Space Shuttle Orbiter Thermal Protection System Processing Assessment

Appendix A

Overview of the Space Shuttle Thermal Protection System

May 1995

Prepared by:

Michael P. Gordon
TPS Orbiter Engineering
Materials & Processes
Rockwell Florida Operations D/830

Summary

The primary purpose of this appendix is to provide an overview of the materials associated with the thermal protection system (TPS) of the Space Shuttle orbiter. Although this appendix is a self-contained informative document, it is intended to accompany the "Space Shuttle Orbiter Thermal Protection System Processing Assessment Final Report". The purpose of the final report is to present the conclusions of a study to assess the processing of the thermal protection system of the Space Shuttle orbiter. This study was initiated on November 10, 1994 in support of the NCC2-9003 Cooperative Agreement Notice (CAN), Lightweight Durable Thermal Protection System, TA-3 Task 6 for the Single Stage To Orbit (SSTO) Lightweight Durable TPS project being performed by Rockwell Downey under the Marshall Space Flight Center (MSFC) NASA Research Announcement (NRA) 8-12 program. This appendix does not contain proprietary information.

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A.1 Space Shuttle Thermal Protection System Overview

The thermal protection system (TPS) of the Space Shuttle Orbiter is unique when compared to other atmospheric reentry vehicles in that it, along with other Orbiter subsystems, is reusable. Of these reusable systems, TPS is unique because the existing design concepts from the aerospace industry could not be utilized in its original development. During a typical reentry heating cycle, the orbiter is subjected to temperatures in excess of 2,300°F as shown in figure A1. The mostly ceramic-based TPS protects the orbiter aluminum and payload bay door graphite epoxy structure and its penetrations from reaching temperatures over 350°F, and the OMS pod graphite epoxy structure from exceeding 250°F. The Shuttle TPS is more than tiles and blankets, as would be the definition from the casual observer. TPS is the integration of all of the materials, development, design concepts, fabrication techniques, installation processes, and refurbishment procedures used to protect a vehicle from the severe heating environment of atmospheric reentry.

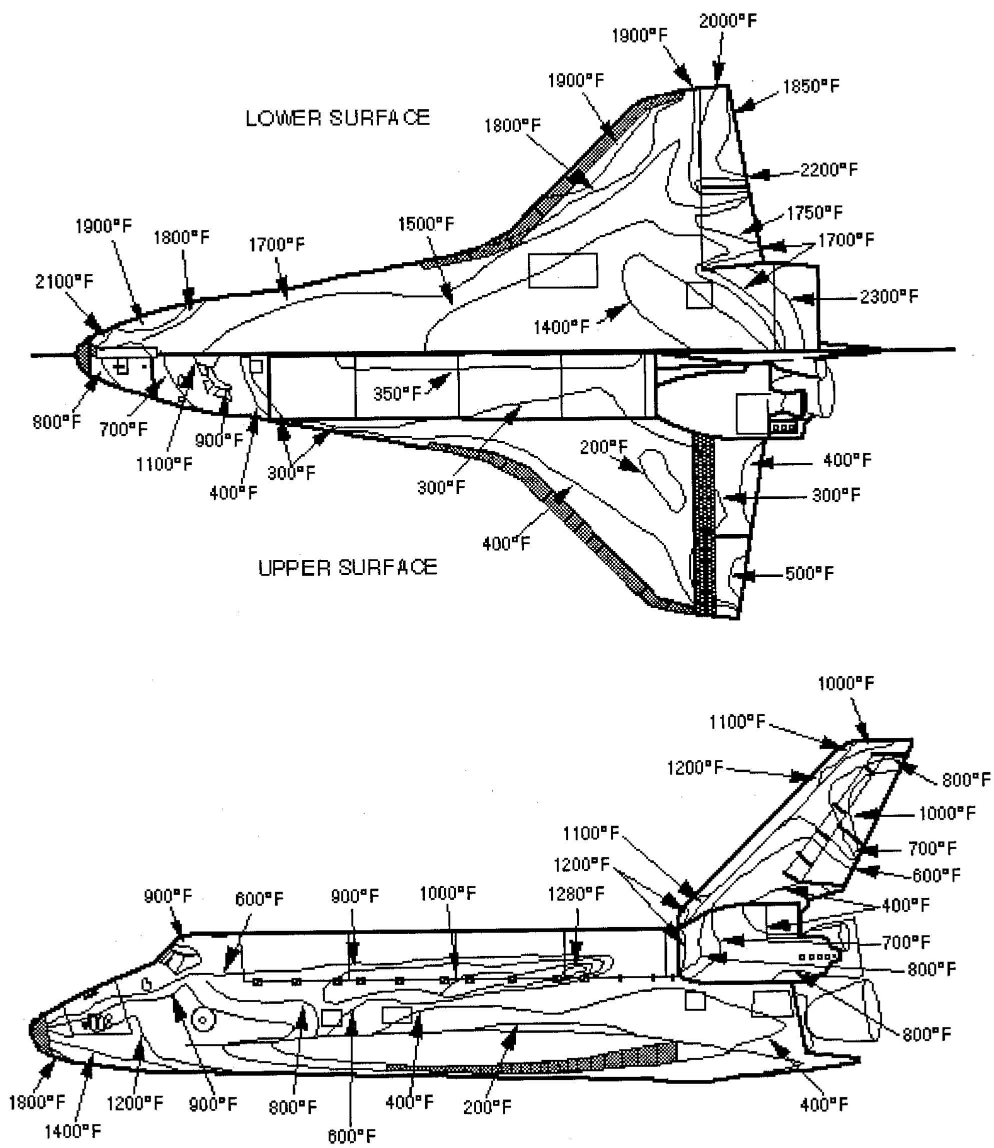


Figure A1. Maximum Recorded OML Surface Temperatures – STS-1 through STS-5

The principle design functions of the TPS are to perform as a radiator (to emit heat), a reflector (to prevent on-orbit heating), and as an insulator (to protect the structure from the residual heat flux). The TPS is primarily white on the upper surface and black on the lower surface to control on-orbit heating from solar radiation and to maximize heat rejection during reentry. By rotating the orbiter so that the more reflective (and less absorbent) white upper surface is towards the sun, the solar heating is minimized. Conversely, directing the black lower surface towards the sun would maximize the solar heating. The high-emissivity black region must be on the lower surface to maximize the heat rejection (in the form of thermal radiation) from the TPS during reentry where this region experiences the highest heat load.

In addition to protecting to structure from heat loads up to 66,000 Btu/ft^2, the outer mold line (OML) of the TPS serves as the aerodynamic shape of the vehicle. This shape is maintained by tight control of the step and gap between installed TPS components. Excessive steps and/or gaps between parts can result in early transition of the laminar to turbulent boundary layer which would result in higher heat loads. Minor steps and/or gaps can result in local overheating which could slump (i.e., melt and deform) tiles or permit subsurface plasma flow, which, in turn, could degrade the TPS bondline or underlying structure.

The Shuttle TPS must also protect the structure from localized heating from plumes of the Space Shuttle main engines (SSME), solid rocket boosters (SRB), orbital maneuvering system (OMS) engines, and reaction control system (RCS) thrusters. In addition to the thermal demands, the TPS also withstands the launch acoustics (up to 166 decibels), structural deflections from aerodynamic loads, on-orbit cold soak temperatures (down to -250°F), environmental exposure at the ocean-side launch pads, and potential damages associated with ground processing.

The primary materials which make up the TPS are as follows:

- Reusable Surface Insulation (RSI) Tiles
- Advanced Flexible Reusable Surface Insulation (AFRSI) Blankets also known as Flexible Insulation (FI) Blankets
- Felt Reusable Surface Insulation (FRSI)
- Reinforced Carbon-Carbon (RCC)
- Gap Fillers
- Thermal Barriers
- Thermal Seals
- Window Thermal Panes

The approximate locations of these materials are given in figure A2 and the specific discussions of each of the materials are provided in the following sections.

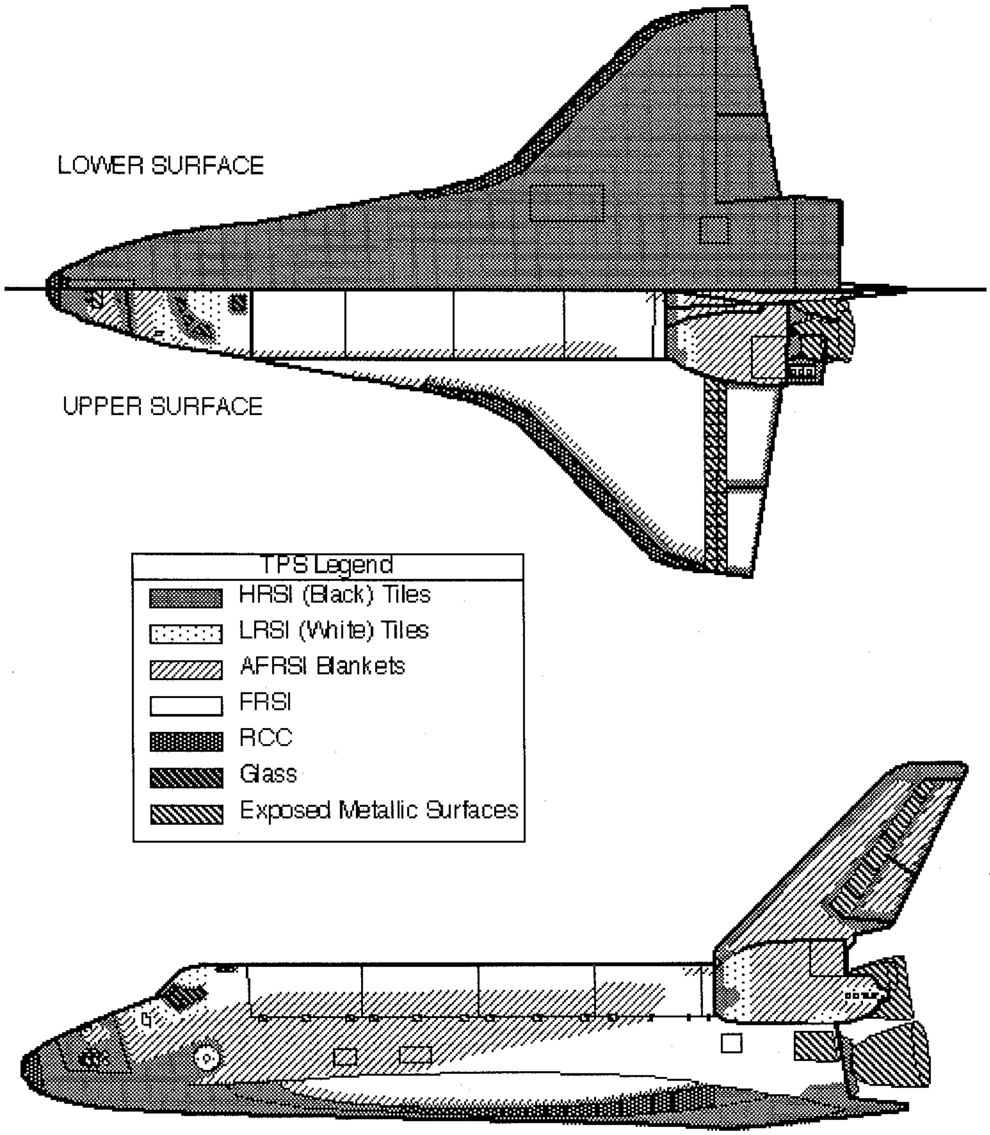


Figure A2. Space Shuttle Orbiter TPS Configuration

A.2 Reusable Surface Insulation (RSI) Tiles

On average there are 24,300 RSI tiles installed on each operational vehicle. It should be noted that there are slightly more tiles on OV-102 (Columbia) due to its original TPS configuration predating flexible blanket technology. RSI tiles are made from one of three substrate materials (LI-900, FRCI- 12, and LI-2200) and are coated with a white or black glass coating. White-coated RSI tiles are referred to as Low-temperature Reusable Surface Insulation (LRSI) and black-coated tiles are known as High-temperature Reusable Surface Insulation (HRSI). LRSI is used in areas where the peak temperatures do not exceed 1,200F, and HRSI is used in regions less than 2,300°F.

The tile substrate material and coating selection are dependent on the mechanical and thermal requirements of the particular location. For example, tiles located on the upper surface of the forward fuselage (some of which are 0.75-inch thick LI-900 LRSI) experience much lower temperatures and require less strength than tiles on the nose landing gear door (which are 2 to 3- inch thick FRCI-12 and LI-2200 HRSI). The thickness of the tiles varies with heat loads and OML contour requirements from less than 1 inch to over 3 inches. The substrate material is machined to the desired shape (usually 6 inch by 6 inch by necessary thickness) prior to coating. The tiles are mostly located on the lower surface on the vehicle, as they have a greater resistance to high heat loads and provide a smoother, more aerodynamic surface than flexible blankets.

Figure A3 depicts a typical RSI tile installation. All of the tiles are bonded to the structure using strain isolator pads (SIP) and room temperature vulcanizing (RTV) silicone adhesives. The IML of the RSI tile is densified prior to SIP bond to uniformly distribute stress concentration loads at the tile-to-SIP interface. The structure beneath tile-to-tile gaps is protected by filler bar. Gap fillers are used in areas of high differential pressures, extreme aeroacoustic excitations, and to passivate over-tolerance step and gap conditions.

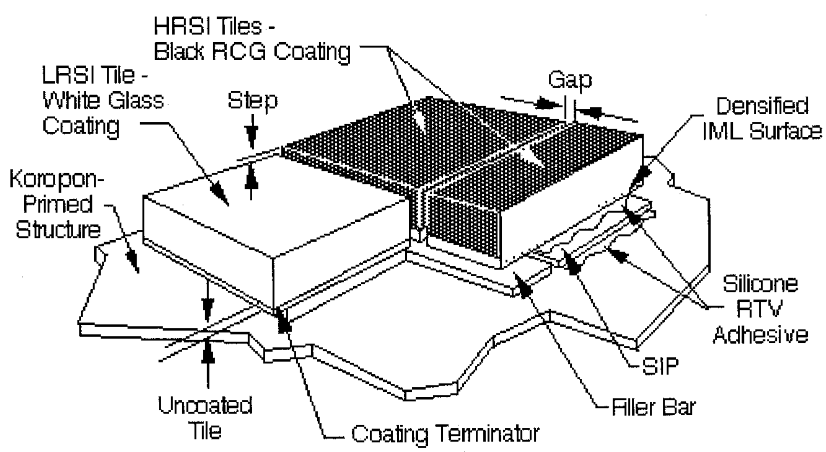


Figure A3. Typical RSI Tile Installation.

RSI tiles require rewaterproofing prior to each mission because the waterproofing compound degrades at temperatures exceeding 1,050'F. The rewaterproofing is accomplished by the injection of at least 2ml of dimethylethoxysilane (DMES) waterproofing compound into each tile. The DMES renders the tile substrate hygrophobic by reactions between the Si-OH groups in the silica and the ethoxy group in the DMES with negligible weight gain. Failure to rewaterproof RSI tiles could result in increased weight (from absorbed water) or tile damage. The damage would be caused by the absorbed water freezing and subsequently contracting on orbit at cold soak temperatures below -70°F, thereby inducing a fracture at the 1,050°F isotherm. During reentry, the absorbed water would convert to steam and complete the failure of the tile by loss of the dewaterproofed region previously fractured. In addition to thermal exposure, the silylated (i.e., waterproofed) surfaces that are not protected by the original tile coating (i.e., damaged or previously repaired areas) could degrade from exposure to atomic oxygen attack on orbit.

A.2.1 RSI Tile Substrate Materials

There are three RSI tile substrate materials currently used on the orbiter, 9 and 22 pcf Lockheed Insulation (LI-900 and LI-2200) and 12 pcf Fibrous Refractory Composite Insulation (FRCI-12). The LI-900 and LI-2200 materials are comprised of high-purity amorphous silica fiber (LI-2200 adds a small amount of silicon carbide powder) made rigid by ceramic bonding. The FRCI- 12 material is similar to the LI-2200 except aluminoborosilicate fiber is added to the silica fiber and silicon carbide powder. The FRCI-12 material is made rigid by boron fusion at the fiber junctions. The fabrication of all three materials is similar, and is shown in figure A4.

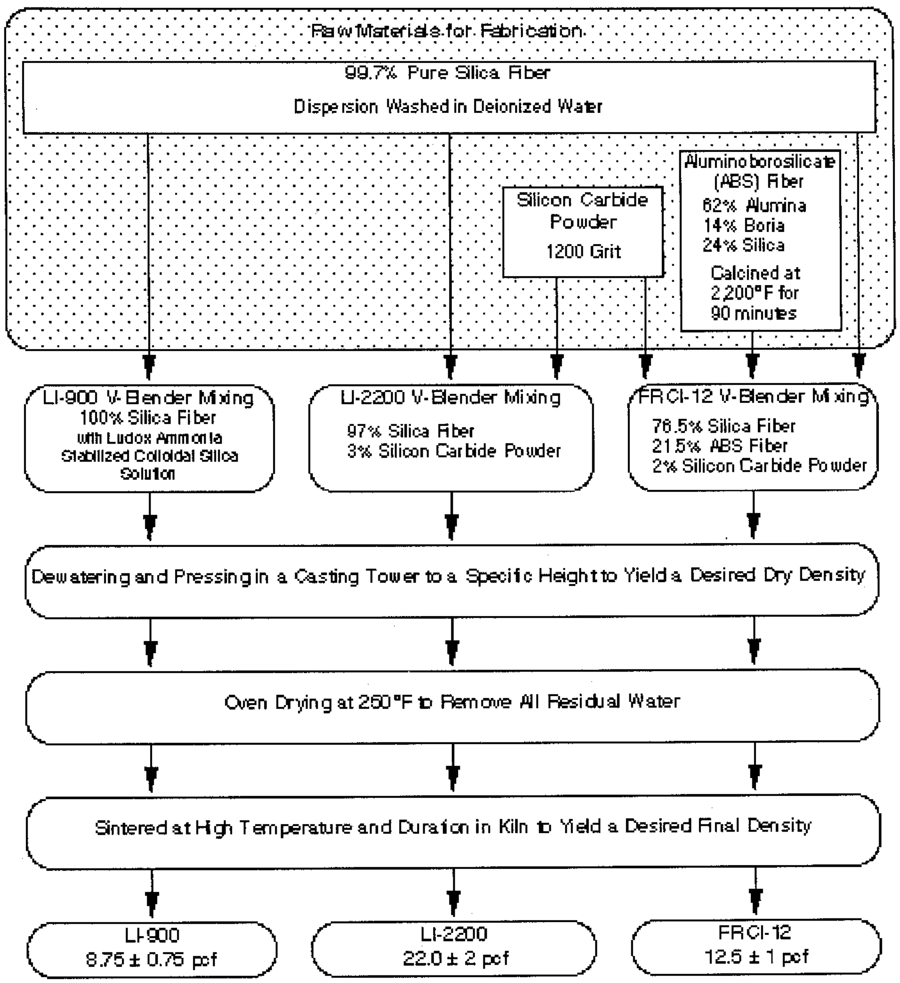


Figure A4. Fabrication Schematic for RSI Tile Substrate Materials.

The fabrication of LI-900 is accomplished in six basic steps. The 99.7% pure silica fiber is dispersion washed in deionized water. The fiber is mixed with Ludox ammonia stabilized colloidal silica solution in a V-blender for a specific duration to obtain the proper length of fibers. The fiber slurry is removed from the V-blender and is poured into a casting tower where excess liquid is removed. The mixture is then pressed in the casting tower to a specific height that will yield the desired dry density. The block is removed from the casting tower and is placed in a low-temperature oven to dry. The dried block is then sintered at a specific high temperature and duration to activate the ceramic bonding and yield the desired final density.

