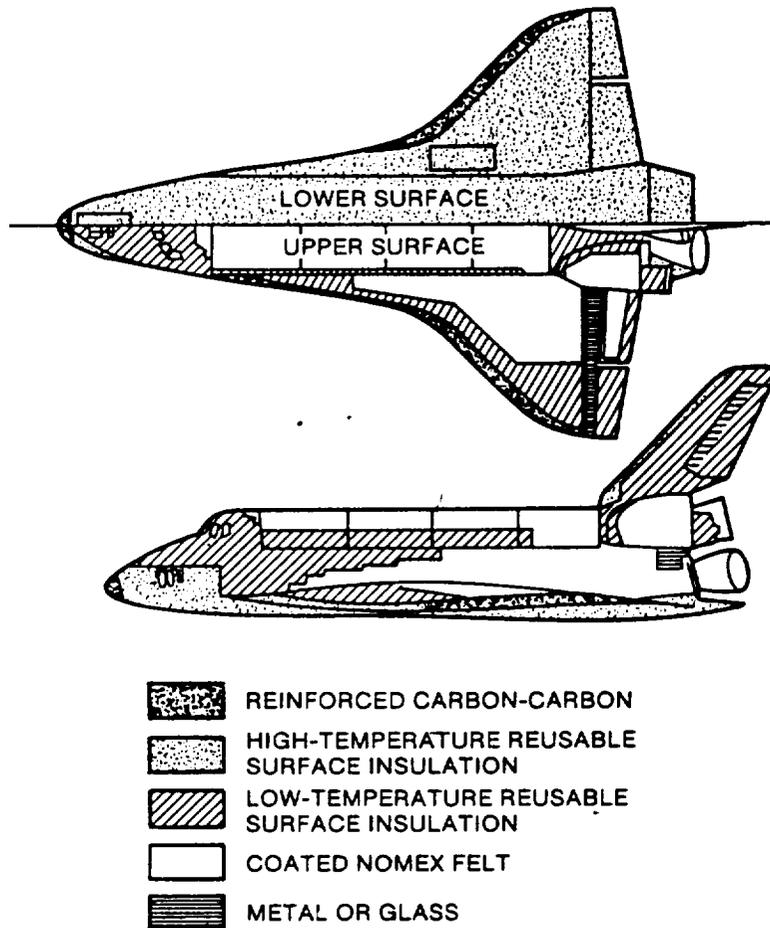


FIGURE 2.- THERMAL PROTECTION SUBSYSTEM.

The basic insulation material for the tile is manufactured in two densities: 9 lb/ft³, which is identified as LI-900 and 22 lb/ft³ which is identified as LI-2200. These materials cover approximately 70% of the surface area of the Orbiter structure. Most of this area is covered with LI-900 tiles, with the higher density material used in areas of door edges and penetrations where use of a stronger more durable material is required.

The basic raw material for the all-silica TPS tile consists of short-staple, 99.6% pure amorphous silica fiber manufactured by Johns Manville. At LMSC, the fibers are felted from a slurry, pressed, and sintered in the form of rigidized blocks of insulation material. Tiles which have been sized for specific thermal environments are then cut from these blocks of insulation material. The majority of these tiles are cut with a square planform. However, other tile shapes are required because of vehicle geometry. The tiles are then coated with a thin borosilicate glass coating. The HRSI tiles have a coating containing a black pigment (silicon tetraboride) for the proper high-temperature emittance value ($\epsilon > 0.8$) which is needed in the high-temperature applications on the Orbiter. This coating also provides a barrier to moisture absorption. The LRSI tiles have a white coating with the proper optical properties (solar absorptance to total hemispherical emittance ratio ($\alpha/\epsilon < 0.4$)) that is needed to maintain the proper onorbit temperatures for vehicle thermal control purposes. After the coating process, the tiles are treated with a water repellent material under controlled heating/vacuum conditions that imparts a hydrophobic film. This prevents water absorption by the low-density insulation. The tile insulation then remains water repellent until exposure to temperatures greater than 1050° F. The tiles are then shipped to Rockwell in the form of large arrays (20 tiles/array) which are applied to the Orbiter structure.

The flexible reusable surface insulation (FRSI), which is the simplest TPS used on the Orbiter, consists of a needled Nomex felt. It is coated with a thin silicone elastomeric film. This material is provided in the form of 3- by 4-foot sheets ranging in thickness from 0.16 to 0.32 in. by the Globe Albany Company. This material is installed on the Orbiter structure by Rockwell International.

RCC MATERIAL

A unique structural material called reinforced carbon-carbon (RCC), manufactured by the Vought Corporation, protects the Orbiter's nose cap and wing leading edge in the regions of highest temperature on the Orbiter.

The fabrication of the RCC begins with a rayon cloth, which is graphitized and impregnated with a phenolic resin. This impregnated cloth is layed up as a laminate and cured in an autoclave. After cure, the laminate is pyrolyzed (baking the resin volatiles out) at high temperature to convert the resin to carbon. The part is then impregnated with furfural alcohol in a vacuum chamber, cured, and pyrolyzed again to convert the alcohol to carbon. This process is repeated three times until the required carbon-carbon density of 90 to 100 lb/ft³ is achieved.

The resulting RCC part is a hard carbon structure possessing reasonable strength and low coefficient of thermal expansion. This provides excellent resistance to thermal stresses and shock. The carbon-carbon is protected from oxidation by converting the outer surface to silicon carbide (SiC) in a diffusion coating process. The oxidation-resistant coating is applied to the part by packing it in a retort with a dry-pack material made up of a mixture of alumina, silicon, and silicon carbide. The retort is placed in a furnace and the coating process takes place in argon with a stepped time-temperature cycle of up to 3200° F. A diffusion reaction occurs between the dry pack and carbon-carbon. This causes the outer layers of the carbon-carbon to convert to silicon carbide (whitish-gray color) with effectively no thickness increase of the uncoated part.

Further oxidation resistance is provided by impregnation with tetraethyl-orthosilicate (TEOS). When cured, TEOS leaves a silicon dioxide (SiO₂) residue throughout the coating and substrate to further reduce the area of exposed carbon. The final step in the fabrication process is the application of a surface sealant (sodium silicate/SiC mixture) to fill any remaining surface porosity or microcracks.

THERMAL PROTECTION DESIGN

RSI DESIGN

The HRSI and LRSI tiles are bonded to the Orbiter structure. A silicone adhesive and an intervening layer of nylon felt material as shown in figure 3 are used. The low-density silica tile is an excellent thermal insulator. However, as it is a ceramic material, it possesses low strength and is brittle. For that reason, a nylon felt material, known as a strain isolation pad (SIP), is used to isolate the structural strains and deflections of the Orbiter airframe from inducing critical stresses in the tile. Tiles are densified by a ceramic slurry process at the inner moldline assuring adequate strength at the tile/SIP interface. Densification was implemented to assure adequate tile structural margins for the predicted load cases. Since so many parts are involved (approximately 31,000 tiles), pull tests were performed to verify that each tile system (tile/SIP/bond) installed on the Orbiter possessed adequate strength margin. Tile-to-tile contact resulting from acoustic-induced tile movement or from contraction of the airframe in the cold extremes of space is prevented by providing gaps between the tiles. The filler bar material in the bottom of the tile-to-tile gaps is used for thermal insulation from tile-to-tile gap heating.

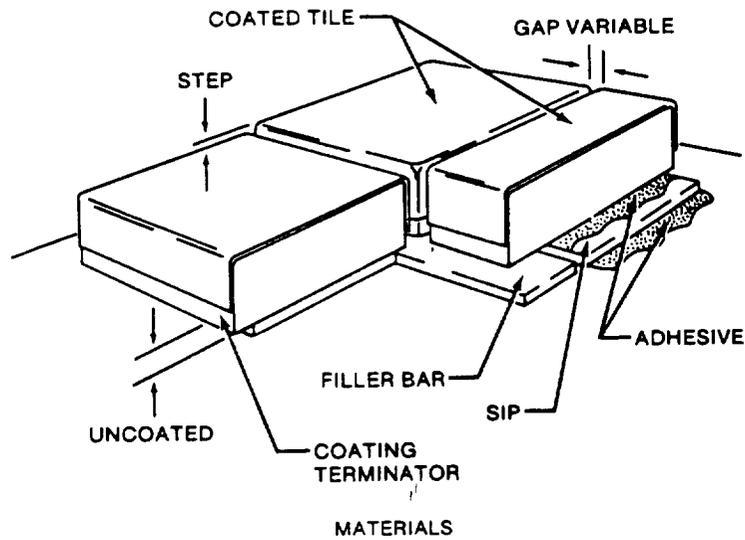
In the higher pressure gradient regions of the Orbiter, open tile-to-tile gaps could result in sufficient ingestion of high-temperature gas flow during entry. This could cause local overtemperature of the various TPS components and the structure. To preclude this from happening, two basic types of gap fillers, "pillow" or "layer", are bonded to the top of filler bar as shown in figure 4. Thermal barriers made from the same cloth and filled with soft insulation and metallic springs are used to fill the larger tile-to-tile gaps around movable hatches and doors.

