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Thermal Protection System of the Space Shuttle

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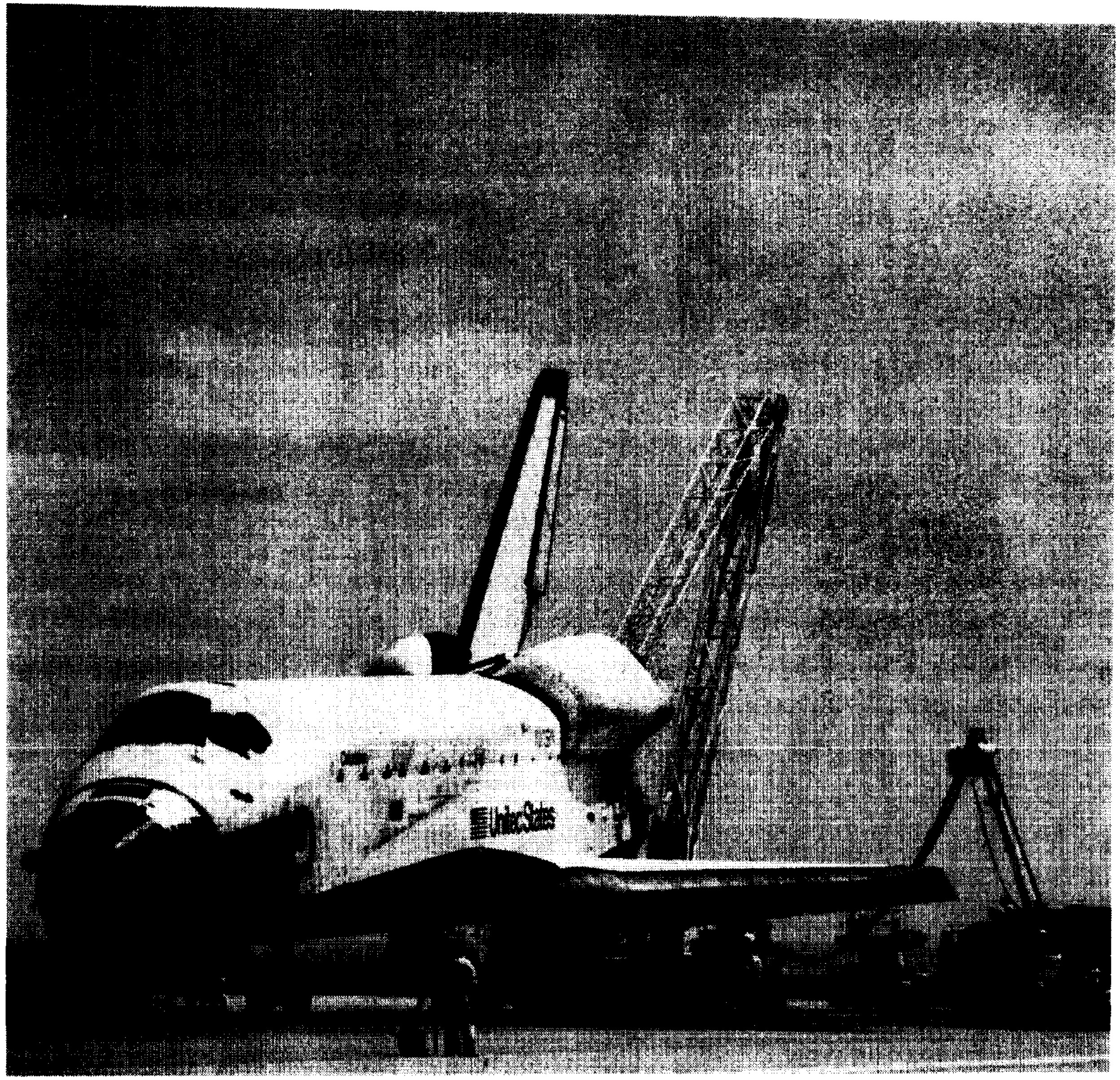

Figure 1. The Space Shuttle Columbia is shown after a successful reentry. The thermal protection system (TPS) has performed as predicted.

INTRODUCTION

The Thermal Protection System (TPS)* of the Space Shuttle is one of the major accomplishments of the National Aeronautics and Space Administration (NASA) Space Transportation System (STS). A casual observer considering heat protection for this spacecraft might think only of the white and black ceramic tiles which cover much of the outer skin of the Shuttle. However, the TPS involves all the materials and their manufacturing methods, design details, testing techniques, and procedures of installation, maintenance and repair developed to protect the Space Shuttle Orbiter Vehicle (OV) from the severe temperature and heat transfer environments encountered in the various phases of each Shuttle mission.

The TPS has also been a very reliable NASA achievement, having performed satisfactorily for 24 missions up to the Challenger accident (a list of these missions is shown in the following page). The Space Shuttle is the first and still the only reusable space vehicle. To be reusable, the vehicle has to be uniquely protected. While most of the major Shuttle systems were adaptations of existing aerospace and aircraft technology, the TPS program engineers, scientists and managers had to meet the launch schedule with a concept never before demonstrated for flight vehicles. Significantly, NASA personnel not only directed the program but also developed and patented insulation system concepts and significant materials breakthroughs.

Many of these developments and those of the NASA contractors which worked on TPS have potential for commercial applications through technology transfers. This report provides, in summary form, fundamental information for understanding the Thermal Protection System and evaluating potential technology transfers to public and private sector interests. NASA and the authors encourage inquiries related to these facts and objectives.

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*Most of the terms, abbreviations, and acronyms used in this document are defined in the Glossary, with their relationship to the TPS context.

Space Shuttle

The Space Shuttle Orbiter, Figure 1, is the principal element in the STS, built by NASA and over 180 United States subcontractors, with contribution by Canadian representatives of the European Space Agency (ESA). The collective components of STS are: 1) the Space Shuttle, 2) the ESA Space Laboratory, and 3) the upper stages of the launch vehicle which are used to boost satellites into geosynchronous orbit. The Space Shuttle is composed of the Orbiter, the external tank, which contains liquid hydrogen and oxygen propellants used by the three main engines of the Orbiter, and two solid rocket boosters (SRB).

The Orbiter and SRB's are reusable. The external tank is expended on each launch, although scenarios for using the tank shell in Earth orbit have been considered. The Orbiter can carry up to seven persons in its two level cabin. Current Shuttle mission plans provide for a maximum duration of 30 days. The original operating Orbiters are Challenger (OV-099), Columbia (OV-102), Discovery (OV-103), and Atlantis (OV-104).

SPACE SHUTTLE FLIGHTS

Flight No.	Date	Orbiter	Duration in days	Flight No.	Date	Orbiter	Duration in days
1	April 12, 1981	Columbia	2	14	November 8, 1984	Discovery	8
2	November 12, 1981	Columbia	2	15	January 24, 1985	Discovery	X
3	March 22, 1982	Columbia	8	16	April 12, 1985	Discovery	5
4	June 27, 1982	Columbia	7	17	April 29, 1985	Challenger	7
5	November 11, 1982	Columbia	5	18	June 17, 1985	Discovery	7
6	April 4, 1983	Challenger	5	19	July 29, 1985	Challenger	7
7	June 18, 1983	Challenger	6	20	August 27, 1985	Discovery	8
8	August 30, 1983	Challenger	6	21	October 3, 1985	Atlantis	X
9	November 28, 1983	Columbia	10	22	October 30, 1985	Challenger	7
10	February 3, 1984	Challenger	8	23	November 26, 1985	Atlantis	7
11	April 6, 1984	Challenger	7	24	January 12, 1986	Columbia	5
12	August 30, 1984	Discovery	6	25	January 28, 1986	Challenger	–
13	October 5, 1984	Challenger	8				

TPS Requirements

The TPS of the Space Shuttle is designed to operate over a spectrum of environments typical of both aircraft and spacecraft, as seen in the mission sequence of Figure 2A. Under normal operations, the TPS is intended to be used for 100 missions with minimum refurbishment, and support a 160-hour turnaround requirement.

The Orbiter structure for the most part is made from 2024-T6 aluminum in a skin-stringer, integrally stiffened, machined-panel or adhesive-bonded honeycomb sandwich structure. A graphite-epoxy (G/E) composite sandwich structure is used for cargo bay doors and the orbital maneuvering system (OMS) pods. Since both aluminum and G/E sandwich degrade significantly above 350 ^{o}F, this temperature became the limit for structural design.

A key design requirement of the TPS was for it to behave as a radiator and reflector (to dissipate heat) and as an insulator (to block the remaining heat from reaching the structure). As a radiator the TPS emits the ascent and reentry heat, while on the upper surfaces it behaves as a reflector to reject most of the incident solar radiation during on-orbit exposure. For example, over 90% of the radiation heat generated at maximum temperatures during reentry is dissipated at the surface due to the emissivity of the black coating used on tiles. The remaining heat (<10%) is blocked using very low conductivity or insulation materials. Additionally, the heat pulse that transfers through the coating is delayed for about 33 minutes so that the Orbiter structure is exposed to its maximum temperature only after landing.

The TPS must protect the Orbiter structure during a variety of missions, with critical mission parameters including orbital inclination (angle to the equatorial plane), altitude, size of payload to be returned, and down range and cross range requirements. Typical parameters and operating environments are shown in Figures 2B to 2D. Figure 2B depicts a typical (though not the most severe) maximum temperature profile experienced by the Orbiter TPS, while Figure 2D presents a corresponding thermal entry environment as a function of time. The TPS must also resist localized heating from the Orbiter main engines, the SRB motors, the Orbiter reaction control system (RCS) engines, and the OMS engines.

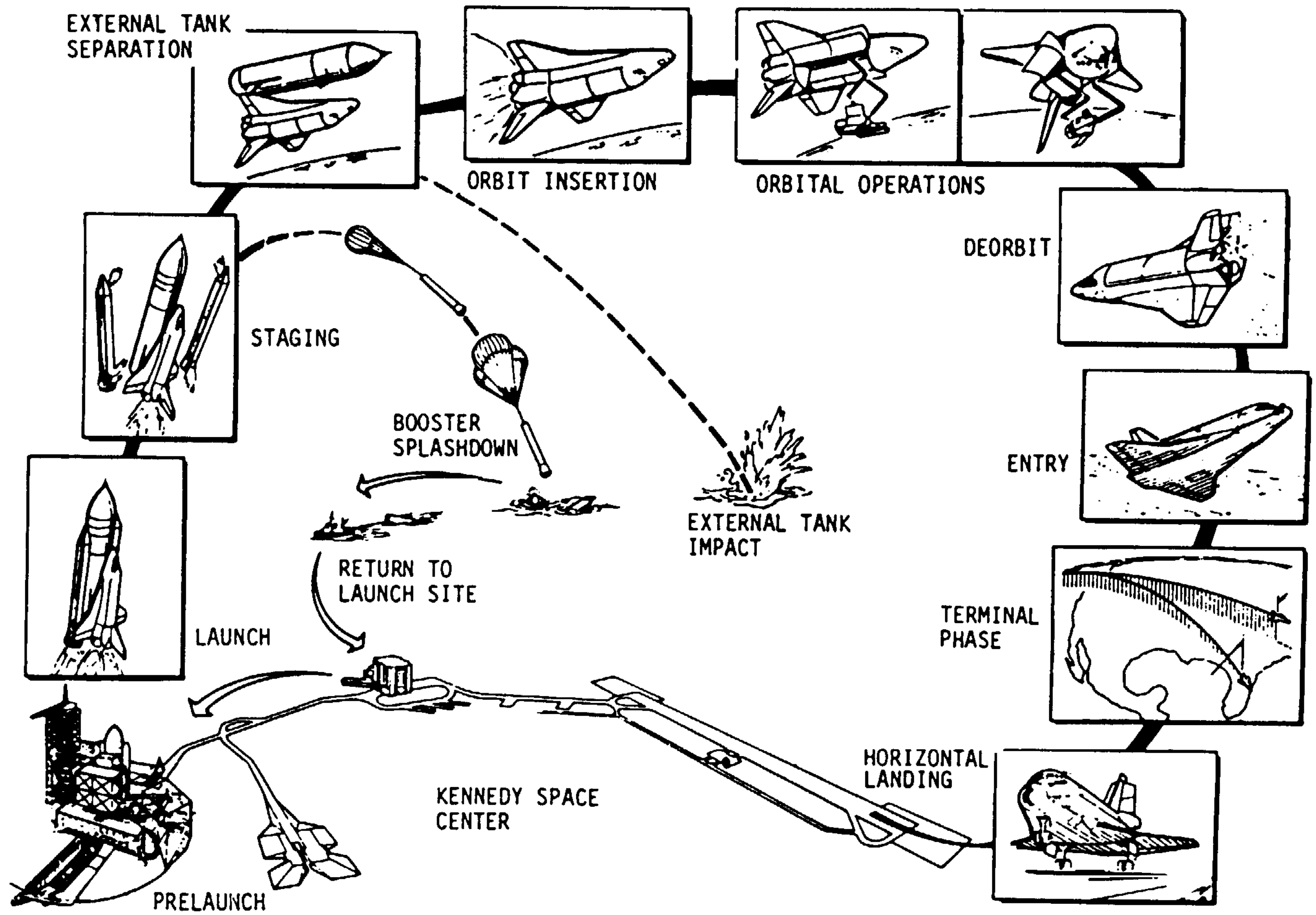
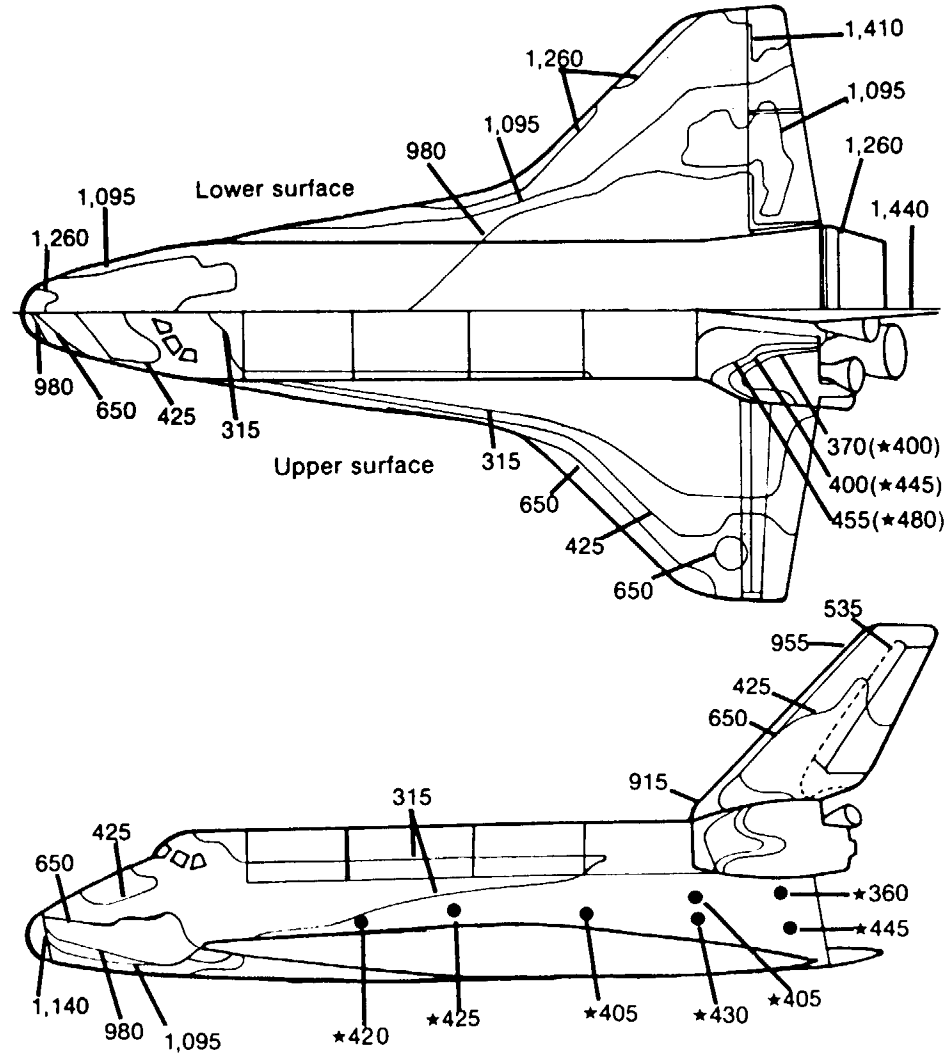


Figure 2. Environments experienced by the TPS of the Orbiter. Figure 2A provides a view of the typical space shuttle mission profile.



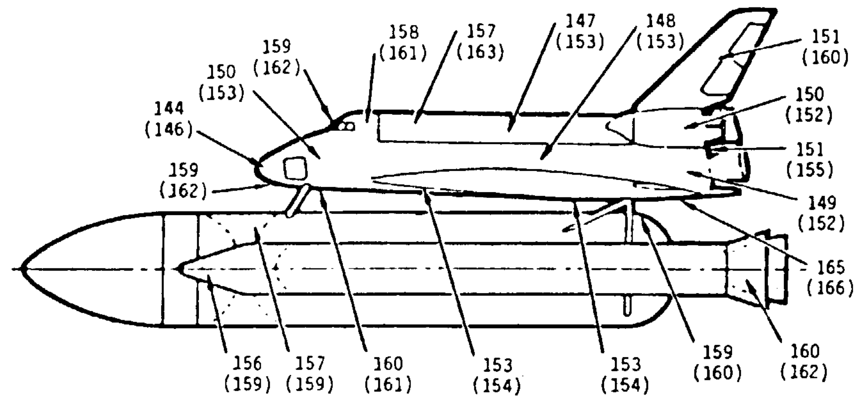
★ Ascent Temperatures

All temperatures in degrees Centigrade
(Maximum YAW 8 degrees)

Figure 2B. Maximum surface temperatures experienced by the Orbiter during normal ascent and reentry trajectories.

WING AREAS

INBOARD				OUTBOARD		
GLOVE	TOP	BTM	ELEVON	TOP	BTM	ELEVON
149 (155)	150 (154)	160 (162)	157 (158)	152 (157)	157 (157)	158 (159)



All values in dB.

Figure 2C. Acoustic environment experienced by Orbiter TPS during initial ascent phase of mission. This environment provides the highest degree of vibrations experienced by the TPS.

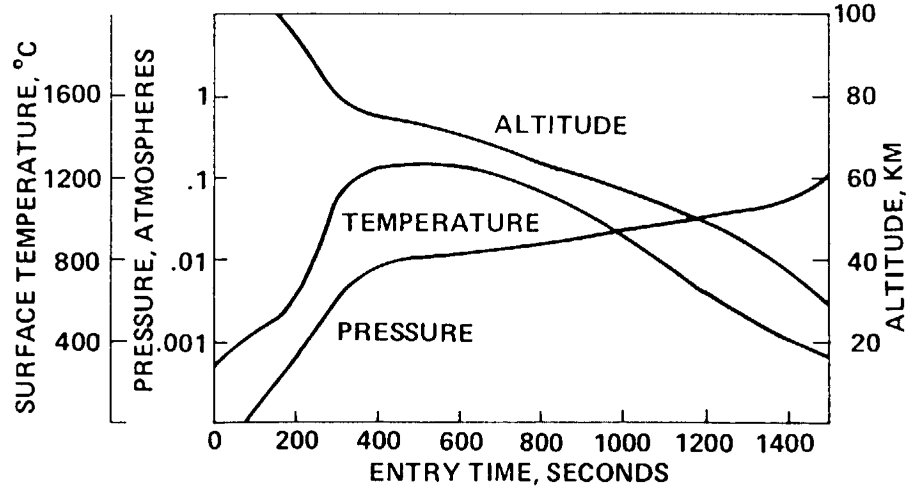


Figure 2D. Altitude, temperature, and pressure experienced by the TPS as a function of time during reentry.

In addition to resisting high temperatures, the TPS must withstand the launch acoustics (Figure 2C), structural deflection generated by aerodynamic loads, the low background temperatures of space while on orbit (cold soak), as well as natural environments such as salt, fog, wind and rain. Because of reuse requirements, the TPS must retain its aerodynamic shape for repeated missions and resist extremely severe thermal shocks and thermal gradients without failure. Aerodynamic smoothness of the installed system is critical. Very tight control of steps and gaps in the TPS must be maintained for two reasons. First, excessive steps or gaps could cause tripping of the aerodynamic boundary layer too early in the entry, resulting in higher heat loads due to early transition from laminar to turbulent flow. Second, relatively minor steps and gaps can result in localized overheating and the possibility of subsurface plasma ingestion, which could destroy the TPS attachment or penetrate the structure.

Major design requirements for the TPS may be summarized as:

- limit structure temperature to 350 ^{o}F;
- provide useful life of 100 missions;
- withstand surface temperatures from -250 to +2750 ^{o}F;
- withstand heat loads to 60,000 Btu/ft^2;
- withstand acoustic levels to 166 dB;
- withstand the aerodynamic pressure (freestream) to 5.7 lb/in.^{2};
- provide the aerodynamic moldline;
- attach to aluminum structure; and
- provide economical weight and cost.

TPS Selection

To meet the challenge of a reusable TPS, new concepts of thermal protection materials and design approaches were necessary. NASA Centers and major aerospace firms took the main TPS concepts from the laboratory to prototype production in the years from 1969 to 1973. At the beginning of this period, four principal thermal protection concepts appeared to have potential application for the Shuttle Orbiter program. These were:

- Replaceable Ablator Panels. Although ablator technology was well-developed at the time, it presented two major problems: First, surface recession of the ablator made reuse a major issue since restoration of the aerodynamic surfaces was required after each flight. For rapid turnaround, the ablator had to be detachable, which was neither weight- nor cost-effective. Second, although ablators would have allowed higher surface temperatures on reentry, those ablators previously qualified for manned flight (such as those used on the Apollo missions) had densities nearly four times greater than ceramic tiles under consideration.

- Metallic Reradiative Heat Shields. For the temperature requirements of the major TPS areas, a metallic system must use coated refractory metals. In spite of the extensive experience with columbium or niobium rocket nozzles on such programs as Apollo and the excellent development efforts on a columbium TPS for the Shuttle Orbiter (funded by NASA-LaRC), the metallic concept had several severe drawbacks. It was heavier than the ceramic tiles. Metallic panels must provide for expansion and contraction without buckling, without distortion of the aerodynamic surface, and without plasma ingestion. The multitude of parts needed for the installation of a metallic system such as clips, beams, standoffs, brackets, and fasteners presented a high degree of manufacturing complexity and coating reliability problems. Local loss of coating could result in burn-through or embrittlement, leading to failure of the columbium substrate. Attachment to curved substructure and dimensional control of the outer mold line presented difficulties as does custom packaging of insulation for each panel section. Finally, thermal structural analysis of the effects of stress, thermal cycling, and creep for various panel geometries would have been a formidable task.

- Carbon-carbon Reradiative Hot Structures. The carbon-carbon material was the only known material that showed potential for providing reuse capability for the areas of the Orbiter TPS exceeding 2300 ^{o}F, such as the wing leading edges and the nose cap region and, thus, was selected for those areas. To insure a multi-mission materials capability, significant developments in coatings to minimize or prevent oxidation of this material had to be accomplished.

- Nonablating, Nonmetallic Insulative-reradiative Ceramic System (the tile system). The nonreceding, ceramic tile heat shield system possessed two unique advantages from the outset: design simplicity and reuse capability. It was recognized that major technology developments would have to be undertaken to bring these materials from the laboratory state to actual vehicle application. However, the significant weight savings and design simplicity inherent with the ceramic heat shield concepts led to their selection as the primary Orbiter TPS.

Due to geometric and temperature constraints, no single concept would have been ideal for all areas of the Orbiter. Because of this, in 1973, the two latter concepts described above were selected as the major ingredients of the TPS. The carbon-carbon hot structure concept was chosen for the leading edges while the tile system was selected for the remaining surfaces.

The tile system endorsed the new approach of a low density insulation which could be tailored in thickness to meet local thermal requirements. This was integrated with a concept of a thin optical coating with minimal weight penalty. Also in innovative fashion, the isolation is bonded directly to the heat skin (i.e. the mass of the structure) through a strain isolation pad (SIP). In addition, structural temperatures were allowed to approach the aging temperature of the material. It has thus been possible to limit the fraction of vehicle weight related to thermal protection to approximately 10% of the landed vehicle weight, less than that associated with previous manned entry vehicles. This is especially impressive since the Orbiter has twice the surface area per unit volume as, for example, the Apollo spacecraft. At the same time, the TPS thermal efficiency equals that of the most advanced ablators despite the lower temperatures to which the Orbiter structure is protected.

It was later realized that the use of tiles with the required minimum thickness of 0.5 in. would have overprotected those surfaces that were not expected to exceed 700 ^{o}F. This would have added unnecessary weight. For example, the upper surface of the Orbiter is shielded from maximum heating by the high angle of attack (30 to 40 degrees) during reentry. Additionally, this upper surface is exposed to maximum solar radiation while on-orbit. This explains the use on this surface of a lower density (5.4 lb/ft^3), lower temperature capability (700 ^{o}F maximum), polymeric base material known as felt reusable surface insulation (FRSI), also, its white coating which similarly to tiles is designed to reflect the direct sunlight while in orbit.