The fabrication of LI-2200 is identical to the procedure for LI-900 except 1200 grit silicon carbide powder is added (3% by weight) to the silica fiber (97% by weight) prior to mixing in the V-blender. The silicon carbide is used to improve the emissivity of the LI-2200 material.

The fabrication of FRCI-12 is similar to LI-2200, with the exception of an additional calcining step for high-boria content aluminoborosilicate fiber (62% alumina/14% boria/24% silica) at 2,200°F for 90 minutes. The calcined aluminoborosilicate fiber is added (21.5% by weight) to the silica fiber (76.5% by weight) and silicon carbide powder (2% by weight) prior to mixing in the V-blender. During the sintering of the material, the boron content in the aluminoborosilicate fuses the fibers together at the junctions resulting in a more rigid structure than the LI-900 and LI-2200 ceramic bonding.

The physical properties and use temperatures of the substrate materials are given below.

Property	**LI-900**	**LI-2200**	**FRCI-12**
Density(pcf)	18.75 ± 0.75	22.0 ± 2	12.5 ±1
IP Tensile Strength (psi)	68	120	141
TTT Tensile Strength (psi)	24	50	52
Thermal Conductivity at 0.001 atm (Btuxin/ft^{2}xhrx°F)	0.021	0.030	0.027
Use Temperature (°F)	2,300	2,300	2,300

A.2.2 RSI Tile Modeling and Machining

There are two distinctly different types of tile machining, tracing a physical model of the cavity on a stylus machine to produce a flight tile or using a numerically controlled (NC) milling machine to create a tile based on a three-dimensional computer model. The use of either method is dependent on the modeling technique employed.

There are four modeling techniques used, a cavity tracer pattern splash per ML0601-9024 process 102, manual computer modeling from master dimension data per process 317, automatic computer modeling on the floor (i.e., by technicians working on the orbiter in the OPF) using tile cavity digitization per process 318, and automatic computer modeling on the floor using the Scanner Closeout Preprocessor And Lofting System (SCOPALS) per process 319.

Following the necessary signatures to authorize the work, the tile is ordered. If applicable, the floor-level cavity modeling is performed by the Shuttle Processing Contractor (SPC), specifically cavity splashes, cavity digitizing, and SCOPALS picture frame fabrication. A tile traveler (i.e., form used to obtain a replacement tile) is issued and, with any additional items (tracer patterns, floppy diskettes, or mylar picture frames), is forwarded to the Thermal Protection System Facility (TPSF) for further processing.

Splashes involve the fabrication of a tile from a tracer pattern using a physical model of the cavity. A tracer pattern is made from a polyisocyanate and polyurethane foam casting of the open cavity. The OML is faired to be flush with the adjacent RSI and, as a result, the technician performing the splash approximates some of the design features, such as the contour of the OML. As a result, the MD configuration of the orbiter is oftentimes not maintained. Drawing defined features, such as delta lips, are not modeled on the tracer pattern. Instead, the sidewall lip is noted on the pattern and the lip is machined to theoretical dimensions following the machining from the tracer pattern. Splashing a tile cavity is a time consuming process, which can take up to a full. shift to produce a tracer pattern. Despite the time consumption, splashes can be performed on all cavities and it is an efficient method for modeling sidewall jogs and other non- design features. Splashes are convenient in that they provide a real-time determination of fit to adjacent tile and, as a result, minimal step and gap rework is required for the bonded replacement tile. The machining of the tile from a tracer pattern is a less accurate process than NC machining as the tracer stylus often chatters on the tracer pattern resulting in poor dimensional stability. The materials that comprise the tracer pattern require the technicians to wear protective equipment. In addition, the tracer patterns are extremely moisture sensitive and can degrade while in storage in the VAB. Therefore, the storage of these patterns results in costly inventory which, oftentimes, returns an unusable product.

The NC machining is a more precise method using current technology such as automation and, in some cases, optical modeling. The cavity models can be created in a variety of ways, but most methods utilize theoretical MD data. Therefore, for most NC related processes, the MD configuration of the orbiter is maintained. NC models can be made of any cavity. The cavity models are saved electronically, whereby they are easy to recall, modify, and used to recreate tiles. One of the disadvantages of NC machining is that the initial programming can be time consuming, especially if sidewall jogs or non-design features are required. This time is easily offset for a recurring replacement tile, such as a landing gear door corner tile or a tile adjacent to a RCS thruster. Sometimes, the theoretical tile IML does not correspond to the vehicle structural configuration. In these cases the NC tile does not fit without corrective rework. Another disadvantage of NC machining is the cost of the associated hardware, but this cost is offset by the time saved in modeling and the high quality of the finished product. NC models are saved as electronic data which require storage on magnetic media. Provided the media is kept in an office environment, the models are highly reliable.

In accordance with the standard process for the installation of replacement RSI tiles (ML0601-9024 process 301), the coated, undensified tile is sent to the SPC for prefit into the cavity and is evaluated for conformance with the installation step and gap criteria outlined in the ML0601-0001 specification and engineering drawing. If the tile is acceptable per this evaluation, the tile fabrication process continues with IML densification, waterproofing, and SIP bonding. If the evaluation indicates the tile is not within the installation requirements, the tile is sent to engineering for further disposition. Refer to figure A5 for a graphical representation of the modeling, machining, and evaluation processes.

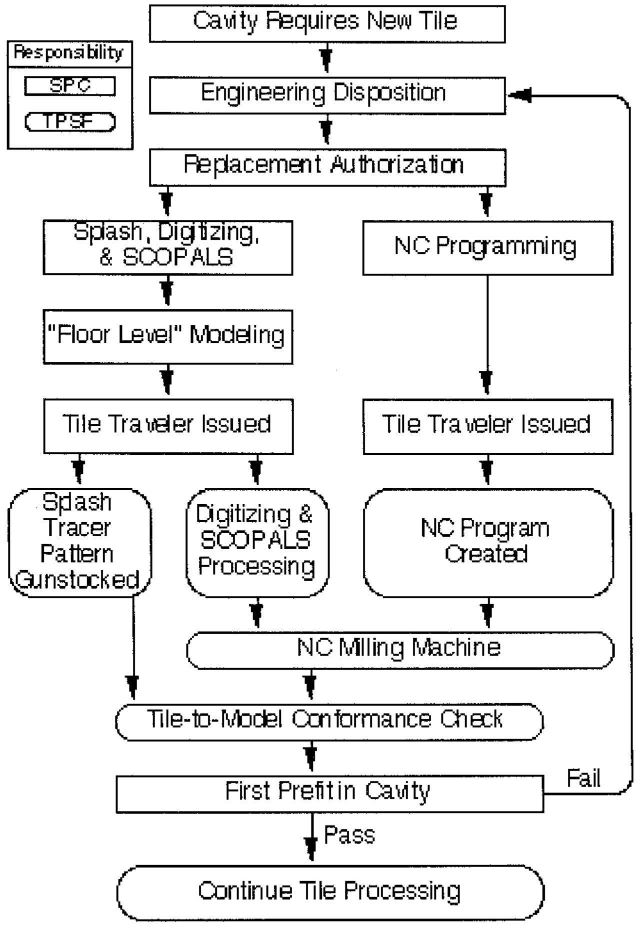


Figure A5. RSI Tile Modeling and Machining Flow Diagram.

A.2.3 RSI Tile Coating, Factory Waterproofing, and IML Densification

There are two types of RSI tile coating materials, a white (for LRSI tiles) and a black (for HRSI tiles) glass coating. Either of the two coatings can be applied to the three substrate materials, and is governed by the engineering drawings.

The white coating is completed by a seal coat, top coat, and firing process. A 10%-Ludox ammonia stabilized colloidal silicaldeionized water seal coat solution is sprayed on the tile OML and sidewalls, leaving the terminator vent zone (an area approximately 0.2 inch above the tile IML) uncovered. The seal coat is dried and the tile is heat cleaned at 1,100°F to 1,450°F for 10 minutes. One coat of the water-based borosilicate glass slurry/acrylate thickening agent top coat is sprayed on the tile. The first coat is air dried and a second coat is sprayed. While the second coat is wet, the tile is oven dried at 1,150°F for 30 minutes. The tile is sintered at 2,100°F for 70 minutes. The fired coating weight is 0.07 to 0.17 lb/ft2 and the coating thickness is 0.007 to 0.011 inch.

The black reaction cured glass (RCG) coating is accomplished by a top coat and firing process. The RCG slurry contains powdered borosilicate glass fit, tetraboron silicide powder, and a methylcellulose suspension agent in a denatured alcohol carrier. The tile is heat cleaned at 1,100°F to 1,450°F. The tile is wetted with denatured alcohol and sprayed with 9 to 13 coats of the RCG slurry. The coating is air dried for 3 hours and the tile is sintered at 2,215°F for 95 minutes. The fired coating weight is 0.09 to 0.17 lb/ft^2 and the coating thickness is 0.009 to 0.015 inch.

Each of the tiles is identified using a black or white very high temperature (VHT) paint in the opposing color to the tile coating. The identification includes the part number from the engineering drawing, the order control number (OCN) to provide traceability, and any other necessary markings (e.g., instrumentation markings, MR designations, etc.).

All of the RSI tiles require factory waterproofing. The original waterproofing is accomplished by the vapor deposition of methyltrimethoxysilane. The tile is placed in a vacuum deposition oven heated to 350°F and is held at a minimum of 27 in. Hg. Heated acetic acid is first injected into the vacuum chamber followed by heated methyltrimethoxysilane. The silane renders the tile substrate hygrophobic by reactions between the SiDOH groups in the silica and the ethoxy group in the silane with negligible weight gain. A minimum of 0.2% silane weight pickup is required to accept the tiles as being waterproofed.

The IML surface of every RSI tile is densified to evenly distribute stress concentrations at the SIP-to-tile interface. The densifying material consists of a mixture of dispersed ground high-purity silica into a Ludox ammonia stabilized colloidal silica solution and tetraboron suicide. The waterproofed IML area is wetted with isopropyl alcohol and the material is applied to a specific weight pickup per unit area. The tile is air dried for 24 hours and then the tile is heated at 400°F for 2 hours to remove any residual acetic acid from the waterproofing process.

A.2.4 RSI Tile Bondline/Subsurface Components

The tiles are bonded to the structure via a strain isolator pad (SIP). The majority of SIP is a non-heat-treated Nomex polyaramid felt pad. The discussion of the Nomex material is given in section A.4. The SIP is available in three thicknesses, 0.090, 0.115, and 0.160 inch. The 0.090-inch SIP is used for high-modulus bonding applications, such as adjacent to thermal barrier installations (where the tile encounters side loading in addition to flight loads). The 0. 160-inch SIP is commonly used in acreage applications. The SIP is bonded to the tile IML following the densification and vapor deposition waterproofing operations. The SIP is bonded to the IML with a RTV silicone adhesive under vacuum pressure. In most situations, the SIP periphery is located one-half inch within the periphery of the tile IML to allow access for the filler bar installation on the structure. The SIP-bonded tile is routed for cavity installation.

The filler bar is bonded to the structure beneath the tile gaps. Filler bar is also used in a similar fashion for AFRSI installations. The heat-treated Nomex felt strips (usually 0.75 inch wide) are bonded with RTV silicone adhesive under pressure in a lattice pattern prior to RSI tile installation. The filler bar provides thermal insulation to the structure from hot plasma flow into the tile gap. The filler bar also provides a seal between the structure and tile IML, protecting the tile bondline. The filler bar can withstand 800'F topside exposure.

In certain regions of the orbiter, pre-cast RTV silicone heat sinks are installed beneath the bondlines of RSI tiles, AFRSI, or FRSI. The heat sinks are used to uniformly distribute backface heat loads to reduce thermal gradients within the orbiter structure.

To compensate for mismatches at structural interfaces or around fasteners, a RTV silicone adhesive (screed) is used to fill voids and provide a smooth surface for RSI bonding.

A.2.5 RSI Tile Removal and Installation

RSI tiles are occasionally removed and replaced as a part of routine TPS maintenance. The reason for the replacement could be in support of a new tile installation on a new vehicle, or more realistically, in support of a TPS reconfiguration modification. Tiles are also removed and replaced due to severe damage or material degradation of the part. A flow diagram of the replacement process is shown in figure A6.

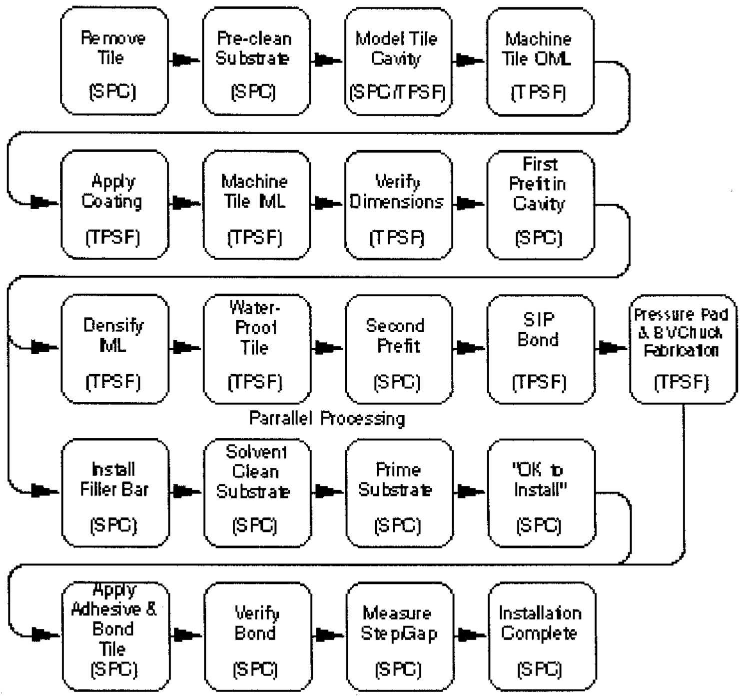


Figure A6. RSI Tile Replacement Flow Diagram.

The RSI tile is removed either destructively or non-destructively per ML0601-9024 process 300. Usually, a knife is used to cut through the SIP to remove the tile non-destructively for future use. If the non-destructive method can not be used, the tile is carefully broken into pieces and removed from the bonded SIP. The remains of the SIP and the residual RTV are skived off with a non-metallic scraper. The cavity is solvent cleaned with 1,1,1-trichloroethane.

The tile installation is performed per ML0601-9024 process 301. The tile cavity is modeled and a flight tile is machined (refer to section A.2.2). The tile is coated with either the white glass or black RCG coating and is identified (refer to section A.2.3). The tile is routed to the SPC for first prefit in the tile cavity. The prefit is used to fit-check the tile and evaluate any step or gap discrepancies that may exist (refer to figure A3). The IML mismatch to the structure is also verified to be within close tolerances prior to continuing the tile processing.

Following the acceptance of the first prefit, the tile and its cavity can be processed in parallel. The tile IML is densified and the tile is waterproofed (refer to section A.2.3). The tile is prefit a second time to verify conformance prior to SIP bond. During the same time as the tile processing, the cavity is prepared. Substrate voids, if any, are filled with a RTV silicone adhesive. The filler bar is installed or reworked as required per ML0601-9024 process 215. The substrate is cleaned per the applicable process (ML0601-9024 process 200-207) and is primed with a silicone primer per process 208.

The SIP is bonded under 1 to 3 psi pressure to the tile IML (refer to section A.2.4). Two customized bonding tools are fabricated, the tile pressure pad and the bond verification (BV) chuck. The pressure pad is a latex foam pad which is calibrated with the required density and thickness for the installation and bonded to a rigid block that matches the OML contour of the tile. The block interfaces between the reaction tooling and tile during bond pressure application. The geometry of the pressure pad directs the pressure uniformly about the tile centroid. The BV chuck is a rigid block that matches the tile OML contour and has a gasket around the OML periphery. The chuck is used to draw a vacuum across the OML surface for tensile testing following adhesive cure.

The tile is prefit a final time and an "OK to install" is obtained when all previous processing has been completed. The tile is bonded with RTV silicone adhesive under 1 to 3 psi pressure as directed through the pressure pad and reaction tooling. Proper pressure is verified by measuring the compressed foam thickness at each corner. Following the cure of the adhesive, the pressure is removed and the bond is tested by a bond verification tensile test per ML0601-9024 process 315. The BV chuck is pressed to the tile OML and a vacuum is drawn through the chuck. The chuck is attached to a threaded shaft or cable assembly to the tensile test unit. The tile is loaded in tension (10 psi of SIP-bonded area for LI-2200 and FRCI-12 and 4 to 6 psi for LI-900) until the specified load is reached. This loading is reduced or eliminated as directed by the engineering drawing for structurally limited areas (e.g., vertical stabilizer, OMS pod). Following bond verification, the step and gap are measured and are verified to be within the ML0601-0001 operational criteria. Following acceptance, the tile installation is complete. Gap filler installation, if required, is performed at this time (refer to section A.6).

A.3 Advanced Flexible Reusable Surface Insulation (AFRSI) Blankets

AFRSI or Flexible Insulation (FI) blankets protect regions of the upper surface of each vehicle where moderate heat loads, pressure gradients, and less air flow are encountered. AFRSI is used where temperatures do not exceed 1,500'F. AFRSI is comprised of quartz fiber batting that is sandwiched between high temperature woven quartz fiber outer fabric and a lower temperature glass inner fabric. The components are stitched together as shown in figure A7 using quartz and glass threads in a one-inch square pattern. The planform size can be up to 30 inches by 30 inches and the thickness varies (with heat load) between 0.41 inch and slightly less than 2 inches. The blanket is bonded directly to the structure using RTV silicone adhesive. Nomex felt ramping, filler bar, and SIP can be used between the AFRSI and structure to allow the installation to fair into adjacent installations. To toughen the outer fabric, the OML surface of the blanket is protected with a ceramic coating. In certain areas, AFRSI requires rewaterproofing to reduce the potential weight increase from absorbed water at launch. This is accomplished by injections of dimethylethoxysilane (DMES) through plastic film on 4-inch centers and covering for 24 hours.