The FRSI TPS installation is the least complex of the TPS materials used on the Orbiter. In this case, 3- by 4-foot blankets of Nomex felt which has been heat-treated and coated with silicone elastomer are bonded with a silicone rubber adhesive to the structure. Figure 5 shows a cross section of this insulation system.

The RSI material characteristics and detailed description of the design applications have been presented in reference 1-3.

LESS DESIGN

The leading-edge structural system (LESS) of the Orbiter consists of the RCC nose cap (figure 6) and wind leading-edge panels (figure 7), the metallic attachments to the Orbiter structure, the inter-



- MATERIALS**
- TILE - 22 LB/FT³ PURE SILICA FIBER-FIRED AT
9 LB/FT³ 2300° F
 - COATING - BOROSILICATE (GLASS) FOR WATER-
PROOF AND OPTICAL PROPERTIES
 - SIP - NOMEX FELT
 - FILLER BAR - COATED NOMEX FELT
 - ADHESIVE - ROOM-TEMPERATURE VULCANIZING
(RTV) SILICONE

FIGURE 3.- TILE SYSTEM CONFIGURATION.

nal insulation system, thermal barriers, and the interface tiles between the RCC and acreage reusable surface insulation.

The wing leading edge and the nose cap are structural fairings which transmit aerodynamic loads to the forward bulkhead or to the wing spar through discrete mechanical attachments. Inconel 718 and A-286 stainless steel fittings are bolted to flanges formed on the RCC components. They are attached to the aluminum wing spar and fuselage forward bulkhead. The fitting arrangement provides thermal isolation, allows thermal expansion, and accommodates structural displacement. The wing leading edge consists of 22 panels joined by 22 T-seals. This segmentation is necessary not only to facilitate the high-temperature fabrication process but also to accommodate the thermal expansion during entry of the leading edge while preventing large gaps or interference between the parts. In addition, the T-seals prevent the direct flow of hot boundary-layer gases into the wing leading-edge cavity during entry. The nose cap seal design and structural attachments are similar to the wing leading edge.

The RCC parts form a hollow shell, which promotes internal cross radiation from the hot stagnation region to the cooler leeward surface area. This reduces the stagnation temperatures and thermal gradients around the shell. The operational temperature range of the RCC is from -200° to 3000° F. Since RCC is not an insulator, the adjacent aluminum and the metallic attachments are protected from internal radiant heating by internal insulation. Dynaflex insulation, contained in formed and welded Inconel foil, protects the metallic attachment fittings from the heat emitted from the inner surfaces of the RCC wing panels. The nose cap internal insulation system consists of blankets fabricated from AB-312 ceramic cloth, saffil and Dynaflex insulation. HRSI tiles protect the nose cap bulkhead access door from the heat emitted from the hot inside surface of the RCC.

The RCC material characteristics and detailed description of the design application have been presented in references 4-7.

ELEVON/ELEVON ABLATOR

Before the first flight of Columbia and late in the Orbiter fabrication cycle, thermal analysis/wind tunnel testing indicated that the areas between the inboard and outboard elevons would

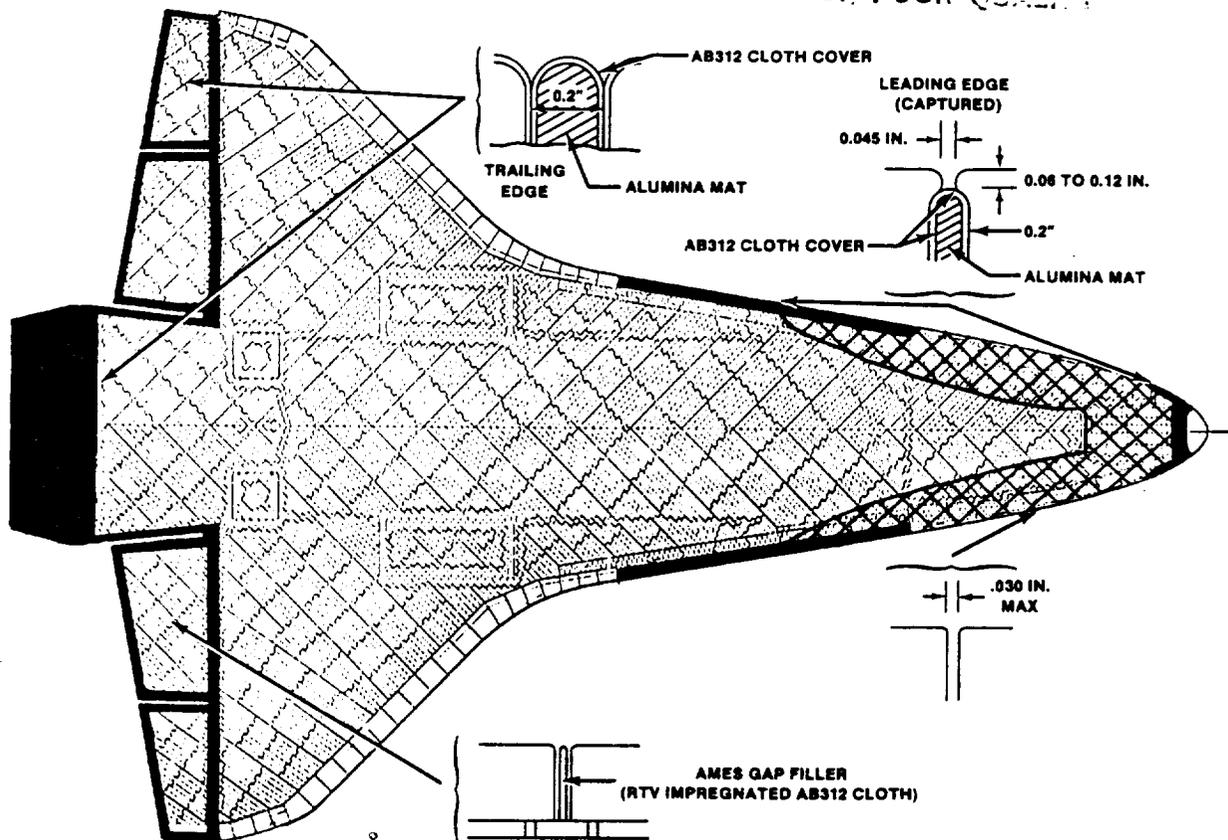


FIGURE 4.- HIGH DELTA-P GAP FILLER ZONES.

encounter temperatures that were too high for tiles to survive even a single mission ($\sim 3200^\circ\text{F}$). Therefore, an ablative material was installed between the split segments of the elevons (outboard end of each inboard elevon and inboard end of each outboard elevon) as shown in figure 8. The ablative material (AVCOAT 5026-39HC/G) was the same as the Apollo heat shield material. It consisted of an epoxy-novolac resin filled with microballoons and a mixture of silica and E-glass fibers. The resin mixture was injected into a fiberglass, open-faced honeycomb core which was bonded to an aluminum plate. Since the ablator chars and undergoes surface erosion, it was replaced after each flight. As a result, the ablator system was mechanically attached to the wing structure for easy installation and removal. The first five flights indicated lower temperatures than initially predicted. All Orbiters are now being returned to the original reusable silica tile design in the elevon split area.

THERMAL PROTECTION SYSTEM TECHNOLOGY

RSI MATERIAL SELECTION

Early in the 1970's, contractors began developing a lightweight, rigid ceramic TPSs for the Shuttle program. The technology programs from 1970 to 1972 consisted of three distinct development phases. The first phase covered materials screening and initial laboratory formulation of materials. Lockheed developed the all-silica system. A mullite system was developed by McDonnell-Douglas. During this phase, initial mechanical and thermal properties were obtained from materials produced in the laboratory. A second phase of development activities was undertaken with the initial contractors as well as with General Electric who was also using mullite materials at the time. During this phase, the design methodology for these brittle materials was developed and a reasonable understanding of the behavior under various load conditions evolved. A major advance occurred with the improvements of the thermal stability characteristics of the silica fiber. A third technology phase was implemented with these contractors. The primary emphasis during this phase was the development of attachment methods for these low density ceramic materials. These materials possessed a low strain

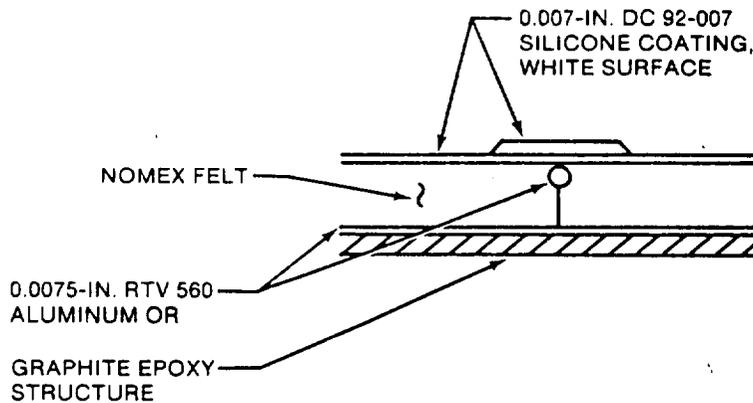


FIGURE 5.- FRSI TPS INSTALLATION.