Other areas of the Orbiter required localized solutions to design problems especially around penetrations. For example, hot radiative metallic panels are used on the flipper doors and as engine-mounted heat shields. Maximum temperatures at these locations are 1,400 to 1,600 ^{o}F.

TPS Components

The TPS can be divided into three major subsystems. These are:

- The Reusable Surface Insulation Subsystem (RSISS);
- The Leading Edge Structural Subsystem (LESS); and
- The Penetration Subsystem (PSS).

A summary of these subsystems and their materials for the various areas of the Orbiter is provided in Figures 3A and 3B.

RSISS

The RSISS, of which the tiles are a part, is the major component of the TPS and includes the methods and materials used to assemble this insulation to the outer skin of the Orbiter. Rigid RSI are the tiles coated with either a white or a black borosilicate glass coating. White coated tiles are called low temperature reusable surface insulation (LRSI) while black coated tiles are know as high temperature reusable surface insulation (HRSI). Flexible RSI includes the thermal blankets developed for the outer surface of the Orbiter. These blankets are felt reusable surface insulation (FRSI), a polymeric base material, and advanced flexible reusable surface insulation (AFRSI), a ceramic fiber base material. A summary of the characteristics of these materials is given on page 27.

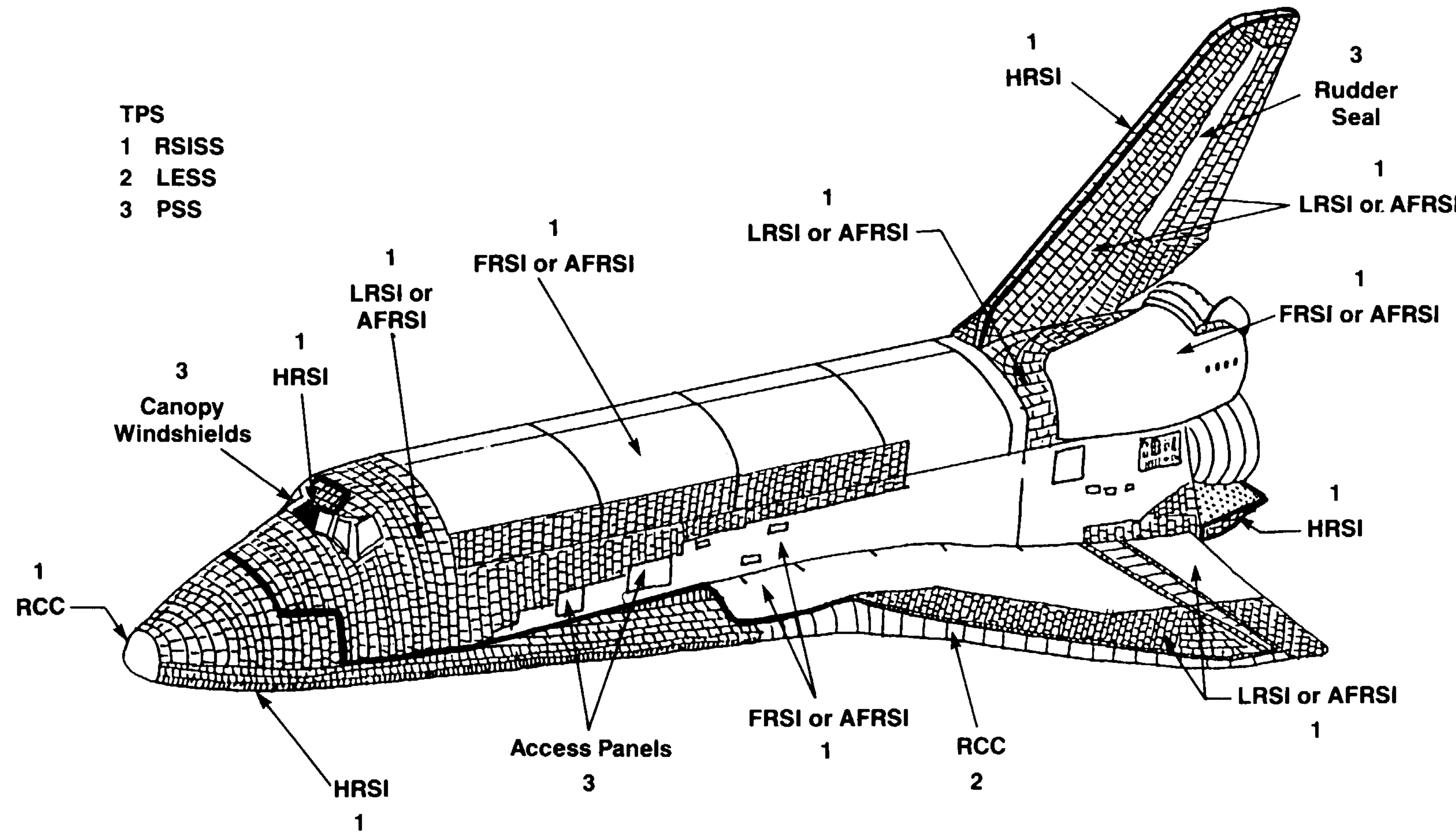


Figure 3. The thermal protection system of the Orbiter. Figure 3A gives a view of the distribution of the various subsystems of the TPS.

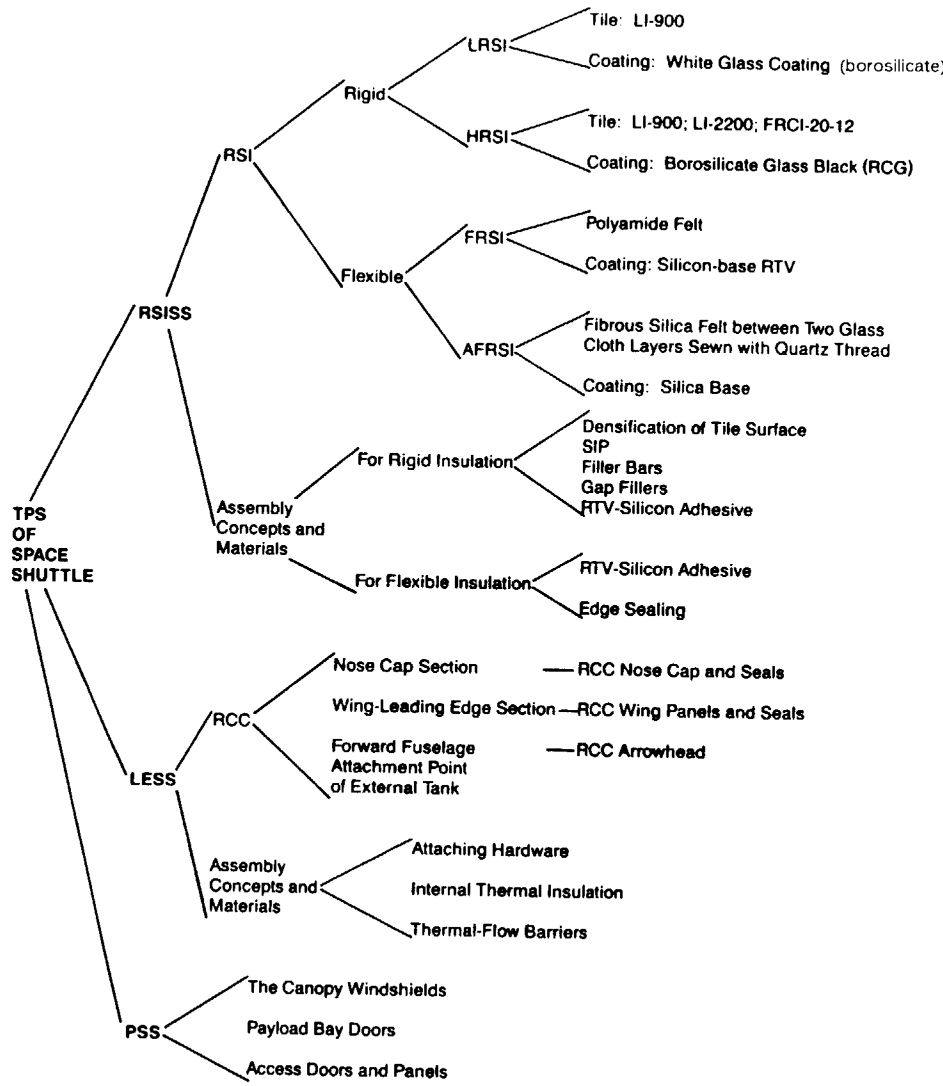


Figure 3B. Description of the TPS.

The rigid ceramic Shuttle tiles are indeed one of the most fascinating aspects of the TPS. Their development can be traced back through work performed by the Lockheed Missiles and Space Company (LMSC) in 1964 on a concept called Lockheat®. By 1971, McDonnell Douglas, General Electric, and Martin Marietta were also being funded by NASA to select the final reusable ceramic tile material. Originally, about 31,000 tiles were used on Orbiter Columbia covering approximately 70% of its surface. Currently, about 24,000 tiles are used instead. Tiles are 93% void with thermal conductivities as low as 0.01 to 0.03 Btu-ft/hr-ft^2-^{o}F under standard, ambient conditions. Each tile has a specific part number and a distinct character configuration contoured to a specific location on the Orbiter. Over 140 manufacturing process controls are applied through stages of automated fiber blending, slurrying/casting, kiln drying, and computer controlled inspection. Tiles are machined with diamond tools on numerical controlled mills.

The amorphous silica fibers used to make tiles known as the silica Q-fiber, are 99.7% pure and heat treated to prevent shrinkage. An amorphous structure results in a very low thermal expansion coefficient. However, a decrease in purity to 99.6% could result in a fiber that tends to crystallize or divetrify at high temperatures. Transformation produces the cristobalite phase which has a high temperature structure with a thermal expansion coefficient up to 30 times higher than amorphous silica. The achievement of a 99.7% pure, amorphous silica fiber (the Q-fiber) was the salient breakthrough that allowed the reusable Shuttle concept to be realized. The efforts of Mr. Robert M. Beasley of LMSC and Mr. John Koch of the Manville Corporation led to this remarkable achievement.

There have been no known thermally induced tile failures in flight. Damage has come from external tank ice impact at launch, runway debris, hailstorms, and handling. Replacement was originally predicted to be 2000 out of 20,000 tiles. This was later revised to 200 and now appears to be 20-50 or less tiles between missions.

LESS

The LESS is generally defined as the insulation concepts adopted for the wing leading edges and nose cap region of the Orbiter. Reinforced carbon-carbon (RCC) is the surface material utilized in LESS. The RCC has been extremely reliable and has provided an initiative for some of the most important current research in high-temperature composites. To date, no RCC panel has had to be replaced following any of the Shuttle missions.

PSS

Finally, the PSS is comprised of the unique insulation concepts and seals required to protect the special areas for the operation and service of the vehicle such as access panels, landing gear doors, and windows. The seals developed for the Orbiter environments are unique and offer interesting potential applications in industry.

Each of the TPS subsystems is described in greater detail in the following Sections of this document. While there is considerable variety in the TPS materials used in these subsystems, there are major characteristics which are common to practically all the primary TPS materials. These include:

- all materials have been made from high purity ingredients;
- all materials are fibrous and anisotropic in nature;
- all materials have a coating of some type;
- all materials have been designed to breathe to equalize pressure; and
- all materials exhibit low thermal expansion, high specific strength, and excellent resistance to thermal shock.

Many properties of TPS materials are included in Appendix II. Most of these reflect the new emphases in materials research and development on composites, ceramics and coatings.

The various upgrades performed on the TPS of operational Orbiters illustrate the continuous evolution of the system. After Columbia (OV-102) flew the initial missions almost entirely protected by tiles, AFRSI blankets were introduced in place of LRSI tiles in some areas on Challenger (OV-099) before its first mission. Also in this Orbiter, some of the LI-2200 tiles (see Figure 3B) were replaced with Fibrous Refractory Composite Insulation (FRCI) tiles. For Atlantis (OV-103) and Discovery (OV-104), practically all of the LRSI tiles were replaced with AFRSI blankets. AFRSI blankets were also substituted for most of the FRSI blankets especially those located in areas exposed to maximum thermal-mechanical loads. For these Orbiters, OV-103 and OV-104, FRCI tiles replaced most of the LI-2200 tiles. These changes were subsequently made for all the Orbiters. Other upgrades made throughout the Shuttle program include, for example, redesign of the original seal and thermal barrier configurations around the nose landing gear doors to address higher-than-expected infiltration of hot gas into the structure during reentry. Another important recent change was the replacement of about five square feet of HRSI tiles located on the lower surface of the Orbiter between the RCC nose cap seal and the nose landing gear doors with a RCC panel known as the chin panel. This change in TPS materials compensates for the large movement experienced by these tiles during reentry which resulted in a hot gas intrusion problem.

TPS Commercial Potential

At least 150 invention disclosures have been related to the lightweight, tile materials alone. More than 50 abstracts related to other potential applications of TPS can be found in issues of the NASA periodical called "NASA Tech Briefs." Possible commercial applications related to the TPS materials include high temperature surfaces and structures for heat cycle and gas turbine engines, furnace heat recovery units (recuperators), piston heads, and electromechanical components.

A few TPS spinoffs serve to illustrate commercial possibilities. For example, Shuttle tiles have been used as a jewelers' soldering base since they absorb so little heat from the jeweler's torch (at 1400-1800 ^{o}F) with no deterioration of the tile material. A bonus here is that the tiles are soft and items to be soldered can be pushed into the surface and held. The tile can also be shaped to mold precious metals into art objects, with no sticking to the tile mold surface.

Another application has been the use of the silica-based tile material as a highly predictable and clean insulator in infrared pyrolysis chambers (typical temperatures to 1130 °F) used in the analysis of organic materials, such as the constituents of raw petroleum and its byproducts. This organics analyzer is known as Pyran System, manufactured by Ruska Instrument Corporation, Houston, Texas.

A spinoff related to TPS manufacturing techniques, rather than TPS materials use, is the development of the Ren Shape® "space block." To develop, verify and maintain quality control over the computer codes needed to cut the tile shapes on numerically controlled machines, test tiles had to be manufactured. It was desired to do this with a material that was inexpensive, easily machinable and which would not produce fumes and dust hazards common with most materials available. Ren Plastic, Ciba-Geigy Corporation, developed an ideal, epoxy-based composite that is now successfully marketed for many other modeling applications.

In general, TPS materials are made from commercially available products which have been modified to meet Space Shuttle requirements. Some of the commercially available products are:

- Q-fiber® (Manville Corporation), a 99.7% pure and amorphous silica fiber which is the most important ingredient of the ceramic fibrous RSI. It is the only fiber used in the LI-900 and LI-2200 tile materials, and also, the thermal insulation felt employed in the AFRSI blankets. It is the primary fiber ingredient of the FRCI-20-12 tile material;
- Nextel 312® (3M Corporation) fiber, used as the secondary ingredient of the FRCI-20-12 tile material. Fabrics made with this fiber are also used to manufacture thermal blankets and gap fillers;
- Vycor® 7930 glass (Corning Glassworks), used as the primary coating frit ingredient;
- RTV-560® (General Electric Company) used to bond all RSI to the structure;
- RTV DC 92-007 (Dow Corning), used as the coating on FRSI blankets; and
- Nomex® fibers (DuPont), used to make the fibrous pad in FRSI, the SIP, and the filler bar.

Little progress has been made in commercializing the actual TPS materials. The various subsystems of the TPS have been designed to provide maximum performance and reliability at the lowest possible weight for 100 missions. Cost was of secondary importance relative to typical industrial and commercial considerations. For example, the cost for a finished, coated tile may range from $4000 to 10,000 per square foot). An un-machined billet of LI-900 (13 in. by 13 in. by 5 in. thick) costs $1000. RCC costs about $16,600 per square foot while AFRSI costs are between $100 to 150 per square foot. In contrast, the cost for the ablator TPS system used in the Apollo capsule was about $30,000 per square foot (all cost values are based on 1988 dollars). To date the entire tile production for all Orbiters has amounted to about 200,000 tiles. Another factor that has not contributed to the commercialization of these materials is that there are a few industrial applications where the stringent TPS performance requirements (i.e., thermal shock resistance from -250 to 2750 ^{o}F and high vibration) are found. However, for such requirements, TPS materials may present a unique solution.

Promising avenues to be explored to increase the commercialization potential of those materials include a decrease on the number of quality control inspections and the use of alternate manufacturing methods such as ceramic injection molding of tiles instead of machining to lower manufacturing costs. FRCI and other rigid RSI materials do not have mechanical characteristics to satisfy many applications. However, these characteristics can be improved considerably by increasing the bulk density of these materials. For example, it has been estimated by NASA researchers at the Ames Research Center (NASA-ARC) that densification of LI-2200 tiles to 30 lb/ft^3 would provide a commercially acceptable product if manufacturing costs can be reduced. Beyond this density value the mechanical properties and thermal stability do not improve at the same high rate. Potential applications are listed and/or described in the examples in later Sections.

The TPS-related techniques of modelling, manufacture, installation, testing, inspection and maintenance offer the most immediate and broadest opportunities for commercial application. These often represent innovations and successful methodologies which might profitably upgrade current industry practices.

Other concepts developed for the TPS, but not applied, exhibit commercial potential. For example, inorganic coatings for mullite to provide water proofing, high emittance and improved handling damage resistance are described in NASA disclosure MSC-14553. The "auger device method" developed by NASA for fastening tiles (U.S. Patent No. 3,936,927) could be used for the installation of soft, fragile materials. This metallic device provides excellent bonding strength and adequate vibration resistance.

No metal fasteners are exposed to the hot face with this technique. A family of advanced external insulation materials has evolved subsequent to the Shuttle TPS development with improved capabilities and strength.

More Information

The following descriptions of the TPS are divided into seven Sections, a Glossary, Bibliography, and three Appendices. These Sections describe the TPS subsystems, and the design, modeling, testing, manufacturing, and installation approaches employed in this program. The last Section entitled "Applications" of TPS materials and techniques, summarizes information most relevant to potential commercial applications of this program.

Four NASA Centers were and are primarily responsible for the TPS technology. The Lyndon B. Johnson Space Center (NASA-JSC) managed the overall program, and made important contributions in materials and testing. NASA-JSC directed the development of LI-900, an LMSC proprietary process, developed and patented FRSI, surface densification and repair techniques for tiles, and several fastening concepts for this insulation. NASA-JSC directed the development of RCC and have published several informative papers on the carbon-carbon materials. NASA-JSC, and on a larger scale NASA-ARC, conducted testing of the TPS materials in large, plasma jet wind tunnels, which are unique NASA facilities. NASA-ARC developed AFRSI and many of the concepts which resulted in rigid RSI. NASA-ARC holds patents on LI-2200 and FRCI tile materials, the RCG tile coating, and the Ames or layer-type gap filler. Kennedy Space Flight Center (NASA-KSC) remains responsible for some of the installation, the maintenance and repair of the TPS. NASA-KSC and NASA-LaRC also conducted extensive environmental testing of the TPS (i.e., for waterproofing and atmospheric agent effects). NASA-LaRC evaluated the structural integrity of the tile-SIP-RTV system. The contributions of the NASA contractors are documented in the literature cited and contained in the Bibliography. Some of these contractors are continuing their own commercial interests related to spinoffs from the TPS program.

Each NASA Center has a Technology Utilization Office which serves the public and private sectors with information on NASA technology. Additionally, the NASA Technology Applications Team at Research Triangle Institute in North Carolina will assist interested parties in assessing potential applications of NASA technology and aid with the initiation of applications engineering projects which incorporate NASA support and that are aimed toward commercialization.

For those interested in samples of TPS materials, the Property Disposal Office at the NASA-JSC may be contacted. The address is Lyndon B. Johnson Space Center, Houston, TX 77058, Mail Code JF-3. An application form to receive excess TPS materials requires that materials be "used for research or technology utilization purposes only, and that the thermal tile will not, under any circumstances, be sold for commercial purposes or as souvenirs." A 16mm film on the manufacture of the TPS tiles may be obtained through the Public Affairs Office of NASA-JSC.

Some of the best references for more detailed examination of the TPS include:

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2. Greenshields, D.H. Orbiter Thermal Protection System Development. NASA Johnson Space Center. AIAA Technical Information Service A77-35304. 1977.

3. Korb, L.J., C.A. Morant, R.M. Calland, and C.S. Thatcher. "The Shuttle Orbiter Thermal Protection System," American Ceramic Society Bulletin, Vol. 60, No. 11 (1981) pp. 1188-1193 (Seven additional articles related to the TPS are also included, in this Bulletin issue, pp. 1180-1217).

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5. Materials Properties Manual. Volume 3. Thermal Protection System Materials Data. Prepared by Laboratories and Test D/284. Rockwell International Space Transportation and Systems Group. Pub 2543-W, Rev 5-79. December 1982.

6. Shuttle Performance: Lessons Learned. Arrington, J. P. and J. J. Jones, compilers. Conference Proceedings. NASA Langley Research Center. March 8-10, 1983. Report No. NASA CP-2283. 759 pages.

7. Dotts, R. L., Curry, D. M., and Tillian, D.J. "The Shuttle Orbiter Thermal Protection System: Materials, Designs, and Flight Performance Overview." SAE 831118. 13th Intersociety Conference on Environmental Systems. San Francisco, CA. July 11-13, 1983. SSN 0148-7191.

Subsystems

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The Thermal Protection System of the Space Shuttle

- Reusable surface insulation subsystem
- Leading edge structural subsystem
- Penetration subsystem

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THE REUSABLE SURFACE INSULATION SUBSYSTEM (RSISS)

The RSISS constitutes most of the TPS and can be divided into rigid and flexible RSI (tiles and blankets) plus the assembly concepts used for this type of insulation. Many of the materials and techniques developed for this subsystem resulted from NASA inventions and ideas. Some of the primary patents related to specific materials and techniques are included with their definitions in the Glossary.

Rigid RSI is divided into low temperature reusable surface insulation (LRSI) or white coated tiles and high temperature reusable surface insulation (HRSI) or black coated tiles. The assembly method developed for rigid RSI (the tile system) includes the Tile-Surface Densification-RTV-SIP attachment method plus the materials used to seal the aluminum structure beneath tile-to-tile gaps. These thermal seals include the filler bar and the gap filler. The filler bar is used underneath all of the tile-to-tile gaps while the gap filler is used between tile gaps located in areas of the Orbiters which experience high aerodynamic pressure gradients.

Flexible RSI is divided into advanced flexible reusable surface insulation (AFRSI), a silica fiber base blanket, and felt reusable surface insulation (FRSI), a polymeric base blanket. The assembly method used for flexible RSI is much simpler than for rigid RSI since it only requires the application of RTV-silicone adhesive between the blanket and the aluminum structure. Edges of flexible RSI are also sealed. Cast films of white RTV silicone adhesive are used to seal the joint between FRSI blankets and also to stiffen these edges while a cigarette shape gap filler is used to fill gaps between corners of AFRSI blankets. The edges of AFRSI blankets are sewn together in areas of the Orbiter which experience high aerodynamic pressure gradients. Also, AFRSI is coated with a silica base coating (C-9) to toughen and stiffen it.

Reusable Surface Insulation Subsystem (RSISS)

- Rigid RSI (tiles)
  - Low temperature reusable surface insulation (LRSI)
  - High temperature reusable surface insulation (HRSI)
- Flexible RSI (thermal blankets)
  - Felt reusable surface insulation (FRSI)
  - Advance flexible reusable surface insulation (AFRSI)
- Assembly Concepts and Materials of the RSISS

– Rigid RSI
(The Tile System)
- RTV Adhesive – SIP – Tile surface densification – Tile
- Gap fillers and filler bars

– Flexible RSI
(The Blanket System)
- RTV adhesive – FRSI
- RTV Adhesive – AFRSI

Rigid RSI

- High Temperature Reusable Surface Insulation (HRSI)
  - Basic Material: Rigidized silica fibers (LI-900), 8.75 $\pm$ 0.75 lb/ft^3
  - Other Materials: (for areas requiring higher thermal stability and strength):
    - A) Rigidized silica fibers (LI-2200), 22 $\pm$ 2 lb/ft^3 (Orbiters Columbia and Challenger)
    - B) Fiber refractory composite insulation (FRCI), 12 $\pm$ 1 lb/ft^3 (Orbiters Discovery and Atlantis)
  - Coating Material: Borosilicate glass with silicon tetraboride emittance agent (RCG)
  - Coating Optical Properties: $\alpha/\epsilon \leq 1.0$; $\epsilon \geq 0.8$
  - Mission Life: 100 missions when used between -250 and +2300°F
- Low Temperature Reusable Surface Insulation (LRSI)
  - Basic Material: Rigidized silica fibers, 8.75 $\pm$ 0.75 lb/ft^3
  - Coating Material: Borosilicate glass (white)
  - Coating Optical Properties: $\alpha/\epsilon \leq 0.4$; $\epsilon \geq 0.7$
  - Mission Life: 100 missions when used between -250 and +1200°F

Flexible RSI

- Advanced Flexible Reusable Surface Insulation (AFRSI)
  - Basic Materials: (Replaced most of LRSI) Quilt-like blanket made with silica (Q-fiber) felt between two layers of glass cloth, and sewn together with silica thread, 11 $\pm$ 1 lb/ft^3.
  - Coating Material: Colloidal silica (Ludox, a product of DuPont) precoat and ceramic (silica) slurry top coat.
  - Coating Optical Properties: $\alpha/\epsilon \leq 0.4$; $\epsilon \geq 0.8$
  - Mission Life: 100 missions when used between -250 and +1200°F
- Felt Reusable Surface Insulation (FRSI)
  - Basic Material: Nomex "E" (product of DuPont) felt; 5.4 lb/ft^3
  - Coating Material: Cast silicone (Dow Corning DC 92-007)
  - Coating Optical Properties: Same as AFRSI
  - Mission Life: 100 missions when used between -250 and +750°F

THE REUSABLE SURFACE INSULATION SUBSYSTEM (RSISS)

A recommended reference list related to the RSISS is:

Dotts R.L., "Orbiter Reusable Surface Insulation Design and Development," NASA/AIAA Fibrous Ceramic Materials Technology Seminar, March 23, 1983.