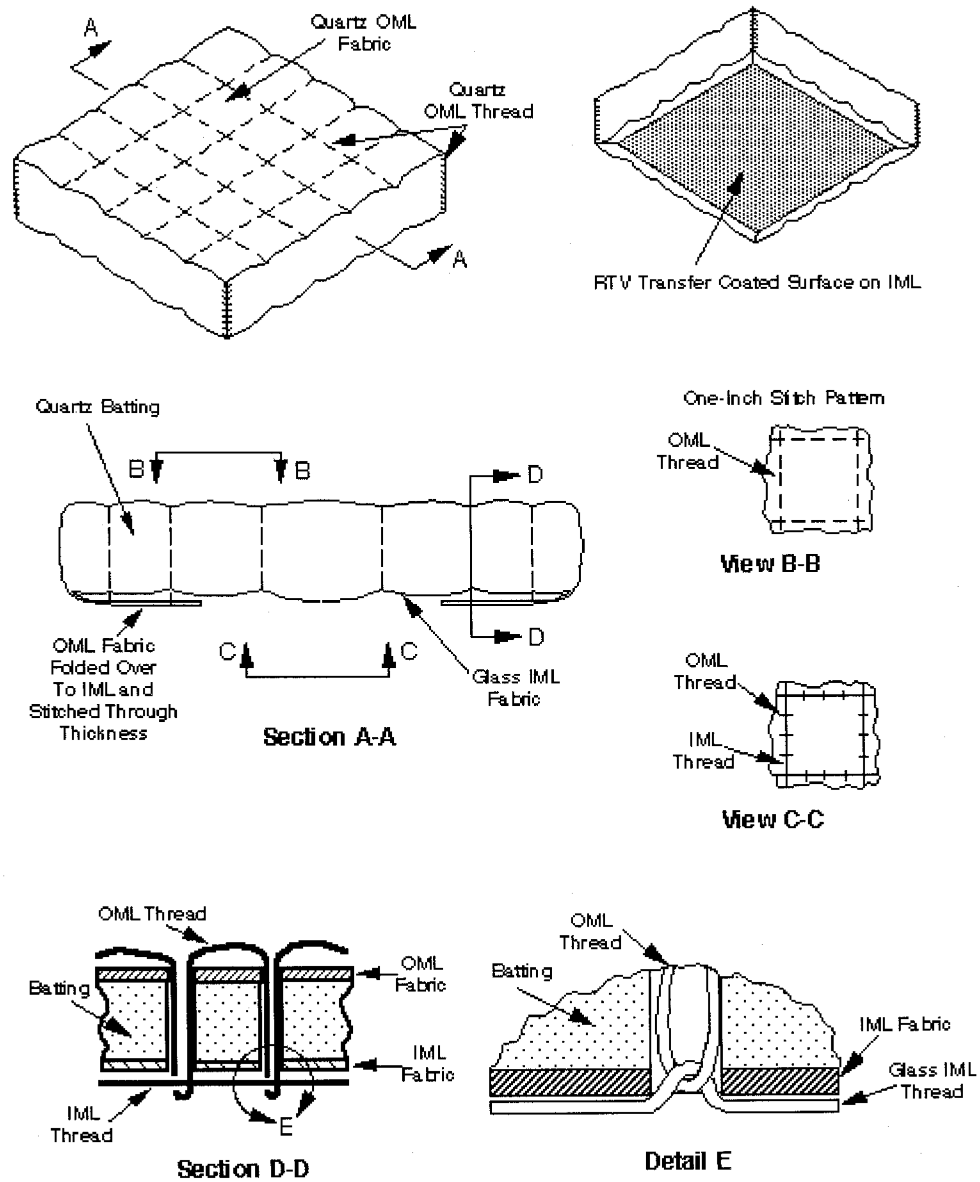


Figure A7. AFRSI Blanket Construction.

The fabrication of the AFRSI blanket primarily involves the assembly of its components. The insulative batting is comprised of 6 pcf quartz fiber. The outer fabric is a 0.027 inch thick quartz fiber woven fabric with an aminosilane binder finish. The inner fabric is a 0.009 inch thick S2-glass yard plain woven fabric with a semi-clean finish. The OML thread is 0.029 inch diameter quartz fiber thread coated with polytetrafluoroethylene (PTFE). The IML thread is 0.020 inch diameter E-glass thread with a liner polyamide coating. The batting is sandwiched between the outer and inner fabrics. The materials are stitched together at 3 to 4 stitches per inch with the two threads interlacing at the IML. The parallel stitchlines are one inch apart in both the length and width directions. The IML fabric and batting are trimmed to a modeled template of the cavity. The OML fabric is folded around the sidewall edges and wrapped around to the IML surface. The corners are looped stitched with the OML thread. The folded OML fabric is stitched to the blanket using a similar two-thread interlacing stitch technique. The blanket is identified by rubber stamping the part number and order control number (OCN) with liquid bright gold ink. The blanket is waterproofed by the vapor deposition of methyltrimethoxysilane (refer to section A.2.3). The part is heat cleaned at 600°F for 2 hours and at 850°F for 4 hours to remove processing aids and oils. The IML of the blanket is transfer coated with 0.010 inch thick RTV silicone adhesive bonded under 3.5 psi of vacuum pressure. A pressure pad consisting of latex foam and Plexiglas is custom made to the particular part. In addition, a 6 inch by 9 inch peel test coupon is fabricated from the identical lots of materials used during the blanket fabrication. The peel test coupon is a process control device that ensures proper adhesion between the transfer coat and structural adhesive. The fabricated blanket, peel test coupon, and pressure pad are delivered for installation.

A.3.1 AFRSI Blanket Installation

The installation of the AFRSI blanket per ML0601-9024 process 501 is depicted in figure A8. The part cavity is precleaned following the removal of the previous part. The cavity is modeled using a template. Following the fabrication of a blanket to the template, the blanket is prefit into the cavity. Ramping or other sub-insulation is installed under pressure using RTV silicone adhesives on a solvent cleaned and primed substrate. The cavity and peel test coupon plate are solvent cleaned, primed, and coated with 0.006 to 0.0 10 inches of RTV silicone adhesive. The transfer coated surfaces on the blanket and peel test coupons are wiped with 1,1,1-trichloroethane and allowed to dry for 2 to 24 hours prior to bonding. The blanket is bonded to the cavity and the peel test coupon is bonded to the plate under 1.5 to 3 psi pressure. Following the cure, the peel test coupon is cut into 1-inch wide strips. A 90° pull test is performed with a force gauge on at least 4 of the strips. The average peel strength of the pulls must be greater than 4 pounds per inch to provide a confidence with the blanket bond.

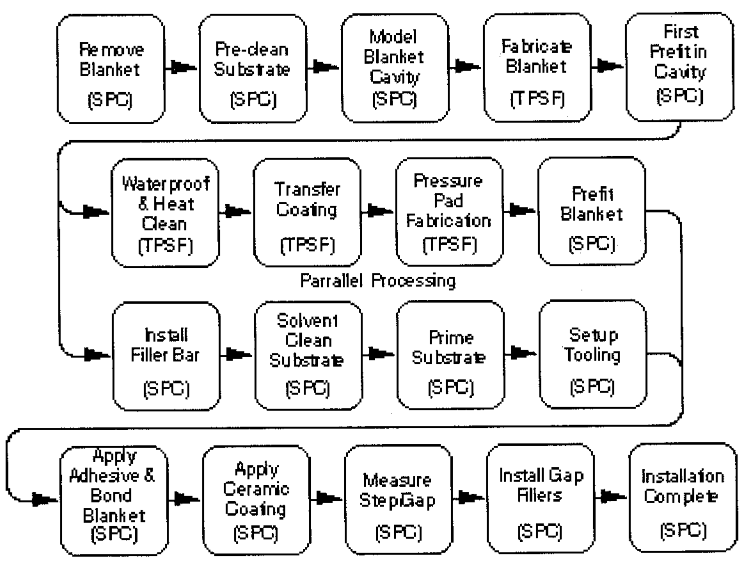


Figure A8. AFRSI Replacement Flow Diagram.

The step and gap of the bonded blanket are measured and any large gaps are filled with an FT blanket type gap filler. There are several types of FT blanket type gap fillers. Primarily they are comprised of 0.040-inch thick high-boria content aluminoborosilicate fiber (Nextel) woven fabric, Nextel braided sleeving, Nextel ceramic fiber cord, alumina fiber (Saffil) insulative batting, and ceramic fiber thread. With these materials, there are essentially three types of gap fillers: folded fabric, stuffed sleeving, and fabric-wrapped cord tadpoles (referencing the cross-sectional appearance). The gap fillers are bonded to the blanket sidewall using RTV silicone adhesives. Following the adhesive cure, the gap fillers are stitched to the adjacent blankets using ceramic thread.

To toughen the outer fabric, a C9 ceramic coating is applied to the outer surface of the blanket in a two-part process. An 80% Ludox ammonia stabilized colloidal silica solution and 20% isopropyl alcohol precoat mixture is applied and air dried for 4 hours. This precoat modifies the fabric to promote the adhesion of the topcoat material. The topcoat consists of a mixture of the Ludox ammonia stabilized colloidal silica solution and silica powder that is applied to the blanket and is air dried for 8 hours. The blanket is reidentified using liquid gold bright ink.

A.4 Felt Reusable Surface Insulation (FRSI)

FRSI panels protect most of the upper surface of each vehicle where temperatures are less than 750°F. FRSI is composed of two materials, a heat treated Nomex felt and a vented white silicone elastomer coating. A typical FRSI component is depicted in figure A9. Additional layers of FRSI or Nomex felt ramping can be used between the FRSI and structure to allow the installation to fair into adjacent installations. FRSI does not require post-flight rewaterproofing because the Nomex polymer is hygrophobic by nature and the silicone elastomer coating inhibits water intrusion into the felt.

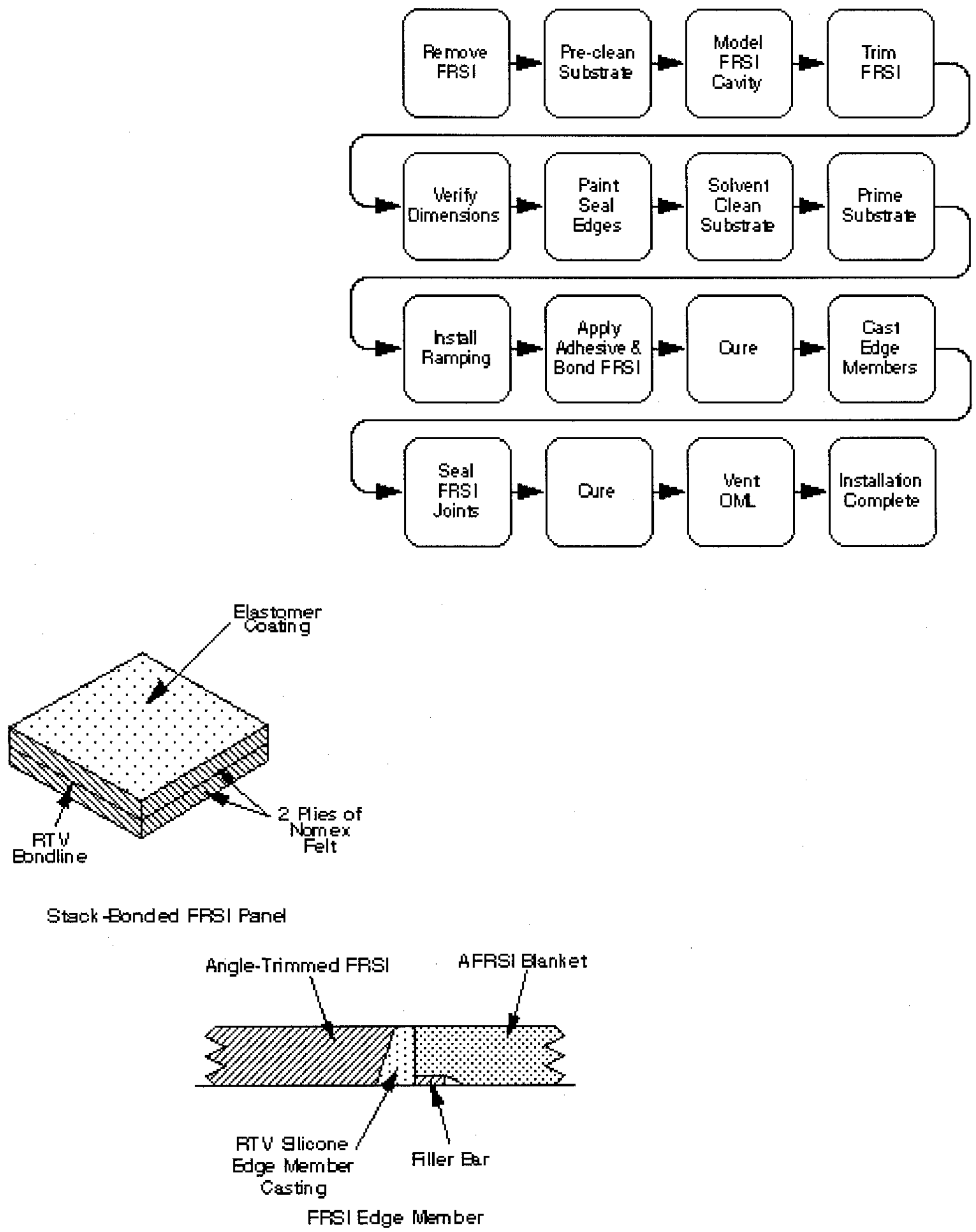


Figure A9. FRSI Detail and Replacement Flow Diagram.

The Nomex felt is made up from 3 inch long, 2 deniers fine, polyamide aramid fibers. The fibers are loaded into a carding machine that combs the tangled fibers into a cross-lapped web. Two webs are placed together and are needlepunched. This sewing-like process passes barbed needles through the webs to compact the fibers into a felt pad of the desired properties. The felt is calendered by passing it through rollers to stabilize the thickness. The felt is heat set at 500°F for 30 minutes to provide dimensional stability. The color of the heat set felt is off-white. This material is used for strain isolator pads (SIP). In all other Nomex felt applications (e.g., FRSI, filler bar, ramping) the felt is heat treated at 700°F for 30 minutes and then at 750°F for 30 minutes to minimize the linear shrinkage at elevated temperatures. The heat treatment darkens the felt to a caramel color.

For FRSI, the heat-treated Nomex felt is transfer coated with a white silicone elastomer. The silicone elastomer is poured and spread to a thickness of 0.006 to 0.008 inch on a screen mesh that was prepared with a parting liquid. The coating is partially cured by air drying for 5 hours. The partially casted coating coating is coated with additional elastomer to provide a wet layer of coating. The Nomex felt is placed in the coating and is bonded under 2 to 3 psi for 2.5 hours. The part is post cured at 650°F for 15 minutes and air dried for 96 hours.

For all other Nomex felt applications (e.g., SIP, filler bar, ramping, sub-surface FRSI), the felt is placed in a 0.006 to 0.010 inch thick layer of red RTV silicone adhesive. The adhesive is bonded to the felt under 2 to 3 psi until cured.

The installation of FRSI per ML0601-9024 process 401 is the least complex of the three RSI material installations as shown in figure A9. The FRSI is trimmed to a cavity template. The exposed edges are paint sealed with a white silicone elastomer. The FRSI is bonded under 2 to 3 psi pressure to a solvent cleaned and primed cavity and/or over sub-insulation. The FRSI-to-FRSI joints are sealed with an RTV silicone adhesive, and other interfaces are filled with an RTV silicone adhesive edge member casting. The coating is vented by 0.035 inch holes made on 6 inch centers. FRSI does not require part identification.

A.5 Reinforced Carbon-Carbon (RCC)

Reinforced carbon-carbon (RCC) is used as a high-temperature aerodynamic structure on the leading edge structural subsystem (LESS) which consists of the nose cap, chin panel, wing leading edge (WLE), and associated expansion seals. In addition, the external tank (ET) forward attach point adjacent structure is protected by an RCC arrowhead component due to the pyrotechnic shock environment of the ET separation mechanism. The RCC material has a maximum use temperature of over 2,960°F and has a density of approximately 103 pcf. The material has a flexural strength of approximately 9,000 psi and a tensile strength of approximately 4,500 psi.

RCC is a structural composite consisting of two discrete carbon-based components, a high-strength substrate and an oxidation protection coating system. The fabrication of RCC is a four-part process as shown in figure Al0.

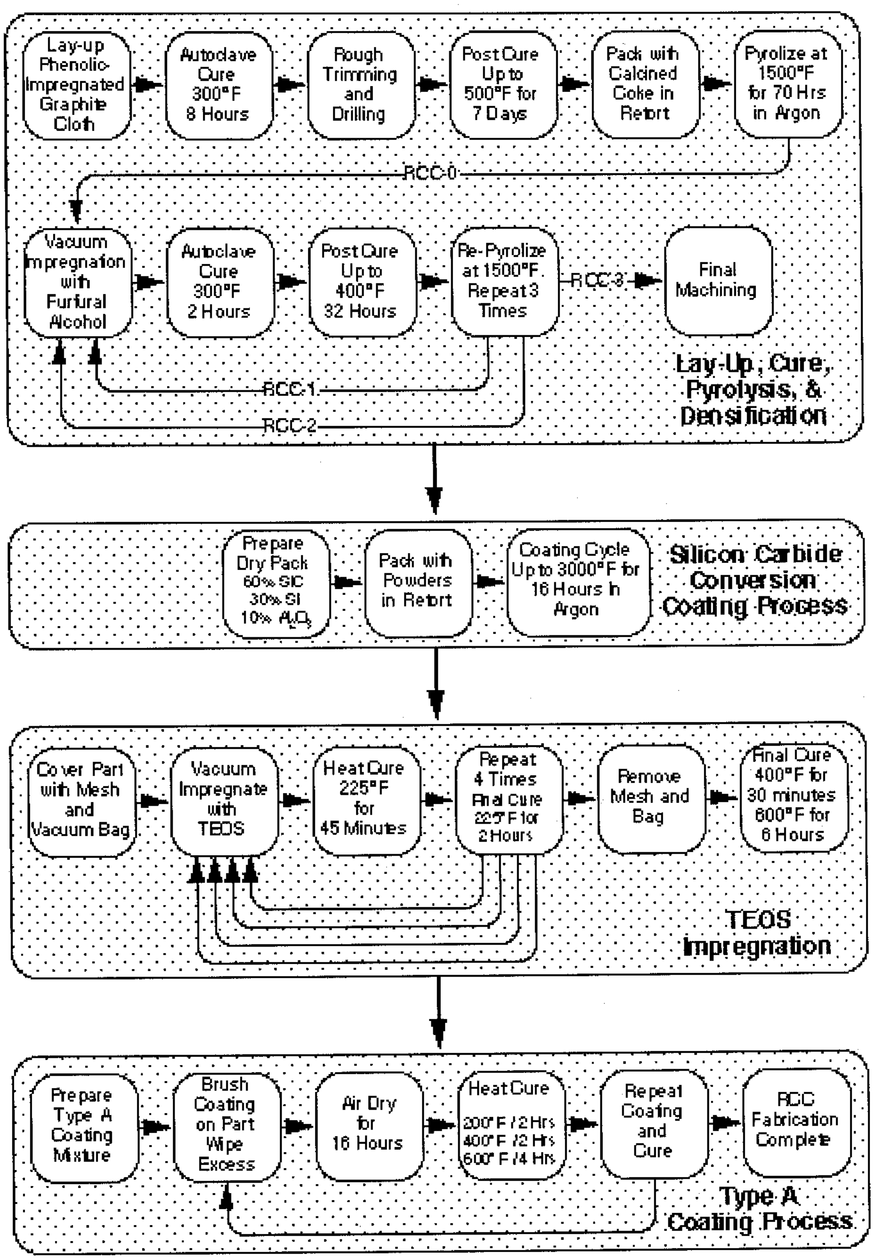


Figure A10. RCC Fabrication Flow Diagram.