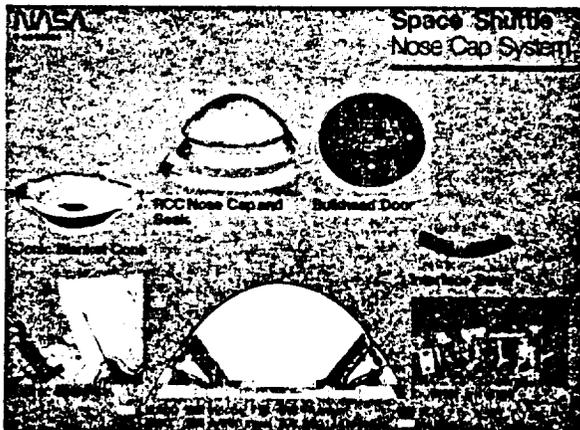


FIGURE 6.- SPACE SHUTTLE NOSE CAP SYSTEM.

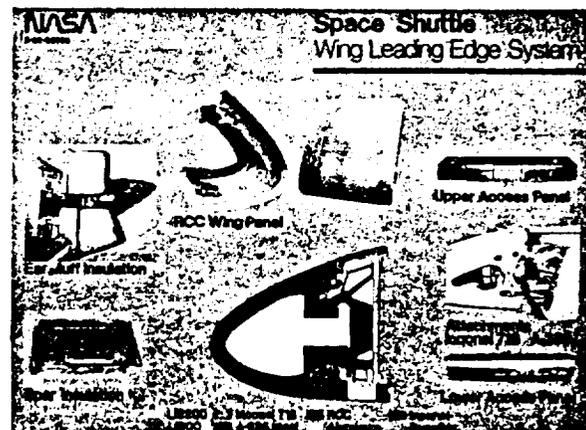


FIGURE 7.- SPACE SHUTTLE WING LEADING EDGE SYSTEM.

to failure capability. The very low density silica material LI-900 proved to be a feasible material. Enough experience had been gained through these technology programs to undertake the selection process for the Orbiter TPS. It was found that the silica strength properties were not adversely affected by high temperature exposure. In contrast, the mullite material had significant strength loss with temperature. Additionally, the mullite contractors failed to strengthen the mullite material to levels compatible with the induced thermal stresses of entry heating. Failures were predicted and experienced during test. Mullite materials initially were expected to have higher temperature capability than silica. However, the low density silica possessed better thermal performance characteristics due to the small fiber diameter material used in its formulation. All of these factors clearly indicated that the rigidized silica ceramic material was the superior product. It was selected in January 1973 as the baseline TPS material. The evolution of the RSI TPS has been presented at a number of conferences (references 8-10).

PRODUCTION

In June 1973, Lockheed was selected to provide the silica RSI TPS for the Orbiter. The reusable ceramic tile engineering, design, and installation to the Orbiter remained a Rockwell responsibility. This allowed for integration of the tile system with the structure design and numerous TPS penetrations that were required. This was in contrast to the leading-edge structural system which was awarded to the Vought Corporation. Vought was responsible for the design, manufacturing, and engineering for the RCC parts. Whereas Rockwell was responsible only for the attachment interface to the structure and the internal insulation. This separation of the design was acceptable because of the configuration of the Leading Edge Structural Subsystem (LESS).

The transition from laboratory and pilot plant production of the silica RSI to a full production status was not accomplished without the usual attendant scale up problems. One key aspect in the ini-

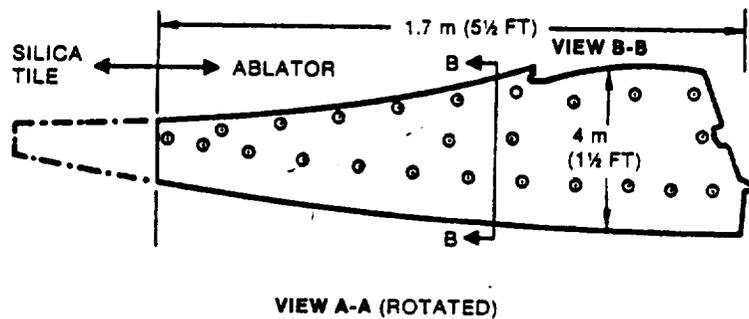
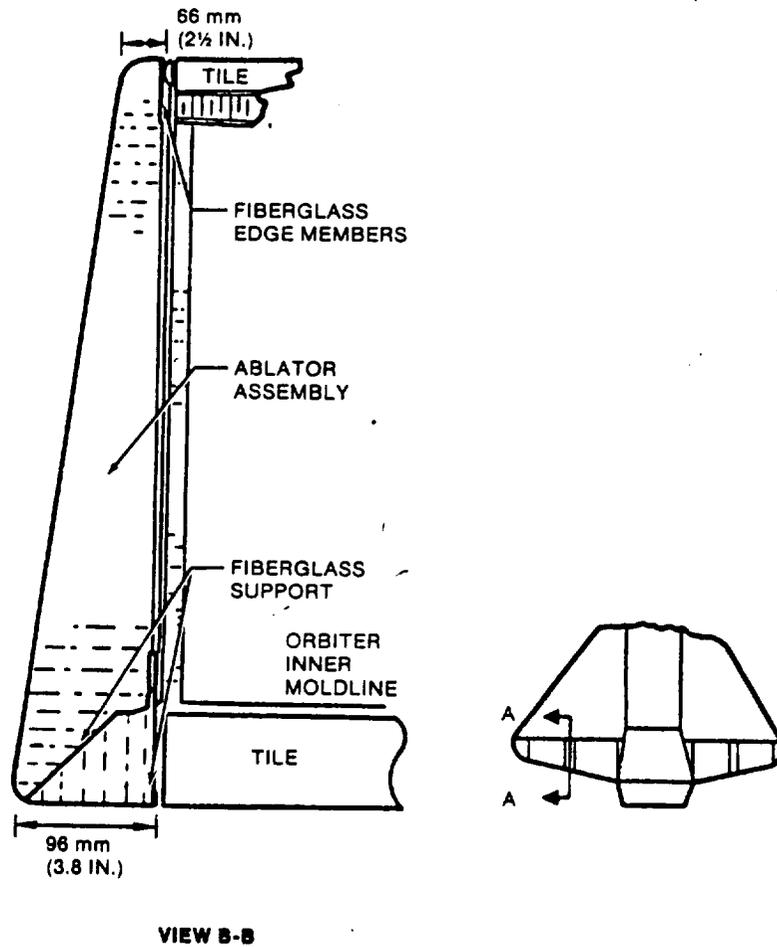


FIGURE 8.- ELEVON/ELEVON ABLATOR INSTALLATION.

tial process developing phase was control of the purity and consistency of the silica fibers. The amorphous silica fiber, a key ingredient in the production of the silica tile, impedes the formation of crystalline forms of the silica. These crystalline forms have a thermal expansion coefficient 30 times greater than the amorphous form. Transformation from the amorphous structure to crystalline form is associated with totally unacceptable shrinkage and distortion of the sintered silica composite. To achieve the required dimensionally stability ultra pure silica fiber greater than 99.6% pure was necessary.

Early in the production phases, fiber with sufficient purity to meet the high production capability for the Orbiter became a major concern. The fiber delivered by Johns-Manville did not meet all the purity requirement and extensive post-treatment was needed. At one point, an alternate high purity fiber source was considered. However, the fiber diameters were larger than the JM fibers which would have adversely affected the thermal performance. Rigid process controls were established from the starting point, that is, from the sand used in making the fiber, through to the fiberizing and cleaning process. These controls minimized contaminants and lead to the delivery of sufficiently pure fibers for the Orbiter tile application.

The next key material development issue encountered during the production phase involved the glass coating. During manufacturing and during thermal tests, the multi-layer glass coating applied to the tile had a tendency to crack or foam. This caused problems related to tile moisture absorption and dimensional stability. A single layer coating, Reaction Cured Glass (RGG), was developed at the Ames Research Center. It did not foam during processing and had a better match with the thermal expansion coefficient of the silica insulation material. This process was eventually implemented in the LMSC production facility after a lengthy verification test program. These tests demonstrated its superior performance as compared to the previous multi-layer coating system.

Rigid process controls were needed to take the tile from the laboratory to production scale-up. A modern manufacturing facility was created at LMSC's main plant in Sunnyvale, California. That plant contained the latest blending and slurry casting and coating equipment, precision controlled kilns and furnaces with low contaminant requirements, and sophisticated numerically controlled machinery to fabricate the TPS tile. Fabricating the tile from a basic block of insulation material was indeed a formidable engineering manufacturing task. The Rockwell master dimension engineering data base and vehicle configuration coordinates were converted into computer tapes that drove the numerically controlled mills that machined the tiles to precise dimensions. After the coating and waterproofing process, Lockheed's job was finished. The tiles in large array form were shipped to Rockwell. Rockwell subsequently bonded them to the Orbiter structure.