LI-900

Banas, R.P., E.R. Gzowski, and W.T. Larsen. 1983. Processing Aspects of the Space Shuttle Orbiter's Ceramic Reusable Surface Insulation. In: Proceedings of the 7th Conference on Composites and Advanced Materials, Ceramic-Metal Systems Division, American Ceramic Society, January 16-21, 1983, Cocoa Beach, FL.

Beasley, R.M., Y.D. Izu, et al. 1973. Fabrication and Improvement of LMSC's All-Silica RSI System. Presented at the NASA Symposium on Reusable Surface Insulation for Space Shuttle, November 1972. NASA TMX-2719, Vol. I, pp. 1-17.

LI-2200

Goldstein, H.E., M. Smith, and D.B. Leiser. "Reusable Silica Surface Insulation Material," NASA Tech Briefs, No. B73-10504, December 1973.

Goldstein, H.E., M. Smith, and D.B. Leiser. "Silica Reusable Surface Insulation," U.S. Patent No. 3,952,083.

FRCI

Leiser, D.B., M. Smith, and H.E. Goldstein. "Developments in Fibrous Refractory Composite Insulation," American Ceramic Society Bulletin, Vol. 60, No. 11, November 1981, pp. 1201-1204.

RCG

Goldstein, H.E., D.B. Leiser, and V.W. Katvala. "Reaction Cured Glass and Glass Coatings," U.S. Patent No. 4,093,771.

Goldstein, H.E., D.B. Leiser, and V. Katvala. 1978. Reaction Cured Borosilicate Glass Coating for Low Density Fibrous Silica Insulation. Presented at the Boron in Glass Ceramics Conference, June 7, 1977, Alfred University, Alfred, N.Y. Published in Borate Glass: Structure, Properties, Applications, New York: Plenum.

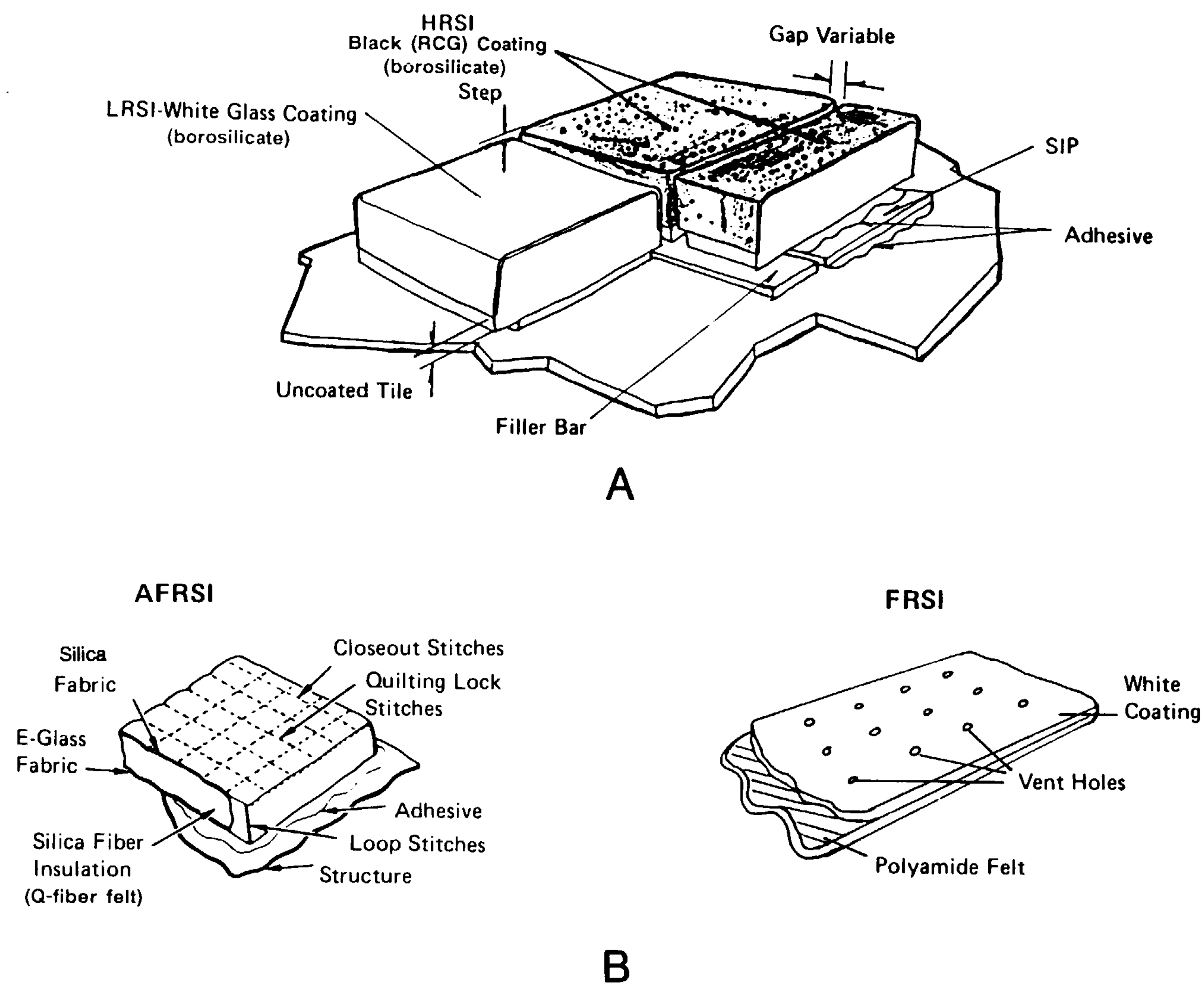


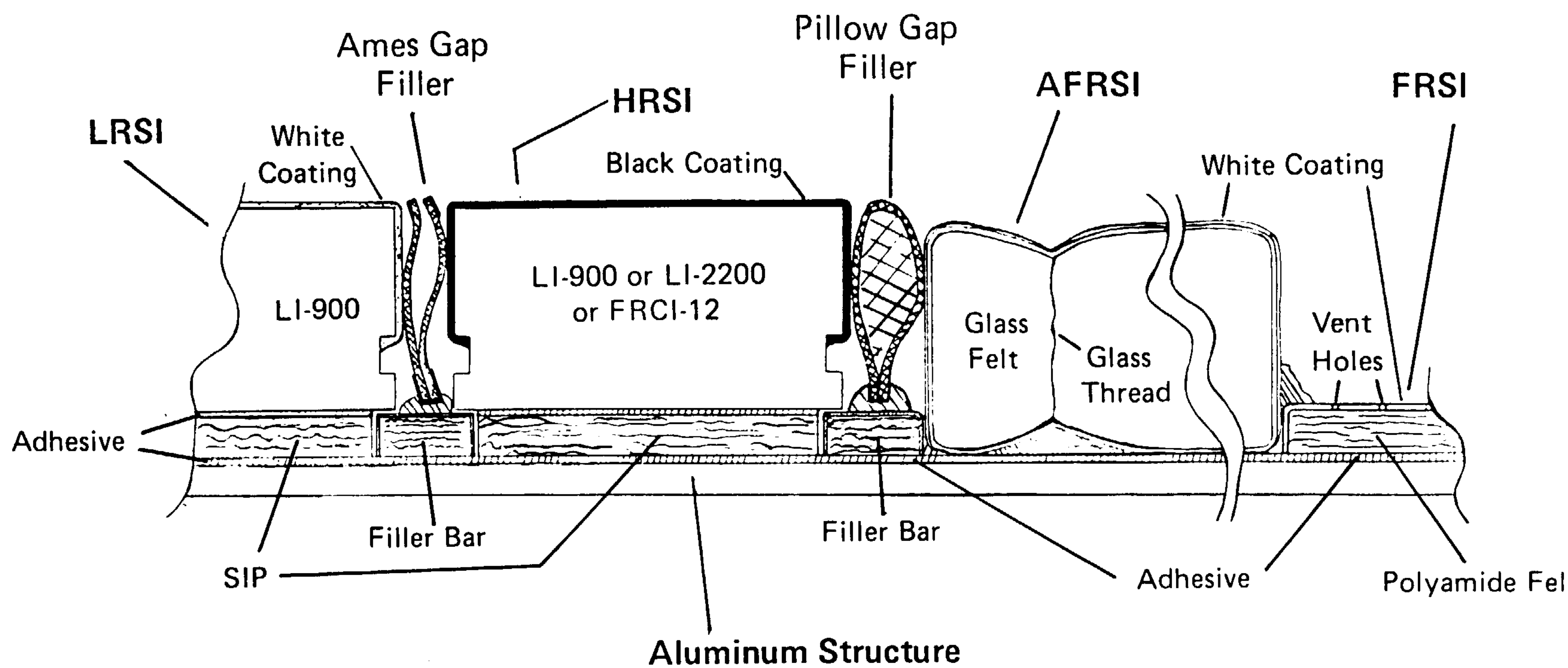
Figure 4. The reusable surface insulation subsystem (RSISS).
Figure 4A gives a view of the rigid RSISS (the tile system).
Figure 4B is the flexible RSISS (the thermal blanket system).

THE TILE SYSTEM

Suggested references on the tile system are:

Cooper, P.A., and P.F. Holloway. "The Shuttle Tile Story," Astronautics and Aeronautics, January 1981, pp. 24-36.

Leger, L.J. "Thermal Insulation Attaching Means," U.S. Patent No. 4,124,732.



C

Figure 4C. Detailed crossection of the RSISS.

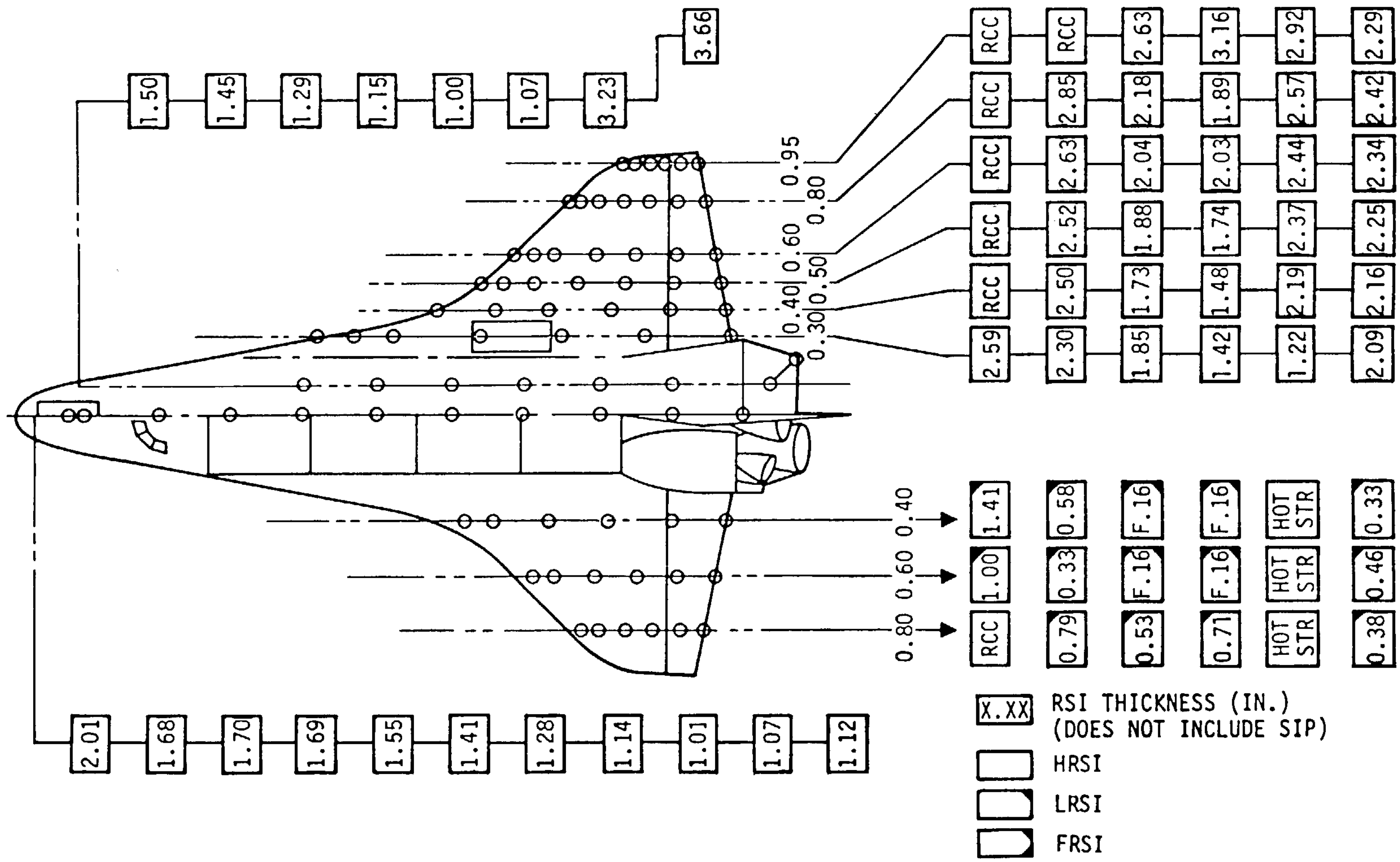


Figure 4D. RSI thickness of initial TPS on Orbiter Columbia.

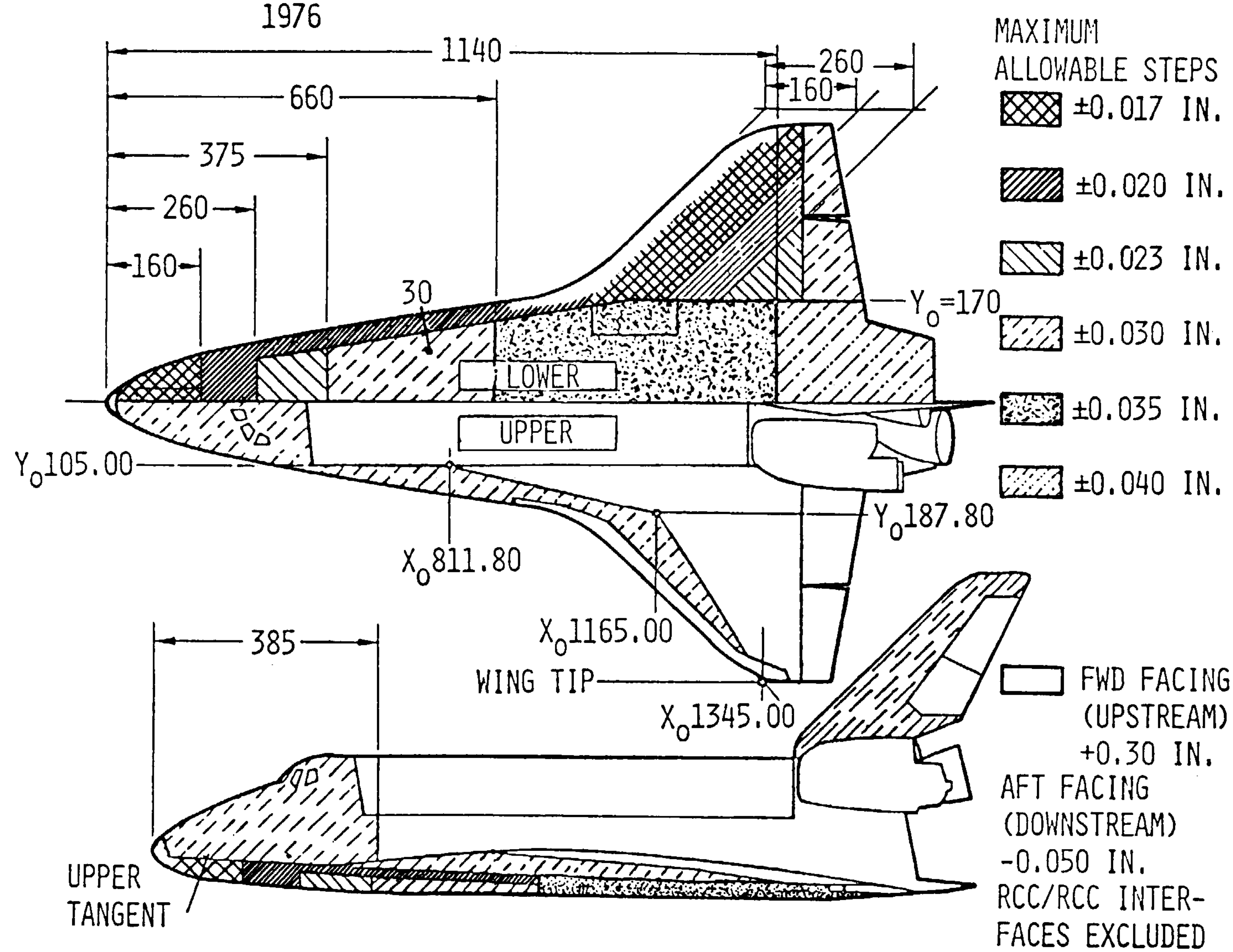


Figure 4E. RSI mold line step criteria of initial TPS on Orbiter Columbia.

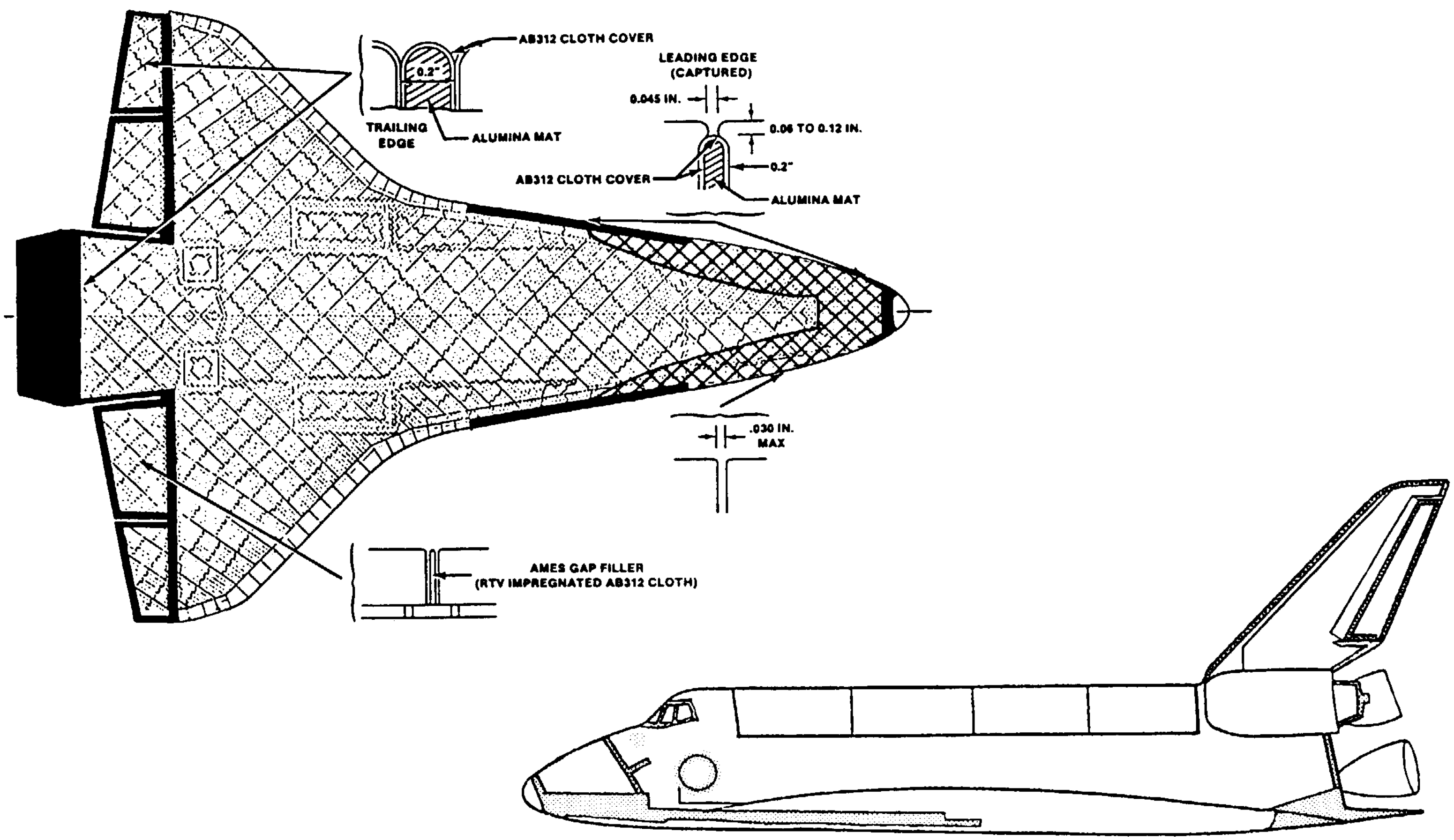


Figure 4F. Locations of gap fillers in high pressure gradient areas of the Orbiter.

FRSI

Suggested references on FRSI are:

Dotts, R.L., B.J. Maraia, J.A. Smith, R.L. Spiker, and G. Strouhal. "Coated-Felt Thermal Insulation," NASA Tech Briefs. Vol. 3, No. 4 (1978) p. 535.

Jones, G.B. "Edge Sealing High Temperature Felt Insulation," (MSC-16382), 1976.

Dotts, R.L., R.J. Maraia, J.A. Smith, and G. Strouhal. "Thermal Insulation Protection Means," U.S. Patent No. 4,151,800.

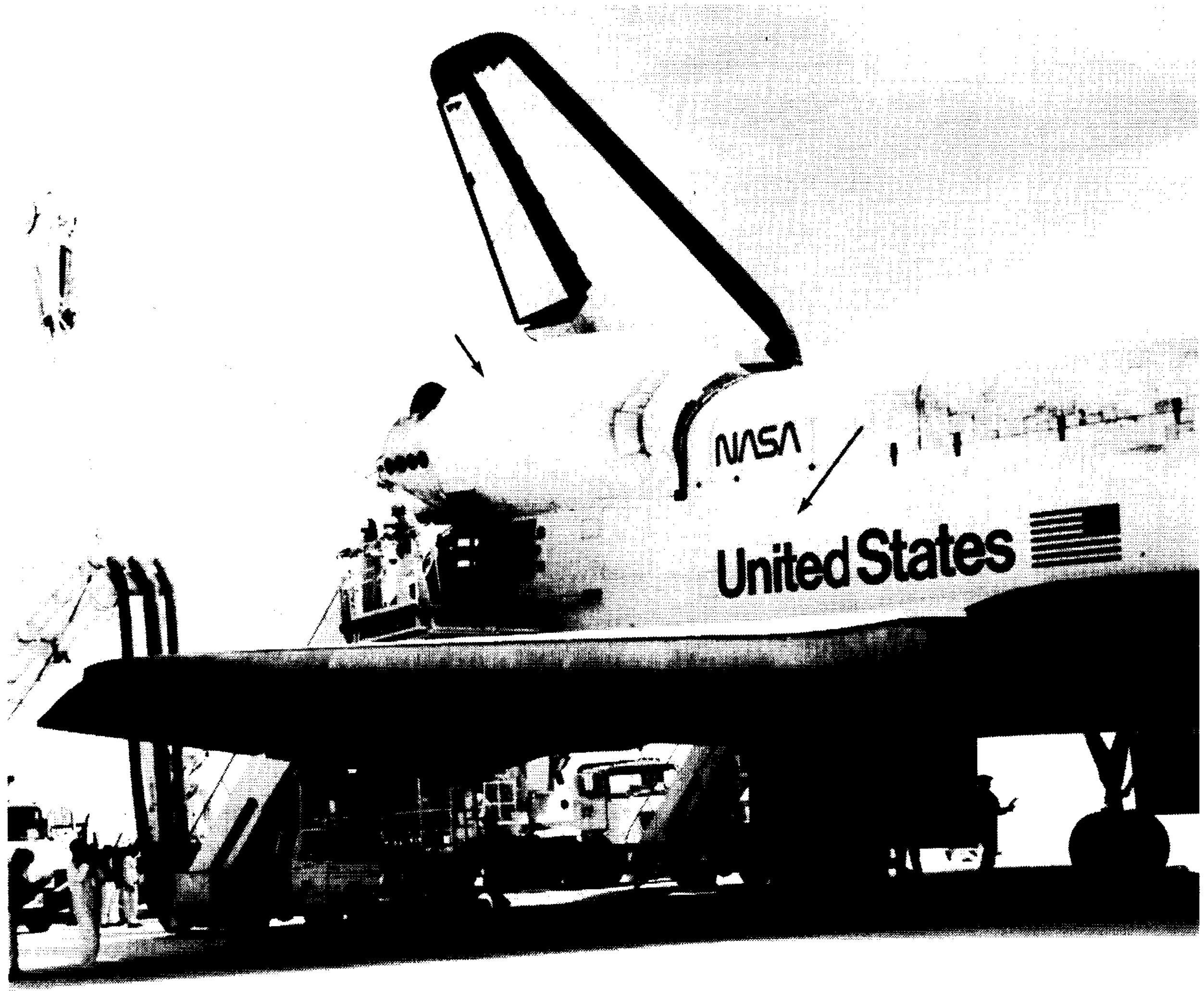


Figure 5. The area of the Orbiter thermally protected mainly with felt reusable surface insulation (FRSI). This NASA-JSC patented polymeric insulation exhibits the best ratio of thermal conductivity to density of all TPS materials.