The carbon substrate is fabricated from 19 to 38 plies of laid-up phenolic-impregnated graphite fiber cloth autoclave cured at 300°F for 8 hours, rough trimmed, and drilled. The part is post cured by heating up to 500°F for 7 days. The part is loaded in a graphite retort with calcined coke and is made rigid by converting the phenolic resin to carbon by a 70-hour 1,500°F pyrolysis cycle in an argon atmosphere. The part is designated as "RCC-O" and has a flexural strength of approximately 3,000 psi. The part is then densified by vacuum impregnation of furfural alcohol and conversion to carbon by pyrolysis. The subsequent pyrolyses are performed by a 2 hour 300°F autoclave cure followed by a 400°F post cure for 32 hours. The furfural alcohol vacuum impregnation and pyrolysis cycles are repeated three times. After the final pyrolysis, the part is designated "RCC-3" and has a significantly stiffer flexural strength of approximately 18,000 psi. The final machining of the part is performed. The pure carbon substrate is subject to oxidation at temperatures over 700°F, well below the service temperature of the component. Therefore, an oxidation protection coating is required. The term "coating's is actually a misnomer as the outer surfaces (0.020 to 0.040 inch) of the carbon component are converted to silicon carbide by a diffusion reaction. The conversion process is accomplished by packing the component into a mix of constituent powders (60% silicon carbide, 30% silicon, and 10% alumina) in a graphite retort and is subjected to a 16 hour heating cycle which includes a 600°F drying cycle and a diffusion coating cycle with temperatures up to 3,000°F in an argon atmosphere. The carbon substrate and silicon carbide materials have a thermal expansion mismatch which results in the formation of very small craze cracks in the silicon carbide layer as the silicon carbide contracts more than the carbon substrate during the cool down period. To provide further protection, the RCC part is vacuum impregnated with tetraethyl orthosilicate (TEOS) and oven cured at 225°F for 45 minutes. The TEOS impregnation and heat curing is repeated four times with the fifth oven cure at 225°F for 2 hours. The part is heat cured at 400°F for 30 minutes and 600°F for 6 hours. The heat cures result in the formation of a protective layer of silicon dioxide residue.

The final fabrication step is to apply Type A sealant to fill any porosity or craze cracks on the RCC part. The Type A sealant is a mixture of silicon carbide powder and a sodium silicate water glass. The mixture is prepared and is brushed on the part. The part is then air dried for 16 hours and heat cured at 200°F for 2 hours, 400°F for 2 hours, and 600°F for 4 hours. The application and subsequent curing is repeated. Once the fabrication is complete, the part is ready for installation.

The LESS is made up of two distinct entities, the nose area and the wing leading edge, as shown in figure All and A12, respectively. The nose area is protected by the RCC nose cap, the chin panel, and nine associated expansion and tee seals. The wing leading edge is protected by 44 RCC panels, 42 RCC tee seals, and 2 angle expansion seals. The parts are mechanically attached to the aluminum forward bulkhead or wing spar using inconel 718 and A-286 fittings on floating joints. The floating assembly is used to prevent excessive loading and to seal the RCC cavity from hot plasma flow. The attachment of the nose cap and chin panel seals allows for circumferential, fore, and aft movement about the nose cap periphery. The angle (located forward of panel 1) and tee seals on the wing leading edge allow for lateral motion and thermal expansion differences between the RCC and wing. To further prevent the flow of hot gas from entering the RCC cavities, alumina-stuffed aluminoborosilicate (Nextel) gap fillers are used on the lower surface between the RCC and HRSI tile interfaces. The open interface gap on the upper surface between the RCC and HRSI tiles allows for venting of the RCC cavity in the thermally benign regions of the LESS.

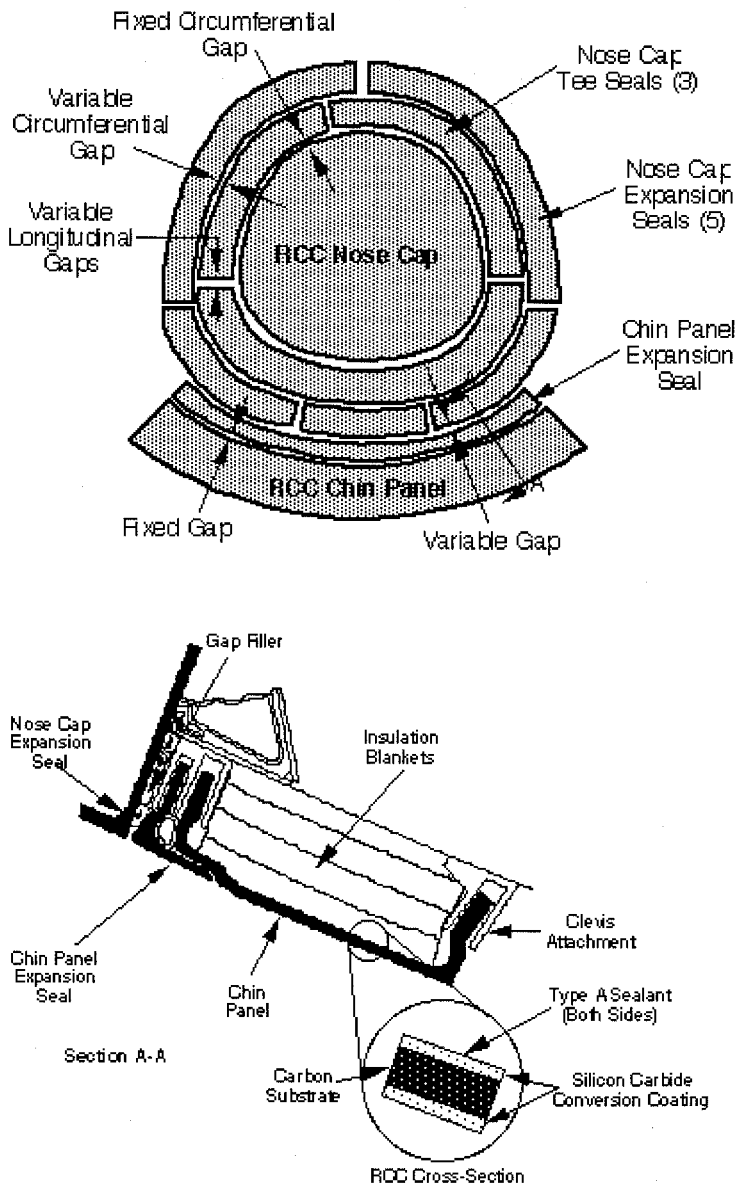


Figure A11. Less Nose-Area RCC Components.

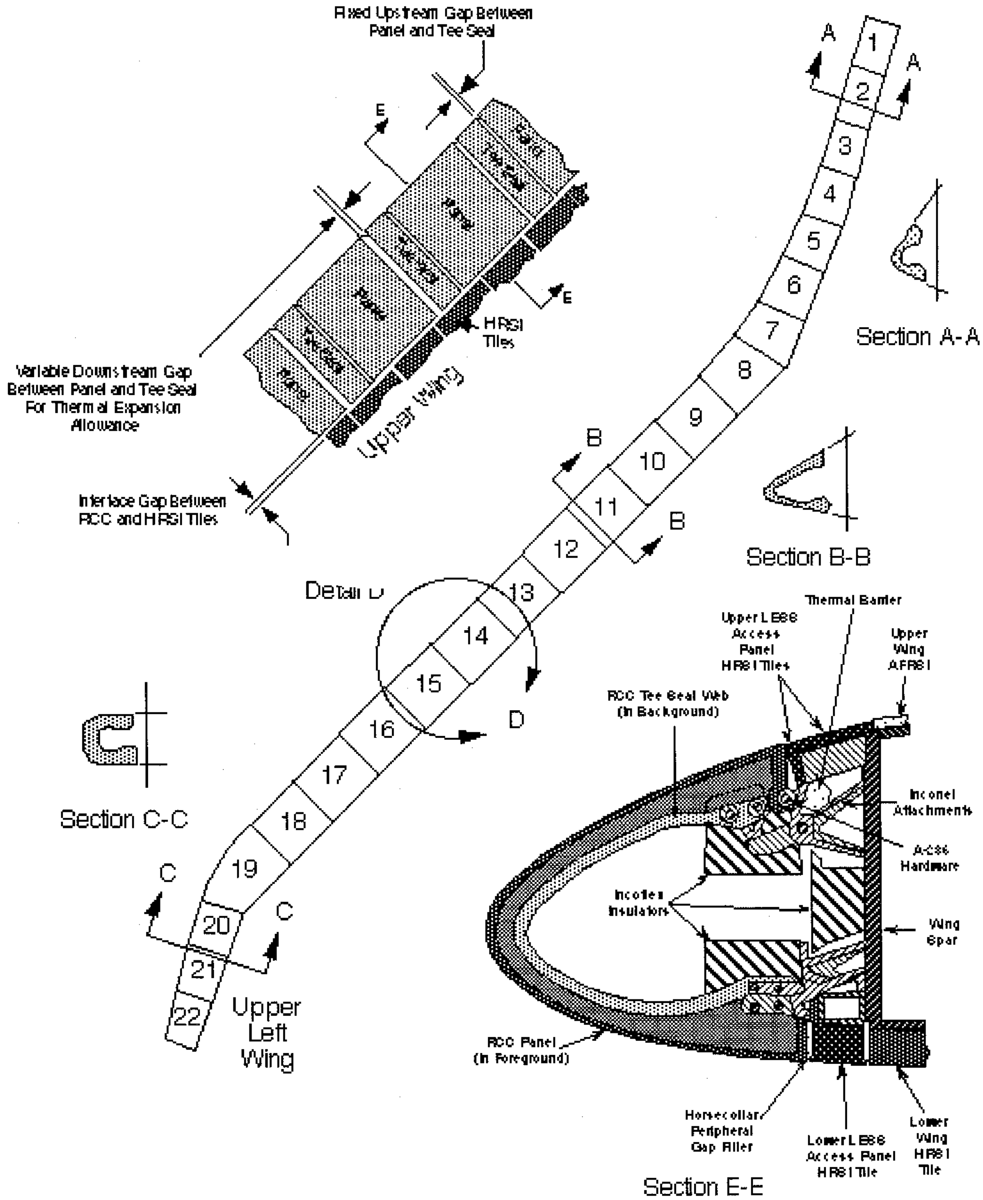


Figure A12. LESS Wing Leading Edge RCC Components.

The RCC material promotes the internal cross radiation from the hot stagnation region at the apex to cooler areas. This cross radiation reduces the temperatures at the apex and increases the temperatures of the cooler regions which, in turn, reduces the thermal gradients around the component. This cross radiation also directs heat back to the structure. Therefore, the structure must be protected by the utilization of backing insulation. The nose cap and chin panel use an uncoated flexible insulation blanket fabricated from aluminoborosilicate fiber fabric (Nextel) and alumina insulation (Saffil) or alumina silica chromia (Cerachrome) to protect the structure. In addition, high-temperature reusable surface insulation (HRSI) tiles are bonded to the forward bulkhead to offer additional thermal protection behind the nose cap. An uncoated AFRSI blanket is used as the insulation beneath the arrowhead. The radiation from the wing leading edge RCC to the wing spar is protected by 0.030 inch thick inconel foil covered Cerachrome batting known as Incoflex insulators. Although the intent of the backing insulation is to protect the structure, it also retards the internal RCC cross radiation and subsequently retards the cooling rate of the RCC lugs adjacent to the backing insulation. This prolonged heating contributes to the undesirable oxidation rate of the RCC which, in turn, reduces the mission life of the component.

A.6 Gap Fillers

Gap fillers are used in areas to restrict the flow of hot gas into the gaps of TPS components. The types and applications of the various types of gap fillers are shown in figure Al3. The predominant gap filler types that are used are the pillow or pad type and the Ames type.

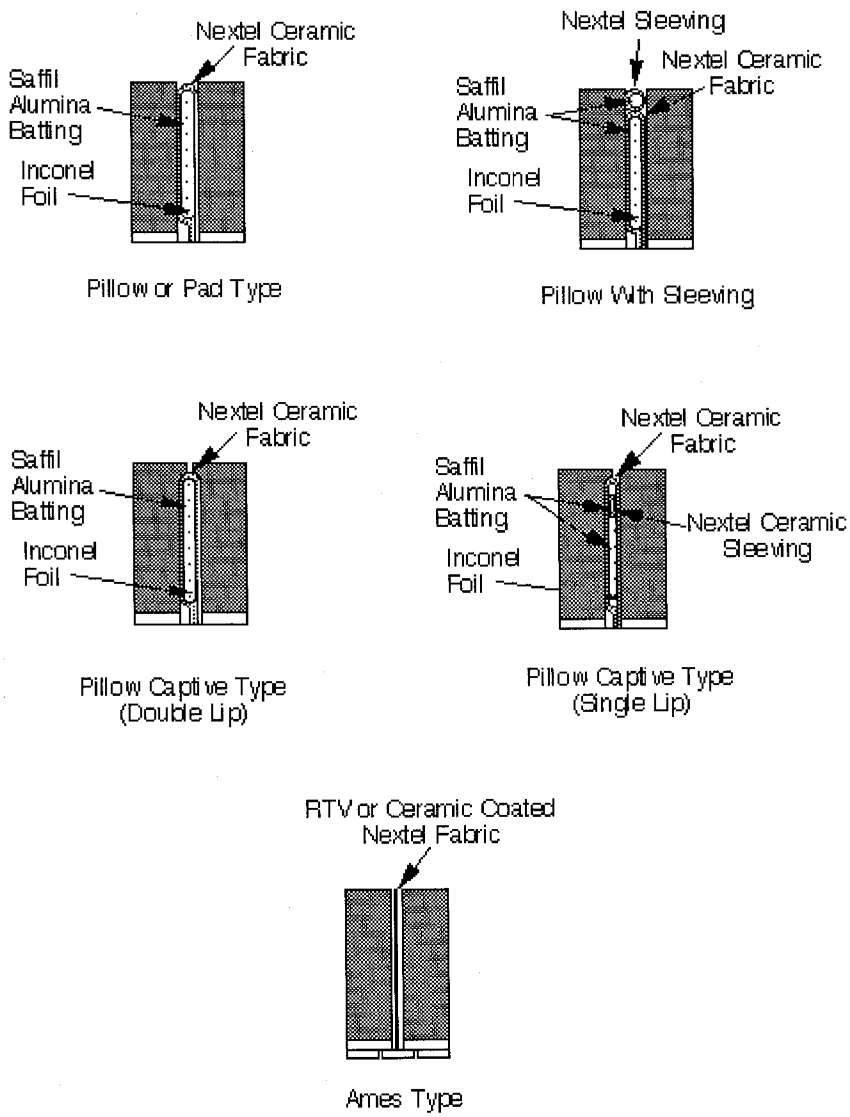


Figure A13. Tile-to-Tile Gap Fillers.

The pillow fabric gap fillers are usually installed to completely fill their intended gaps. The basic pillow gap filler is fabricated from a template (depicting the contour, height, and width required) of the gap with specific thickness requirements recorded on the mylar. The gap filler fabrication begins with trimming a 0.001-inch thick sheet of Inconel 601 alloy to the shape of the gap to be filled. The aluminoborosilicate fiber (Nextel) fabric is folded over the inconel, and the fabric is stuffed with an alumina fiber (Saffil) batting to obtain the desired thickness. The gap filler is stitched with Nextel thread. The tail of the gap filler is stiffened with RTV silicone adhesive. The other types of stitched gap fillers are derivations of the basic pillow type. The derivations include the use of Nextel ceramic fiber braided sleeving. The sleeving can be added to the exterior or interior of the folded area of the gap filler fabric.

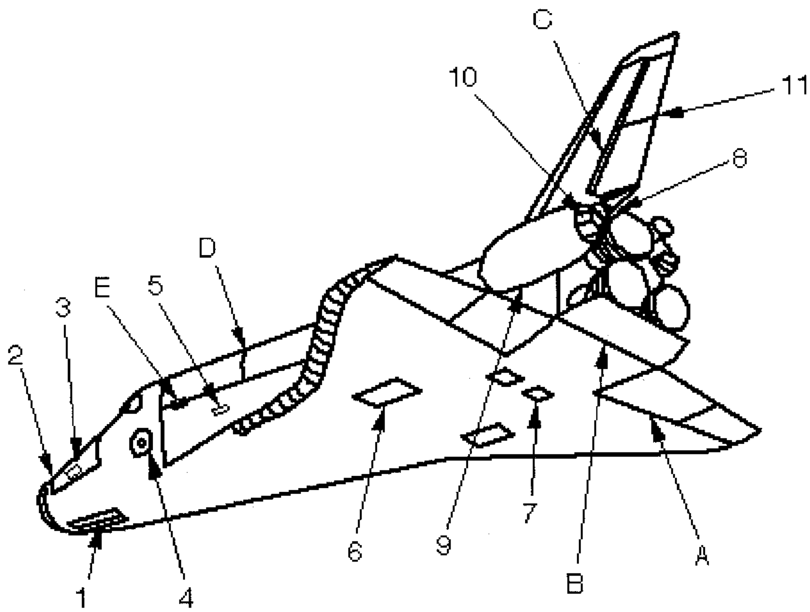
The majority of gap fillers are installed following the installation of RSI tiles. The gap filler is bonded to the underlying filler bar or tile sidewall with RTV silicone adhesive. Following the cure of the adhesive, the gap filler is friction tested to ensure the proper compression within the gap and to validate the integrity of the gap filler bond. Pillow and pad-type gap fillers are coated with a high emissivity ceramic coating in a two-part application procedure similar to that of AFRSI blankets. A 85% Ludox ammonia stabilized colloidal silica solution, 12% isopropyl alcohol, and 3% silicon carbide powder precoat mixture is applied and air dried for 4 hours. This precoat modifies the fabric to promote the adhesion of the topcoat material. The topcoat consists of a mixture of the Ludox ammonia stabilized colloidal silica solution, silica powder, silicon carbide powder that is applied to the exposed area of the gap filler and is air dried for 8 hours.

There are three varieties of Ames gap fillers comprised of two fabric types and two coating types. The fabric is available in a non-vacuum baked and vacuum baked condition. The non-vacuum baked fabric can be coated with black RTV for upper surface use and ceramic coating for lower surface use. The vacuum baked variety can only be fabricated with the black RTV coating for upper surface use.

The Ames gap filler is nominally 0.020 inch thick and is cut to fit a gap mylar. Up to 6 layers of Ames gap fillers are installed to fill a gap partially or completely. A mylar is made of the gap which duplicates the length, width, and contour of the gap with gap measurements recorded in the corresponding locations on the mylar. The gap filler is prefit and pull test loops are installed. The gap filler is installed by RTV bonding onto a primed surface, and the bond is verified by pulling on test loops after the adhesive cure.

A.7 Thermal Barriers

Thermal barriers are used around penetrations and in the closeout areas between the major components of the orbiter. The primary purpose is to restrict hot gas flow to the underlying cavity or structure. The locations of the orbiter thermal barriers (and aerothermal seals, section A.8) are shown in figure A14.

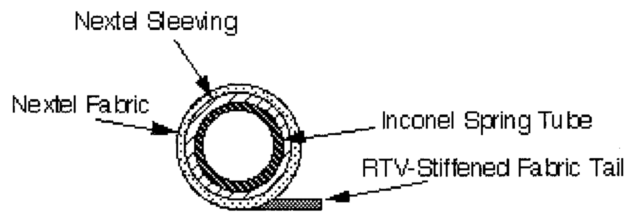


Thermal Barriers	
1	Nose Landing Gear Door
2	FRCS Module/Fuselage Interface
3	Forward RCS Thrusters
4	Crew Hatch
5	Vent Doors
6	Main Landing Gear Doors
7	External Tank Doors
8	Vertical Stabilizer/Fuselage Interface
9	OMS Pod/Fuselage Interface
10	OMS Pod RCS Thrusters
11	Rudder Speed Brake Split Line

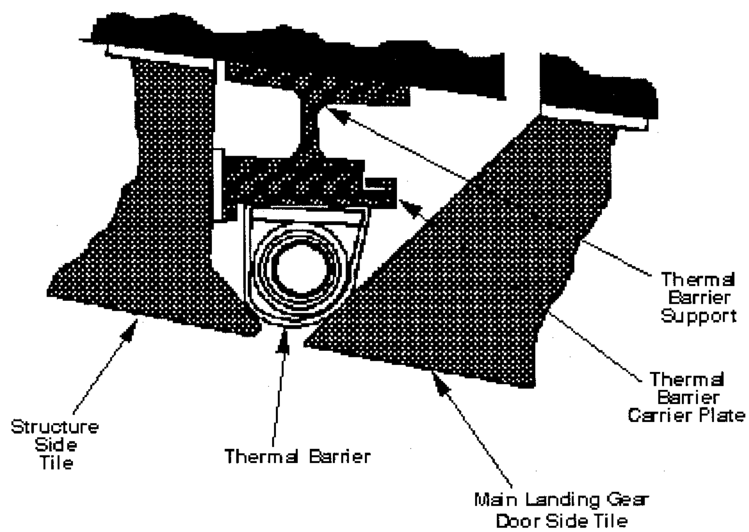
Aerothermal Seals	
A	Wing/Elevon
B	Aft Fuselage/Body Flap
C	Vertical Stabilizer/Rudder Speed Brake
D	Payload Bay Door Expansion Joints
E	Payload Bay Door Hinge Covers

Figure A14. Thermal Barrier and Aerothermal Seal Locations.