RSI DESIGN CHALLENGE

First the RSI material is produced in the form of tiles with dimensionally stable planforms. The next major technical problem became one of assuring adequate attachment to the Orbiter vehicle structure. As mentioned previously, the RSI ceramic TPS is a relatively brittle, low strength material. Therefore, a strain isolation system is needed providing mechanical isolation of the tile from structure deformations. To accomplish this function, the tile was bonded to a low-modulus nylon felt pad with a silicone adhesive. Then the composite was bonded to the structure with the same adhesive. For the load conditions predicted during the initial design study there was every reason to believe that the design approach would function properly. However, as the mission requirements became better defined, the updated environments indicated higher load conditions. These loads would have exceeded the strength capability of the low density tile/SIP system. In addition, structural integrity tests of the tile/SIP system indicated the presence of discrete stress concentrations. They were caused by the needling characteristics of the SIP, which significantly reduced the strength of the system. Many different approaches were explored to solve this problem. The most effective procedure involved a densification process. The inner mold line of the tile was densified by filling the voids in the tile fiber material with a ceramic slurry. It was found that this densification process distributed the loads induced by the individual fiber bundles in the SIP. It also provided adequate strength such that the failure of the tile/SIP system always occurred in the tile and not at the SIP to tile joint. With this deficiency corrected, a truly functional design of the Orbiter TPS had been achieved. The various sources of tile stresses, the stress analysis methodology, description of the various structural integrity tests, and verification activities were described in reference 11.

REINFORCED CARBON-CARBON

Materials Selection

Carbon-carbon material development during the Shuttle technology phase programs in the early 1970's was conducted at the McDonnell-Douglas and Vought Corporations (reference 12 and 13). These early material investigations were divided into two classes, substrates and coatings. The ultimate test of the materials was their compatibility as a system. Many combinations of carbon filaments, binder materials, inhibiting and coating materials were developed and tested. Both yarns and cloths of carbon and graphite, with phenolic and epoxy as initial binders were evaluated for basic strength properties. Further strengthening was accomplished by the chemical vapor deposition (CVD) reimpregnation process and by reimpregnation with pitch and furfural alcohol. Various metal and boride oxidation inhibitors were considered as diffused-in coatings and as additives to the initial binders. The

add-mix oxidation inhibitors reduced the interlaminar strength properties. Extensive oxidation and structural testing indicated that the cloth with carbon binder and a silicon carbide diffusion coating provided the highest temperature and strength reuse capability.

The selected material used on the Orbiter is an all-carbon composite produced by the Vought Corporation. Graphite fabric, preimpregnated with phenolic resin, is laid-up in complex shaped molds and cured. Once cured, the resin polymer is converted to carbon by pyrolysis. The part is then impregnated with furfuryl alcohol and pyrolyzed three more times to increase its density and strength. The carbon-carbon is protected from oxidation by converting the outer carbon plies to silicon carbide in a diffusion coating process. Further oxidation protection is provided by (1) impregnation of the laminate with tetraethyl orthosilicate (TEOS) which, when cured, leaves a silicon dioxide residue throughout the coating and substrate and (2) a final surface sealant treatment consisting of sodium silicate and graphite fibers.

LESS DESIGN CHALLENGES

The primary purpose for the LESS is to provide thermo-structural capabilities for the regions of the Orbiter that exceed 23000° F. Operational requirements include the retention of aerodynamic shape of the outer moldlines and interface control between the RCC and the RSI tiles. Additionally, control of the aluminum structure temperature to less than 350° F., servicability for easy access and removal of the RCC components, and the capability to sustain 100 missions with minimal refurbishment are required.

The life expectancy of the RCC system is reached in one of two ways. The first is by erosion of the SiC coating which exposes the unprotected carbon substrate to direct oxidation. The second is by subsurface oxidation where the substrate is oxidized with the coating intact in such a way that either the strength of the substrate is diminished or the adherence of the coating is lost.

During the NASA technology phases (1970), coating erosion performance was primarily emphasized. At temperatures above the oxidation threshold of silicon carbide (2700° F), the coating will ablate. Thus it will tolerate only a limited number of entries (ref. 14). Therefore, a post-coating heat treat process was applied to the coated RCC part at 3200° F for 45 minutes in argon. This process step was a standard procedure to enhance the coating resistance to oxidation. However, plasma arc tests performed in 1972-73 at NASA/JSC and NASA/Ames indicated that non-heat treated RCC specimens experiences less mass loss than the heat treated RCC. Further investigation revealed oxidation of the substrate at the coating substrate interface. This was a result of inherent microcracks in the silicon carbide coating. Extensive air oxidation tests over a wide range of temperature and pressures, microanalyses, and mechanical property tests were conducted. These tests characterized the subsurface oxidation and its impact on RCC mission life (ref. 15 and 16).

The resultant strength degradation caused by the substrate mass loss restricts the mission life capability through the inability of the RCC to sustain the predicted loads. Therefore, the TEOS impregnation process was developed which infiltrated the silicon carbide coating and carbon substrate. This resulted in increased oxidation protection. Before delivery of the Columbia, element tests revealed the possibility of getting porous substrate in some areas of the production parts. High porosity in the substrate reduces the effectiveness of the basic SiC coating and the TEOS impregnation. Consequently, the oxidation rate in the porous region is increased. In some cases, the mission life of the affected part is reduced. A post-coating treatment of sodium silicate and graphite fiber sealing the surface porosity has minimized this undesirable surface condition.

Critical manufacturing challenges were to hold ± 10 mil tolerance on large molded parts. Additional challenges were high temperature tooling and developing Non Destructive Evaluation (NDE) techniques to monitor process and insure consistent hardware.

The most difficult technical problem was maintaining dimensional tolerances for the gap and step requirements (fig. 9) in the LESS design. An extensive manufacturing tolerance and process program characterized the dimensional changes of an RCC part. Dimensional fit and control was then achieved by designing the growth and expansion of the RCC into all stages of tooling. Since all trim and drilling of the RCC parts is performed before the coating process, the ability to predict any growth or shrinkage after coating is most important. A final assembly fixture (fig. 10) was used for fit-up of the RCC wing panels and seals just before coating as well as for final assembly of the coated panels with the attach fittings. A similarly employed assembly fixture is used for the nose cap and seals.

Continual inspection, acceptance testing, and weight measurements are performed during the RCC fabrication process. NDE inspection (visual, x-ray, ultrasonic and eddy current) of the parts along with strength and mass loss testing of control panels processed with each part assures acceptable substrate in all hardware.

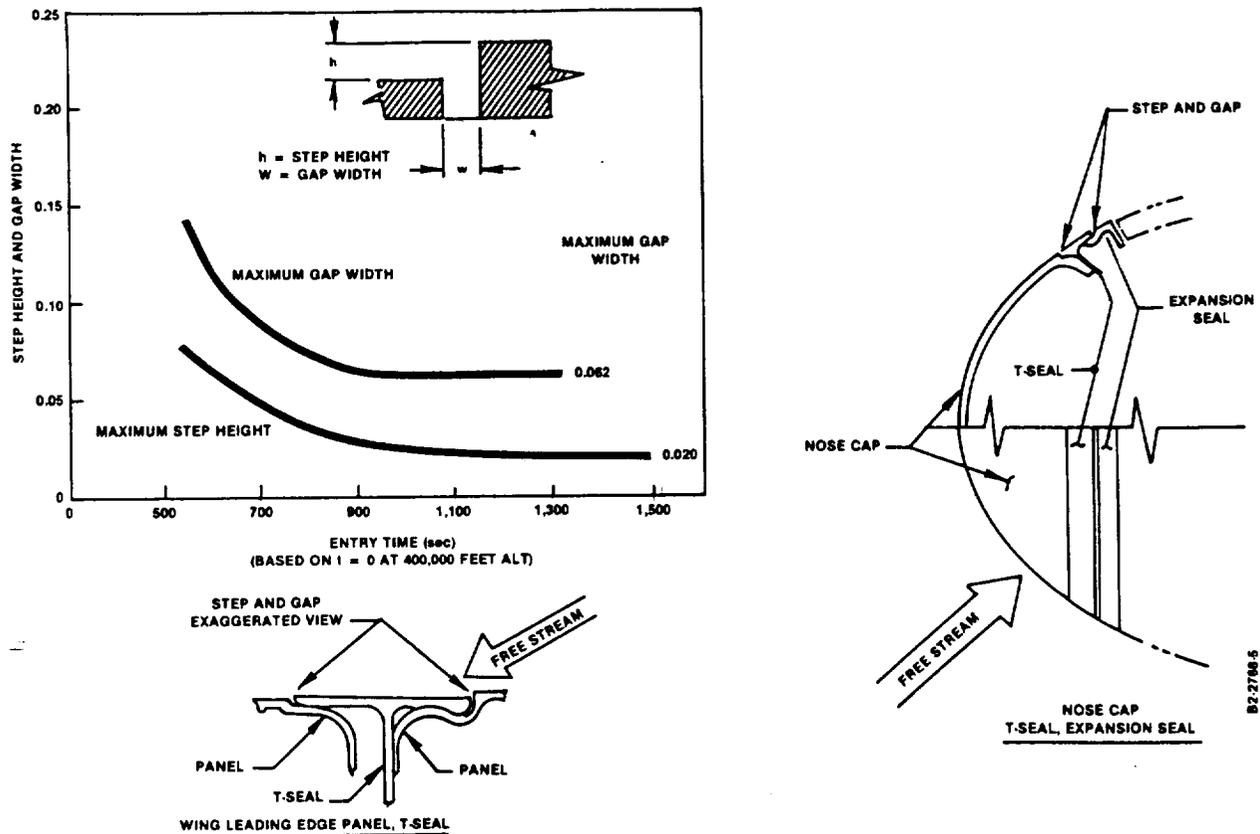


FIGURE 9.- SPECIFICATION REQUIREMENTS (STEPS AND GAPS).