Suggested references on ceramic fiber flexible RSI are:

AFRSI

Goldstein, H., D. Leiser, P.M. Sawko, H.K. Larson, C. Estrella, M. Smith, and F.J. Pitoniak, "Insulation Blankets for High-Temperature Use," NASA Tech Briefs, Vol. 9, No. 4, Winter 1985, pp. 107-108.

Wang, D.S. "Silicone-Rubber Stitching Seal," NASA Tech Briefs, Vol. 9, No. 2, Summer 1985, p. 154.

Goldstein, H.E., "Fibrous Ceramic Insulation," NASA/AIAA Advance Material Technology Conference. Nov. 16-17, 1982.

Mui, D. "Abrasion-Resistant Coating for Flexible Insulation," NASA Tech Briefs, September/October, Vol. 10, No. 5, 1986, p. 80.

Tailorable Advanced Blanket Insulation

(TABI)

Sawko, P.M. "Tailorable Advanced Blanket Insulation (TABI), NASA Tech Briefs, Vol. 11, No. 7, September 1987, p. 45.

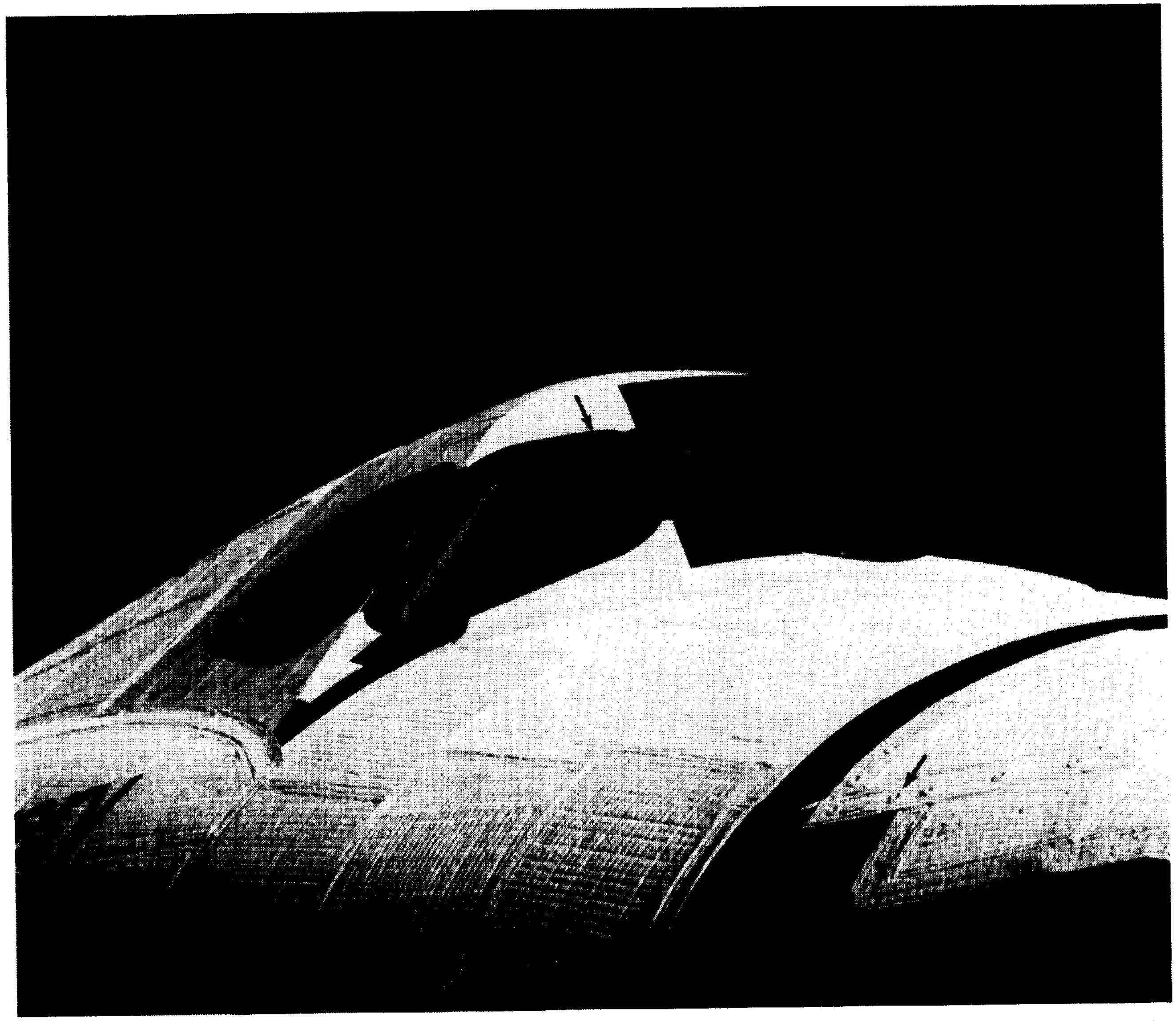


Figure 6. Advanced flexible reusable surface insulation (AFRSI) is a NASA-ARC developed quilt-like thermal insulation blanket (large arrow) with similar thermal characteristics to those of low temperature reusable surface insulation (LRSI). Developed later in the Shuttle program, AFRSI blankets have replaced most of LRSI tiles mainly because it is much easier to install and exhibits a greater resistance to impact damage.

As with many NASA inventions, a description of one of the gap fillers has been given in the publication entitled "NASA Tech Briefs". Gap fillers are a very important element of the RSISS.

High-Temperature Filler for Tile Gaps

Procedure using ceramic fabric can be used in kilns and furnaces.

NASA Tech Briefs, Winter 1982

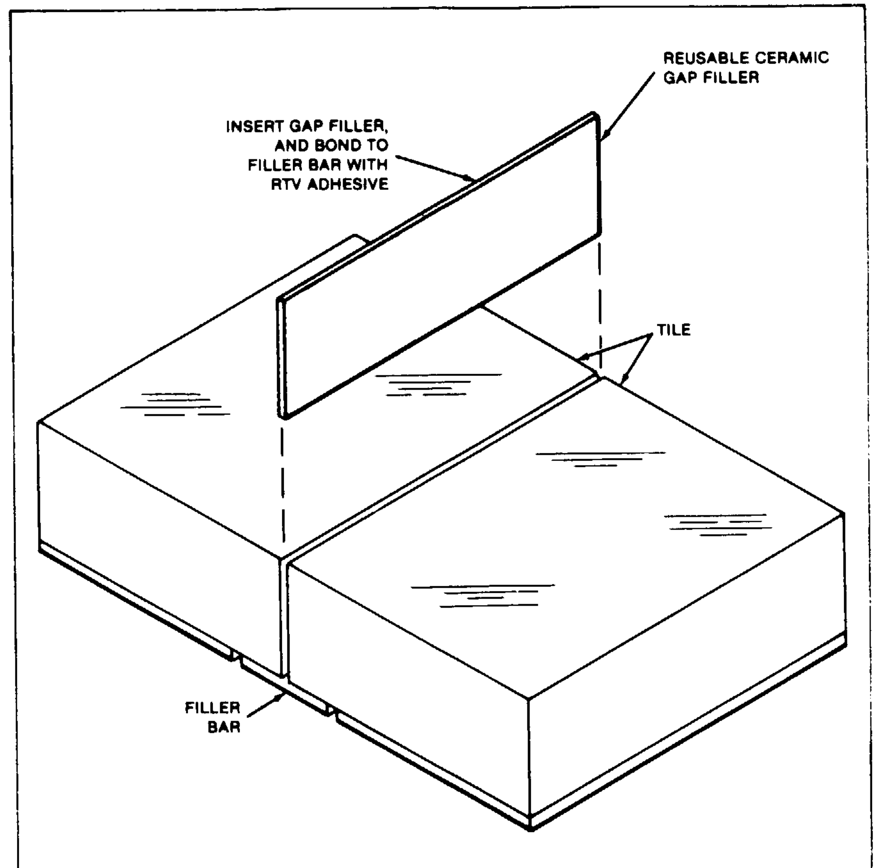
Lyndon B. Johnson Space Center, Houston, Texas

Gaps between ceramic tiles can be filled with ceramic-coated fabric that withstands temperatures as high as 2,400° F (1,300° C). The caulking procedure with this material supplements existing gap fillers between surface insulation tiles on the Space Shuttle; it saves time by permitting the repair of fillings already in place, without the need to remove, rework, or replace them. High-temperature liners in kilns, furnaces, and other applications of heat-resistant insulation can also be caulked by this technique.

The ceramic fabric used for the gap-filler shim has a thickness of either 0.012 or 0.040 inch (0.3 or 1.0 mm). It is mounted in a metal frame and sprayed with colloidal silica, then sprayed with ceramic coating, and dried. It is then cut to size for the gap and inserted in place. Finally, it is bonded in place with a room-temperature-vulcanizing (RTV) adhesive, as indicated in the figure.

These gap fillers maintain the required pressure against the sidewalls of the tiles, are flexible, and can withstand airloads and high-temperature exposure for repeated missions.

This work was done by Jack W. Holt and David S. Wang of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 49 on the TSP Request Card.*
MSC-20137



The **High-Temperature Gap Filler** is reusable. It is made of fabric coated with a ceramic slurry and bonded in place with room-temperature-vulcanized adhesive.

THE LEADING EDGE STRUCTURAL SUBSYSTEM
(LESS)

The LESS is composed of the RCC surfaces of the nose cap, wings leading edges, and the arrowhead (the forward attachment point of the external tank), plus the thermal insulation, thermal barriers, seals and attaching hardware for the RCC surfaces. The RCC surfaces are provided by the Vought Corporation, a unit of LTV Missiles and Defense Corporation. The remainder of the LESS is the responsibility of Rockwell International Corporation.

Recommended references describing LESS and its functions include:

1. Curry, D. M., J. W. Latchem, and G. B. Whisenhunt. Space Shuttle Orbiter Leading Edge Structural Subsystem Development. Prepared for AIAA 21st Aerospace Sciences Meeting, AIAA Paper No. 83-0483. Reno, Nevada. January 10-13, 1983.

2. Curry, D. M., D. W. Johnson, and R. E. Kelly. "Space Shuttle Orbiter -- Leading Edge Structural Design/Analysis and Material Allowables." Submitted for publication in 1986.

3. Curry, D. M., D. W. Johnson, and R. E. Kelly. Space Shuttle Orbiter – Leading Edge Flight Performance Compared to Design Goals. Shuttle Performance: Lessons Learned. NASA CP 2283 (October 1983).

4. Curry, D. M., H. C. Scott, and C. N. Webster. Material Characteristics of Space Shuttle Reinforced Carbon-Carbon. Proceedings of the 24th National SAMPE Symposium. Vol. 24, Book 2. 1979.

Another RCC surface known as the chin panel is being retrofitted to the original tile design. This panel fits the area between the lower section of the RCC nose cap seals and in front of the nose landing gear doors. This replacement is due to the large movement from original positions experienced by tiles in this area after each flight.

Leading Edge Structural Subsystem (LESS)

- Nose cap
- Wing leading edge
- External tank attaching point (the arrowhead)

LESS
- Reinforced Carbon-Carbon Composite (RCC)
- Thermal insulation
- Thermal barriers and seals
- RCC attaching hardware

NASA National Aeronautics and Space Administration

S81-30874

Lyndon B. Johnson Space Center
Houston, Texas 77058

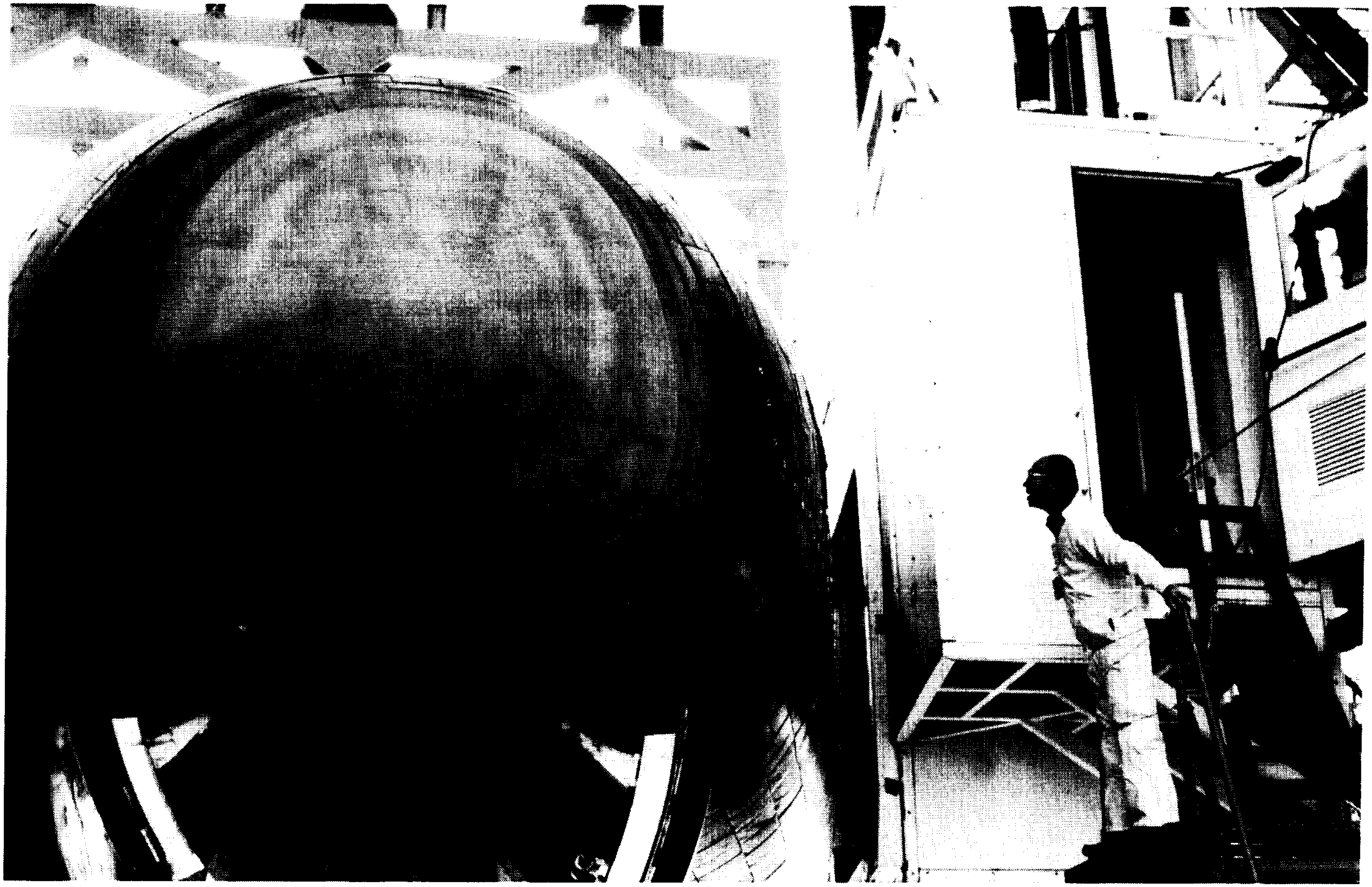


Figure 7. The nose cap section of LESS. Figure 7A presents a view of the reinforced carbon-carbon (RCC) nose cap. The RCC parts experience the most aggressive aerothermal environment encountered by the Orbiter during rentry.

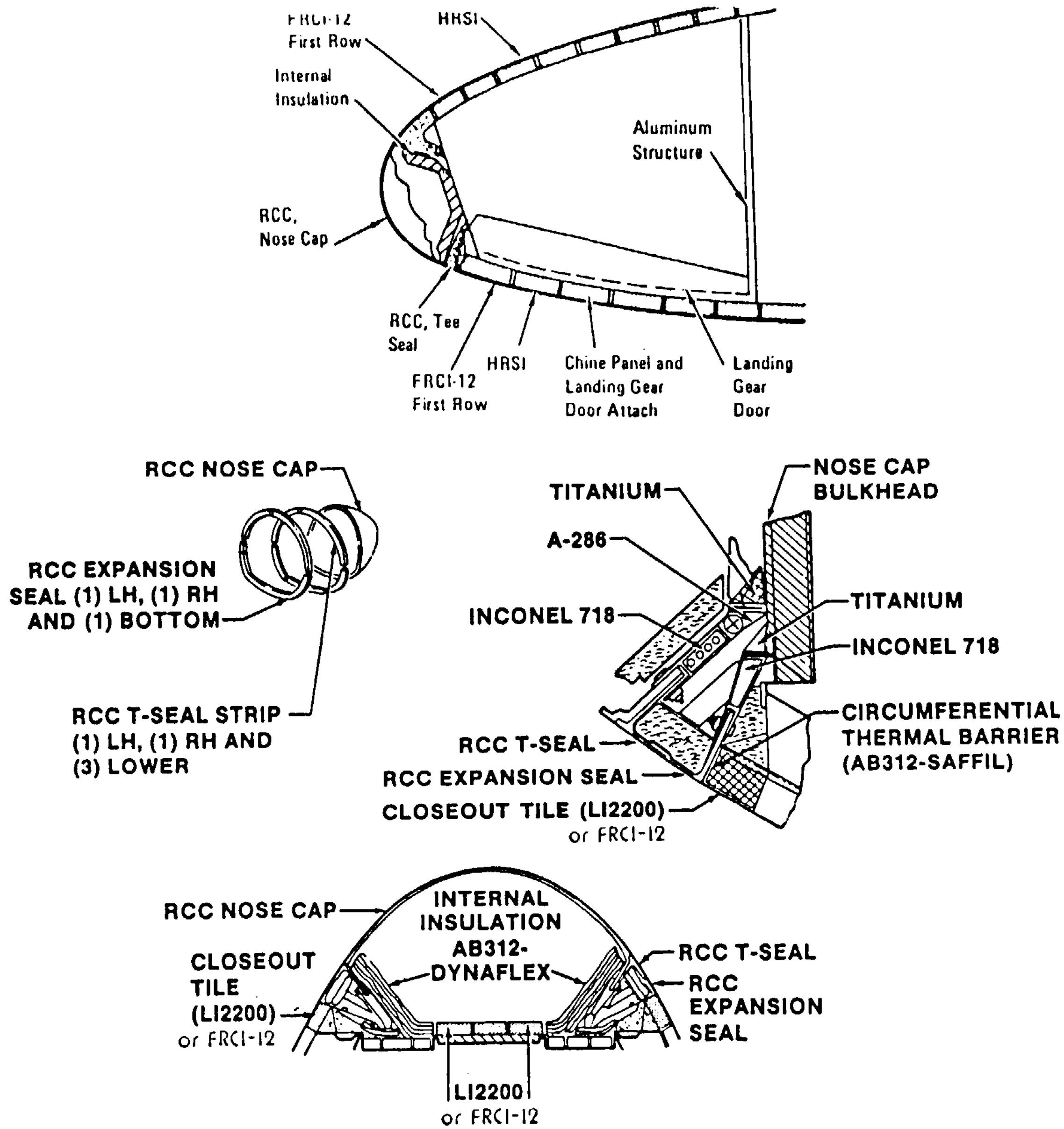


Figure 7B. Detailed view of the nose cap section of LESS.

Figure 8. The wing leading edge of LESS. There are 22 RCC panels and 22 RCC T-seals (arrow in Figure 8A) per wing in each Orbiter. Oxidation inhibitors and coatings used in RCC parts have performed as predicted.

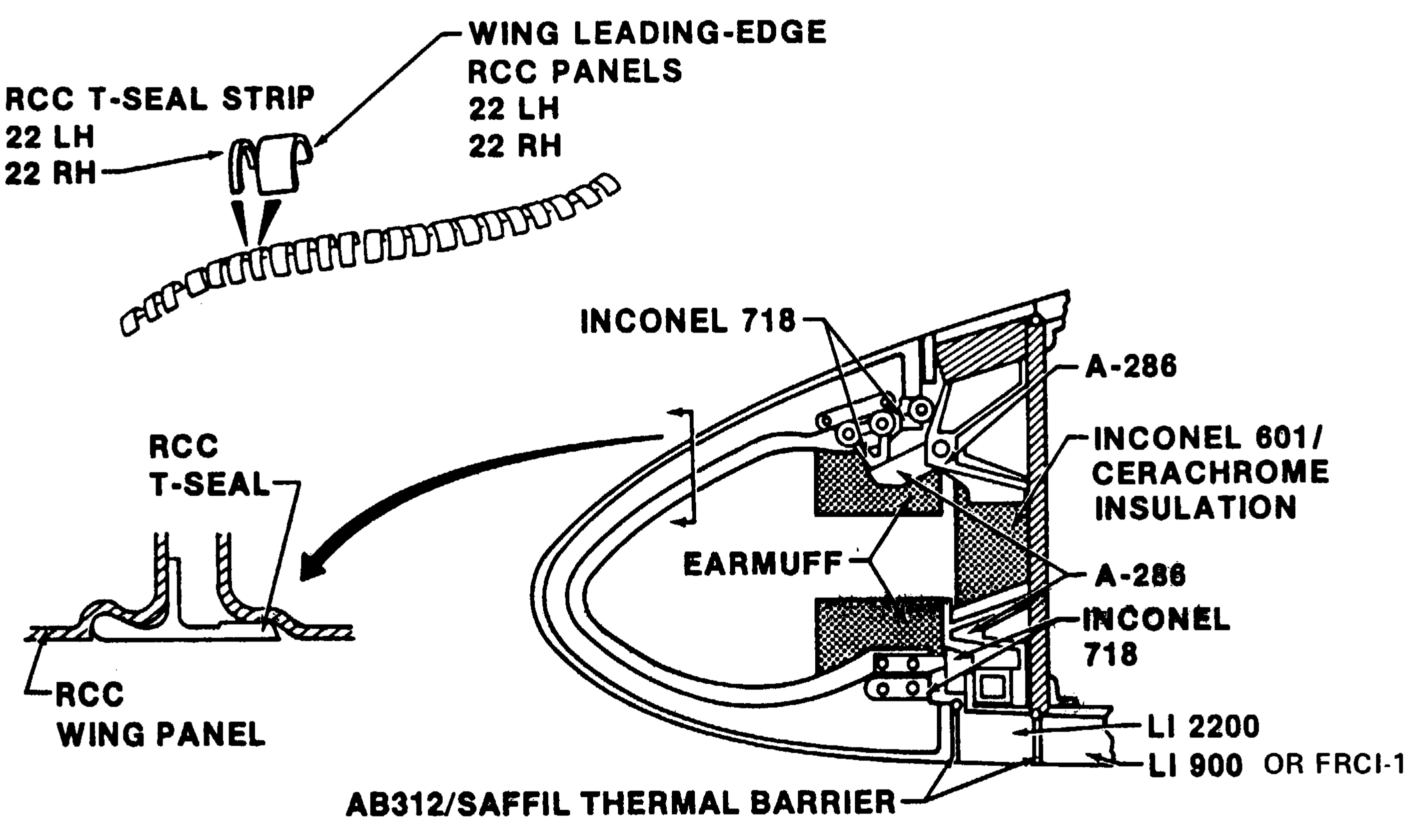


Figure 8B. Detailed view of the wing leading edge section of LESS.

Surface Seal for Carbon Parts

An antioxidation coating extends part life at high temperature.

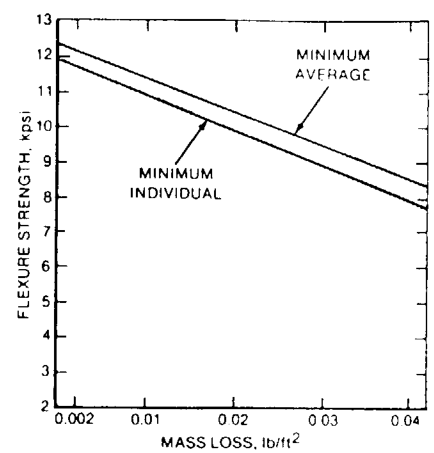
Lyndon B. Johnson Space Center, Houston, Texas

Surface pores in parts made of graphite or reinforced-carbon/carbon materials are sealed by a silicon carbide-based coating. The coating inhibits subsurface oxidation and thus lengthens the part life.