The majority of thermal barriers are constructed from spring tube, insulative batting, sleeving, and ceramic fabric. The spring tube is a tubular inconel wire mesh. The part is inserted into aluminoborosilicate fiber (Nextel) braided sleeving. The thermal barrier is then covered with a Nextel ceramic fiber fabric outer cover. The thermal barrier is bonded by its ceramic fabric tail to its intended cavity (for adhesive bonded types), attached to the structure by the use of hardware (for mechanically attached types), or attached to a carrier plate (for mechanically attached carrier panel types). Figure Al5 depicts the mechanically attached carrier panel type gap filler installed around the periphery of the main landing gear doors.



Typical Thermal Barrier Detail



Main Landing Gear Door Thermal Barrier

Figure A15. Thermal Barrier Detail.

Thermal barriers are installed per specific processes for the particular design. They are usually bonded under pressure to a solvent cleaned and primed structural substrate with RTV silicone adhesive. The outer thermal barriers in the thermally extreme nose landing gear door area are bonded to the peripheral HRSI tile sidewalls and RCC surfaces with a ceramic adhesive. The ceramic adhesive is a two component mixture. The first component is a 75% deionized water and 25% Ludox ammonia stabilized colloidal silica solution. The second component is a ceramic adhesive powder. The thermal barriers on the main landing gear and external tank doors are bonded to a solvent cleaned and primed carrier panel using RTV silicone adhesive. The carrier panel is clipped into a retaining fixture affixed to the orbiter structure. The thermal barriers around the nozzles of the reaction control system (RCS) thrusters are attached to the structure using fasteners.

Following installation the thermal barrier outer fabric is coated. The coating is made of a polyethylene or a black RTV silicone adhesive. The coatings provide improved thermal performance and durability.

A.8 Aerothermal Seals

Aerothermal seals are used to restrict hot gas flow into the control surface cavities and payload bay door areas. A14 depicts the locations of the aerothermal seals.

The wing trailing edge/elevon leading edge (i.e., the elevon cove) and the aft fuselage trailing edge/body flap leading edge (i.e., the body flap cove) are thermal seals. Figure A16 depicts the aerothermal seal in the elevon cove region. The primary seal in this region is the spanwise polyimide seal which contacts the the elevon rub tube. This seal requires a precise fit against the rub tube to limit the flow into the cavity during control surface movement. Within the cavity, there are heat sinks and additional insulative material to increase the thermal mass and reduce structural thermal gradients. At the inboard and outboard ends of the control surfaces, there are spring loaded columbium seals to prevent hot flow from entering the cavity and potentially overheating the underlying structure and mechanisms. This spring loaded seal allows for the inboard and outboard floating of the elevon due to thermal expansion mismatches between the wing and elevon. The upper surface of the elevon cove is sealed with inconel flipper doors. These flipper doors are hinged on the wing trailing edge and move in concert with the elevon to ensure a proper seal with the rub panels on the upper elevon. The exposed metallic surface is coated with Pyromark coating to optimize the thermal emissivity of the part.

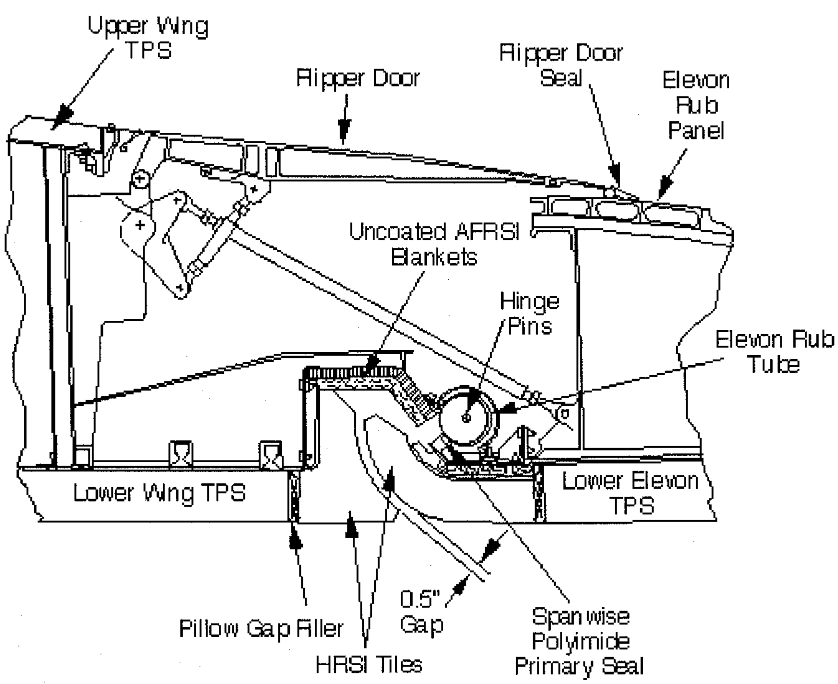
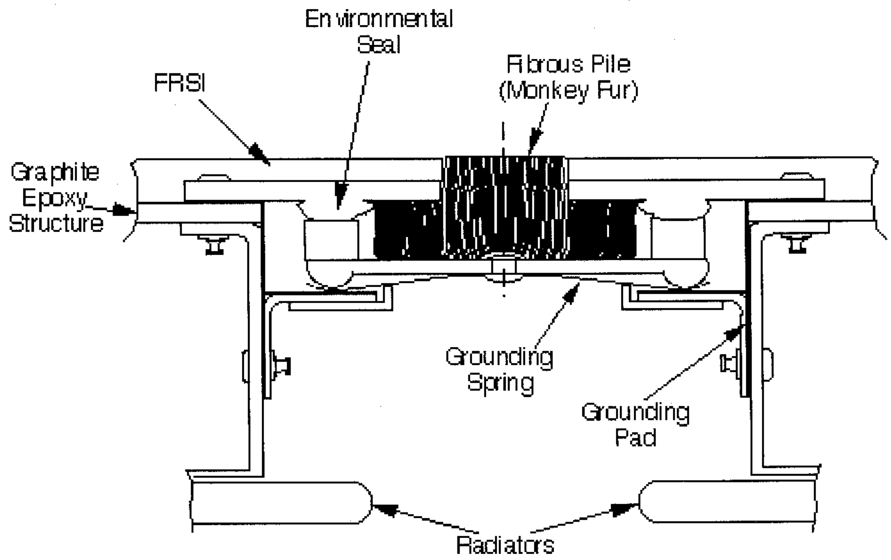
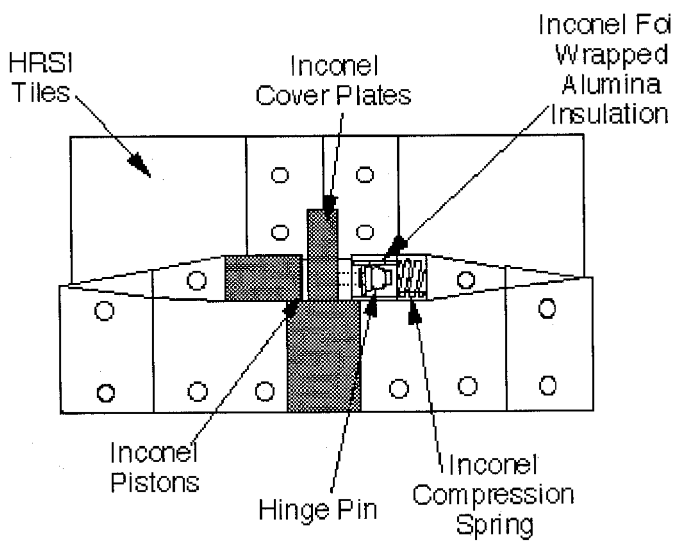


Figure A16. Elevon Cove Aerothermal Seal.

The payload bay door area is protected by two types of aerothermal seals as shown in figure A17. The expansion joints are sealed by environmental bulb seals. These FEP Teflon seals are protected during reentry by a quartz fibrous pile thermal barrier. The sealing surfaces are coated with a fluorinated grease to prohibit water intrusion into the payload bay. The payload bay door hinge area is protected by a spring loaded inconel 718 cover assembly. This assembly is used on the first six hinges on OV-102 (Columbia) and the first ten hinges on OV-103 (Discovery) and subsequent orbiters (Atlantis and Endeavour). The design allows for floating as the spring loaded piston is driven inward towards the center clevis cover. This floating design allows for fore and aft movement of the graphite epoxy composite payload bay doors for the thermal expansion mismatch with the aluminum alloy midfuselage. The exposed surfaces of the hinge cover are coated with the high emissivity Pyromark coating.



Payload Bay Door Expansion Joint Seal



Payload Bay Door Hinge Cover Seal

Figure A17. Payload Bay Door Aerothermal Seals.

A.9 Windows

There are eleven windows on the orbiter to provide visibility for mission operations. There are six forward windows, two overhead windows, two aft flight deck windows, and one crew hatch window. The window locations and their designations are shown in figure A18. The forward, overhead, and crew hatch windows consist of three panes of glass held in a pressure sealed retainer. The outermost pane is attached to the forward fuselage structure and the inner two panes are attached to the crew module. The aft flight deck windows have only two panes of glass attached to the crew module. The outermost pane is the only window component of the thermal protection system. The window installation configuration is shown in figure A18.

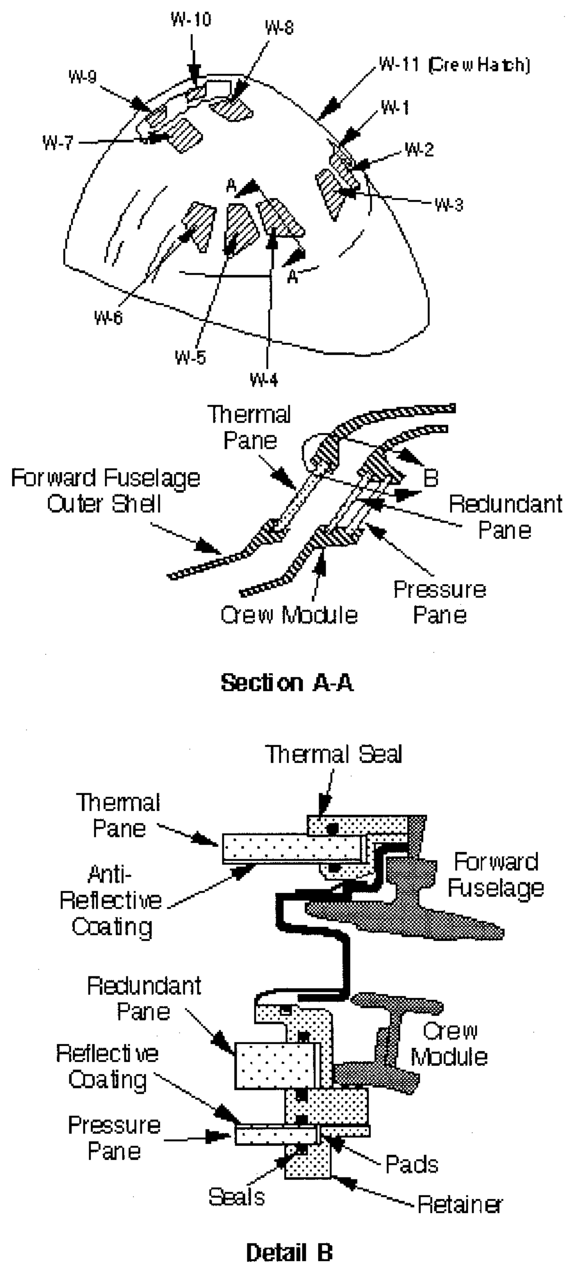


Figure A18. Orbiter Window Locations and Installation Detail.

The innermost pane is the pressure pane. It is fabricated from an aluminosilicate glass which is tempered to provide the strength required to withstand the crew compartment on-orbit pressure differential. The pressure pane, along with the thermal pane, is designed to withstand a pressure of 8,600 psi at 240°F. The outer surface of this pane is coated with an infrared reflective coating. This pane is 0.625 inch thick on the forward windows, 0.450 inch thick on the overhead windows, 0.300 inch thick on the aft flight deck windows, and 0.250 inch thick on the crew hatch window.

The center pane is the redundant pane. It is fabricated from a low-expansion fused silica glass. This uncoated pane is 1.300 inch thick on the forward windows, 0.450 inch thick on the overhead windows, 0.300 inch thick on the aft flight deck windows, and 0.500 inch thick on the crew hatch window.

The outermost pane is the thermal pane. It is fabricated from the same fused silica glass as the redundant pane. This pane is designed to withstand the same pressure as the pressure pane. The interior of this pane is coated with a high-efficiency anti-reflective coating to improve light transmission. This pane is 0.625 inch thick on the forward windows, 0.680 inch thick on the overhead windows, and 0.300 inch thick on the crew hatch window.

Space Shuttle Orbiter Thermal Protection System Processing Assessment

Appendix B

Overview of the Space Shuttle TPS Operations

May 1995

Prepared by:

Michael P. Gordon
TPS Orbiter Engineering
Materials & Processes
Rockwell Florida Operations D/830

Summary

The primary purpose of this appendix is to provide an overview of the materials associated with the thermal protection system (TPS) of the Space Shuttle orbiter. Although this appendix is a self-contained informative document, it is intended to accompany the "Space Shuttle Orbiter Thermal Protection System Processing Assessment Final Report". The purpose of the final report is to present the conclusions of a study to assess the processing of the thermal protection system of the Space Shuttle orbiter. This study was initiated on November 10, 1994 in support of the NCC2-9003 Cooperative Agreement Notice (CAN), Lightweight Durable Thermal Protection System, TA-3 Task 6 for the Single Stage To Orbit (SSTO) Lightweight Durable TPS project being performed by Rockwell Downey under the Marshall Space Flight Center (MSFC) NASA Research Announcement (NRA) 8-12 program. This appendix does not contain proprietary information.

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B.1 Space Shuttle Processing Overview

The processing of the Space Shuttle Orbiter begins at landing and ends at launch. The orbiter processing can be divided into four principle areas as shown in figure B1. The first area is the orbiter landing operations which lasts for 1 day to 1 week or more depending on landing location. Following this operation the vehicle is towed to the orbiter processing facility (OPF). The approximately 10-week long OPF processing flow is the longest duration of all processing activity. Following the OPF operations, the vehicle is rolled over to the vehicle assembly building (VAB) on the orbiter transport vehicle (OTV). In the VAB, the orbiter is mated to the external tank (ET) and solid rocket boosters (SRBs) on the mobile launch platform (MLP). The orbiter processing flow in the VAB is approximately 1 week in duration. The MLP, with the stacked Space Shuttle vehicle, is rolled out to the launch pad by a crawler-transporter vehicle. The MLP is affixed to the launch pad service structure. The final preparations for launch are performed for approximately three weeks. The vehicle is launched and the mission duration lasts 1 week or more.

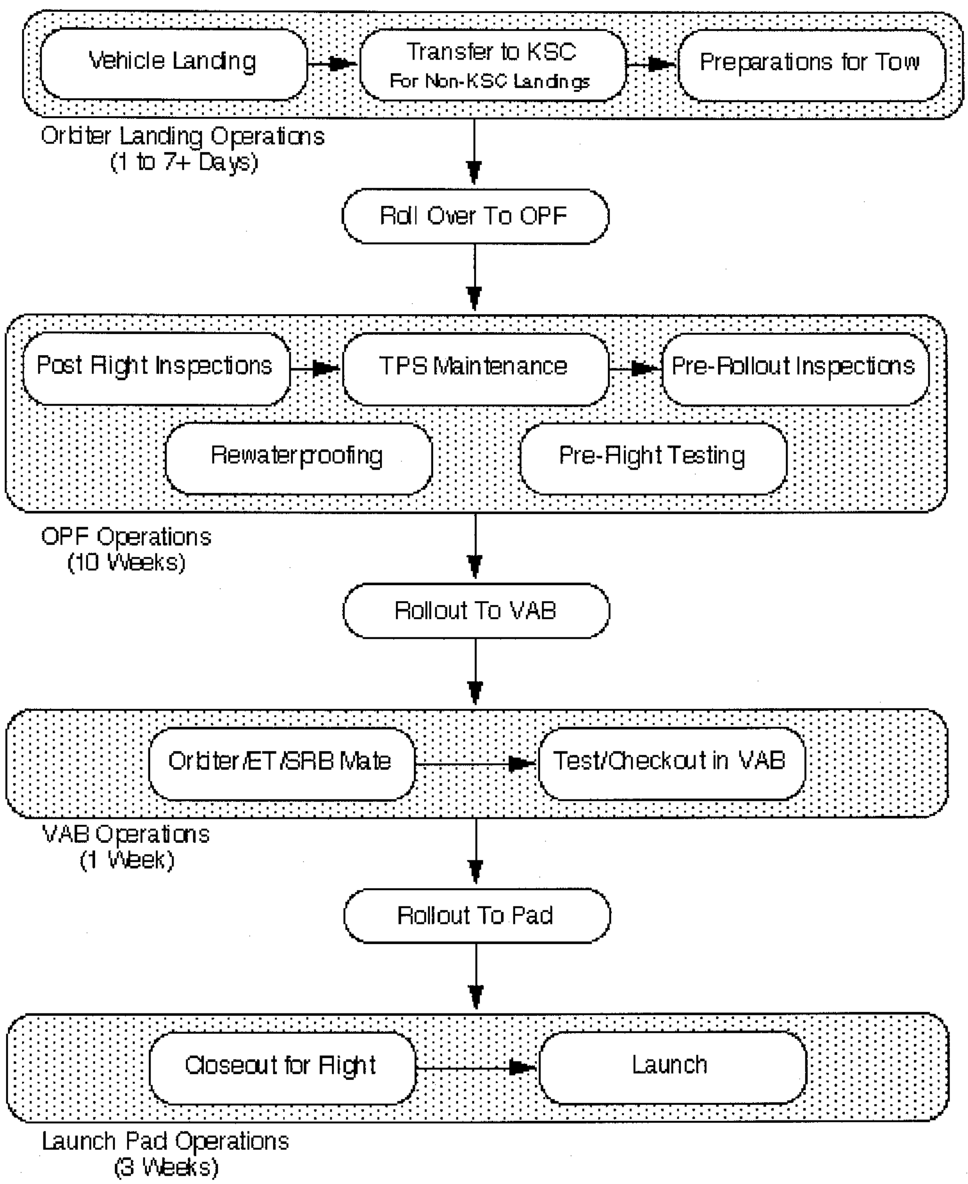

Figure B1 Space Shuttle Processing Overview

B.1.1 Orbiter Landing Operations

The processing operations begin immediately following the wheel stop of the vehicle on the runway. The vehicle is serviced by a landing convoy team. This convoy team is primarily associated with safeing the various subsystems and preparing the vehicle for towing to the deservicing area. For landings at the shuttle landing facility (SLF) at the Kennedy Space Center (KSC), the vehicle is towed to the OPF. For non-KSC contingency landings at Edwards Air Force Base (EAFB) in California, the vehicle is towed to the mate-demate device (MDD) at the NASA Dryden Flight Research Center (DFRC).