Elevated temperature is the primary factor in the design of the attach fittings, the internal insulation system for the protection of these attachments, and the adjacent aluminum structure. Thermophysical properties of the RCC material (i.e., good conductor) and the hollow shell design promote internal cross-radiation from the hot stagnation region to the inherently cooler regions. Since structural fasteners were required to be held below 1200° F., they needed to be located inside the outer moldline (OML). However, as the attachment flange was lengthened, it also intercepted more cross-radiation from the panel surface. Since the attach hardware also intercepted the OML radiation, it was necessary to provide insulation to limit the temperatures. Heat resistant metals such as Inconel 718 and A-286 steel interface between the RCC and aluminum support structure. These metal components are protected with various insulation packages composed of Dynaflex, AB-312 ceramic cloth, saffil or RSI tiles. Dynaflex contained in formed and welded Inconel 601 foil, is the primary insulation system used in the wing leading edge. Blankets of Dynaflex and saffil wrapped with AB-312 cloth are used in the nose cap cavity along with RSI tiles on the forward face of the access door. Paradoxically, the internal insulation which prevents exceeding the maximum temperature for the metallic attachments also retards the cooling rate of the RCC lugs. This contributes to the undesirable oxidation rate.

ORBITER FLIGHT PERFORMANCE

RSI FLIGHT PERFORMANCE

The RSI tile performs its required thermal protection function during entry by two primary means of heat dissipation. A large percentage (~98%) of the heat energy is reradiated to the atmosphere by the high-emittance glass coating on the tile surface. The remaining heat energy is effectively retarded by the low diffusivity of the basic insulation material. The basic thermal performance of the RSI tiles can be evaluated by three important parameters: the induced surface temperature profile, the transient response of the RSI interior, and the structural temperature response.

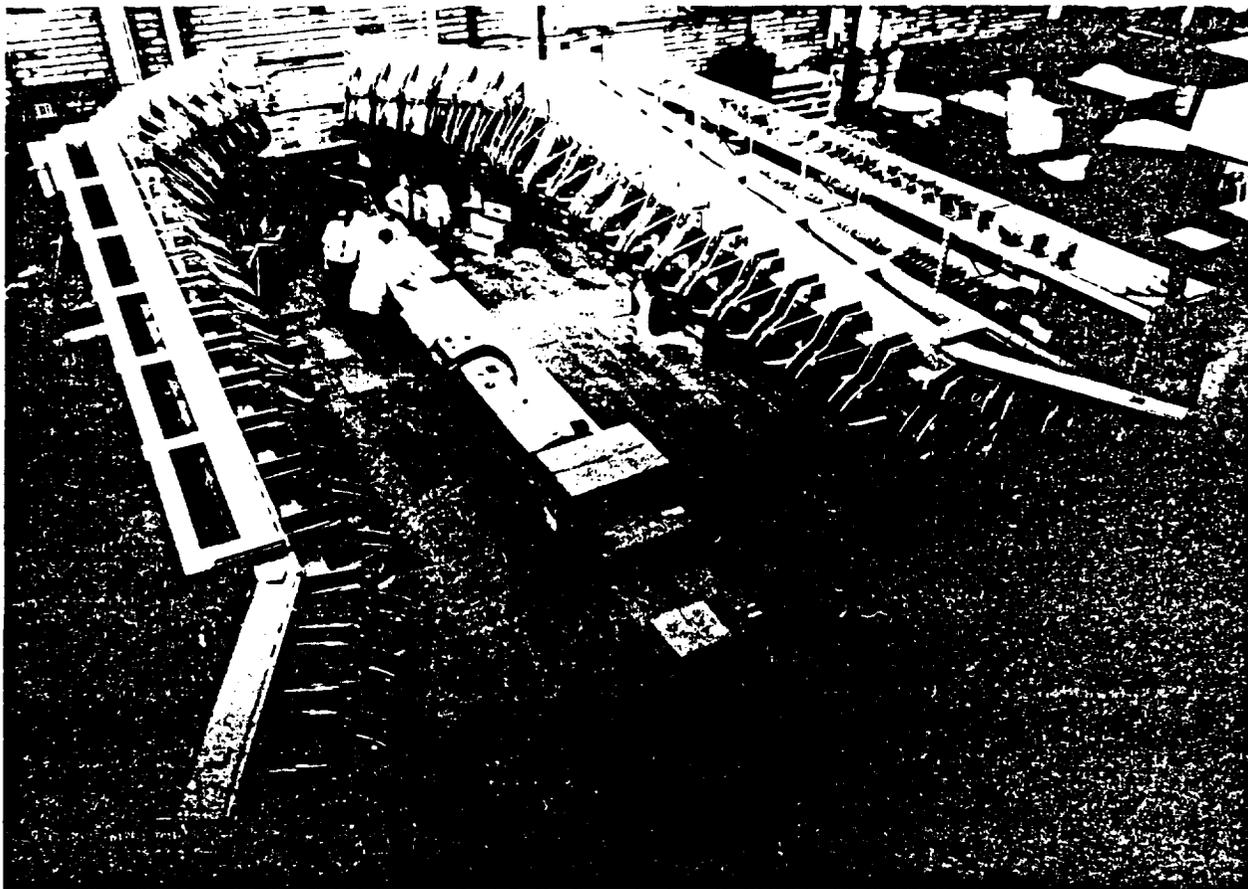


FIGURE 10.- WING LEADING EDGE FINAL ASSEMBLY FIXTURE.

Figures 11 to 13 show comparisons of flight data from STS-5 with predicted performance. In this case, the measured RSI surface temperature was the driving function for the thermal math model. In general, good agreement is shown between the flight data and predictions. Figures 14 and 15 show the distribution of peak surface temperatures and structural temperatures experienced during representative Orbiter test flights. The temperatures on some localized areas such as the OMS pods were higher than expected. Therefore, modification of the TPS was required for the later vehicles that will be flown at the more severe entries typical of the Western Test Range. For the most part, the measured surface temperatures during entry have been lower than expected for the flight tests flown from the Eastern Test Range. The lower surface temperatures indicate lower surface heating rates and a lower heat load into the structure. The lower heating rates are attributed to the noncatalytic effects of the TPS tile coatings. Other contributing factors include later than expected transition from laminar to turbulent flow and internal convective cooling effects on the structure during the later part of entry which had not been accounted for in the thermal math models. These effects are summarized in figure 16.

Tile structural integrity for the most part has been excellent during the flight test program. Since there are so many parts (~31,000 tiles), the tile attachment to the structure has always been one of the major concerns. None of the HRSI tiles on the lower surface have been lost during the flight test program. Some undensified white tiles were lost on the OMS pods during STS-1. This was because of improper machining operations of the diced LRSI tiles in this area. Also, loss of undensified tiles during the STS-3 mission on the upper forward fuselage area and the upper body flap area was attributed to excessive application of the tile rewaterproofing agent. Keeping moisture out of the tile has been one of the major technical problems encountered during the flight test program. The anomalies due to moisture absorption that have occurred during the flight test program, the corrective actions, and ongoing improvement activities are discussed in reference 17.