The starting material for the coating is graphite felt, which is converted to silicon carbide felt by processing it according to a prescribed time/temperature schedule. The converted felt is pulverized in a ball mill, and the resulting powder is mixed with an equal weight of black silicon carbide powder. Finally, the powder mixture is combined with an equal weight of an adhesive to form a paste.

The part to be sealed is prepared by rubbing with fine abrasive paper and wiping with an alcohol-moistened cheesecloth. The sealant is applied with a brush, spatula, or blade.

After the part has dried in air, the coating is ready to be cured. The part is placed in a furnace, where it is subjected to temperatures up to 600° F (315° C) for several hours. A second coating is applied to the part after it has cooled, and the curing procedure is repeated.



The **Acceptability of an Antioxidation Seal** is determined by flexural strength tests on specimen bars that have been heated at high temperatures for prolonged periods. Strengths of specimens should exceed the minimum values defined by the straight lines. The lines represent data for coatings 0.029 to 0.031 inch (0.74 to 0.79 millimeter) in thickness; a correction factor is applied for thinner or thicker coatings.

The coated parts are expected to pass difficult tests of their oxidation resistance. Specimens are exposed to high temperatures for prolonged periods, and their before-and-after weights are compared. The maximum mass loss per unit of surface area after exposure at 1,000° F (540° C) for 18 hours must not exceed 0.05 lb/ft2 (0.24 kg/m2). After exposure at 2,300° F (1,260° C) for 18 hours, mass loss must not exceed 0.03 lb/ft2 (0.15 kg/m2). After these heat treatments, the parts are tested for flexual strength (see figure).

This work was done by David M. Shuford and John P. Spruiell of Vought Corp. for **Johnson Space Center.** *For further information, including a detailed process specification, Circle 39 on the TSP Request Card.*
MSC-18898

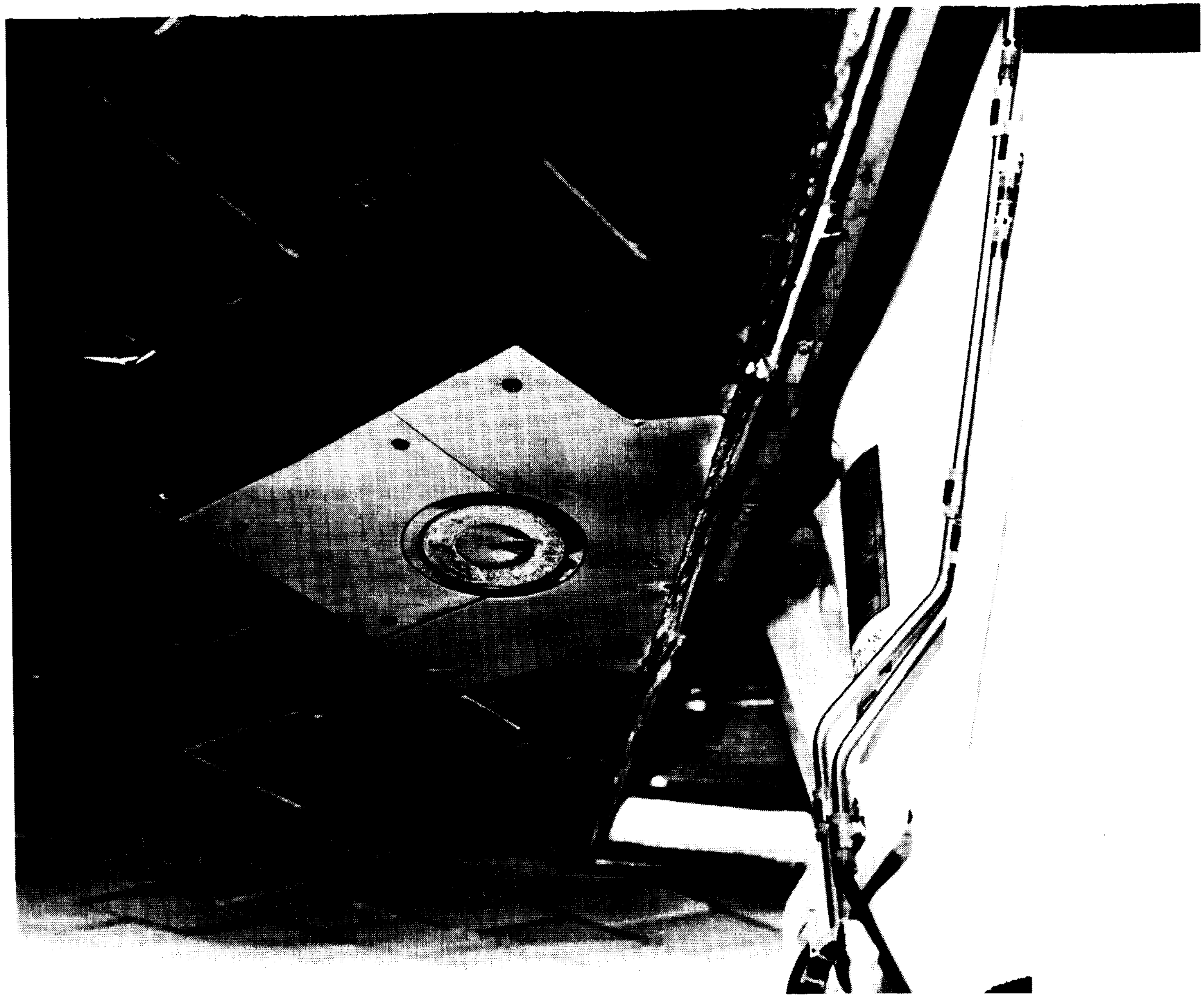


Figure 9. Arrow head. This RCC panel, denoted as the "arrow head," was a retrofit to the original high temperature reusable insulation (HRSI) tile design which failed the qualification test of the explosive separation of the external tank.

PENETRATIONS
(PSS)

Penetrations are the many areas that are indispensible for the operation and servicing of the Orbiter, such as service access panels, landing gear doors and windows. The PSS include the unique thermal insulation concepts and seals required to protect these penetration areas. A list of these penetrations and detailed view of some of them are given in the following pages.

A recommmended reference list for the PSS is:

1. Hughes, J. T., and D. U. McBride. TPS is More Than Tiles. AIAA Paper 83-1486. Prepared for AIAA 18th Thermophysics Conference, Montreal, Canada. June 2, 1983.

2. McBride, D.U. Subsurface Flow Considerations in Thermal Protection Design. Prepared for AIAA/ASME 4th Annual Thermophysics and Heat Transfer Conference. Boston, Massachusetts. June 2-4, 1986.

3. Chang, J., A. M. Khilnani, and D. U. McBride. Thermal Analysis of Shuttle Orbiter Wing/Elevon Seals. Prepared for AIAA 19th Thermophysics Conference No. 84-1760. Snowmass, Colorado. June 1984.

4. Neuenschwander, W. E., D. U. McBride, and G. A. Armour. "Shuttle TPS Performance and Analysis Methodology." Shuttle Performance: Lessons Learned. NASA CP 2283. October 1983.

Penetration Subsystem (PSS)

- Canopy windshields
- Rudder-speed break conic seal
- The pay load bay door hinges
- Landing gear doors

PSS
- RSI
- Thermal barriers
- Seals

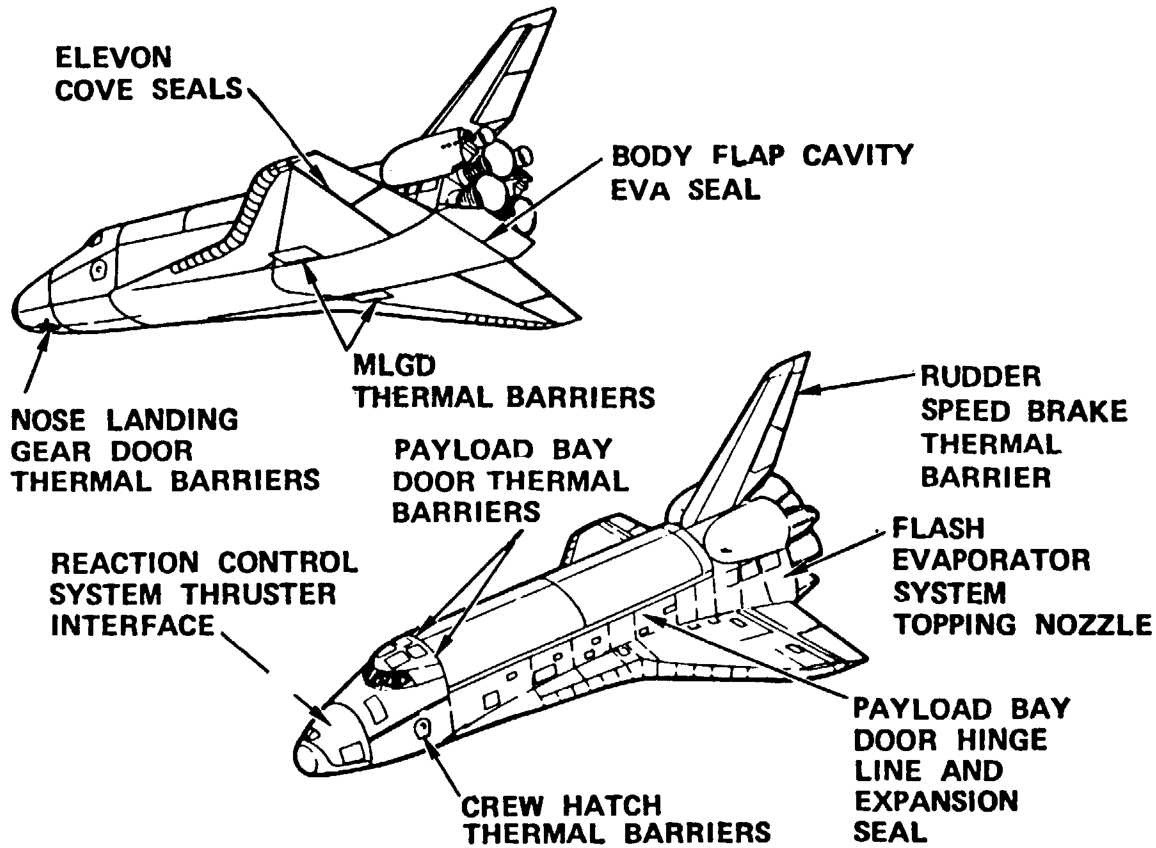


Figure 10. Typical penetrations of the penetration subsystem (PSS).

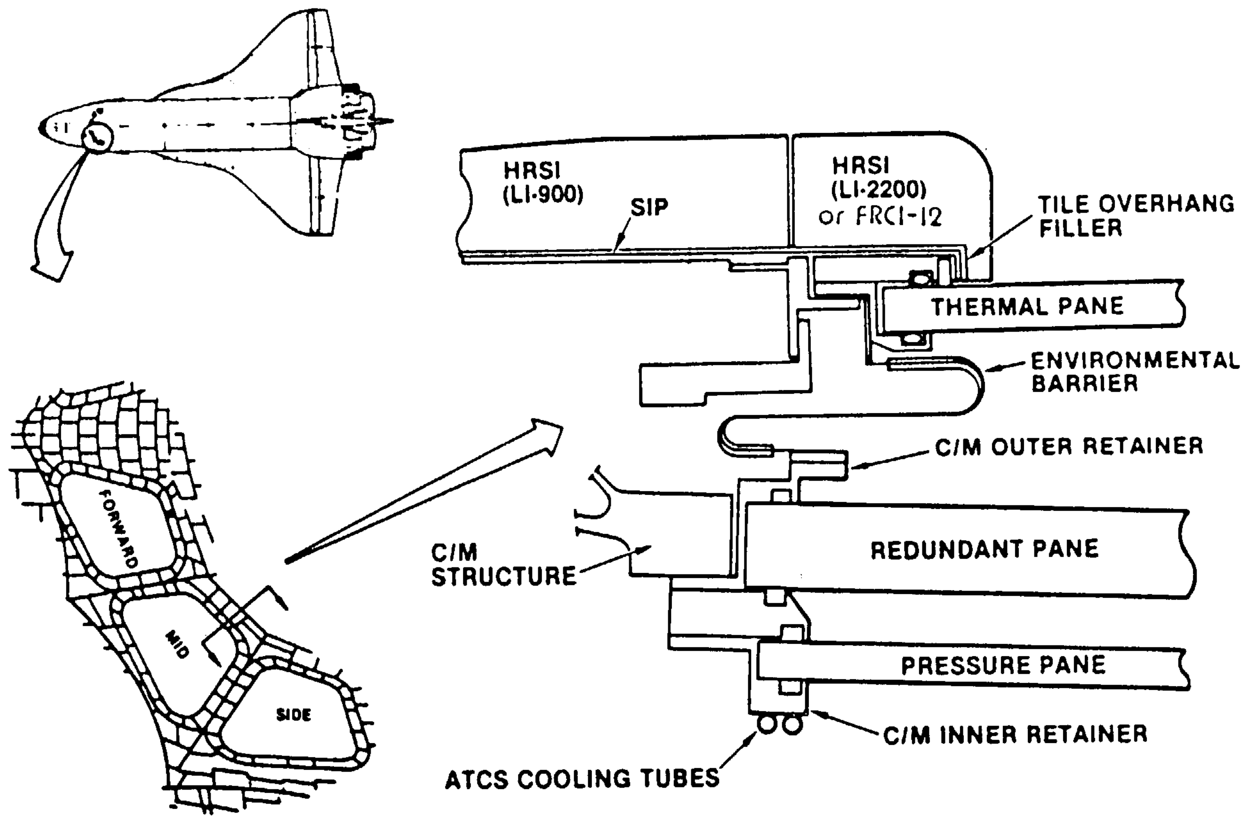


Figure 10B. The canopy windshields of the PSS.

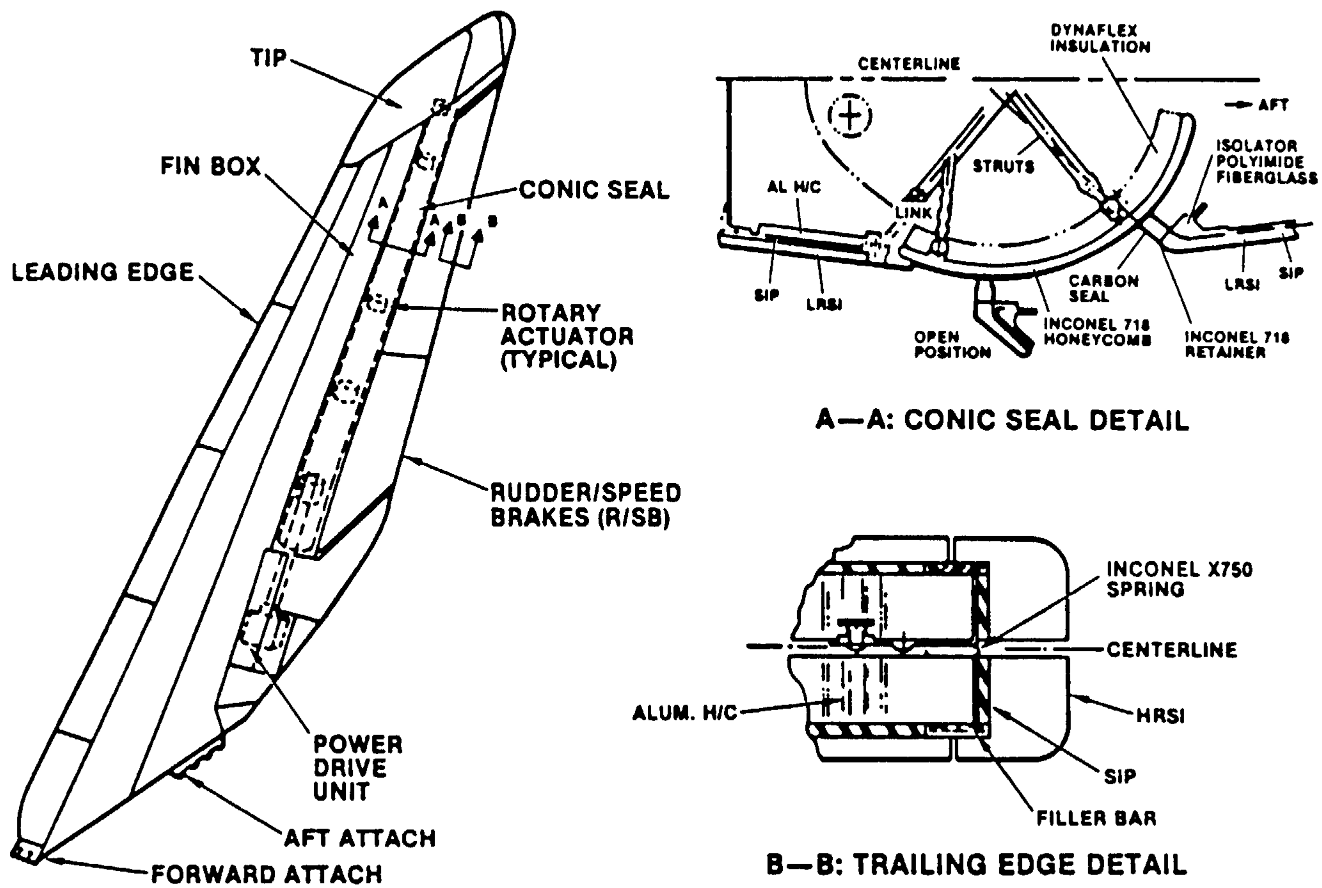


Figure 10C. The rudder-speed break conic seal and structure of the PSS.

Materials Design and Development

COMMON CHARACTERISTICS OF TPS MATERIALS

- All materials have been made from high purity ingredients.
- All materials are fibrous and anisotropic in nature.
- All materials have a coating of some type.
- All materials have been designed to breathe to equalize pressure.
- All materials exhibit low thermal expansion, high specific strength, and excellent resistance to thermal shock.

TILE MATERIAL DESIGN AND DEVELOPMENT

(RSISS)

Figure 11 presents a summary of the evolution of the tile material architecture used in the rigid RSISS. The initial tile material developed for NASA by LMSC, designated as Lockheed Insulation (LI-900), exhibited little bonding at fiber junctions, as illustrated in Figure 11A. Any bonding present in these sites was due mainly to a small amount of colloidal silica that was added for this purpose during manufacture. The high purity, amorphous silica fiber known as the Q-fiber developed by LMSC and the Manville Corporation led to the selection of the LI-900 as the primary insulation material. The Q-fiber is also the main fiber ingredient in all the successive generations of tile materials.

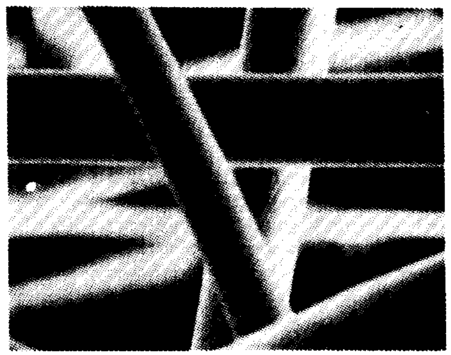
The second generation of tile material, designated as fibrous refractory composite insulation (FRCI), had greater strength and temperature stability than the LI-900. The greater thermal strength of this NASA material was due to the formation of borosilicate bonds at fiber junctions, achieved by the fluxing of boron oxide from the aluminoborosilicate (Nextel 312 - 3M Corporation) fiber during firing, see Figure 11B. These bonds are much greater in number and stronger than the LI-900 silicate bonds.

Finally, a more current generation of tile materials, designated as high temperature performance (HTP) insulation, and the alumina enhanced thermal barrier (AETB) exhibit even higher temperature capabilities than obtained with previous materials. This increased thermal capability is due to the use of a more refractory alumina fiber in both materials which is also smaller than the Nextel 312 fiber. On the HTP file material, boron nitride powder is used to achieve the borosilicate bond at fiber junctions, Figure 11C. AETB tile material uses a smaller Nextel 312 fiber to achieve a similar result. The use of the smaller alumina fibers also increases the number of fiber junctions compared to those obtained in FRCI which is made with the larger, less stable at high temperatures Nextel 312 fibers. As illustrated, a description of the HTP material has already appeared in the Winter 1985 issue of NASA Tech Briefs. A description of AETB material will appear soon in the NASA Tech Briefs publication.

In conclusion, the evolution of the tile material has provided the scientific and industrial communities with a novel strengthening mechanism for fibrous composites. This strengthening mechanism can be used to increase the strength of most fibrous composite materials (ceramic, metallic, or polymeric) in which one of the constituents behaves as a bonding agent to create the required bonding at fiber junctions.

Material Design and Development
(RSISS)

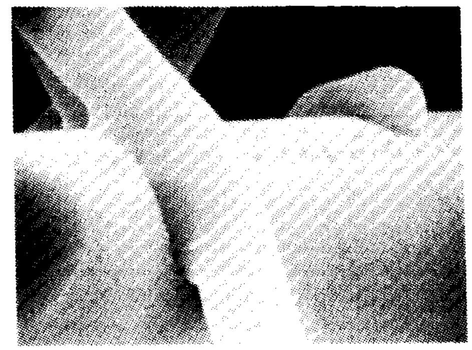
LI-900
1970 - 1976



- Silica fibers
- Colloidal silica (binder)

A

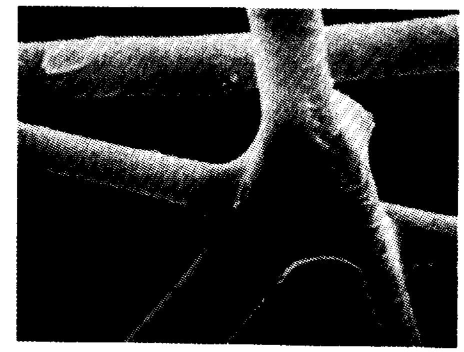
FRCI
1974 - 1978



- Silica fibers
- Alumino-boro-silicate fibers

B

HTP
1980 - Present



- Silica fibers
- Alumina fibers
- Boron nitride powder (binder)

C

Figure 11. Evolution of the tile materials.

High-Strength, Low-Shrinkage Ceramic Tiles

Flexural strength and other properties are improved by additives.

NASA Tech Briefs, Winter 1985

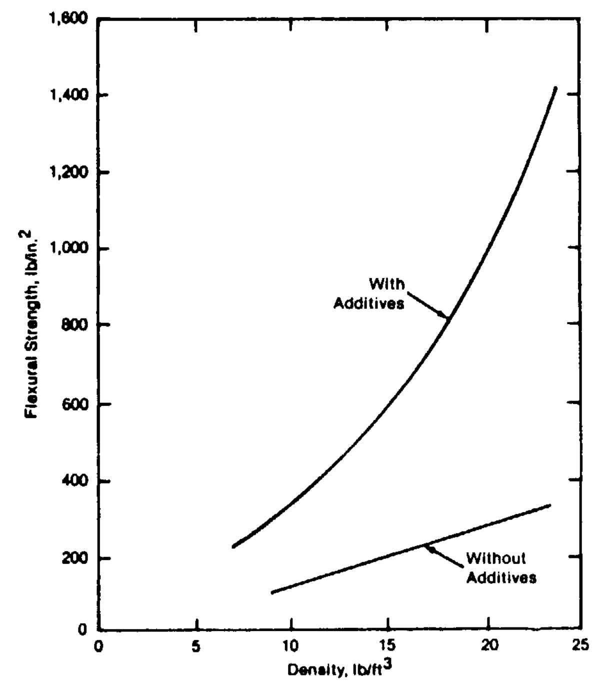
Lyndon B. Johnson Space Center, Houston, Texas

The addition of refractory fibers and whiskers to insulating tiles composed primarily of fibrous silica, such as those used on the skin of the Space Shuttle orbiter, greatly improves their properties. For example, modifying the formulation with 15 percent alumina fibers and 2.85 percent boron nitride multiplies the flexural strength by a factor of 2 or 3 while maintaining the same low density as that of the all-silica tiles (see figure). Added benefits are an increase in the modulus of elasticity and less deformation and shrinkage at temperatures above 2,300° F (1,260° C).

Although originally developed to provide the Space Shuttle with tiles of greater strength and dimensional stability, the new composition is also suitable for lightweight, thermally-stable mirror blanks and as furnace and kiln insulation. The improved tiles are made with current tile-fabrication processes.

The tile ingredients are blended with deionized water to form a thick slurry and cast to shape. After drying, the tiles are heated. Boron nitride is especially useful because it resists oxidation up to about 1,830° F (1,000° C), then slowly dissociates to boron trioxide and nitrogen gas during subsequent heating to higher temperatures.