Contingency landing operations performed at the MDD are primarily involved with the preparation of the orbiter for ferry flight to KSC on board the shuttle carrier aircraft (SCA). Ferry flight ground support equipment (GSE) is installed. Hardware is installed on the ET attach points to enable mating to the SCA. Other GSE is installed to provide for a smooth orbiter contour to provide a more aerodynamic shape for atmospheric flight. An aluminum tail cone is installed enclosing the main engines and ferry flight doors are installed in the ET door cavities. Significant TPS discrepancies are repaired to prevent damage propagation which could result in costly removal and replacements. Following the mate to the SCA, the orbiter is ferried with an additional "path finder" aircraft. The pathfinder aircraft is flown ahead of the SCA and is utilized to locate a flight path that is free of potentially damaging weather systems and precipitation. The pathfinder aircraft also transports a small contingency maintenance crew should the orbiter need maintenance during the ferry flight operation. Upon landing at the SLF at KSC, the orbiter is removed from the SCA by means of another MDD. The orbiter is then towed to the OPF for further processing.

B.1.2 Orbiter Processing Facility Operations

The orbiter processing facility (OPF) is the primary location for the majority of vehicle servicing. There are three OPF high bays for vehicle processing. A typical OPF processing flow is shown in figure B2. Following transport from the SLF, the vehicle is jacked and leveled. The orbiter is powered up for system integration with ground support systems. Payload bay door strong back GSE is installed and the doors are opened by means of the OPF zero-g payload bay door opening mechanism. System inspections, problem reporting, and maintenance are performed (refer to section B.2 for TPS processing). The payload is removed and the payload bay is reconfigured for the next payload. Vehicle functional checks are performed on the orbiter subsystems. The three space shuttle main engines (SSMEs) are removed for servicing in the VAB engine shop. Maintenance is performed on all major subsystems. The refurbished drag chute and its pyrotechnic mechanism are installed. Non-hazardous payloads that can accommodate horizontal ground processing are installed into the payload bay. The refurbished SSMEs are installed and integration checks are performed. Aerosurface and landing gear door hydraulic functional checks are performed. The ET door functional checks are performed. The payload bay doors are closed and the strong backs are removed. Final OPF inspections and component installations for flight are performed. The ET attachment hardware and associated mechanisms are installed. The orbiter is jacked down and weight and center of gravity locations are measured. The orbiter is then mated to the orbiter transport vehicle (OTV). The orbiter is powered down and rolled over to the VAB.

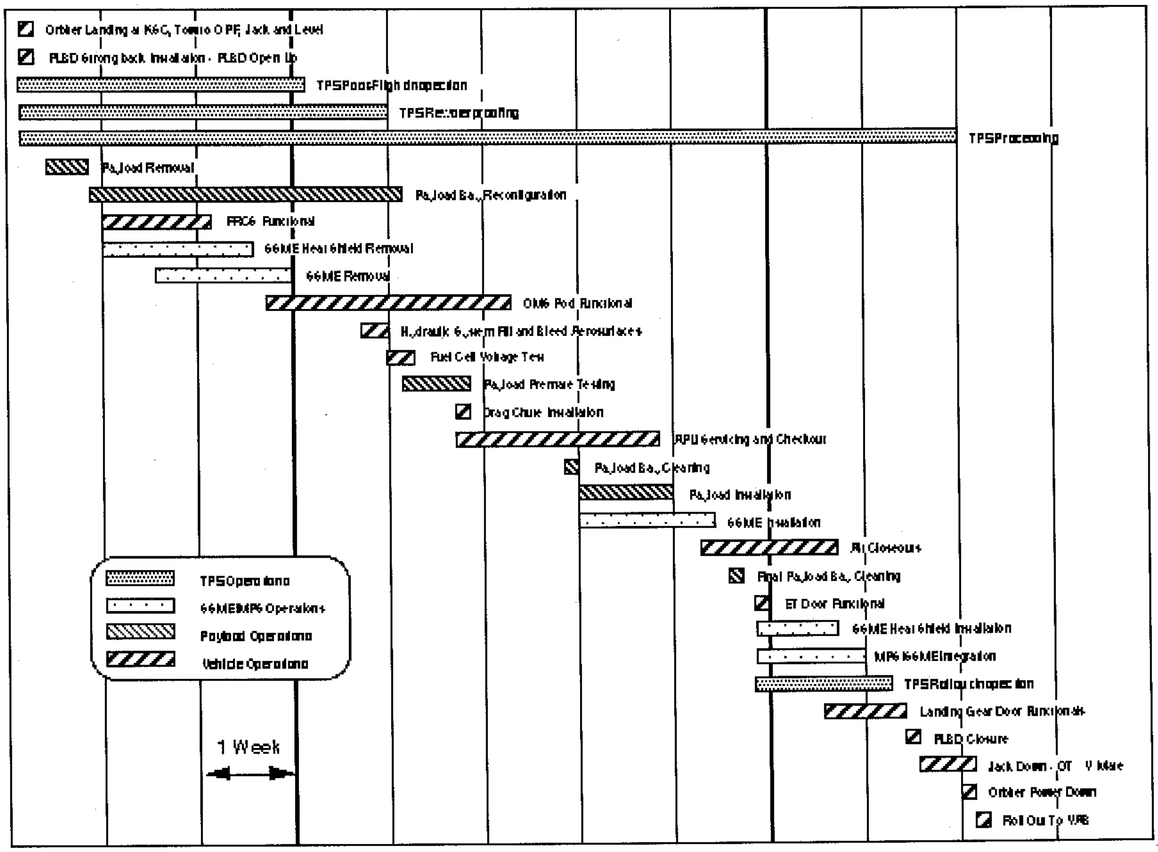

Figure B2. Orbiter Processing Facility Operations

B.1.3 Vehicle Assembly Building Operations

The vehicle assembly building (VAB) is the location for orbiter mating to the other Space Shuttle vehicle components. During non-orbiter mating operations, VAB activity is centered around the processing of the SRBs and ET. The SRB segments are stacked on an MLP which is located in one of two high bays. The external tank is mated to the SRBs on the MLP. The completed assembly is prepared for orbiter mating.

Following orbiter delivery from the OPF, a dual-crane operated lifting sling is attached to the orbiter structure and the vehicle is hoisted from the OTV. The orbiter is rotated from a horizontal position to a vertical position. The suspended vehicle is moved to one of the two VAB high bays for mating to an SRB/ET set on the MLP. Following the mating of the Space Shuttle components, nose landing gear door flow path inspections and rework are performed as necessary. Final checkouts are performed, and a crawler-transporter vehicle is positioned beneath the MLP. The crawler-transporter vehicle transports the Space Shuttle vehicle over the gravel crawlerway to one of two launch pads, LC-39A or LC-39B.

B.1.4 Launch Pad Operations

The launch pad is the location for the final preparations for flight. The launch pad is comprised of three major components: the fixed service structure (FSS), the rotating service structure (RSS), and the mobile launch platform (MLP). The FSS is the primary launch pad structure and the RSS is attached to the FSS by means of a large hinge. The RSS rolls around the FSS to enclose the Space Shuttle vehicle upon arrival. The RSS provides access to the critical vehicle locations for flight closeout activity and protects the vehicle from potentially damaging rain and hail. The RSS also has a payload changeout room (PCR) which is used for final payload activity. The PCR may also be utilized for vertically-oriented or hazardous payload installations. The exposed cryogenic propellant lines between the ET and orbiter are sprayed with a insulative foam to prevent ice formation prior to and during launch. The monomethylhydrazine (MMH) and nitrogen tetroxide (N2O4) hypergolic propellants are loaded into the FRCS, OMS pods, and auxiliary power units (APUs). The protective covers are removed and final component installations are performed. The RSS is rolled back and the ET is filled with liquid hydrogen (LH2) fuel and liquid oxygen (LO2) oxidizer. The flight crew is assisted during ingress and the crew hatch is closed. The orbiter processing is complete when the vehicle is launched.

B.2 Orbiter TPS Processing

The majority of TPS processing is performed in the OPF. TPS processing can be divided into four major areas: inspection, documentation, maintenance, and rewaterproofing.

B.2.1 Inspection

An engineering inspection is performed immediately after landing to provide a "quick look" report to the TPS community. This quick look report is to assess the general condition of the TPS, to determine the processing needs, to provide information for the research of a particular anomaly, and to differentiate flight related damage from ground handling damage.

After the vehicle is jacked and leveled in the OPF, quality assurance personnel perform a detailed inspection of the TPS. This inspection is performed per operations maintenance instruction (OMI) V6028. The vast majority of the inspection is complete within the first three weeks of the processing flow. The inspection is performed by two methods, a general macro-inspection and a detailed micro-inspection. The macro-inspection consists of a walk-around inspection from 3 to 5 feet away for damage, discoloration, missing parts, or over-heated areas. The micro-inspection consists of a detailed, hands-on inspection for charred filler bar, hot spots, subsurface flow, thermal barrier discrepancies, missing parts, and gap filler damage. A summary of the inspection methods to be used for specific areas on the vehicle are presented in table B1. The inspectors use the acceptability criteria as defined by the Rockwell Design Center in the ML0601-0002 specification. From the inspection, the quality assurance personnel generate problem/discrepancy reports (PR/DRs) or matrix discrepancy reports (MDRs).,

Table B1. Post-Flight Inspection Criteria for Specific Orbiter Locations

Orbiter Area	3 to 5 Ft Macro Inspection	Hands-On Micro Inspection
Lower Forward Fuselage		
Lower Surface Acreage	X	X
Nose Landing Gear Door Thermal Barriers		X
External Nose Cap and Chin Panel RCC		X
Upper Forward Fuselage		
Air Data Probe Door and Adjacent Tile		X
Upper Surface Acreage	X	
Sidewall Acreage	X	
FRCS Module Acreage	X	
Crew Hatch Thermal Barrier		X
External Nose Cap RCC		X
Nose Cap Thermal Barrier		X
FRCS Thruster and Plume Shield Thermal Barriers		X
FRCS Thermal Barrier Interface		X
AFRSI/Tile Interfaces		X
Lower Midfuselage		

Lower Surface Acreage	X	X
Upper Midfuselage		
Midbody Sidewall Acreage	X	
Payload Bay Door Acreage	X	
All Vent Doors		X
Payload Bay Door Expansion Joints		X
FRSI/Tile/AFRSI Interfaces		X
Payload Bay Door Hingeline AFRSI/FRSI		X
Lower Wings		
Lower Wing Acreage	X	X
Lower Elevon Acreage	X	X
Elevon Cove Area	X	X
Wing Tip Area	X	X
Elevon Split Sidewalls	X	X
Elevon Columbium Seals		X
Extemal Wing Leading Edge RCC		X
Lower LESS Access Panels, Tiles, Gap Fillers		X
Main Landing Gear Door Thermal Barriers		X
Upper Wings		
Upper Wing Acreage	X	
Upper Elevon Acreage	X	
FRSI/Tile/AFRSI Interface		X
External Wing Leading Edge RCC		X
Upper LESS Access Panels, Tiles, Gap Fillers, Thermal Barriers		X
Lower Aft Fuselage		
Lower Surface Acreage	X	X
Lower Body Flap	X	X
Upper Aft Fuselage		
Upper Body Flap		X
Base Heat Shield	X	
OMS/RCS Thruster Thermal Barriers		X
OMS Pod/Aft Fuselage Interface		X
Aft Fuselage Sidewall Acreage	X	
FRSI/Tile/AFRSI Interface		X
Vertical Stabilizer		
Vertical Stabilizer Acreage	X	
Leading Edge	X	

Vertical Stabilizer/Aft Fuselage Interface		X
Rudder Speed Brake Split Line Thermal Barrier		X
Rudder Speed Brake Perimeter Springs and Tabs		X
AFRSI/Tile Interfaces		X
SILTS Pod Tile and Gap Fillers (OV-102 Only)		X
OMS Pods		
OMS Pod Acreage	X	
AFRSI/Tile Interfaces		X
OMS Leading Edge Tiles		X

Prior to the roll out from the OPF, the vehicle is inspected an additional time per OMI V6037. This inspection usually takes 10 days and is predominately a macro-inspection to identify any damages that are the result of ground processing.

In additional to general TPS inspections, there is a series of preflight testing of critical areas. The thermal barrier seal around lower surface penetrations and lower LESS access panels are checked for potential flow paths. A surface contour inspection is performed in the aerothermal roughness critical area in the vicinity of the nose landing gear door if door rerigging or major rework was performed in the area.

Final inspections are performed at the launch pad. The TPS adjacent to the RCS thrusters is inspected for contamination from the hypergolic propellants. The TPS in the vicinity of access doors and protective covers are inspected for damage during the flight closeout operations. The last inspection is performed 4 hours prior to launch by the ice and debris inspection team. The team identifies ice formations on the ET that could potentially damage the TPS during launch.

B.2.2 Documentation

All of the work authorizing documents (WADs) written and completed during a processing flow are recorded in a test assembly inspection record (TAIR) index. A WAD is a document which directs technicians to perform work with a set goal, usually to rework a discrepancy. There is a TAIR category for each of the work areas as shown in table B2. In addition to locations, there are indexes for each of the documentation types. The three types of WADs are KSC form 2-151A: Problem/Discrepancy Reports (PR/DR), KSC form 4-496: Matrix Discrepancy Reports (MDRs), and KSC form 4-124B: Test Preparation Sheets (TPSs).

The PR/DR form is used for a vast majority of discrepancies. Minor discrepancies detected during the inspections are documented on discrepancy reports (DRs). The disposition of DRs is performed by qualified technicians. More significant discrepancies are documented on problem reports (PRs). PRs require engineering evaluation and disposition.

Matrix discrepancy reports (MDRs) are used to document minor chips and gouges in RSI tiles. MDRs are a simplified discrepancy report, disposition, and process control record (PCR) in one (refer to section B.2.3). The data recording on the MDR is greatly simplified. Each MDR form can accommodate up to four tile discrepancies. The vast majority of the four tiles on an MDR are all within close proximity of each other. This permits the ability to work on these tiles in parallel, improving rework efficiency.

In addition to MDRs, there are commonly three bucket PRs (i.e., a problem report on a general area documenting several components with similar discrepancies), known as slurry logs, which address the chips on the base heat shield and the base of each of the OMS pods. Slurry logs are similar to MDRs in that the paperwork is greatly simplified. The rework efficiency is equal to, or better than, the MDRs. As the name suggests, slurry logs can only be used to perform RSI tile densification slurry-type repairs (refer to ML0601-9025 procedure TPS-207).

The TPSs are used for configuration change (A-type) and non-configuration change (B-type). The A-type TPS is commonly used for TPS modifications. The B-type can be used for non-discrepancy work, usually to gain structural access. A majority of the removal and replacements are performed on TPSs. The TPS can also be utilized to perform testing, as the name suggests, and other non-rework type labor.

TPS record keeping is performed primarily by the Thermal Information Processing System (TIPS) database. The TIPS database records all of the TPS removal and replacements and all PCRs used during the rework of a discrepancy.

Table B2. TPS TAIR Categories

Category TAIR	Orbiter Area Affected
AFT	Aft Fuselage (Aft of Xol307 Bulkhead), Base Heat Shield, Body Flap, External Tank Doors
FRSI	All FRSI Protected Areas (Predominately Upper Wings and Payload Bay Doors)
FWD	Forward Fuselage (Forward of Xo582 Bulkhead), Canopy, Crew Hatch Area
LWNG	Upper and Lower Left Wing Areas Main Landing Gear Doors
MID	Lower Midbody Fuselage (Between Xo582 and Xo1307 Bulkheads), Midbody/ Payload Bay Sidewalls
PLBD	Payload Bay Doors Areas Other Than FRSI Protected Areas
RINST	Instrumented TPS Components
RSI	General TPS (Generic Vehicle Discrepancies)
RWNG	Upper and Lower Right Wing Areas, Main Landing Gear Doors
TES	Thermal Element System (RCC and Windows)
TFRC_	Forward Reaction Control System (FRCS) Module FRC Module Specific (e.g., TFRC2)
TLP0_	Left Hand Orbital Maneuvering System (OMS) Pod Left OMS Pod Specific (e.g., TLP05)
TRP0_	Right Hand Orbital Maneuvering System (OMS) Pod Right OMS Pod Specific (e.g., TRP05)
VERT_	Vertical Stabilizer, Rudder Speed Brake, and Shuttle Infrared Leeside Temperature Sensing (SILTS) Pod (OV-102 Only)

B.2.3 Maintenance

TPS maintenance is performed by certified technicians using WADs. The WAD can be to rework a discrepancy (e.g., PR/DRs, MDRs), to perform a removal and replacement of a component (e.g., PRs, TPSs), to perform mandatory refurbishment of specific components (e.g., operations maintenance nstructions (OMIs), TPSs). The vast majority of the WADs utilize the processes and procedures documented in the ML0601-9024, ML0601-9025, ML0601-9026, and ML0601-9028 specifications. Each of these processes and procedures have an abbreviated process control record (PCR) that is used to record the buys for all work steps. Other WADs utilize detailed work steps to direct the technician to perform the specific maintenance.

B.2.4 Rewaterproofing

The RSI tile and FI blankets must be rewaterproofed prior to every mission. Due to the unpleasant side effects of the exposure to the dimethylethoxysilane (DMES) waterproofing compound, the operation is often performed on third shift with technicians utilizing hard-line supply breathing air and inspectors wearing air-purifying respirators. The majority of the rewaterproofing task is complete within the first three weeks of the processing flow provided that the vehicle was dry prior to rewaterproofing. In the event that the vehicle is rained upon, infrared thermography scans are used to detect the latent cooling from water evaporation to locate wet areas. This method is used to confirm the TPS is dry prior to authorizing the rewaterproofing effort.

TPS internal rewaterproofing with DMES waterproofing compound is performed each flight to satisfy the operations maintenance requirements specification document (OMRSD). The pneumatic injection of DMES into each RSI tile is performed per ML0601-9024 process 113, and the injection into FI blankets is performed per ML0601-9024 process 114. The tiles on the base heat shield, upper body flap, base of the OMS pods, and drag chute area are not rewaterproofed as these areas are not exposed to steady-state temperatures that exceed the degradation temperature for DMES. In addition, these areas are not in the direct path for rain water contact while at the launch pad. There are AFRSI areas on the upper wings, payload bay door, midfuselage sidewall, and vertical stabilizer that do not require rewaterproofing every processing flow. These areas have a low DMES burnout depth and would not present a significant weight impact if the blankets were saturated at launch. There is no need to rewaterproof FRSI, as the polymeric nature of the Nomex renders it waterproof. In addition, the elastomeric coating on the FRSI provides an additional moisture barrier to the Nomex felt. The rewaterproofmg for TPS soft goods (i.e., gap fillers, thermal barriers, thermal pads, etc.) is not required as the potential for weight pickup from water at launch is negligible.

B.3 TPS Processing Personnel at KSC

There are personnel at KSC working under different contracts that are assigned to quality assurance, engineering, maintenance, manufacturing, operations, and off-site technical tasks.