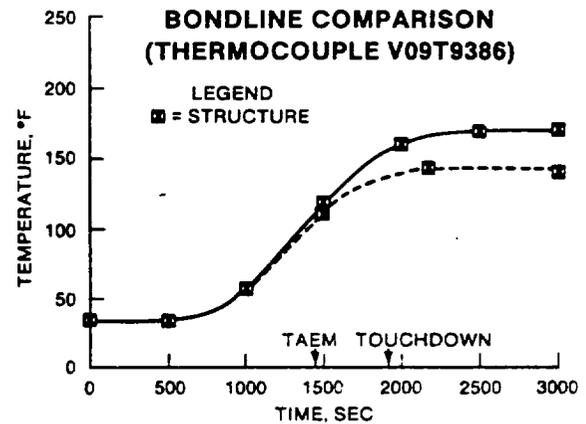
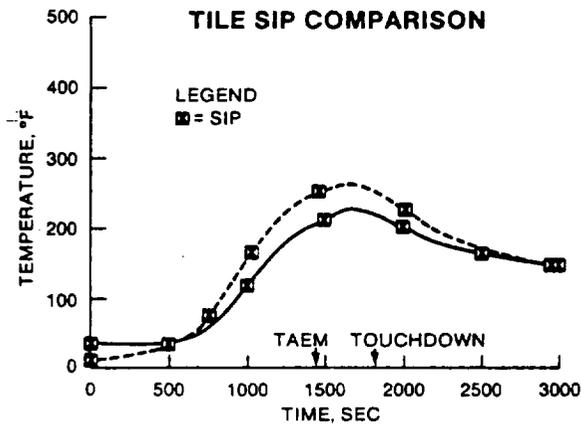
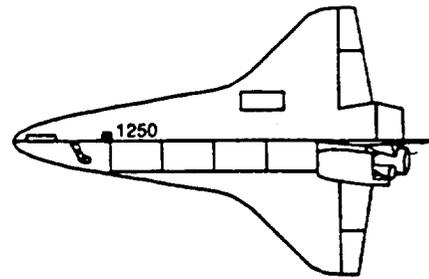
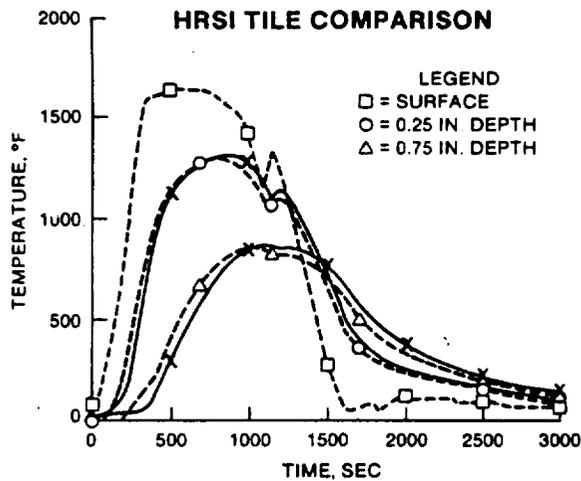


FIGURE 11.- STS-5 FLIGHT DATA (DASHED LINES) VERSUS ANALYSES (SOLID LINES): BODY POINT 1250.

Overall, the TPS tiles have performed exceptionally well despite exposure to adverse weather conditions and debris damage during ascent. Because this material has low impact resistance, minor surface damage in the form of dents, gouges, and coating chips has occurred during all of the flights. This damage is attributed to ascent debris from the external tank and the solid rocket boosters (SRB). A worse case example after entry is shown in figure 17. Some of the conditions have been corrected. A smaller amount of damage occurred during the STS-4, STS-5 and STS-6 flights. The damage that has occurred has not resulted in any significant degradation of the overall Orbiter tile thermal performance. The majority of the damaged areas were readily repairable by use of ceramic filler agents. There has also been some surface contamination of the TPS outer surface. This contamination comes primarily from the decomposition of silicone materials that are used in the gap fillers. Additional contamination is from the aluminum oxide in the SRB plumes and decomposed external tank insulation. The deposits for the most part are surface effects and have not resulted in any loss of thermal performance or life.

Excessive tile-to-tile gap heating has occurred in a number of locations and has been a continual problem during the flight test program. Extensive analyses of the flight data and ground test results have been undertaken to understand this complex flow phenomenon. The excessive gap heating, attributed primarily to excessive tile-to-tile steps and gaps, has resulted in tile sidewall shrinkage, filler bar charring, and localized severe structural temperature gradients in one instance. Gap fillers that have been installed in those areas where excessive gap heating occurred have been completely effective.

In conclusion, even though a number of anomalies have occurred during the first five flights of the Orbiter Columbia and the first flight of the Challenger, the overall thermal-structural performance of the ceramic silica tiles has been far better than expected. The flight performance characteristics of the RSI TPS during the STS-1 through STS-5 missions has been presented in reference 2, 3, and 18.

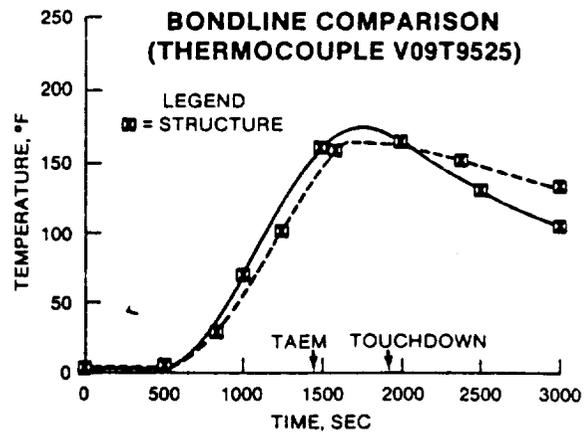
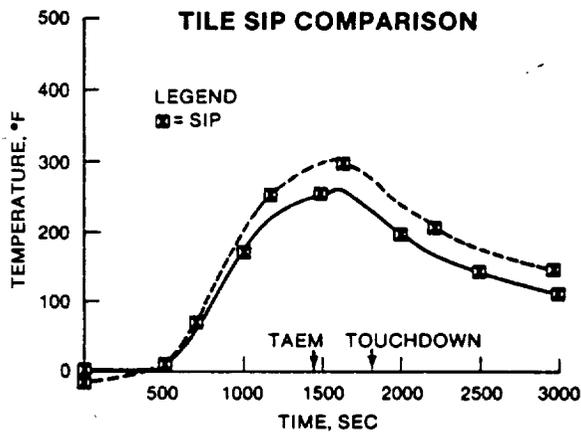
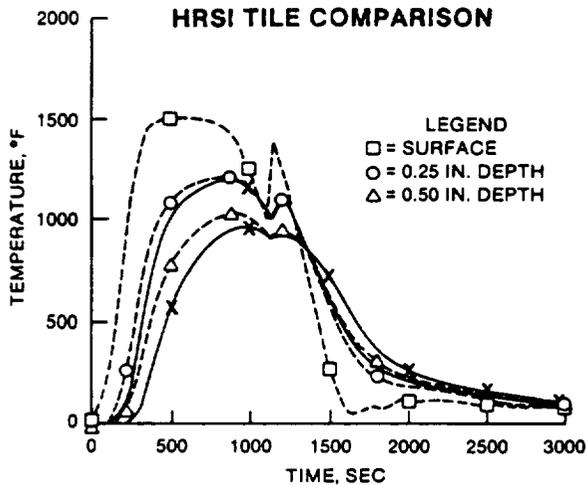


FIGURE 12.- STS-5 FLIGHT DATA (DASHED LINES) VERSUS ANALYSES (SOLID LINES): BODY POINT 1500.

LESS FLIGHT PERFORMANCE

Successful completion of the first five development flights of the Orbiter Columbia have provided sufficient engineering data, coupled with postflight inspections after each flight, to assess the thermostructural design and capability of the LESS. Peak nose cap radiometer temperature measurement (STS-5) of the inner moldline (IML) of the RCC shell are presented in figure 18. A comparison of the predicted and measured stagnation point-IML transient temperature histories (fig. 19) shows the STS-5 flight data 180° F. higher than the prediction but approximately equal to the predicted design temperature using the 14414.1C design trajectory.

Nose cap internal insulation, attachment, and bulkhead measured temperatures for the STS-2, 3 and 5 flights are shown in figure 20. These temperatures agree with predictions made using the thermal math models developed to support flight certification. Postflight visual inspections of the RCC external surface and the nose cap conic internal insulation blankets have indicated no anomalies or degradation. However, damage to the nose cap's lower windward surface interface tiles has occurred as a result of excessive tile-to-tile and RCC-to-tile gap heating. Corrective action in terms of redesigned gap fillers and flow stoppers has been incorporated into this interface area. This eliminates the overheating condition. Radiometer measurement of wing leading edge IML temperature as a function of semi-span is shown in figure 21. The maximum heating zone (45% to 55% semi-span) results from the interaction of the nose cap bow shock and wing shocks producing higher boundary-layer pressures and heat rates. Comparing predicted and measured RCC shell IML temperatures indicates heat flux levels substantially lower than predicted for panels 4 and 22. There was excellent agreement for panel 16 and heat flux levels higher than predicted for panel 9. Wing panel 9 is located in the "double shock" heating zone. The results shown in figure 22 indicate the measured RCC IML tempera-