The boron addition promotes eutectic sintering and provides a strong borosilicate bond at fiber junctions. At the tile-firing temperature of 2,350° F (1,290° C), the slowly-released boron trioxide has time to react as it diffuses throughout the tile to all the silica fibers, producing uniformly bonded intersections. The boron trioxide content also inhibits crystallization of the silica during exposure to high temperatures. Crystallization — which would make tiles brittle — does not exceed 5 percent when boron-containing tiles are heated to 2,300° F (1,260° C) for 15 hours.



For a Given Density, tiles containing silicon carbide and boron additives are stronger in flexure than are tiles made from silica alone. In addition, the tiles with additives are nearly immune to heat distortion, whereas pure-silica tiles shrink and become severely distorted.

This work was done by William H. Wheeler and John F. Creedon of Lockheed Missiles & Space Co., Inc., for **Johnson Space Center.** *For further information, Circle 64 on the TSP Request Card.*

MSC-20654

Light, Strong Insulating Tiles

Experiments with composition yield a desirable combination of properties.

Lyndon B. Johnson Space Center, Houston, Texas

Improved lightweight insulating silica/aluminum borosilicate/silicon carbide tiles combine increased tensile strength with low thermal conductivity. The new tiles replace older all-silica tiles of approximately equal thermal conductivity and higher mass density.

As originally formulated, the new tiles were composed of silica and aluminum borosilicate fiber with 2 percent silicon carbide particles (as an emissivity-controlling agent) in 320-grit (about 41-μm) size. These tiles had lower density than their all-silica predecessors — 8 lb/ft^3 vs. 9 lb/ft^3 (128 kg/m^3 vs. 144 kg/m^3) as well as higher tensile strength. However, insulating properties were not adequate, the thermal conductivity being much higher than that of the all-silica tiles.

Material	SiC Grit Size/ Concentration (Percent by Weight)	Dry (Unfired) Density lb/ft^3 (kg/m^3)	Silica/Aluminum Borosilicate Fiber Ratio	Final Density lb/ft^3 (kg/m^3)	Effective Thermal Conductivity Relative to All-Silica
Original Composition	320/2	5.0 to 6.5 (80 to 104)	78/22	8.0 to 9.0 (128 to 144)	Much Higher
Improved Composition	600/3	7.0 (112)	85/15	8.5 (136)	Essentially Equivalent

Changes in Composition substantially improve the heat-insulating properties of silica-based refractory tile. The silicon carbide particles act as high-emissivity radiation scatterers in the tile material.

After some experimentation, a new formulation was found to reduce the thermal conductivity: the concentration of aluminum borosilicate fiber was reduced, the dry (unfired) density of the tile billets was increased, the proportion of silicon carbide was increased to 3 percent, and the silicon carbide particle size was reduced to 600 grit (14 μm), as shown in the table. Although the density increased slightly with this formulation, it was still less than that of the all-silica tiles, and thermal conductivity was essentially the same as that of the all-silica tiles.

Apparently the smaller silicon carbide particles reduce the mean free path for thermal radiation, thereby increasing scattering in the material. Increasing the dry density and reducing the aluminum borosilicate fiber content have a similar effect.

This work was done by E. Cordia and J. Schirle of Lockheed Missiles & Space Co. for **Johnson Space Center**. *MSC-20601*

HRSI COATING DESIGN AND DEVELOPMENT MODEL

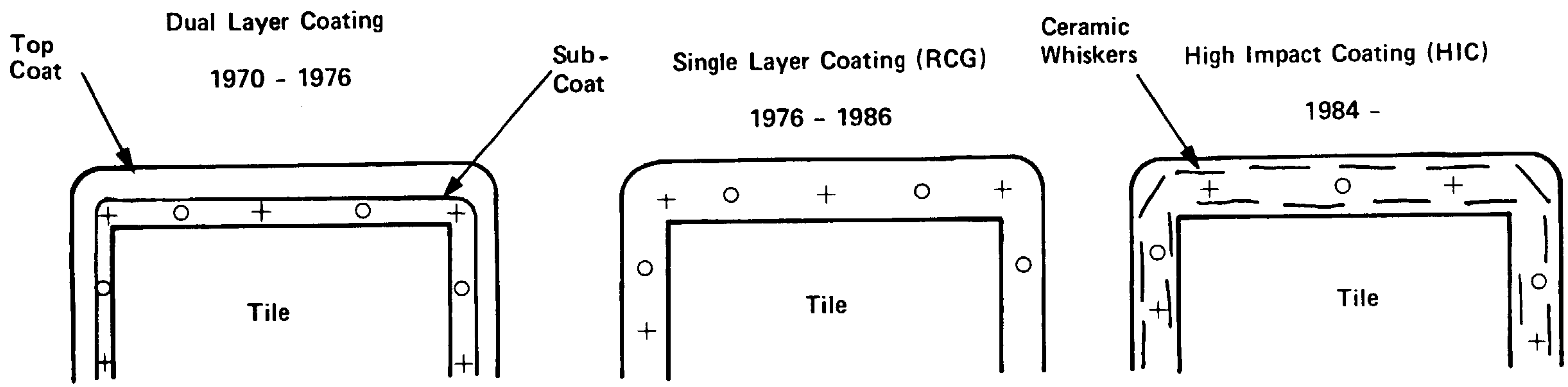
(RSISS)

Figure 12 presents a summary of the evolution of the material used to coat the HRSI tiles. Initially, the coating was made of a dual layer system, Figure 12A. In 1976, the dual coating system was replaced with a single, NASA-Ames Research Center (ARC) patented reaction cured glass (RCG) coating. This coating met all the optical properties requirements, had a lower residual tensile strain, minimized a water imperviousness problem, and was much easier to apply and repair than the dual coating system, Figure 12B. Additionally, RCG coating can be used at a higher temperature (2350 ^{o}F) than its temperature of manufacture (2190 ^{o}F). Both of these coatings, however, exhibited a low tolerance to impact damage, a consequence of the thin cross section requirement for this material.

A new coating system identified as high impact coating (HIC) was proposed in 1984 (MSC-20829) which has shown in laboratory testing to exhibit significantly improved strength properties, Figure 12C. This increased strength is achieved by the addition of refractory fibers or whiskers such as aluminum silicate fibers or silicon carbide and silicon nitride whiskers added to the RCG matrix. The controlled addition of any of these fibers or whiskers has doubled the impact strength while maintaining all the other properties of the unreinforced RCG coating. As shown on the following page, a description of this coating system and possible applications has already appeared in the March/April 1986 issue of NASA Tech Briefs.

Continuing coating development at NASA-ARC has resulted in another new coating called Toughen Unipiece Fibrous Insulation which will appear soon in NASA Tech Briefs. This coating is 20 to 100 times as impact resistant as the RCG coating.

HRSI Coating Design and Development (RSISS)



- Two Coating System
  - Topcoat: Clear, waterproofing layer
  - Subcoat: Porous, optically adjusted layer
- Difficult and Slow to Apply
- Low Waterproofing
- Lowest Impact Strength

A

- All Components in a Single Layer
  - Reaction cured glass (RCG)
- Easier to Apply
- Adequate Waterproofing
- Low Impact Strength

B

- All Components in One Layer
  - RCG
  - Ceramic Whisker Reinforcement
- High Impact Strength
- Adequate Waterproofing

C

Figure 12. Evolution of HRSI tile coatings.

Impact-Resistant Ceramic Coating

Refractory fibers more than double the strength of the coating.

NASA Tech Briefs, March/April 1986

Lyndon B. Johnson Space Center, Houston, Texas

A coating for ceramic insulating tiles has increased impact strength as well as high thermal emittance. The addition of whiskerlike ceramic fibers in controlled amounts to the standard coating for Space Shuttle insulating tiles increases the impact strength to about 2½ times that of the standard coating alone. The new coating can also be used to improve the thermal and mechanical properties of electromagnetic components, mirrors, furnace linings, and ceramic parts for advanced internal-combustion engines.

The whiskers may be aluminum silicate, silicon carbide, or silicon nitride. They should have a high fiber aspect ratio (the ratio of length to diameter) for greatest effectiveness.

In comparative tests, each of the three types of whiskers was blended into a slurry of the standard coating material in proportions of 5, 10, 15, and 20 percent by weight. The mixtures were applied to tiles and baked at 2,250° F (1,232° C) for 90 minutes. The impact strengths of the coated tiles were then measured in an impact-testing apparatus in which the fractures of the coatings were detected acoustically.

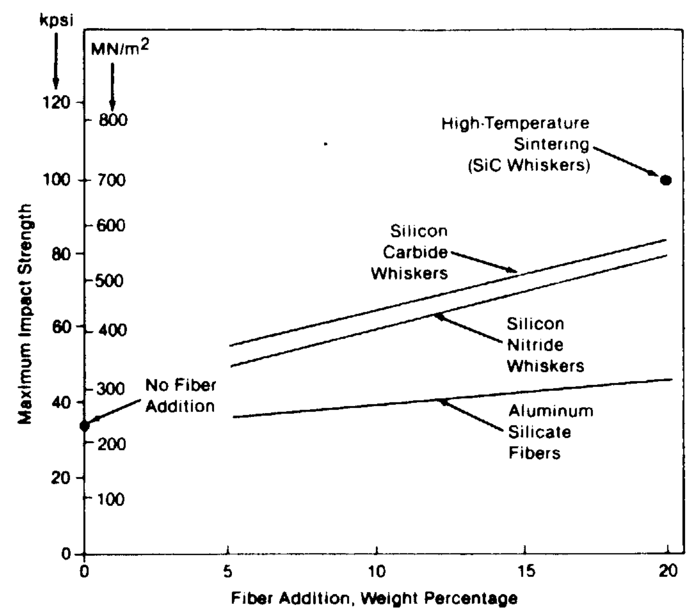
For all whisker materials, the impact strength increased with the amount of the additive (see figure). Silicon carbide whiskers produced the greatest increase in impact strength. The strength of the silicon carbide coating was increased even further by sintering at 2,400° F (1,300° C) for 60 minutes.

The formation of cristobalite, which causes microcracking of the coating, was not a problem, even after long exposures to high temperatures. X-ray diffraction tests of the coatings after 15 hours at 2,300° F (1,260° C) showed that the concentration of cristobalite was within acceptable levels.

The whisker-reinforced coating can be applied and processed by methods that are compatible with existing production equipment. The reinforcement concept is adaptable to other refractory coating and can lend them improved impact resistance without detracting from other properties.

This work was done by William H. Wheeler, John F. Creedon, and Yamato D. Izu of Lockheed Missiles & Space Co., Inc. for **Johnson Space Center.** *For further information, Circle 57 on the TSP Request Card.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Johnson Space Center [see page 29]. Refer to MSC-20829.



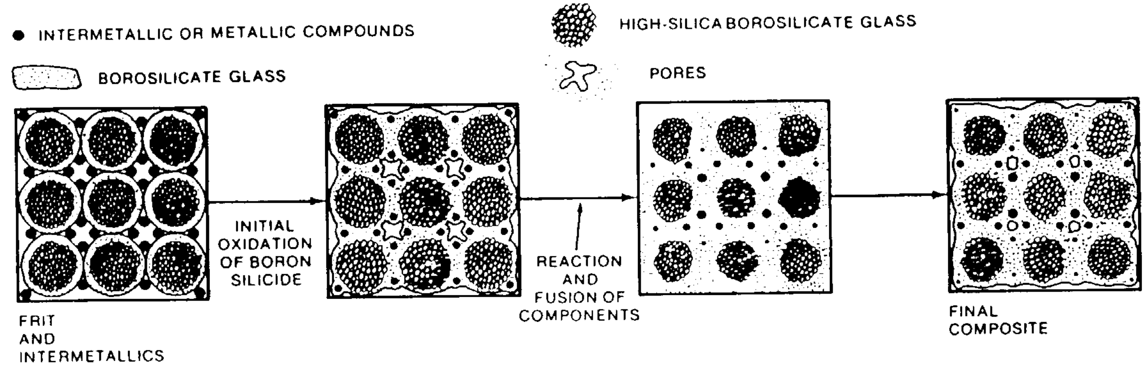
The Impact Strengths of Ceramic Coatings increase with increasing whisker content. Silicon carbide whiskers clearly produce the largest increase, and the improvement grows even more with high-temperature sintering.

High-Temperature Glass and Glass Coatings

Family of reaction-cured glasses resists thermal shock and maintains properties above 1,450° C.

NASA Tech Briefs, Spring 1977

Ames Research Center, Moffett Field, California



The **Composite-Glass Forming Reaction** of the dispersion of glass frit and intermetallic compound is characterized by an initial oxidation of the intermetallic compounds that react with the outer boron-oxide-rich layers of the frits. Upon further heating, the components fuse into a continuous mass. The boron-oxide-rich matrix prevents devitrification of the more-refractory high-silica core in each particle.

A new group of composites, called reaction-cured glass, have outstanding thermal and physical properties that make them excellent materials for high-temperature glassware and tubing or for coatings for porous materials, such as porous insulation tile. They can be used over a temperature range of -150° to 2,700° F (-100° to 1,480° C), have low thermal-expansion coefficients to resist thermal shock, and can be made highly opaque with a high surface emittance, or translucent, or transparent.

The glass is a composite material comprising a borosilicate glass matrix containing regions of high-silica borosilicate glass with intermetallic or metallic compounds dispersed throughout. The intermetallic compounds may be silicon tetraboride or other boron silicides, boron, or a mixture of these.

The glassy composites are prepared by combining about 100 parts of high-silicate borosilicate glass with two to ten parts of boron oxide. These are mixed to form an aqueous dispersion and are dried, dispersed, screened, and fired at 2,000° F (1,090° C) for over an hour after grinding and powdering, the resulting sintered material, or frit, is a two-phase glass with a very-reactive boron-oxide-rich borosilicate layer on the outside and a high-silica borosilicate core in each particle (see figure).

This frit is used to form a glass coating or glass article by reacting it with the appropriate intermetallic compound. Typically, a glass coating is prepared by blending the glass frit and 2.5 percent, by weight, of silicon tetraboride with an ethanol carrier and a methylcellulose prebinder. The mixture is ball milled and sprayed on the surface to be coated with an airbrush or a spray gun [coatings have been applied in thickness of 0.1 to 0.3 lb/ft^2 (0.05 to 0.15 g/cm^2)]. The coated article is dried and glazed at around 2,225° F (1,215° C) for about 1-1/2 hours.

This process forms an opaque, waterproof coating with an emittance of about 0.90 to 0.93 from room temperature to above 2,300° F (1,260° C). Its thermal-expansion coefficient is below 9×10^{-7} cm/cm/°C from room temperature to 1,500° F (820° C). The coating resists devitrification and does not bubble, foam, or change dimensions during glazing.

This work was done by Howard E. Goldstein and Victor E. Katvala of **Ames Research Center** *and Daniel B. Leiser of Stanford University. For further information, Circle 48 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Ames Research Center [see page A8]. Refer to ARC-11051.

GAP FILLERS
(RSISS)

Most of the TPS problems encountered in early Shuttle test flights were related to the unexpected behavior of gaps between tiles, particularly in high pressure, high temperature areas of the Orbiter. These problems were responsible for filler bar burning, loosening, and consequent damaging of tiles. Gaps in these high pressure areas were generally fitted with gap fillers to prevent hot plasma gas from reaching the aluminum structure. Figure 13 shows the types of gap fillers used for this purpose while Figure 4F gives a view of gap filler locations on the Orbiter. A description of one of these gap fillers (No. 1 in Figure 13) has already been given in page 41 of this document. The RTV-coated ceramic fabric, or Ames-type gap filler (No. 4 in Figure 13) has been patented by NASA (U. S. Patent No. 4,308,309). See "the glossary" for a more detailed description of gap fillers.

The design presented in Figure 14, which is an enlarged view of the encircled area in Figure 13, is an outer mold line (OML), wedged-in (captive) gap filler engineered to stop migration problems encountered with the original gap filler around the lower wing leading edge (MSC-18966). This gap filler exhibits greater edge stability due to the incorporation of two different size ceramic (Nextel) tubes (0.25 and 0.50 in. in diameter, respectively) filled with a dense alumina mat (0.70 to 0.75 kg/m^2). The outside is partially enclosed with a metal foil (INCONEL 601), 0.001 in. thick to provide additional stiffness and to ease its installation.

Gap fillers are also used in some areas between flexible and rigid RSI. A special purpose gap filler made of ceramic sleeving known as a Cigarette Gap Filler because it resembles a cigarette shape, is used to fill gaps between corners of AFRSI blankets.

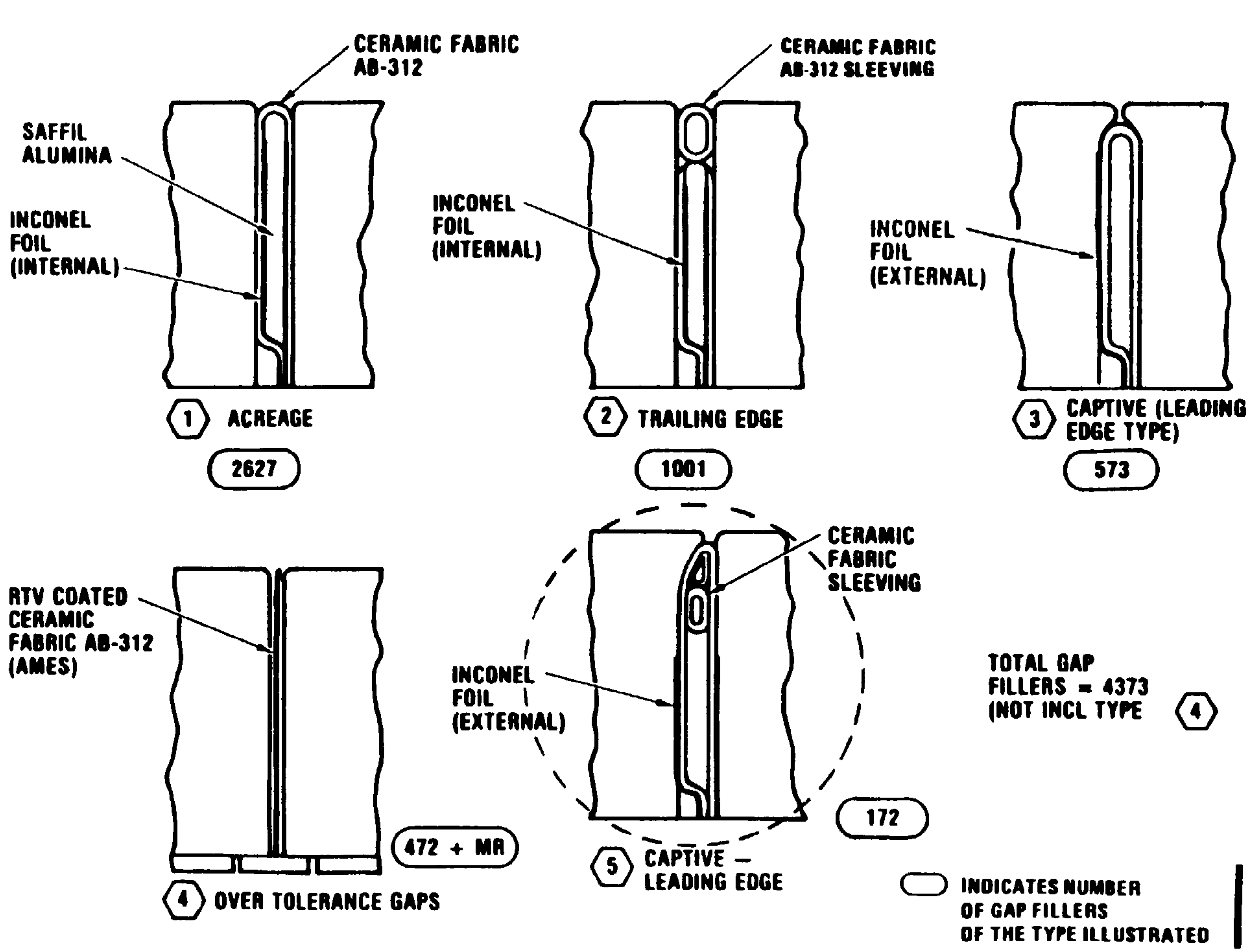


Figure 13. Various types of gap fillers used between tiles in high pressure, high temperature regions of the RSISS to prevent hot plasma gas from reaching the aluminum structure.

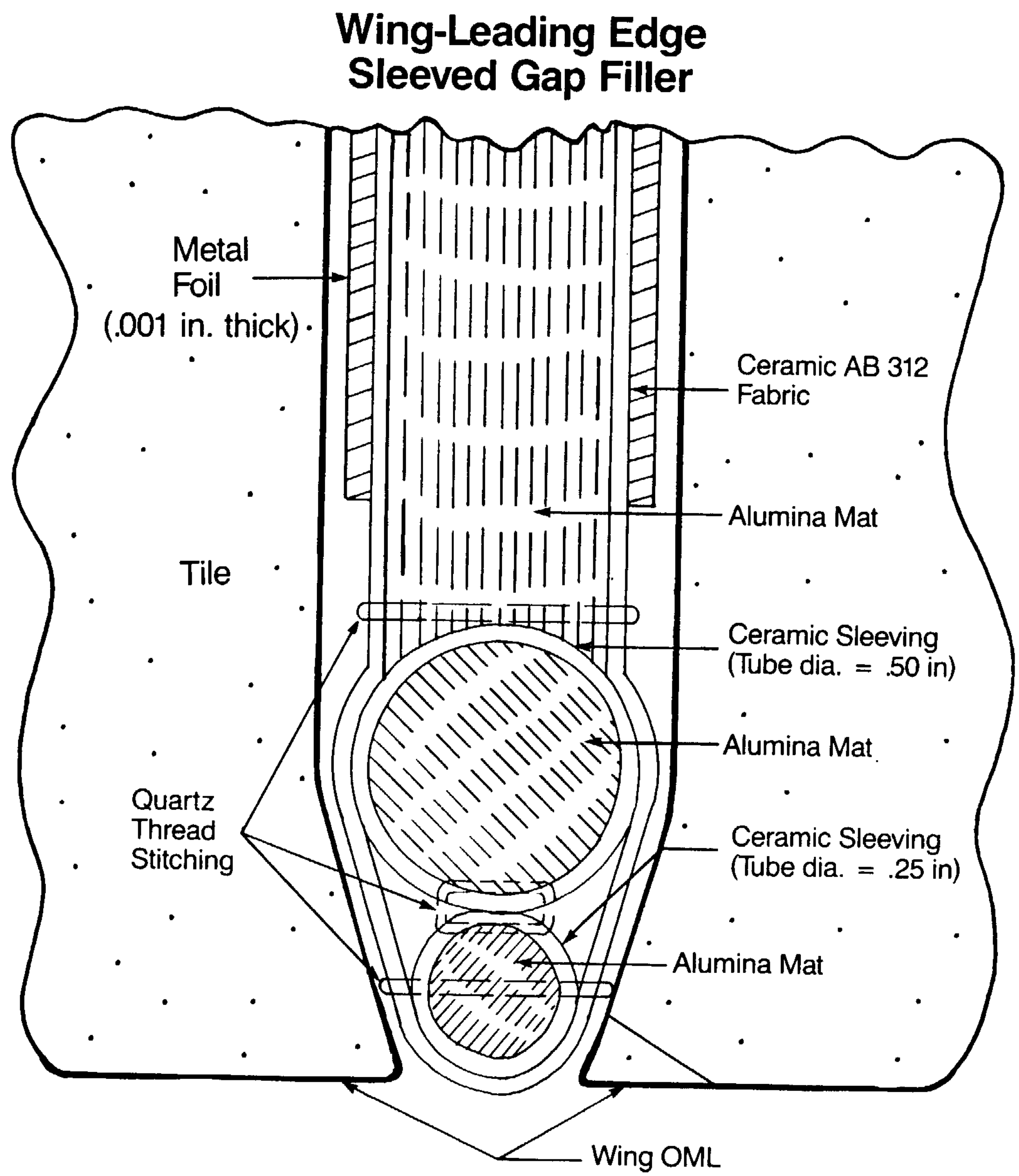


Figure 14. Detailed view of a wing leading edge, sleeved (captive) gap filler.

Theoretical Models

THEORETICAL MODELS AND TECHNIQUES DEVELOPED FOR THE THERMAL PROTECTION SYSTEM

Various models and techniques were developed to analytically predict the behavior of materials and assembly concepts of the TPS. Some examples of this modeling are contained in such publications as:

1. Williams, S. D. and D. M. Curry. "Effective Thermal Conductivity Determination for Low-Density Materials," NASA TP 1155, February 1978.

2. Williams, S. D., and D. M. Curry. "An Analytical and Experimental Study for Surface Heat Flux Determination." J. of Spacecraft and Rockets, Vol. 14, No. 10, pp. 632-637. October 1977.

3. Green, D. J. and Lange, F. F. " Micromechanical Model for Fibrous Ceramic Bodies." J. of the American Ceramic Society. Vol. 65, No. 3, pp 138-141. March 1982.

4. Bollard, R. J. H., and J. I. Mueller. Development of a Mathematical Model for FRCI. University of Washington under NASA Grant No. NAG 2-120 from the NASA Ames Research Center. December 1985.