B.3.1 Quality Assurance

Quality assurance personnel inspect the TPS and ensure conformance to the specifications on all maintenance performed at KSC. The SPC Quality operates under a NASA- KSC contract and is the primary quality assurance organization. They perform all of the major vehicle inspections. The SPC Quality also perform in-process inspections to witness or verify, as applicable, the repair and replacement of the TPS components. There is also a NASA-KSC Quality Assurance group, however, this group is not TPS specific nor is it as numerous as the SPC group. The NASA Quality organization is included in the acceptance of a cavity surface preparation for RSI tile bonding.

B.3.2 Engineering and Review Boards

There are three separate engineering organizations involved in the routine maintenance of the orbiter TPS.

The SPC TPS Systems Engineering is the largest engineering organization and is responsible for the disposition of the discrepancies reported by Quality. Normally, the SPC TPS Engineer refers to: the ML0601-9025 specifications and chooses a repair method that is the most appropriate for the damage. Oftentimes, the extent of the reported damage requires the use of repairs within the ML0601-9026 specification which requires Rockwell Design Center evaluation. The organization within the SPC TPS Engineering group is primarily vehicle specific with one exception; there is a group dedicated to the operational requirements imposed by the TPS maintenance specifications. This group is responsible for the interpretation of the specifications and the maintenance of the processing control records (PCRs) which are the documents that the technicians record the necessary information of each rework procedure.

The NASA-KSC TPS Systems Engineering group reviews all problem report dispositions, performs inspections, and provides oversight to the contractor operations.

The Rockwell Orbiter TPS Engineering group, which operates under a NASA-JSC contract, represents the Design Center for all operations at KSC. The responsibility of the Orbiter Engineering group is to ensure the Design Center requirements are satisfied through the technical maintenance of the operational specifications and the material review of work performed at KSC. The Orbiter Engineering group also performs detailed analyses to assess out- of-tolerance conditions and material compatibility. The organization of the Orbiter TPS Engineering group is discipline (i.e., aerothermal, stress, materials and processes, design, and manufacturing) specific with individuals representing each vehicle as well.

The engineering community also provides material review (MR) support. Material review is a system that authorizes a one-time repair or disposition of a discrepancy which can not or will not be returned to engineering drawing, specification, or standard repair requirements. Material review items are approved by qualified Rockwell Orbiter Engineers, SPC TPS Engineers, NASA Systems Engineers, and SPC and NASA Quality Assurance serving as the material review board (MRB). The MRB disposition can not authorize the rework of an unacceptable part to accommodate for a discrepant part, nor can it authorize the substitution of an unauthorized part for a discrepant part. The disposition can not correct a design deficiency or authorize a part deletion. There are three basic types of MRB dispositions. Discrepancies that do not adversely affect the safety, reliability, durability, performance, or interchangability are dispositioned to be "acceptable as is for unrestricted use." Repairs that exceed the standard rework allowables or procedures but return the part to an acceptable, but non-design, configuration are dispositioned to be "repaired for unrestricted use." The third MRB disposition is the "acceptable as is or repaired for restricted use." This disposition is used for discrepancies or repaired configurations that adversely affects the durability or performance and requires post- flight evaluation and/or correction. The restricted acceptance is limited for a specific period, usually one flight. The restricted dispositions require additional approval from the senior level Rockwell Orbiter Engineers, SPC TPS Engineers, NASA Systems Engineers, and SPC and NASA Quality Assurance serving as the prime material review board (PMRB).

B.3.3 Maintenance

All of the TPS maintenance is performed by SPC technicians. Technicians are involved in the material preparation for TPS operations as well. The technicians are assigned to a specific work area with the exception of the splash shop technicians being assigned to all processing areas. All technicians are periodically evaluated and are required to successfully pass certification courses. The certification courses, the certification duration, and the recertification requirements are summarized in table B3. The recertification is obtained by a combination of the following, as shown in table B3: successful completion of a written test, completion of a recertification class, approval of the technician or Quality Assurance supervisor, and approval from SPC engineering.

Table B3. TPS Certification Summary

Certification		Certification Title	Certification Duration (months)	Recertification			
Tech	Quality			Test	Class	Supv OK	Eng OK
	376	Pre/PostFlight TPS Inspection	24	X		X	X
387	387	Tile Damage Evaluation	12	X		X	
118	394	Step and Gap Measurement	12	X		X	X
430	430	Laser Step and Gap Measurement	24	X	X		
337	372P	Mix Crib (Material Preparation)	12	X		X	
349	399P	Dew Point Measurement	12	X		X	
116-1	370P	RSI Tile Installation on 0.160 SIP	12	X	X		
311	332P	RSI Tile Installation on 0.090/0.115 SIP	12	X	X		
117-2	377	RS1 Tile Standard Repair	12	X		X	X
117-1	377	RSI Tile Standard Repair & RTV Repairs	12	X		X	X
092	379	On-Vehicle Tile OML Machining	12			X	X
179	179	RSI Plug Installation	24	X	X		
102	373	Tile Bond Verification	12	X		X	
180		Hot Wire Tile Removal	12			X	X

338	339	Ceramic Shim Bond	12	X		X	
345		Heat Sink Bonding	12	X	X		
483	483	Working with Beryllium	12	X	X		
298	396	Gap Filler Design and Installation	12	X		X	X
347	398	Elevon Cove Flow Restrict	24	X		X	X
003-1	369P	FI Blanket Installation & Repair Level 1	12	X	X		
003-2	340P	FI Blanket Installation Level 2	12	X	X		
065	341P	FRSI Installation	12	X	X		
432	432	Rewaterproofing	12	X	X		
500		Forward RCC Installation/Removal/Repair	24	X	X		
592		Wing RCC Installation/Removal/Repair	24	X	X		
115	395	RCC Coating Repair	12	X		X	
104	104	Window Polishing	24	X	X		
241	367P	Instrumentation	12	X	X		
009	009	Bonded Sensors	24	X	X		
001-1	001-Q	Structural Bonding	6	X	X		
001-2	001-2	Structural Bonding	12	X	X		
567	567	Drag Chute Handling	12	X	X		

B.3.A Manufacturing

The personnel in the Thermal Protection System Facility (TPSF) are responsible for the manufacturing of most of the TPS components used on the vehicle. The manufacturing is performed by Rockwell International under a NASAKSC Logistics contract. The TPSF Quality Assurance group ensures conformance to the fabrication specifications. The TPSF Manufacturing Engineers prepare planning documents, disposition problems, and develop processing tools. The logistics personnel monitor the flow of component orders and ensure the punctual delivery of the manufactured part. The Rockwell TPS Orbiter Engineering group is involved in the daily manufacturing activities and ensures the Design Center requirements for component manufacturing are satisfied.

B.3.5 Operations

The management of all of the work performed on the orbiter is regulated by Operations. There are two distinct organizations, the SPC Operations group, operating under a NASAKSC contract, and a smaller NASA-KSC group. The Operations group determines how to allocate technician resources to best accomplish the processing goals for the flow.

B.3.6 Off-Site Technical Personnel

The off-site technical tasks are performed by personnel from NASA-JSC, Rockwell-Houston, and Rockwell-Downey. Engineers from NASA-JSC and Rockwell-Houston perform the technical evaluations of the TPS during Shuttle missions. Rockwell-Downey is the Space Shuttle Design Center and is ultimately responsible for the design configuration of the TPS, problem analyses, and certification for flight.

Space Shuttle Orbiter Thermal Protection System Processing Assessment

Appendix C

Overview of TPS Material Degradation and Failures

May 1995

Prepared by:

Michael P. Gordon
TPS Orbiter Engineering
Materials & Processes
Rockwell Florida Operations D/830

Summary

The primary purpose of this appendix is to provide an overview of the degradation and failures associated with the thermal protection system (TPS) of the Space Shuttle orbiter. Although this appendix is a self-contained informative document, it is intended to accompany the "Space Shuttle Orbiter Thermal Protection System Processing Assessment Final Report." The purpose of the final report is to present the conclusions of a study to assess the processing of the thermal protection system of the Space Shuttle orbiter. This study was initiated on November 10, 1994 in support of the NCC2-9003 Cooperative Agreement Notice (CAN), Lightweight Durable Thermal Protection System, TA-3 Task 6 for the Single Stage To Orbit (SSTO) Lightweight Durable TPS project being performed by Rockwell, Downey under the Marshall Space Flight Center (MSFC) NASA Research Announcement (NRA) 8-12 program. This appendix does not contain proprietary information.

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References

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C.1 TPS Material Degradation

The materials associated with the thermal protection system are designed to protect the structure from the extreme reentry thermal environment while being as light as possible. The mostly ceramic-based materials all exhibit a balance between thermal performance and physical strength. Implementation of these materials into a system design involves a compromise of the thermal/physical properties of the particular materials. Despite this design effort, the system is oftentimes susceptible to degradation by one, or both, of these factors. The thermal limitations and physical limitations of the TPS materials are discussed in the following sections.

C.1.1 Thermal Limitations

Most of the TPS materials are subjected to the highest heat loads upon reentry. The exceptions are the base heat shield areas (i.e., orbiter base heat shield, OMS/RCS pod base heat shield, upper body flap, and vertical stabilizer rudder/speed brake trailing edge) which are subjected to the highest heat loads on ascent from SSME and SRB plume convection and radiation heating.

All of the materials used in TPS have thermal limitations. The maximum temperature constraints for all TPS materials are presented in table C l. It should be noted that the thermal limitation for one particular material may not be the limitation for the finished component. For example, each of the components for AFRSI (i.e., quartz fabric, thread, and batting) can be exposed to temperatures near 2,000°F, however the fabricated blanket degrades rapidly at temperatures in excess of 1,500°F. The temperature limitations are given for a 100-mission exposure and a single-flight exposure. Should any material be exposed to temperatures in excess of the 100-mission limitations, it should be evaluated post flight and replaced as necessary. The temperature operating range allowables for TPS components are given in table C2. This table includes the critical item if the temperature limitations are exceeded.

Table C1. TPS Maximum Temperature Constraints

TPS Element	Material Composition	Temperature Constraints (°F)	
		100 Mission	1 Mission
RSI Tiles	LS-900	2,300	2,600
	FRC1-12	2,300	2,700
	LI-2200	2,300	2,900
HRSI Tile Coating	Reaction Cured Glass	2,300	2,700
LRSI Tile Coating	Borosilicate Glass	1,200	2,100
AFRSI Blanket	Quartz Fabric, Batting, & Thread	1,500	1,800
FRSI (Ascent/Entry)	Elastomer-Coated Nomex Felt	750/700	900/850
FRSI Edge Member	RTV-577 Silicone	700	850
LESS Components	Reinforced Carbon Carbon (1)	2,960	3,220
Filler Bar	Heat-Treated Nomex Felt	800	1,100
SIP	Heat-Set Nomex Felt	550	625
RSI Bondline	RTV-560/566 Silicone	550	625
RTV Fill-Type Repairs	RTV-560/566/577 Silicone	800	1,200

Structural Substrate	Aluminum (Al-2024 or similar)	350	
	Graphite Epoxy (Payload Bay Door)	350	
	Graphite Epoxy (OMS Pods) (2)	250	
Gap Fillers and Thermal Barriers	Nextel Cover Fabric & Sleeving	2,000	2,600
	Saffil Insulative Batting	2,000	2,600
	Inconel Spring Tube	1,000	1,300
	Fibrous Pile (Monkey Fur)	1,400	
Pressure Seals	Silicone Rubber	500	
	PTFE (Teflon)	600	
Metals	Beryllium	1000	
	Columbium	2,400	
	Inconel 718	1,250	
	Inconel 625	1,400	
	Inconel 750	1,500	
	Titanium	800	
	Stainless Steel 316	1,200	
	Haynes 188	1,800	
Ceramics	Alumina	3,000	
	Fused Silica	2,100	
	Macor Glass	1,800	
	Silica Glass	1,270	1,753
	Polyimide 700	700	

Table C2. TPS Component Temperature Limits

TPS Item	Minimum Operating Temperature (°F)	Critical Item and Results if Exceeded	100-Mission Maximum Operating Temperature (°F)	Critical Item and Results if Exceeded	Single-Mission Maximum Operating Temperature (°F)	Critical Item and Results if Exceeded
HRSI Tiles	-200	RTV bondine adhesive becomes brittle below -170°F with possible failures below -200° F	2,300	Performance degradation, possible surface cracking and shrinkage	2,600 (L1900) 2,700 (FRCI-12) 2,900 (LI-2200)	Severe shrinkage, opening of gaps, and structural over heating
		RTV bondine				Severe

LRSI Tiles	-200	adhesive becomes brittle below -170°F with possible failures below - 200°F	1,200	Performance degradation, possible surface cracking and shrinkage	2,100	shrinkage, opening of gaps, and structural over heating
AFRSI Blankets	-200	RTV bondine adhesive becomes brittle below -170°F with possible failures below -200°F	1,500	Performance degradation, fabric and thread embrittlement, possible failure due to erosion	1,800	Fusion of fibers in thread, failure of thread, loss of fabric and batting
FRSI	-200	RTV bondine adhesive becomes brittle below -170°F with possible failures below -200°F	700 (Reentry)	Performance degradation, possible surface cracking and shrinkage	850 (Reentry)	Charring of FRSI, and structural over heating
RCC	No lower limit identified	Not applicable	2,960 (mission life is not a direct function of operating temperature)	Performance degradation, increased carbon oxidation and mass loss	3,220 (mission life is not a direct function of operating temperature)	Silicon carbide and carbon oxidation and erosion, loss of structural integrity
Thermal Window Pane	No lower limit identified	Not applicable	1,270	Performance degradation, glass becomes viscous	1,753	Loss of structural integrity

The thermal limitations of the materials are affected by a series of aeroheating factors. The factors include aerodynamic heating factors from flight profiles, thermo-physical heating factors of the TPS materials, external surface heating factors of the TPS coatings, and external roughness heating factors of the TPS installation. These factors are discussed in the following sections and are summarized in table C3.

Table C3. Summary of Orbiter TPS Thermal Limitation Factors

Heating Factor	Nominal or Preferred Value	Negative or Undesirable Change and Effect
Aerodynamic Factors		
Angle of Attack	40° above horizontal	Decreasing will increase upper surface heat load
Orbital Inclination	28.5° to 63° (KSC launch)	Launching from VAFB (700 to 104°) will increase orbital velocity, increasing maximum heating rate and overall heat load.
Orbiter Weight	202,000 pounds	Increasing weight (payload dependent) will increase heat load
Center of Gravity Location	Xo1077 to Xo1109 (inch)	Changes will require control surface deflections. Extreme forward location is least desirable and will require up position on control surfaces which will increase overall heat load
Reentry Ground Track	Descending (from north)	Ascending (from south) ground track will result in higher heat load by high air density variations
Crossrange Distance	Close proximity between orbital ground track and landing location	Greater crossrange distance results in more maneuvers, longer reentry time, and increased heat loads.
Thermo-Physical Factors		
Thermal Conductivity	Material dependent	Increasing will increase heat load by increasing heat flux through material.
Density & amp; Specific Heat	Material dependent	Decreasing will decrease thermal mass of material which will decrease heat capacity of part.
Fiber Direction	Parallel to bondline	Non-parallel direction will increase thermal conductivity thereby increasing heat load. Non-parallel grain direction also increases the strength of the part.
External Surface Property Heating Factors		
Emissivity	Material dependent	Decreasing will inhibit radiation from surface increasing heat load.
Absorptivity	Material dependent	Changing would inhibit reflection or absorption of solar radiation on orbit affecting thermal control.
Catalycity	Material dependent	Increasing catalycity would promote atomic recombination and increased heat load.
View Factor to Space	Configuration dependent	Decreasing view factor (by obstructions) will decrease total net radiation from orbiter by transferring heat to other locations.
External Surface Roughness Heating Factors		
Steps	Flush	Forward facing steps can cause drastic early boundary layer transition increasing heat load. Aft facing steps have a less drastic effect.
Gaps	Location dependent	Large gaps induce flow paths and affect early boundary layer transition thus increasing heat load. Tight gaps can increase loading within components during on-orbit cold soak.

Waviness	Theoretical OML contour	Non-theoretical waviness can cause early boundary layer transition increasing heat load.

C.1.1.1 Aerodynamic Heating Factors

Aerodynamic heating factors are based on the flight profile of the orbiter during reentry. The pertinent reentry heating factors are the angle of attack, the flow field Reynolds number, the free-stream velocity or Mach number, control surface deflections, the orbital inclination, the orbiter weight, the location of the center of gravity, the reentry atmosphere, the reentry ground track, and the crossrange distance.

The initial angle of attack (i.e., the tilt of the vehicle above a horizontal reference) at initial entry is 400. This results in attached flow on the lower surface and detached and separated flow on the upper surface. This angle protects the upper surface from extreme temperatures and remains somewhat constant for the majority of the reentry phase. Should this angle be reduced during the reentry heating cycle, the upper surface heat load would increase and the lower surface heat load would decrease. When the free-stream velocity decreases to approximately 12,000 feet per second (at an approximate altitude of 170,000 feet), the angle of attack decreases for landing maneuvers.

The orbital inclination (i.e., the angle between the orbital path and the equator) is dependent upon launch location. For KSC Eastern Test Range (ETR) launches, the inclination can vary between 28.5° and 63°. For VAFB Western Test Range (WTR) launches, the inclination can vary between 70° and 104°. Higher inclination orbits will increase the orbital velocity, increasing the maximum heating rate, and, as a result, increase the overall heat load.

Orbiter weight and location of the center of gravity are primarily dependent upon the payload, the payload bay hardware, and their distribution. The nominal weight for an orbiter reentry is 202,000 pounds. This weight can be as high as 248,000 pounds. Increasing the orbiter weight will increase the heat load. The location of the center of gravity can be between X01077 and X01109. The location of the center of gravity affects the deflections of the elevons and body flap. Aft center of gravity locations will require the control surfaces to be in a down position and, conversely, forward locations will result in upward control surface positions. Extreme forward locations are least desirable as the upward control surface positions expose the upper surface and aerosurface penetrations resulting in increased heat loads.

The reentry ground track is one of two paths the orbit travels during the reentry phase. The descending track is from the north-west and travels in a south-east direction. The ascending track is from the south-west and travels in a north-east direction. The descending track is preferred as there are less atmospheric density variations than the ascending track, resulting in a lower overall heat load.

The crossrange distance is the perpendicular distance from the orbital ground track to the landing location. A low crossrange is preferred as it minimizes orbiter maneuvers, decreases the reentry time, and, as a result, decreases the overall heat load.

C.1.1.2 Thermo-Physical Heating Factors

The thermo-physical heating factors are dependent upon the TPS material used. The most important property is the thermal conductivity of the material. The higher the thermal conductivity, the higher the heat flux through the material, which results in higher structural temperatures. Increasing the thermal conductivity of the component can be accomplished by altering the fiber grain direction, material type, or by a fill-type repair.

The alteration of the fiber direction can have a significant effect on the thermal conductivity and strength of the tile. Ideally, the grain direction is parallel to the bondline to minimize the thermal conductivity. However, the strength is minimized with the parallel grain direction and is maximized when the grain direction is in the most-thermally-conductive perpendicular direction to the bondline. Therefore, the component must be cooperatively designed.

Two other thermo-physical heating factors that are material dependent are the density and specific heat. The higher these factors are, the greater heat capacity the component has as the thermal mass is increased. However, the higher these values are, the higher the thermal conductivity is as well. A lightweight, low thermally conductive component is preferred during high heating phases of reentry, and a high thermal mass component is preferred for the transient cooling phases of reentry. The component must be designed with both considerations.