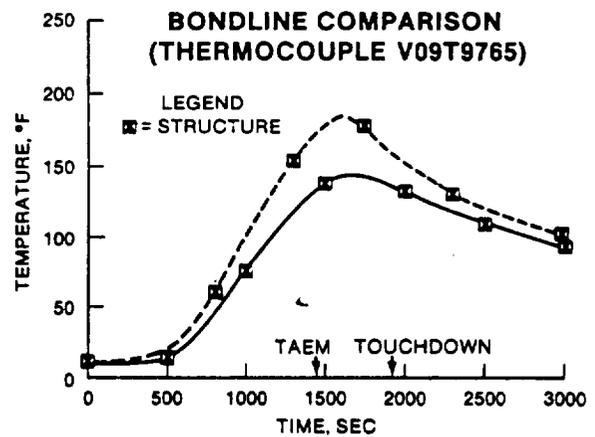
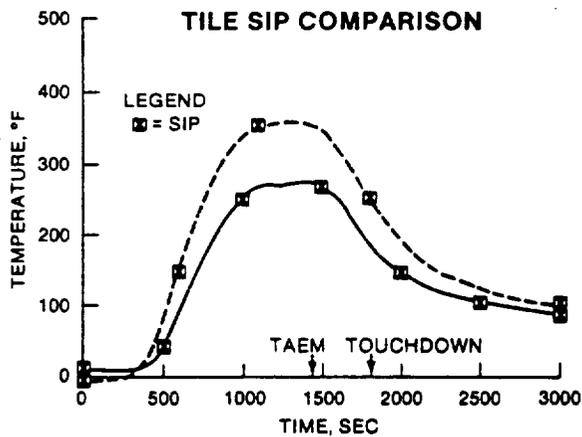
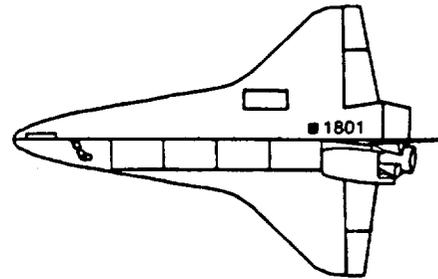
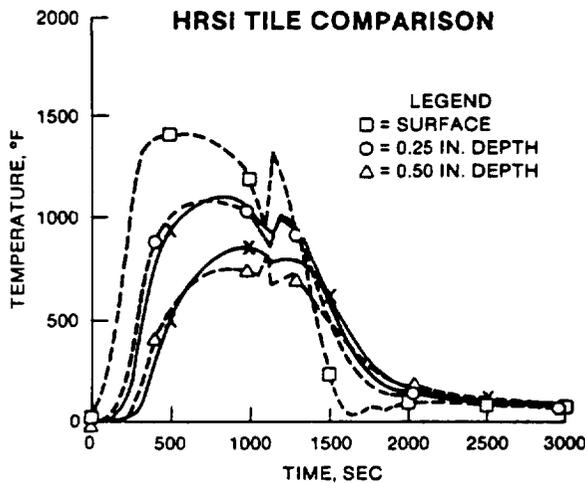


FIGURE 13.- STS-5 FLIGHT DATA (DASHED LINES) VERSUS ANALYSES (SOLID LINES): BODY POINT 1801.

ture (2601° F.). A comparison of measured temperatures of the RCC attachment hardware, internal insulation, and wing spar structure for the STS-2, 3 and 5 flights are shown in figure 23. These temperatures agree with the preflight/postflight predictions made using the math models verified from certification tests.

The wing leading edge panels have all been examined externally for evidence of any anomalies, i.e., coating chips, cracks, etc., after each STS flight. In addition, selected wing panels have been removed for detailed internal inspection of the interior surfaces of the RCC, attachment hardware and Inconel/Dynaflex insulation. The associated upper and lower access panels have also been inspected after flight for evidence of deterioration.

During the STS-2 postflight inspection, areas of discoloration were evident on the wing leading-edge upper access interface panel at RCC left-hand panels 9 through 13 and right-hand panels 10 through 13. This discoloration (white deposit/streaking on black tiles) was the result of gas flow through the subject panels entering the RCC/RSI lower access panel interface. An examination of the lower access panels show gap filler/thermal barrier heating, embrittlement, and discoloration. Inspection of the wing leading-edge front spar showed evidence of gas flow streaking and heating of the tile filler bar (discoloration, scorching, burning) at the RCC panel/T-seal joints. Before the STS-3 flight, larger diameter corner gap fillers were added to the four corners of the lower access panels. Postflight inspections of STS-3 - 5 indicate virtual elimination of hot gas flow-through in those RCC panels which had experienced extensive flow-through heating on STS-1 and 2. The flight performance characteristics of the LESS during the STS-1 through STS-5 missions has been presented in references 6, 7 and 19.

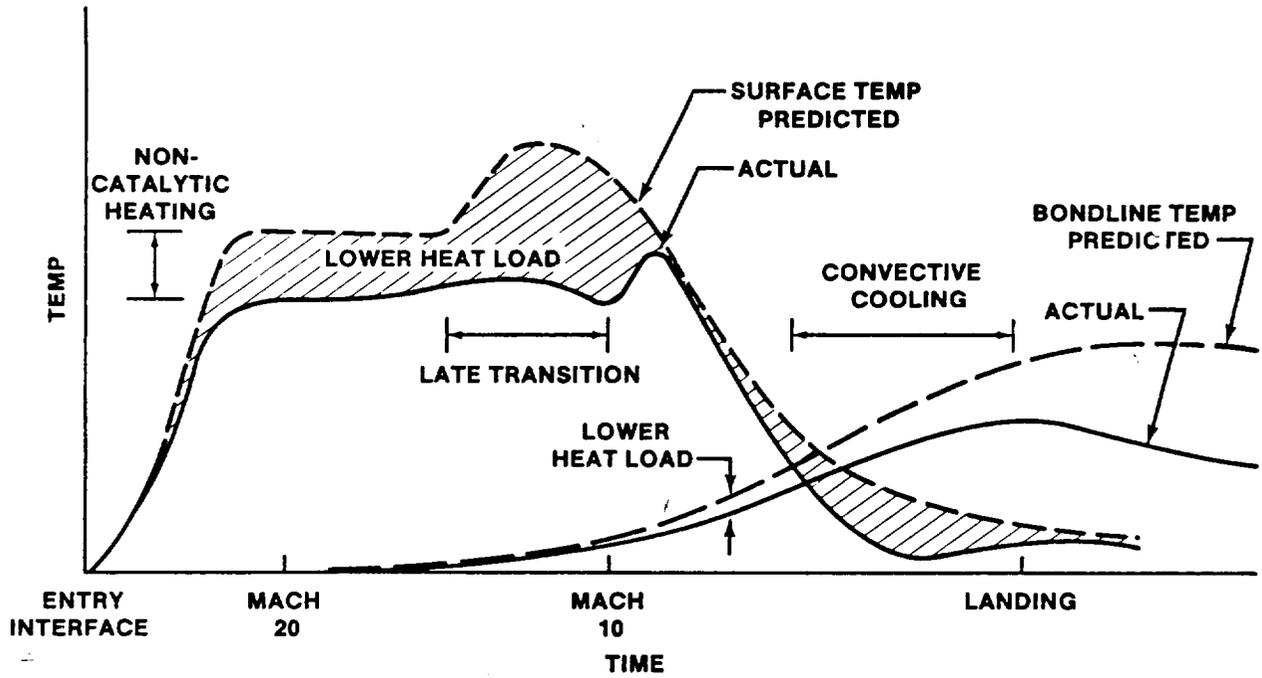


FIGURE 16.- GENERALIZED LOWER SURFACE THERMAL PERFORMANCE RESULTS: STS-1 TO STS-5.

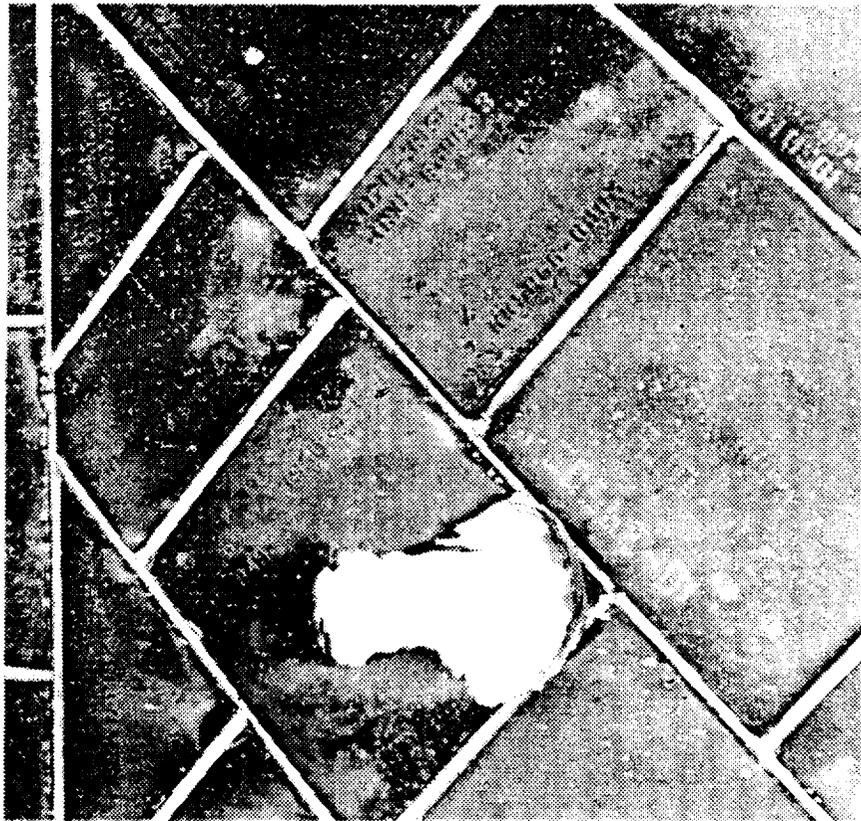


FIGURE 17.- STS-1 DAMAGE AND ENTRY HEATING EFFECTS, BODY FLAP.