5. Cooper, P. A. and J. W. Sawyer. " Life Considerations of the Shuttle Orbiter Densified-Tile Thermal Protection System." Shuttle Performance: Lessons Learned. NASA, CP 2283. October 1983.

An example of one such model is given in some detail in the following pages.

PREDICTION OF THERMAL PERFORMANCE

(RSISS)

MSC-16897

Ordinarily, the thermal insulation properties of a material are obtained by conducting extensive thermal conductivity tests for a specific density, and across many temperatures and pressures within the range encountered in the application environment. These tests are generally time consuming, difficult, and sometimes impossible to perform with acceptable accuracy, particularly at high temperatures (>1600 ^{o}F) and in a vacuum.

Attempts were made to model the thermal insulation performance of the bulk ceramic fibrous materials used to protect the internal structure behind the RCC nose cap. These attempts resulted in a theoretical expression for the effective thermal conductivity which has been verified by extensive testing over a broad range of temperatures and pressures. Currently, the theory is being used at Rockwell International to supplement other methods for the determination of the required thickness of tiles. This theoretical expression can reliably predict, interpolate or extrapolate the effective thermal conductivity of ceramic fiber insulation of densities up to 24 lb/ft^3, temperatures to 2000 ^{o}F, and from hard vacuum to one atmosphere. The application of this expression depends mainly on the use of absorption coefficients or backscattering cross-sectional data derived from blackbody radiation transmittance measurements for these ceramic fiber insulations. The major advantage is that data can be relatively easily and accurately obtained for high temperature, blackbody radiation while the specimen is maintained at room temperature. This information is necessary to determine the effective thermal conductivity (k_{eff}) related to radiation transmission.

A description of the theoretical expression is given as follows:

$$k_{eff} = k_{sc} + k_{rt} + k_{gc}$$

where

k_{eff} = effective thermal conductivity of the ceramic fiber insulation, Btu in./hr ft^2 ^oF, for densities up to 24 lb/ft^3, temperatures to 2000 ^oF, and pressures from hard vacuum to 1 atmosphere;

k_{sc} = effective conductivity due to solid and solid-to-solid contact conduction, Btu in./hr ft^2 ^oF;

k_{rt} = effective conductivity due to radiation transmission, Btu in./hr ft^2 ^oF; and

k_{gc} = effective conductivity due to gas conduction, Btu in./hr ft^2 ^oF.

The various components of the effective thermal conductivity are obtained from the following expressions:

$$k_{sc} = 0.05\,\rho_B$$

where

ρ_B = bulk density of the insulation, lb/ft^3

It should be noted also that the coefficient value of 0.05 is only applicable to alumino-silicate fiber insulation with an average fiber diameter of 3.6 microns. Coefficients for other materials may be obtained from low temperature measurements on insulation at atmospheric pressure.

$$k_{rt} = \frac{4\sigma T^3 L}{(2/\epsilon - 1 + NL)}$$

where

σ = Stefan-Boltzman constant = 1.713×10^{-9} Btu/hr ft^2 $^oR^4$;

T = mean temperature of insulation, ^oR;

L = thickness of insulation or spacing between the bounding surfaces, in;

N = backscattering cross-section, in.$^{-1}$; and

ϵ = emittance of the bounding surfaces

$$k_{gc} = \frac{(K_G)_T}{(1-f)}\left(\frac{L_f}{L_f + L_G}\right)_{P,T}$$

where

$(K_G)_T$ = thermal conductivity of the gas at temperature T, Btu in./hr ft^2 ^oF

f = fiber volume fraction

L_F = mean free path (at low pressures) for "molecule-fiber" collisions, μm

L_G = mean free path of gas molecules at pressure P, μm

The mean free path terms are found from the expressions:

$$L_F = \frac{\pi D}{4f} \qquad L_G = 0.131\,\frac{T}{P}$$

where

D = effective fiber diameter, m; and

P = pressure, torr.

The conductivity of the gas may be expressed as

$$(K_G)_T = \frac{a\sqrt{T}}{1 + \left(\frac{b}{T \times 10^{c/T}}\right)}$$

where

$a = 0.01368$
$b = 441.7$ (for air)
$c = 21.6$

The general approach of this expression can be useful in obtaining initial design data on the thermal insulation performance of existing or future ceramic fibrous materials specifically when radiation is the major mode of heat transfer at temperatures greater than 1600 ^{o}F. A typical application would be to predict jet engine insulation systems for high altitude aircraft. This approach should also be applicable to other similar mineral fiber insulation used in high temperature vacuum furnaces which have many industrial applications.

Thermal Response of Composite Insulation

An engineering model gives useful predictions.

A pair of reports presents theoretical and experimental analyses of the thermal responses of multiple-component, lightweight, porous, ceramic insulators. Although the particular materials examined were destined for use in the Space Shuttle thermal protection system, the test methods and heat-transfer theory will be useful to chemical, metallurgical, and ceramic engineers who need to calculate the transient thermal responses of refractory composites.

The bulk materials studied were mixtures known as fibrous refractory composite insulation (FRCI), consisting mostly of oriented silica and aluminoborosilicate fibers. A wide range of properties can be obtained by varying the proportions of the different types of fibers; for example, the addition of alumina reduces the shrinkage at high temperature, but it also reduces the tensile strength. An integrated insulating tile with low shrinkage at high temperatures might be made by bonding an alumina-enhanced layer to the front surface of a block of FRCI.

The thermal responses of the composite insulators were calculated from an engineering model described in detail in one of the reports. In the model, interlocking walls of fibers form cubic pores. The pore size, effective fiber diameter, and other model parameters are defined, using governing equations, from the composition and the physical and mechanical properties of the corresponding real composite material. The weak and strong directions of the composite are determined by tensile-strength measurements. In the model, the number of fibers under load within the walls of the model in a given direction is set proportional to the strength in that direction. It is important to incorporate this anisotropy into the model because the thermal conductivity is higher in the strong direction than in the weak direction.

The effective thermal conductivity is calculated as a weighted sum of solid, gaseous, and radiant components. The weighting parameters of the sum take into account the rule of mixtures and the corrections for the variations in fiber bonding and optical properties of each composite.

Insulator samples were assembled into flat-face, 40°-half-angle cones and were instrumented with thermocouples at various depths. To simulate a spacecraft reentry heat pulse, the cores were exposed to arc-heated airjets for 300 s, followed by cooling for 600 s in a vacuum of 190 μm Hg (25 Pa). The engineering-model prediction of temperature as a function of time agreed closely with the thermocouple measurements.

This work was done by David A. Stewart, Daniel B. Leiser, and Marnell Smith of **Ames Research Center** *and Paul Kolodziej of Informatics, Inc. To obtain copies of the reports, "Thermal Response of Integral Multicomponent Composite Thermal Protection Systems" and "Characterization of Thermal Conductivity for Fibrous Refractory Composite Insulations," Circle 76 on the TSP Request Card.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Ames Research Center [see page 12]. Refer to ARC-11680

Laboratory Testing

LABORATORY TESTING OF TPS MATERIALS

Equipment and techniques were developed to test a wide variety of materials, some very fragile, from cryogenic to high temperatures, destructively and nondestructively. A brief list of references related to the testing of TPS materials is given below.

Rigid RSI

1. Larson, H.K. and H.E. Goldstein. "Space Shuttle Orbiter Thermal Protection Material Development and Testing," Proceedings of 4th Aerospace Testing Seminar. Published at the Institute of Environmental Sciences, Aerospace Co., (1978), pp. 189-194.

2. Smith, M. and C.A. Estrella. "Furnace for Tensile Testing of Flexible Ceramics," NASA Tech Briefs. V. 10, No. 6, (1986) pp. 44-45.

3. Reference No. 5 on page 71.

DEVELOPMENT OF FIXTURING AND EXTENSIOMETRY FOR ELEVATED AND CRYOGENIC TEMPERATURE TESTING OF BRITTLE SILICA INSULATION MATERIALS

(RSISS)

MSC-20731

The testing of brittle, low strength materials such as those of the tiles, has always presented a challenge to industry, specifically in the design of grips and fixtures to accommodate the low density specimen material. Both grips and specimens for testing must be designed to minimize "end effects" that create localized stress concentrations and induced bending moments which contribute to premature failure.

When testing at either elevated or cryogenic temperatures, the problems are further complicated over room temperature testing, especially when the material exhibits as near a zero thermal expansion coefficient as do the silica based tiles. Since the fixtures dimensions may vary considerably relative to the specimen, a technique to isolate specimen variation is needed.

Commercially available testing systems and procedures could not accurately characterize these new materials in the severe temperature range they would encounter in service (-250 ^{o}F for on-orbit soak condition to +2300 ^{o}F for re-entry peak condition). The lack of these testing systems resulted in the design of new testing procedures and fixtures such as the one presented in the NASA Tech Briefs article entitled "Clip-on Extensometer," on the following page.

Features which qualify the test system described in the MSC-20731 disclosure as a novel technology item are:

- The use of self-tightening grips compensates for differential thermal expansion characteristics between grips and specimens.

- Averaging the strain reading from extensometers accounts for possible slippage in one side.

- Flowing helium through the lower frame assembly maintains low extensometer temperatures. Flowing helium through the furnace also reduces oxygen embrittlement of the furnace components.

- A water-cooled baffle plate separates the specimen environment from extensometry and read-out devices.

- The use of quartz pins with the same thermal expansion characteristics as the specimen, maintain contact between reference bars and specimen.

The overall efficiency of the system has been proven through test results obtained over the past decade in which the statistical (gaussian) distribution of data obtained at elevated and cryogenic temperatures has been shown to be remarkably similar to those obtained from room temperature tests.

Clip-On Extensometer

A flexural clamp eliminates problems of operator variability.

NASA Tech Briefs, January/February 1986

Lyndon B. Johnson Space Center, Houston, Texas

A clip-on extensometer developed for testing Space Shuttle insulation tiles accurately measures length changes of loaded specimens. The new design reduces measurement errors caused by variability among test operators.

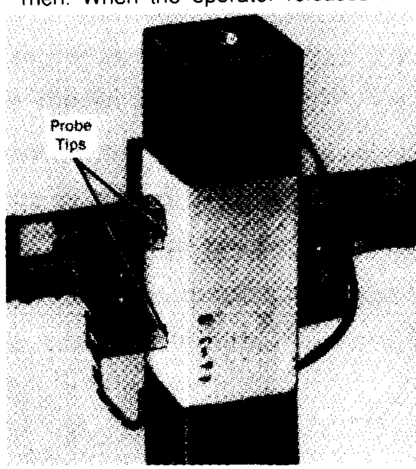
The extensometer is intended for use on ceramic, metal, or polymeric specimens (see figure). It consists of three flexural units. The central unit serves as a spacer, and the outer two units clamp on the specimen.

The operator attaches the extensometer to a specimen by squeezing together two screws on the central flexure unit. This moves the outer units apart so that the extensometer can be slipped over the specimen. When the operator releases the screws, the outer units move toward each other, pressing on the specimen sides with a force of 1.2 pounds (5.3 N). Each outer unit contacts the specimen with two small probe tips on flexing arms that are instrumented with strain gages.

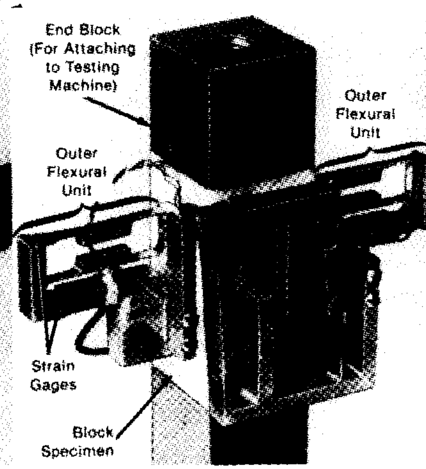
As a tensile or compressive load is applied to the specimen, the corresponding change in specimen length is transmitted to the strain gages. Each gage is previously calibrated separately, and an appropriate amount of resistance is added to the gage producing the highest output; each gage thus ends up with the same sensitivity. The two outputs are summed to give an average strain value.

Previously, a strain-gage fixture had to be installed on the specimen by alternately tightening the screws in each of a pair of plates to embed sensing tips. The amount of tightening varied among operators and created uncertainty about the distance between the tip and the strain gage. Now, with the clothespin extensometer, strain measurements are repeatable within 1 percent for a test-machine stroke of 0.005 in. (0.127 mm).

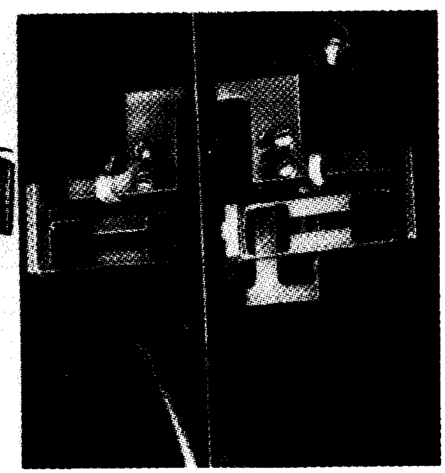
This work was done by Alan M. C. Holmes and Michael C. Duggan of Lockheed Missiles & Space Co., Inc., for **Johnson Space Center.** *For further information, Circle 21 on the TSP Request Card.*
MSC-20710



VIEW FROM FRONT OF BLOCK SPECIMEN



VIEW FROM REAR OF BLOCK SPECIMEN



EXTENSOMETER CLIPPED TO STRIP SPECIMEN

The **Extensometer Opens and Closes Like a Clothespin** and can thus be placed easily on a specimen. The dimensions of the block specimen are 1 by 1 by 2.2 inches (2.54 by 2.54 by 5.59 centimeters). By constructing central flexure units of various widths, one can adapt the extensometer to handle specimens ranging from thin strips to those many inches thick.

UNIQUE HEATING AND TEMPERATURE MEASURING TECHNIQUE
(RSISS)
MSC-20834

This innovation consists of developing a methodology for utilizing thermocouple probes in establishing the surface temperature of flexible ceramic thermal insulation (AFRSI). The surface of AFRSI (quartz cloth) does not allow thermocouples to be installed without damage to the cloth.

By placing a tent of silicon carbide cloth between the test specimen and the heat source, all components inside the tent are heated to the same temperature. Thus, a measurement of the air temperature inside the tent is a true representation of the actual surface temperature of the article. As illustrated on the following page, this technique has already been described in the NASA Tech Briefs, Fall 1985 issue. The technique would be useful to any laboratory performing thermal property testing of materials.

Another technique developed to quickly and simply estimate the thermal conductivity of carbon-carbon composites above 3000°F should be applicable to a variety of materials at a wide range of temperatures. It can be used to quickly rate candidate materials for applications in which thermal conductivity is of prime consideration.

The technique compares the thermal conductivity of candidate materials to that of a material with a known conductivity using a parallel cell type arrangement. Once the selection has been reduced to one or two candidates, formal conductivity test methods can be used to verify and quantify the results of the inexpensive comparative method. A description of this testing method can be found in the Article (MSC-20980) entitled "Comparative Thermal Conductivity Test Procedure," NASA Tech Briefs, Vol. 10, No. 2, pp. 75-76, March/April 1986.

Noncontacting Measurement With a Thermocouple

A tentlike covering brings the thermocouple to within
a few degrees of the surface temperature.

NASA Tech Briefs, Fall 1985

Lyndon B. Johnson Space Center, Houston, Texas

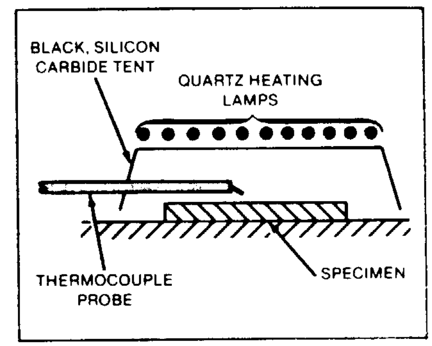
A technique originally developed for measuring the surface temperature of quartz fabric under radiant heating requires no direct contact with the heated surface. The technique is particularly useful when measuring the surface temperatures of materials that might be damaged if a thermocouple or other temperature sensor were to be attached.

As shown in the figure, a thermocouple is positioned above a specimen, and a "tent" of black SiC cloth is placed over the thermocouple. A heat source, such as an array of quartz lamps, heats the tent.

The tent acts as a black body, transforming the shorter wavelength radiation from the quartz lamps into longer wavelength infrared radiation. Everything in the tent is heated to nearly the same temperature. By bathing the experimental apparatus in black-body radiation, the tent prevents the differences in radiant heating that would result from the differences in emissivity and absorptance among the objects directly exposed to the heating lamps.

In an experiment to verify the technique, a thermocouple was installed on the surface of an insulating material, while another thermocouple was suspended just above the surface. Without the tent, there was a difference of 100° to 200° F (50° to 111° C) between the temperatures measured by the two thermocouples. With the tent, the two thermocouples agreed within a few degrees.

This work was done by William T. Weatherill, Cecil J. Schoreder, and Heinz J. Freitag of Rockwell International Corp. for **Johnson Space Center.** *No further documentation is available.*
MSC-20834



Ceramic-Fiber Cloth covers a test specimen. Despite the separation between them, the thermocouple and the specimen reach nearly the same equilibrium temperature.

PEEL TEST FIXTURE FOR AFRSI INSULATION BLANKETS

(RSISS)

MSC-20755

The fixture illustrated in Figure 15 was used to perform adhesion tests on specimens of AFRSI blankets. It consists of a peel test specimen bonded to a panel and mounted in a test fixture having a 90° guide bar for pull orientation. A (tension/compression) testing machine is employed to load the specimen.

The use of this fixture consistently assures controlled testing of the peel strength between the blanket and the coupon plate. The resulting numbers, which are recorded on autographic charts, are representative of the bond strength for a specific Shuttle insulation blanket or group of blankets. The novel feature of this fixture is the use of a 90° guide bar, and a free floating test panel contained by rows of bearings. This design assures a constant 90° pull direction of the peel strip in relation to the panel to which it is bonded. Potential uses for this system are to test the adhesive strength or tear strength of materials.

Peel Test Fixture Used for AFRSI Insulation Blankets

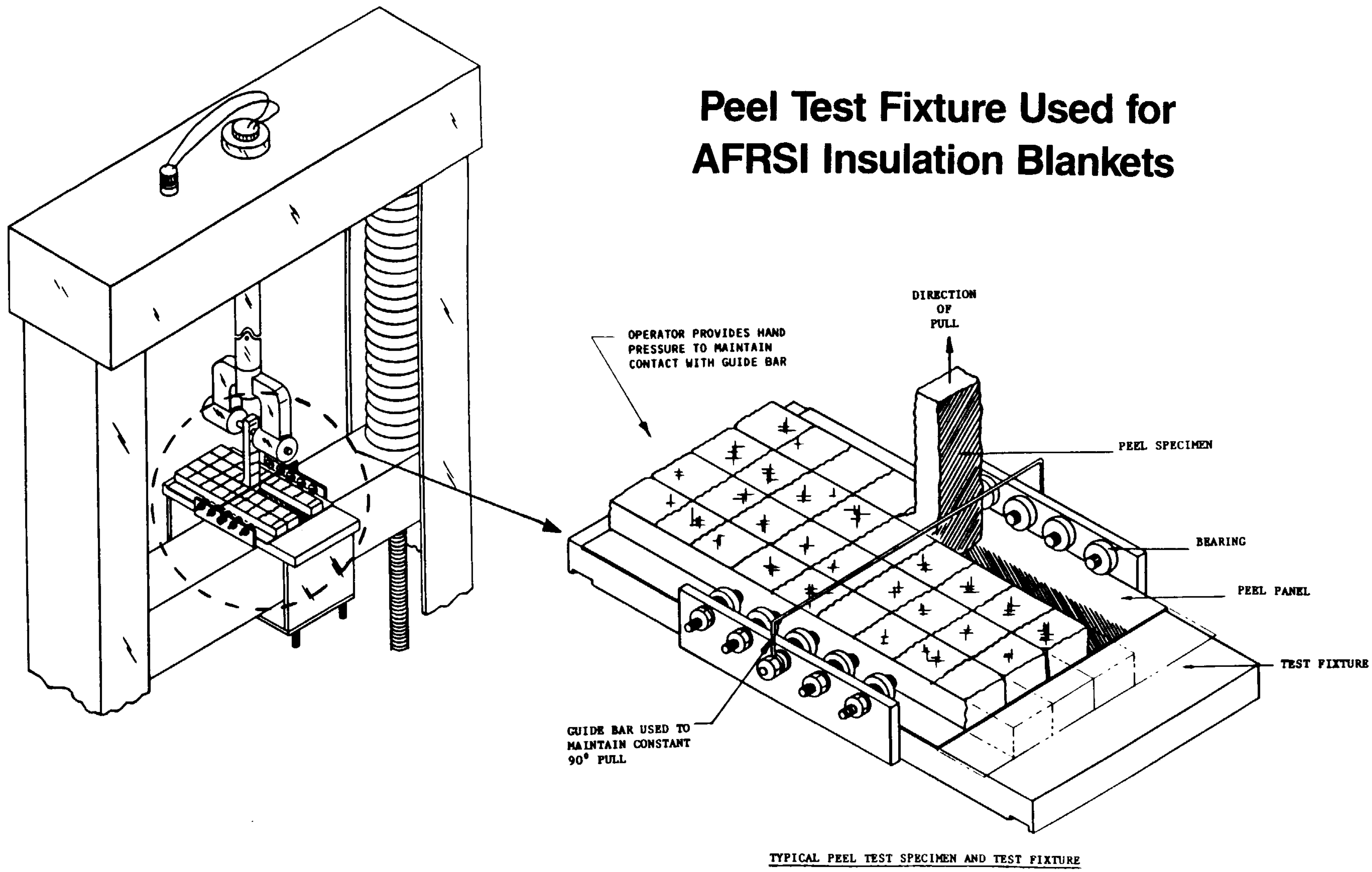


Figure 15. Fixture developed to test the bond strength of AFRSI blankets.

Zone-Controlled Resistance Heater

Temperature varies with time and location under computer command.

NASA Tech Briefs, Fall 1979

Lyndon B. Johnson Space Center, Houston, Texas

A geodesic array of heaters produces controlled temperatures over many independent zones. The array can be arranged to conform to and enclose virtually any shape, with close thermal coupling; and it can be programed to reproduce almost any desired time/temperature distribution. Each of the 18 graphite resistance heaters in the array is powered by a separate electrical supply unit and silicon-controlled-rectifier (SCR) control unit. Under the command of a computer the SCR's produce the time-varying temperature distribution.

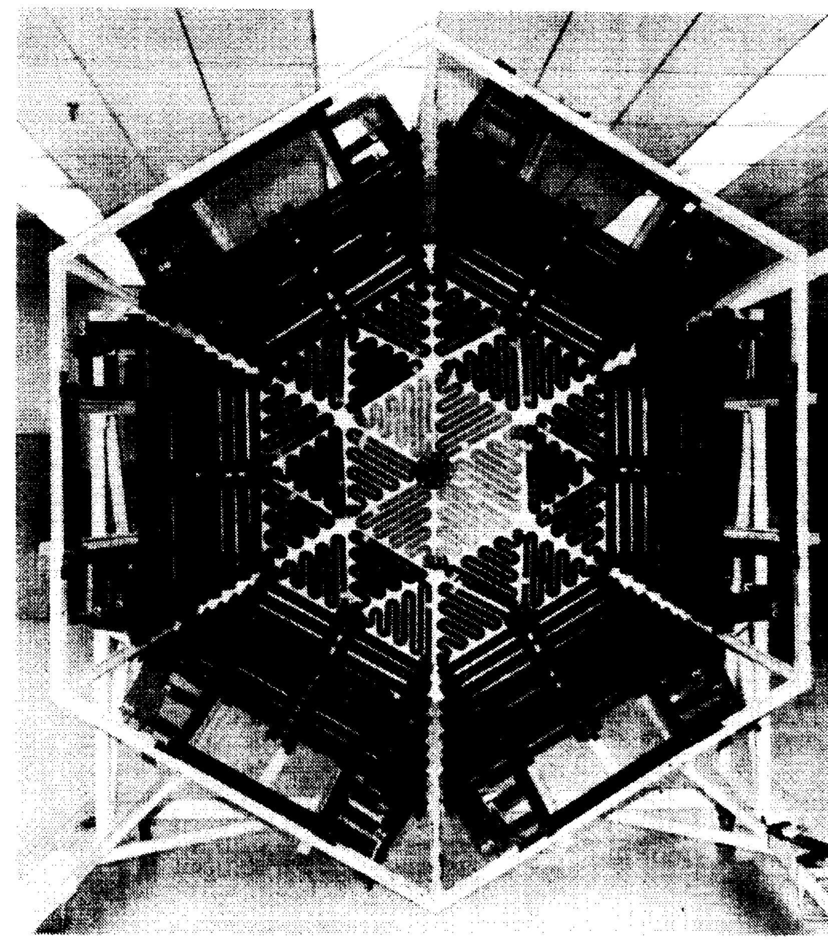
The heating system was developed to simulate reentry temperatures in tests of the Space Shuttle nose cap (see figure). The computer-controlled heater array may also be useful in producing zone-controlled heat reactors for coal gasification, in testing ceramic materials, and as a simulated heat source for testing energy generators and collectors. The heaters can create temperatures up to 2,600° F (1,400° C).