The fill-type repairs for RSI tiles utilize pulverized tile material and tetraethyl orthosilicate (TEOS) or silicone RTV adhesive. These repairs restore the contour but change the thermo-physical properties within the tile, as the density, heat capacity, and thermal conductivity of the repair material are greater than the RSI tile material. For this reason, there are certain depth, volume, and area limitations on these types of repairs.

C.1.1.3 External Surface Property Heating Factors

The external surface properties of a TPS component are dependent on the coating used. The significant properties from a thermal standpoint are the emissivity, absorptivity, catalycity, and the view factor to space.

The emissivity is most important during peak heating cycles. A high emissivity promotes thermal radiation from the tile, thereby removing a greater portion of the thermal energy from the component and subsequently reducing the heat load.

The absorptivity is most important during on-orbit direct and earth-reflected solar radiation exposure. A high absorptivity promotes heating from solar radiation. A lower absorptivity promotes reflection of this radiation. Prudent use of high and low absorptivity materials enables thermal control while on orbit by rotating either the low absorbant upper surface or the high absorbant lower surface towards the solar radiation.

The catalycity is most important during reentry. During reentry, the altitudes and velocities involved induce non-equilibrium disassociated air flow and cause diffusion of atomic oxygen and nitrogen to the orbiter TPS surface. A low-catalytic surface (e.g., RCG coating) inhibits atomic recombination which reduces the heat load from this exothermic chemical reaction. Conversely, a high-catalytic surface (e.g., tile repair) promotes this recombination and increases the heat load.
The view factor to space is most important during reentry. The majority of the TPS acreage has an ideal view factor to space and is able to radiate heat from the vehicle to unrestricted space. Certain areas of the vehicle (e.g., elevon-elevon gap, aft fuselage sidewall-elevon gap, base heat shield, etc.) have structural obstructions and, as a result, do not emit heat as efficiently.

C.1.1.4 External Surface Roughness Heating Factors

During reentry, the air flow is attached to the lower surface and is detached from the upper surface. The air flow on the lower surface is laminar during the initial stages of reentry. As the free-stream velocity and atmospheric density increase, the flow changes, as a function of Reynolds number (about Re=8x 106 ft-1), from laminar to turbulent. The heat load increases rapidly after this boundary layer transition. The transition usually occurs at approximately 1,180 seconds after the 400,000 feet entry interface. The surface roughness can have a significant effect on the heat load by causing the boundary layer transition to occur earlier than designed. The surface roughness is a measurement of relative tile misalignment (i.e., step and gap) along with any deviation of the faired surface contour (i.e., waviness and repairs) resulting from installation or processing. Roughness can cause large changes in transition location causing a forward movement of flow turbulence and have increased heat load downstream.

Steps between TPS components are mostly caused by installation. Ideally, steps should be flush, but there are installation tolerances documented in the ML0601-0001 specification. Forward facing steps can cause a drastic early transition of the boundary layer. Aft facing steps have a lesser effect. The tile corner roughness effect would produce the dominant disturbance because the tile step and gap values are generally larger at the corner.

Gaps between TPS components and contour waviness are caused by installation. The size of the design gap is dependent on the stress and aerothermal loads for the particular location of the component. Most lower surface locations have a gap requirement of 0.045 inch. As with steps, the allowable installation tolerances are documented in the ML0601-0001 specification. Large gaps increase roughness effects which affect boundary layer transition. Tight gaps can increase the loading within the component during on-orbit cold soaks (refer to section C.1.2.1) Deviations from the theoretical OML contour (i.e., installation waviness and tile repairs) also have an affect on boundary layer transition.

C.1.2 Physical Limitations

The TPS materials are subject to several physical limitations. The most common limitation is low impact resistance. Impact damages are the most common discrepancy for TPS materials. The components are designed to be lightweight and have a low thermal conductivity. As a result, the physical properties of these materials are less than ideal and damages frequently occur. The physical limitations of each TPS component are discussed in the following sections.

C.1.2.1 Tile Damages

Tile damages are the most common of all TPS discrepancies. The discrepancies include chips, gouges, coating damage, edge and corner slumping, breaching, and erosion. Chips are small impact damages with the coating and a minor amount of tile substrate missing. Gouges are larger impact damages but are similar in nature to chips. Exposed tile substrate material is susceptible to erosion as the protective layer of glass coating is lost. In areas of high vibroacoustic loads, sidewall chatter-type damage occurs. This damage consists of lost sidewall coating and a minor amount of lost tile substrate material. A similar type damage is caused by tight gap tile-to-tile contact during on-orbit cold soak where the aluminum substrate contracts more than the gap between the TPS materials. Coating cracks are load-related fractures of the coating extending over large areas of the OML and down sidewalls. Occasionally, the cracks travel through the tile substrate to the SIP bondline breaking the tile into two or more pieces. Slumping is a condition where the tile coating and substrate material melts and deforms. Slumping is caused by overheating of the OML surface most likely due to a protruding corner or edge forward facing steps, or subsurface flow down a tile gap. In severe cases, the slumping can evolve to breaching where the coating material has been penetrated and rapid melting and erosion of substrate follow.

The tile repairs predominately use pulverized tile material, densification solutions, silicone RTV fillers, and plugs made from tile material. These repairs can fill a void with a solid material to restore the contour or densify a shallow damage to harden and restore optical properties.

C.1.2.2 AFRSI Damages

AFRSI damages include coating loss, embrittlement, fabric frays, tears, broken threads, or blanket debonds. Ceramic coating loss occurs by mechanical or aerodynamic loading on the part and erosion. Coating loss can propagate to frays and tears if not reapplied. Frayed fabric consists of damaged yams within the fabric weave. Frays can propagate to tears if not repaired. Tears consist of broken fabric. Tears can propagate rapidly if not repaired. Embrittled threads can break from flight loads. Blankets can become debonded from sideloading during flight.

The AFRSI blanket discrepancies can be repaired by using quartz fabric patches, stitching with thread, ceramic coating, and silicone RTV adhesive.

C.1.2.3 FRSI Damages

FRSI damages are the least frequent of all TPS discrepancies. The damages include coating overheating, coating tears, joint seal damages, and edge member damages. The coating overheating is characterized by discoloration and bubbling. The tears in the coating or breaks in the joint seals expose the felt, and edge member damages can expose the structural substrate.

Most of the damages can be repaired by filling the damages with a silicone RTV adhesive.

C.1.2.4 RCC Damages

RCC damages occur much less frequently than other TPS components. The damages include chips and cracks in the silicon carbide conversion coating, flaking of the type A sealant, sealant loss and pinholes, and severe damages that expose the underlying carbon substrate. Minor chips and cracks are repaired by filling with a putty-type mixture. The flaking of the type A sealant is a cosmetic discrepancy and does not require refurbishment. The sealant loss and formation of pinholes in the silicon carbide conversion layer reduce the oxidation protection and result in an increased mass loss rate of the component, which can not be easily characterized. These anomalies, along with substrate-exposing damages, require refurbishment by the vendor.

C.1.2.5 Gap Filler and Thermal Barrier Damages

Gap filler and thermal barrier damages are two of the major items that require rework. Damages include lost coatings, frays, fabric breaching, tears, charring, and protruding or lost gap fillers. The coating anomalies and frays can propagate to further damage if not repaired. They are repaired by recoating with the appropriate material. Breaching and tears on thermal barriers can be repaired by patching or stitching and coating. Improper repair can result in subsurface flow or tile slumping. All other discrepancies are repaired by removing and replacing with new parts.

C.1.2.6 Filler Bar Damages

Charred filler bar discrepancies result from overheating caused by combinations of out-of-tolerance steps, gaps, and heating environments, and are characterized into three stages. The charring mechanism starts as the RTV membrane loses its gloss by exposure to 950°F to 1,100°F (category 1), the RTV membrane chars at 1,100°F to 1,375°F (category 2), and the filler bar chars at temperatures m excess of 1,375°F (category 3). At these temperatures, structural overheating is possible and the tile bondline is vulnerable to overheating. The minor chars are cleaned and the gaps are filled with Ames gap fillers to preclude recurrence or further degradation. Excessively charred filler bar is removed and replaced.

C.2 Mission-Induced Degradation

Aside from the thermal and physical limitations of the TPS materials, there are reoccurring damages that occur in the same locations following each flight. It should be noted that these damages occur across all vehicles (i.e., they are not unique to a particular location on a particular vehicle due to a vehicle-specific anomaly). The damages are summarized in table C4 and are discussed, as a function of mission phase, in the following sections.

Table C4. Mission-Induced Damage Summary

LTPS Item	Discrepancy	Potential Cause	Mission Mode	Critical Items
RSI tile	OML/sidewall chips and gouges	Impacts (debris, ice, & micrometeorites)	All phases	Filler bar, SIP, and structure
RSI tile	Sidewall/IML chips and gouges	Cold-soak contact, chatter	Launch, on-orbit	Filler bar, SIP, and structure
RSI tile	Tile slumping	Overheating from roughness	Reentry	Filler bar, SIP, and structure
RSI tile	Overtolerance step or gap	Processing tolerances	Reentry	RSI tile, filler bar, SIP, and structure
RSI tile	Out-of-tolerance roughness/waviness	Processing tolerances	Reentry	RSI tile, filler bar, SIP, and structure
RSI tile	Non-design alterations	Processing tolerances	All phases	Filler bar, SIP, and structure
RSI tile	Misoriented grain direction or wrong material	Processing error	All phases	RSI tile, filler bar, and structure
AFRSI	Outer ceramic coating loss and/or fraying	Erosion, installation	Reentry, atmospheric flight	Structure
AFRSI	Tears, broken stitches, lost batting	Impacts, overheating, installation	Reentry, atmospheric flight	Structure
FRST	Coating cracks and/or charring	Overheating, cold soak	Reentry, on-orbit	Structure
FRSI, Filler Bar, SIP	Lost material without overheating	Erosion	Reentry, atmospheric flight	Structure
Filler bar	Coating cracks and/or charring	Overheating, subsurface flow	Reentry	RSI tile, SIP, and structure
Gap filler	Ames OML charring	Overheating, protrusion	Reentry	RSI tile, filler bar, and structure
Gap filler	Pillow gap filler frays, tears, embrittlement	Overheating, installation	Reentry	RSI tile, filler bar, and structure
RCC	Chips and gouges	Impacts (debris, ice, & micrometeorites)	All phases	Structure

RCC	Carbon oxidation and mass loss	Sealant loss, overheating	Reentry	Structure
Thermal barriers	Fraying, tears, overheating	Overheating, installation	Reentry	RSI tile, filler bar, and structure
Aerothermal seals	Overheating	Subsurface flow, installation	Reentry	Seals and structure
Windows	Scratches, chips, gouges, and bruises	Impacts, ground handling	All phases	Structure, crew cabin pressure

C.2.1 Launch-Induced Degradation

At approximately T-6 seconds in the Space Shuttle launch countdown, the SSMEs are started. At this time, launch pad debris and paint chips are entrained into the air circulation around the engine bells due to engine thrust. This debris impacts the base heat shield area causing widespread tile damage. This damage, known as peppering, is moderately-sized gouges in the tile OML. These damages are repaired by a slurry-type repair (refer to ML0601-9025 procedure TPS-207) which densifies the exposed area but does not restore the contour. This type of damage can be reduced by improved maintenance of the launch pad or by advanced tile materials.

At liftoff, the aft areas of the vehicle are exposed to acoustic vibrations in excess of 166 decibels. As a result, the tiles installed on 0.160-inch thick SIP vibrate. This results in tile-to-tile chatter damage in areas of tight gaps. A similar type of damage occurs in the acreage areas around the ET attachment structure. This area, known as the ET buffeting area is subject to excitations from the disturbed flow field around the forward ET attach point and aft ET umbilicals. This damage is repaired by shaving the sidewalls to open up the gap to the upper limits of the installation criteria, densifying the surfaces that were exposed during the sidewall machining, and installing 1 ply of Ames gap filler into the affected gap.

During the launch countdown, ice forms from the atmospheric humidity freezing on the external tank and associated cryogenic lines. During ascent, the small ice pieces fall from the tank and lines (especially on the liquid hydrogen (LH2) feed line on the left side) and contact the tiles resulting in damage. In addition, other debris related damage occurs in the forward fuselage and canopy regions. These damages are usually repaired with a plug or putty-type repair (refer to ML0601-9025 procedures TPS-202 or TPS-211) which fills the area and restores the OML contour. Window damages are most likely to occur at this time as the flow impingement in this area is the highest during ascent. The damages could be reduced by improving the cryogenic insulation adhesion and performance to reduce ice formation.

Between the first and second stages, the SRBs are jettisoned from the external tank. The separation mechanism utilizes pyrotechnic devices. This mechanism causes hazing on the windows and could be the cause of window related damages as well. This type of damage is difficult to eliminate as it would require a redesign of the SRB component.

C.2.2 On-Orbit Degradation

During orbit, the vehicle is traveling at extremely high velocities. Orbiting debris (e.g., micrometeorites, previous satellite or space vehicle-related debris, etc.) can impact the orbiter. Impacts of this type can cause significant damage as the impact velocity is very high. Moderate damages are repaired per standard procedures. The occurrence of impacts of this type is rare, but is difficult to avoid altogether.

The orbiter vehicle is exposed to extremely low temperatures during a periods of no solar heating. These "cold soaks" can decrease the bondline temperature with on-orbit temperatures as low as -250°F. This cold soak causes the aluminum structure to contract resulting in tile-to-tile contact in areas of tight gaps. This damage is repaired by shaving the sidewalls to open up the gap to the upper limits of the installation criteria and densifying the surfaces that were exposed during the sidewall machining.

Failure to rewaterproof RSI tiles could result in severe tile damage if the tiles were to absorb water prior to launch. The dimethylethoxysilane (DMES) waterproofing compound degraded at temperatures exceeding 1,050°F. The damage would be caused by the absorbed water freezing and subsequently expanding on orbit at cold soak temperatures below -70°F, thereby inducing a fracture at the 1,050°F isotherm (i.e., the dividing line between the waterproof and dewaterproofed areas of the tile). During the reentry, the absorbed water would convert to steam and complete the failure of the tile by loss of the previously fractured dewaterproofed region.

C.2.3 Entry Interface Degradation

The predominant source of reentry damage is a result of the extreme heat loads that are encountered on the lower surface after laminar-to-turbulent boundary layer transition. Typically damages occur to components installed to out-of-tolerance criteria. Forward facing steps of tiles are susceptible to slumping and the adjacent filler bar is subject to high temperatures from increased flow due to the step condition, potentially charring the filler bar. Tile impact damages can occur. AFRSI, thermal barrier, and gap filler damages are more likely to occur as the strength of the textiles decreases rapidly at high temperatures and the material becomes embrittled. Damages to the aerothermal seals can occur from subsurface flow due to excessive control surface deflections or out-of-tolerance conditions. Other damages are associated with the materials being exposed to temperatures in excess of their thermal limitations (refer to tables C1 and C2). The damages are repaired by removal and replacement or by utilizing one of the standard maintenance procedures. The damages are kept to a minimum by maintaining the original design configuration.

C.2.4 Atmospheric Flight Degradation

The predominant TPS component that is damaged during atmospheric flight is AFRSI. The effects of erosion are much greater in more dense air. The embrittled fabric and previous damages propagate into much more severe damages. Erosion can also degrade the exposed edges of the Nomex felt used in FRSI, filler bar, ramping, and SIP. In addition, the exposed tile substrate within a gouge is susceptible to erosion.

Prior to landing, the landing gear doors open, and occasionally, the Ames gap fillers that were installed to seal flow paths fall out. These gap fillers can cause minor damage to tiles. The corner tiles on these doors, especially on the nose landing gear door oftentimes break. This could be a result of the high degree of loading against the corners from the thermal barriers. This loading could also cause minor damage to the tile lips around the periphery of the door tiles. These damages are repaired per the standard maintenance procedures.

C.3 Processing-Induced Damages

In addition to mission-induced damages, there are processing-induced damages. In the event of a non-KSC landing, the vehicle must be ferried back to KSC. The ferry flight damage causes are the same for the atmospheric flight damages (refer to section C.2.4 with two exceptions: the duration is on the order of several hours, not several minutes, and the significant damages are repaired prior to the ferry flight to preclude serious damage propagation. The other sources of processing-induced damages are ground handling contamination.

C.3.1 Ground Handling Damages

Ground handling is primarily caused by technician contact with the TPS. Certain damages are unavoidable, such as minor adjacent tile damages that occur while a discrepant tile is being removed, or AFRSI damages that occur when a carrier panel is removed. Other damages that occur, such as gouges in high traffic areas, non-TPS tooling damages, or other mishaps, can be avoided through careful activity around the orbiter.

C.3.2 Contamination

Contamination is another processing-related discrepancy that can be avoided. The orbiter TPS can withstand most environmental exposure without corrective action. Examples of this are rain water, atmospheric salt, dust, and bird droppings. Contamination of the materials with orbiter processing fluids presents a problem requiring extensive clean-up operations, and potentially, removal and replacement of the affected components. Spillage of either of the hypergolic propellants, monomethylhydrazine (MMH) or nitrogen tetroxide (N2O4) in vapor or liquid form is damaging to the TPS. The oxidizer N2O4 is especially damaging to the Nomex felt and silicone RTV adhesives. Contamination requires removal and replacement after the spillage is cleaned by OMS/RCS technicians. Orbiter TPS exposure to potable tap water prior to rewaterproofing presents a problem with the rewaterproofing operation. The presence of a sodium cation with a chlorine anion "poisons" the tile (i.e., inhibits the rewaterproofing of the tile with DMES). An MR procedure utilizing a triple injection with DMES must then be performed in an effort to overcome the poisoning effects of the ions. Another waterproofing-related contamination occurs from the past use of hexamethyldisilazane (HMDS) waterproofing compound. It was learned that the HMDS causes a reversion (i.e., irreversible shortening of the polymer chains) of the silicone RTV adhesives which decreases the strength of the bondline. RTV sampling is performed on a flight-to-flight basis to detect further degradation. Spillage of hydrocarbons, such as hydraulic fluid or Freon, has the potential to cause the silicone RTV adhesive to swell and degrade bondlines. Seriously contaminated components must be removed, heat cleaned or replaced, and reinstalled.

Space Shuttle Orbiter Thermal Protection System Processing Assessment

Appendix D

Overview of TPS Maintenance Specifications

May 1995

Prepared by:

Michael P. Gordon
TPS Orbiter Engineering
Materials & Processes
Rockwell Florida Operations D/830

Summary

The primary purpose of this appendix is to provide an abbreviated version of the approximately 1,800 pages of the TPS maintenance specifications. Although this appendix is a self-contained informative document, it is presented as a reference only and is not intended to be used as a replacement for the corresponding specifications. This appendix is intended to accompany the Space Shuttle Orbiter Thermal Protection System Processing Assessment Final Report. The purpose of the final report is to present the conclusions of a study to assess the processing of the thermal protection system (TPS) of the Space Shuttle orbiter. This study was initiated on November 10, 1994 in support of the NCC2-9003 Cooperative Agreement Notice (CAN), Lightweight Durable Thermal Protection System, TA-3 Task 6 for the Single Stage To Orbit (SSTO) Lightweight Durable TPS project being performed by Rockwell Downey under the Marshall Space Flight Center (MSC) NASA Research Announcement (NRA) 8-12 program. This appendix is not proprietary information.

Table of Contents

Referenced Specifications

ML0601-0001 Reusable Surface Insulation (RSI) Tile Step/Gap/Waviness Requirements and Acceptance Criteria for Orbiter Operational Vehicles.

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