ST-2, ST-3, ST-5

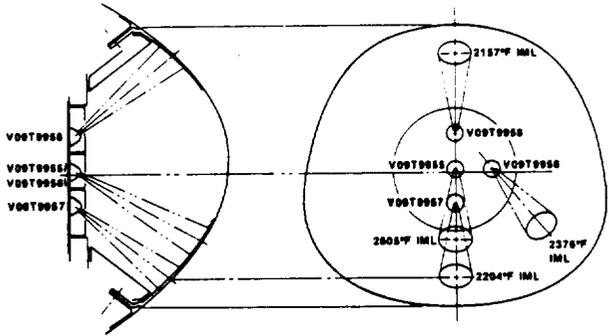


FIGURE 18.- STS-5 NOSE CAP FLIGHT DATA: PEAK RCC IML TEMPERATURES.

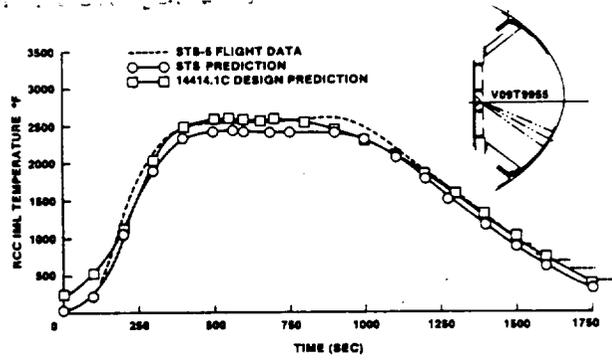


FIGURE 19.- NOSE CAP RCC IML TEMPERATURE.

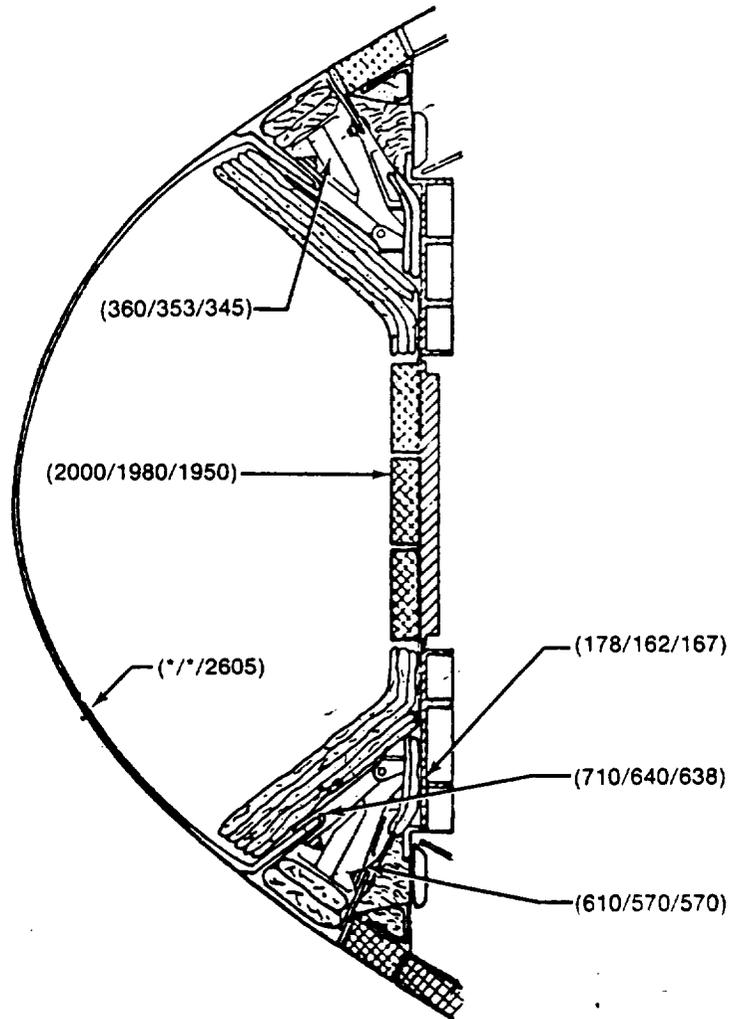


FIGURE 20.- STS-2, STS-3, AND STS-5 PEAK TEMPERATURE (°F): LESS NOSE CAP.

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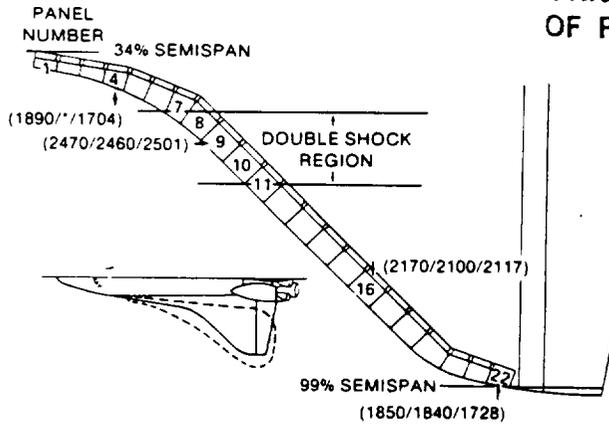


FIGURE 21.- STS-2, STS-3, AND STS-5 PEAK TEMPERATURE (°F): LESS WING LEADING EDGE.

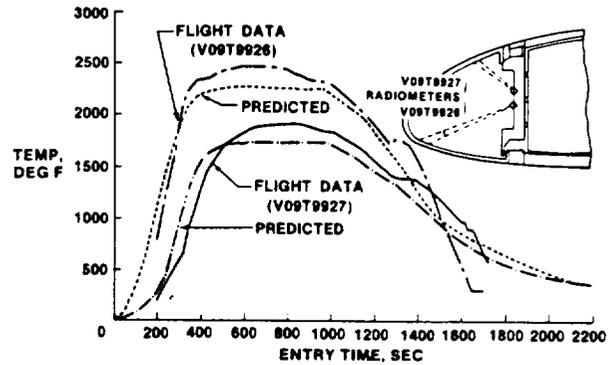


FIGURE 22.- STS-2 FLIGHT RESULTS: LESS-RCC PANEL 9 IML TEMPERATURES.

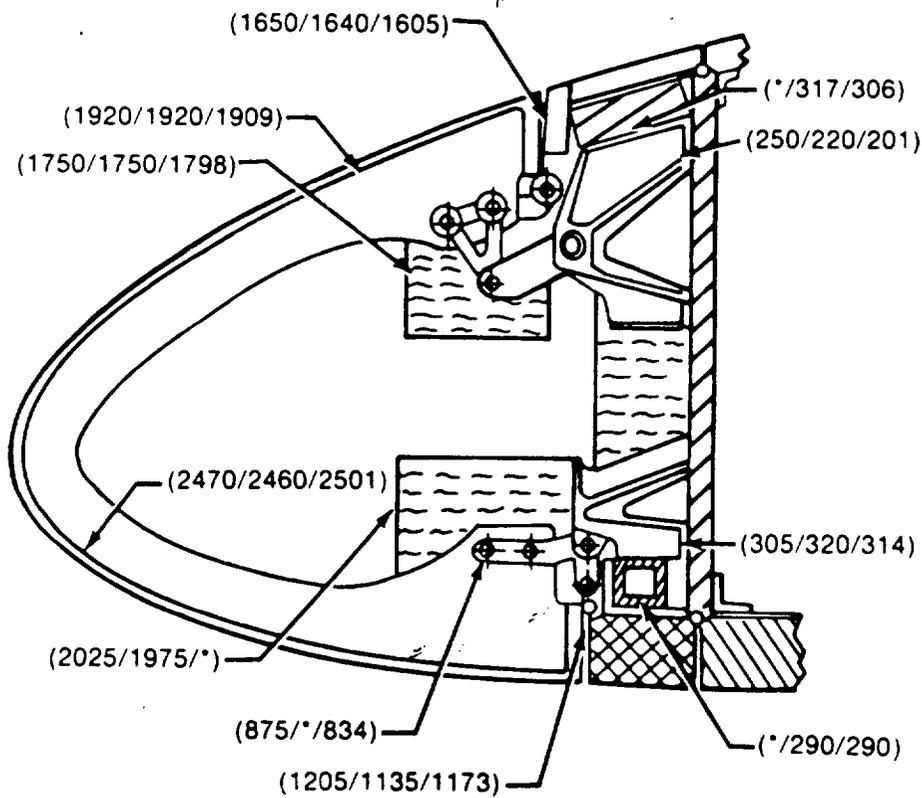


FIGURE 23.- STS-2, STS-3, AND STS-5 PEAK TEMPERATURE (°F): LESS PANEL 9.

CONCLUDING REMARKS

Five successful flights have been accomplished on the Orbiter Columbia TPS and one on the Orbiter Challenger. Satisfactory thermal/structural performance during these flights indicates that the proper thermal protection materials and design approaches were selected for the Orbiter. There are some minor localized areas of the Orbiter where heating or damage exceeded expectations. However, these areas are amenable to minor design modifications. Satisfactory performance is expected during the operational phases of the Shuttle. Degradation of the Orbiter thermal protection system (RSI and LESS) has been minimal, and satisfactory vehicle turnaround operations have been accomplished.

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