Two types of graphite heater elements are used in the array. Serpentine elements are used in 12 zones over the nose-cap semihemispherical surface. Straight rectangular bars are used around the nose-cap skirt. The serpentine heaters were specially developed to produce the required power density with minimum separation between the heater and the nose-cap surface.

The zone temperatures are sensed by thermocouples affixed to the back surfaces of the nose cap — one thermocouple in the center of each cap. The thermocouples furnish closed-loop feedback to the controller. Air blowers on the back surface allow the heating elements to be cooled off quickly in response to commands from the controller.

The graphite elements were selected over quartz lamps and silicon carbide bars because those heaters cannot produce the required high temperatures and cannot be oriented to produce the required temperature distribution. With a thin ceramic coating, the graphite heaters can be used in an oxidizing environment.

This work was done by Phillip R. Bagwell of Vought Corp. for **Johnson Space Center.** *No further documentation is available.*
MSC-16251



Heater Array for Testing Nose Caps utilizes serpentine graphite elements at the curved end of the cap and straight graphite bars around the skirt of the cap.

Manufacturing Tools and Techniques

MANUFACTURE OF TPS MATERIALS

The manufacturing techniques developed under the TPS program have important effects on the overall material properties. For example, a study of the effects of various compositional and processing variables on the properties of the FRCI-20-12 tile showed that the apparent thermal conductivity can be affected by several factors. The most important factor is the density change which occurs during the billet sintering cycle. By reducing the density change during sintering, a FRCI-20-12 tile with an acceptable apparent thermal conductivity was obtained. The reduction in density change during sintering was accomplished with the use of two techniques. First, the density of the unfired billet was increased to the highest attainable value (the high dry-density concept). Secondly, the sintering cycle was performed at a lower temperature for a shorter time. The high dry-density concept was achieved by a combination of compression, vacuum dewatering, and drying of the unfired billet. This procedure resulted in a material with more fibers per unit volume which explains the enhanced thermal conductance exhibited by tiles manufactured from high dry-density concept type billets.

Another beneficial technique incorporated into the tile billet manufacture was to replace the original five-sided heating with six-sided heating. This approach produced a more uniform strength distribution within the billets and increased the tensile strength of production units to near that which had been obtained in the pilot plant.

Manufacture of TPS materials remains distributed over several manufacturers. Some major materials, manufacturers and documentation of the manufacturing processes include:

Rigid RSI billets and array frame assembly tiles– Lockheed Missile and Space Company. See Banas, R. P., et al, "Lessons Learned from the Development and Manufacture of Ceramic Reusable Surface Insulation Materials for the Space Shuttle Orbiters," Shuttle Performance: Lessons Learned. NASA CP 2283. October 1983.

Close Out Tiles – Rockwell International. See same reference as above.

AFRSI – Manville Corporation. See Yung, E., "Unique Softgoods Applications for the Space Shuttle," Proceedings of the 26th National SAMPE Symposium. April 28-30, 1981. See also "Insulation Blankets for High Temperature Use," NASA Tech Briefs, Vol. 9, No. 4, Winter 1985, pp. 107-108. and "Silicon Rubber Stitching Seal," NASA Tech Briefs, V. 9, No. 2, Summer 1985, p. 154.

FRSI – Globe-Albany. See Thermal Insulation Protection Means. Dotts, R. L., et al., U. S. Patent 4,151,800.

RCC for the LESS – Vought Corporation. Curry, D. M., et al, Materials Characteristics of Space Shuttle Reinforced Carbon-Carbon. Proceedings of the 24th National SAMPE Symposium, Vol. 24, Book 2. 1979.

Seals and Thermal Barriers for PSS, Several manufacturers. Consult Rockwell International Manufacturing Process Procedures (MPP).

MANUFACTURE OF THE FRCI-20-12 BILLETS

(RSISS)

FRCI-20-12 is a composite fiber material containing amorphous silica fiber (Q-fiber) and approximately 20% alumino-boro-silicate (Nextel 312) fiber. Silicon carbide powder is added to enhance emittance and thus reduce overall thermal conductance. Bulk silica and Nextel fibers are heat treated at 2200 ^{o}F and 2000 ^{o}F, respectively, to stabilize and standardize fiber properties. Hydrocyclone cleaning is performed on the silica fibers to remove particulate contaminants, called "shot", followed by drying. The silica and Nextel fibers are intermixed and cast into billets, and then dried by using a microwave and convective air ovens. The dry casting is sintered in an elevator kiln at 2400 ^{o}F. The final size of production billets for all tile materials is 13 in. by 13 in. by 5 in. The billet process is summarized in Figure 16A.

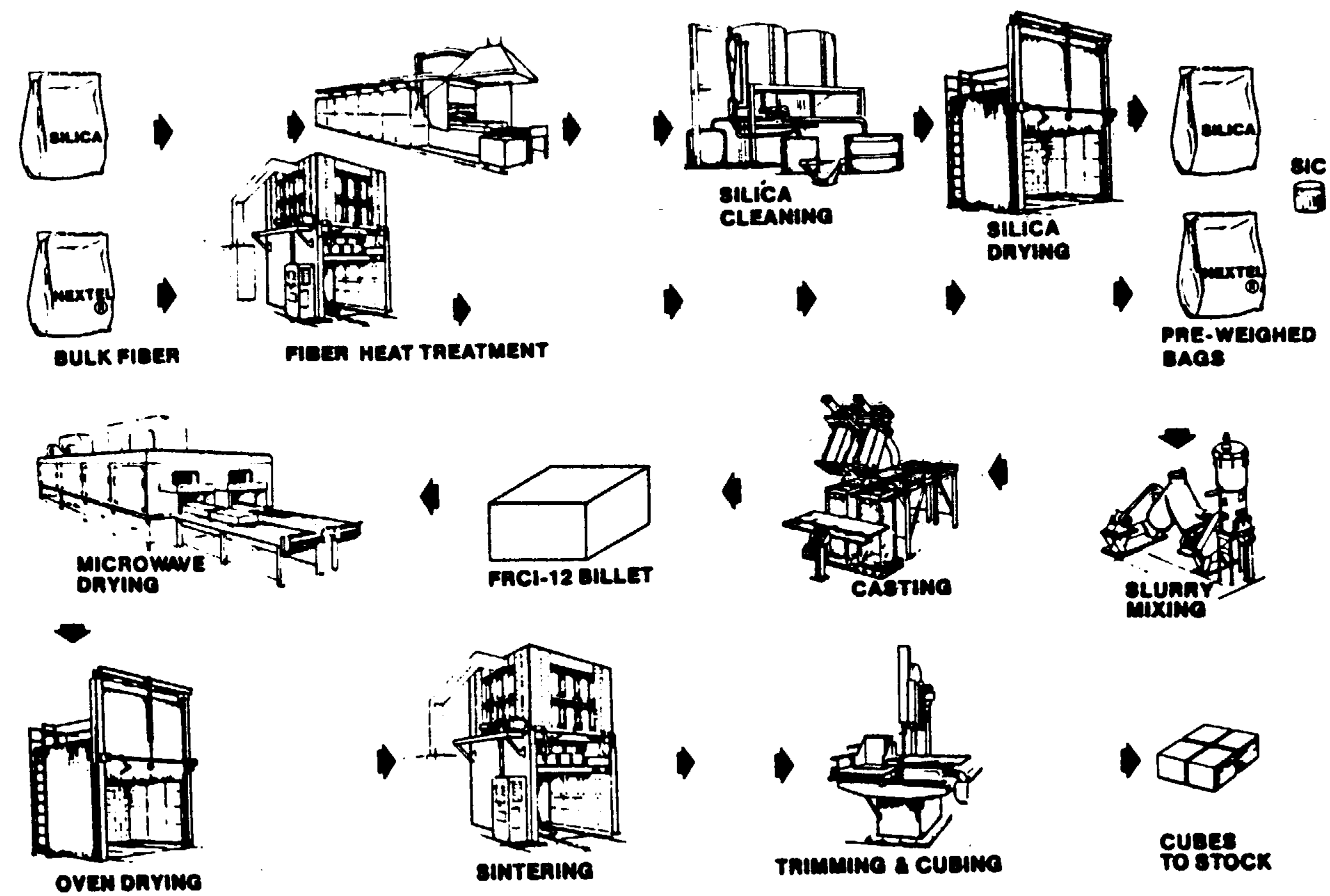


Figure 16. Manufacturing process of rigid RSI. Figure 16A shows the manufacture of the FRCI-20-12 billet (LMSC, from Reference No. 1 on page 92).

MANUFACTURE OF TILES
(RSISS)

The sequence of operations performed after the insulation material is cut into cubes is shown in Figure 16B. After the tiles are machined, they are heat cleaned to remove organic contaminants, masked to allow an uncoated breather space area along the lower perimeter adjacent to the inner mold line (IML), sprayed with an LRSI or HRSI coating and sintered at 2100 to 2250 ^{o}F in an Impsen tunnel-hearth roller kiln. The tiles are then waterproofed with a methyl trymethoxy silane (Dow Corning DC 6070) using a vacuum deposition technique. After waterproofing, the tile identification number is painted on the outer mold line (OML) and the tiles are checked dimensionally. Now, the tile IML cuts are performed, either individually which is called a nested tile or on a group in an array frame assembly (AFA) or array tile assembly (ATA). Dimensions of the AFA tiles are checked based on their fit in the polyurethane array frame (e.g., tile-to-tile gaps of 0.045 ± 0.016 in. for the lower wings and fuselage and 0.055 ± 0.016 in. on the upper wings, fuselage and vertical fins) . The dimensions of a nested tile are checked as required prior to the IML cut. The majority of the tiles have a square planform with inner mold curvature. More complex tile shapes are defined in the Rockwell master dimension specification book.

The tiles are manufactured using numerical controlled machines. Interactive graphic systems such as CATIA or CADAM are used for the design of tile geometries and machine control. Special offset methods are introduced to compensate for tile shrinkage during glazing operations following machining, Figure 17.

Recommended references describing the manufacture process of tiles are:

1. Banas, R.P., et al, "Lessons Learned from the Development and Manufacture of Ceramic Reusable Surface Insulation Materials for the Space Shuttle Orbiters," Shuttle Performance: Lessons Learned. NASA CP 2283. October 1983, pp. 967-1008.

2. O'Lone, R.G., "Thermal Tile Production Ready to Roll," Aviation Week and Space Technology, November 8, 1976, pp. 51-54.

3. Drossel Margaret. "A Production Routine for Shuttle Tiles," American Machinest, December 1982, pp. 65-67.

4. Fitchett, B.T. 1983. Dimensional Control of Space Shuttle Tiles During Manufacture. In: Proceedings of the 7th Conference on Composites and Advanced Materials, Ceramic-Metal Systems Division, American Ceramic Society, Vol. 4, No. 7-8, January 16-21, 1983, Cocoa Beach, FL.

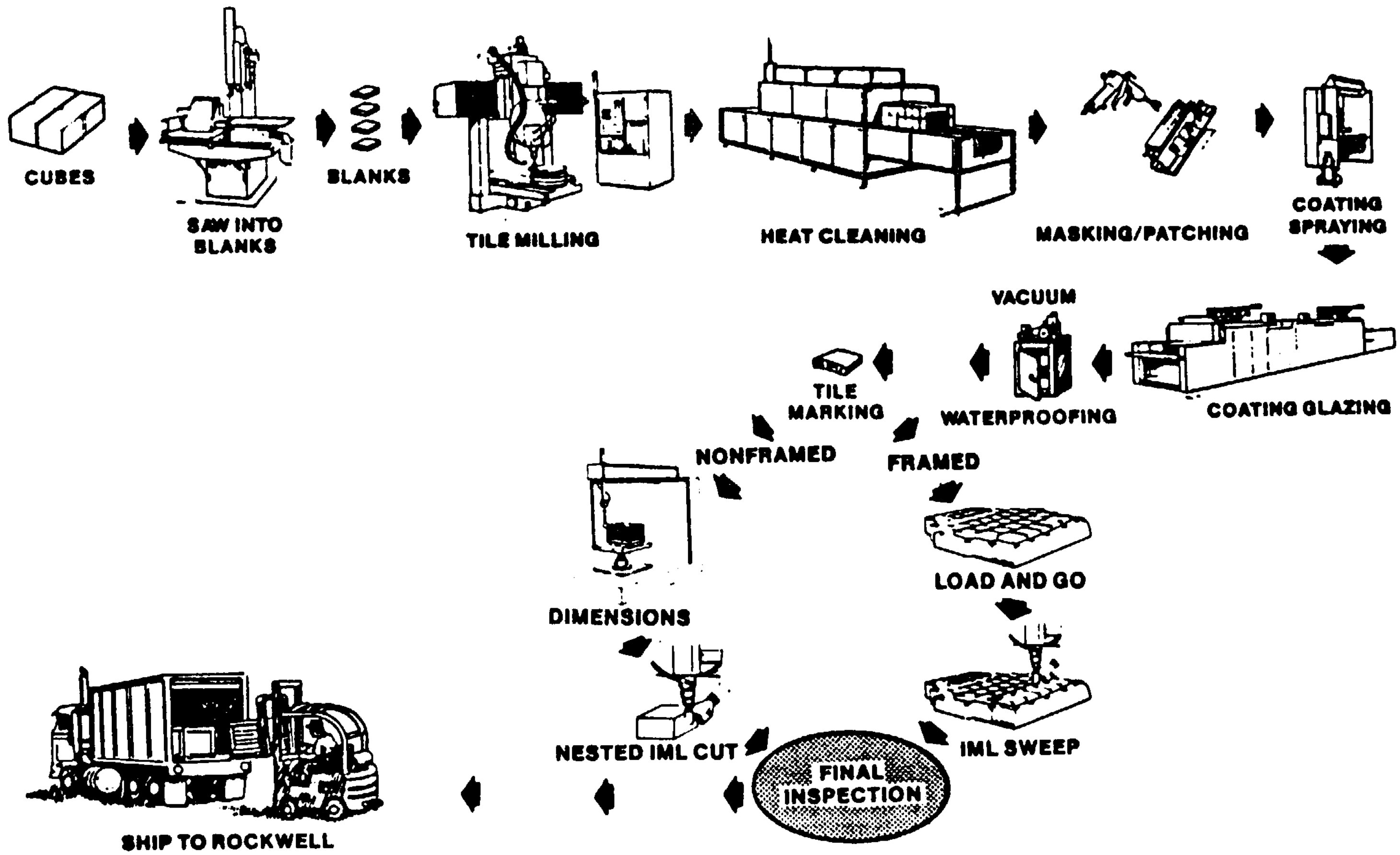


Figure 16B. Manufacture of tile from billets of any type of rigid RSI (LI-900, LI-2200, or FRCI-20-12) LMSC process.

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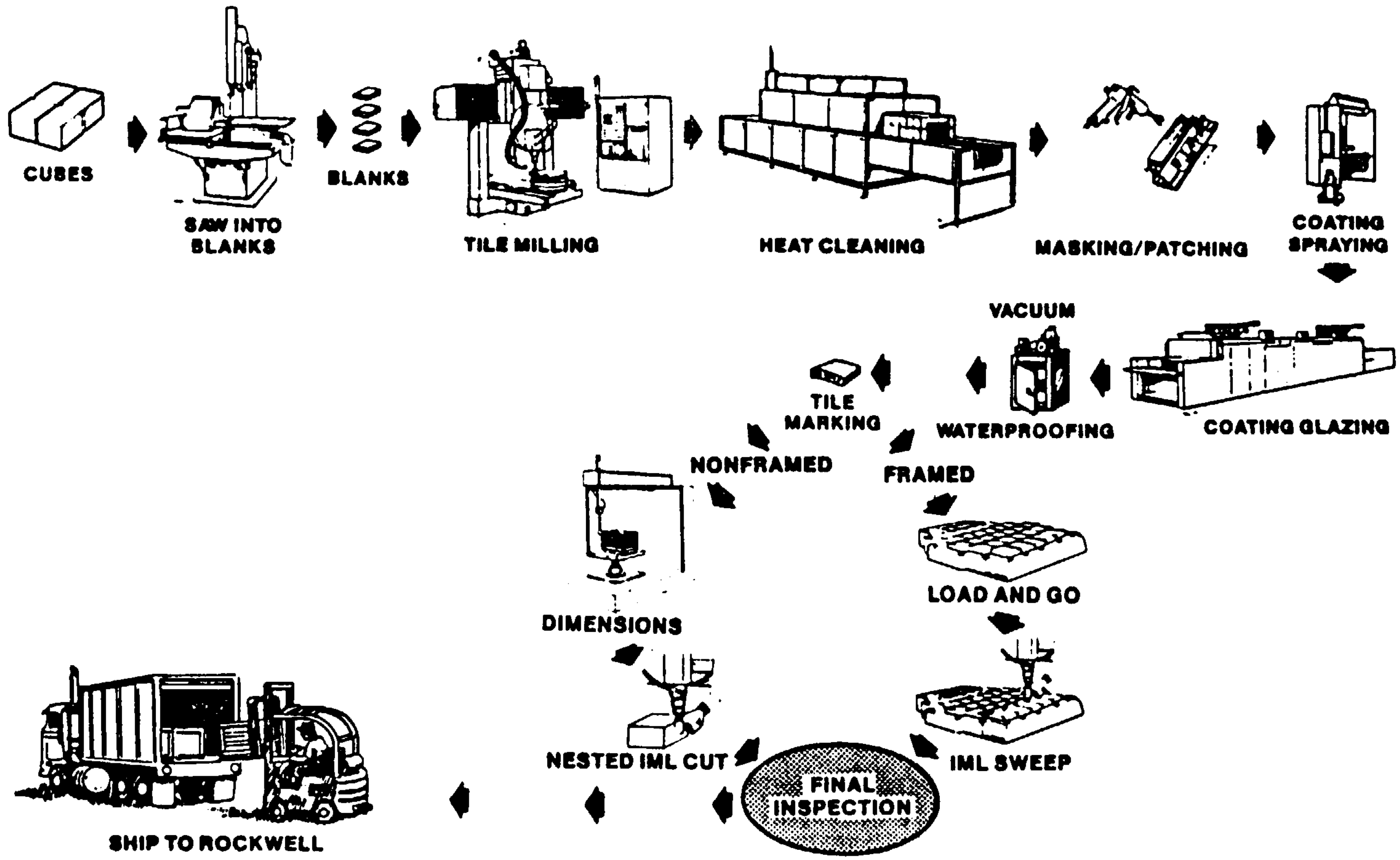

Figure 16B. Manufacture of tile from billets of any type of rigid RSI (LI-900, LI-2200, or FRCI-20-12) LMSC process.

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DEVELOPMENT OF THE CONICAL CUTTING TOOL

(RSISS)

MSC-20679

Initially, the machining of the HRSI tiles was accomplished using a straight-sided, diamond-embedded, cylindrical cutter one inch in diameter. Although this produced a tile whose precoated dimensions were within tolerances, the postcoated or glazed tile assumed a pillow shape due to excessive shrinking at the top corners (see Figure 17). This pillow shape was due to the upper surfaces being exposed to more radiant heat than the lower surfaces which sat on a silica fiber setter plate. The glazed tiles also had from 0.002 to 0.010 in. more shrinkage at the upper surfaces than at the lower or inner mold line surface. In general, there was about 0.004 in. of excess shrinkage at the upper surface per inch of tile depth. This meant that tiles greater than or equal to two inches thick would exceed the specification allowance of ± 0.008 in. per side just due to the coating glazing cycle. The development and subsequent use of a conical cutting tool with a 0.25 degree cone half angle and the machining of the tile sides with a radius cut solved the problem. The advantages of the conical tool are shown in Figure 17B. Figure 17A illustrates the problems with the original cylindrical cutting tools.

Another problem associated with the original cutting technique was that the coating witness line with a step caused many coating cracks and, consequently, many scrapped tiles. A second innovation which was designed into the conical tool was the undercut witness line. This undercut witness line minimized coating cracks at this location. A 0.020 in. undercut was utilized for this purpose.

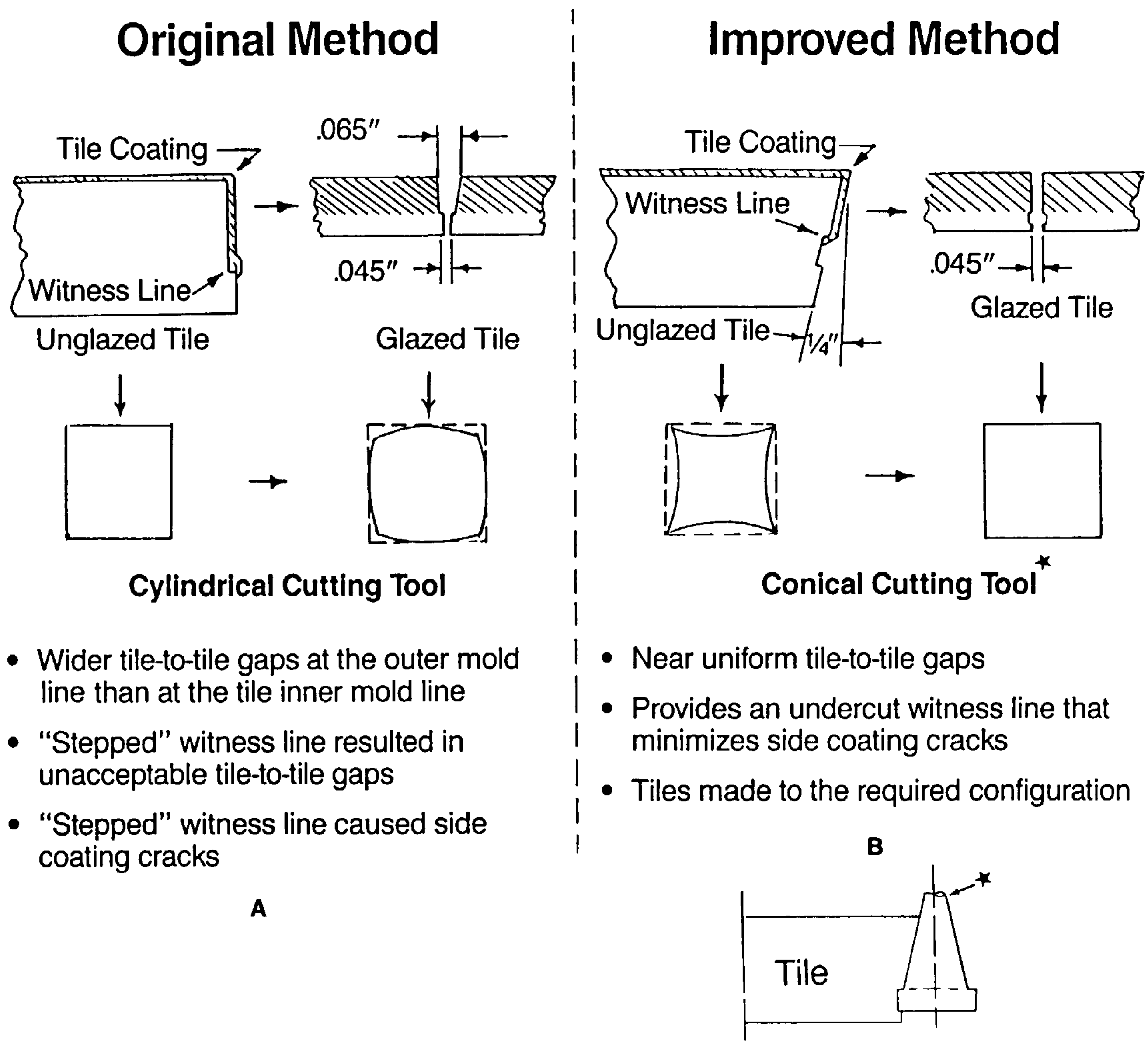


Figure 17. Tile machining procedure using the conical cutting tool. The implementation of this tool considerably improved tile yield (from Reference No. 1 on page 92).

FILLER BAR GUILLOTINE CUTTING FIXTURE

(RSISS)

MSC-20178

The filler bar is a strip of the same Nomex felt material used to make FRSI and the SIP, which after cutting is coated with RTV-560 adhesive. Its function is that of a seal to thermally protect the structure of the Orbiter beneath all of the tile-to-tile gaps.

Filler bars are made from Nomex felt strips of various thicknesses, 0.090, 0.115 and 0.160 in., but of the same width, 0.750 in. The thickness of the Nomex strip or filler bar depends on the thickness of the SIP used in the specific tile assembly.

In general filler bars are not bonded to the tile. However, those located under tile assemblies which experience the highest mechanical loads are bonded to the tile to increase bonding surface. Filler bars work better as a seal on flat surfaces than on curved ones.

There were over 60,000 filler bars or over 120,000 filler bar cuts required for Orbiter Columbia. Initially, the filler bars were cut with scissors, paper cutters and "Exacto" knives with rollers. However, these methods were time consuming and generally did not provide either a square-end cut or controlled dimensions because of the soft nature of the Nomex felt.

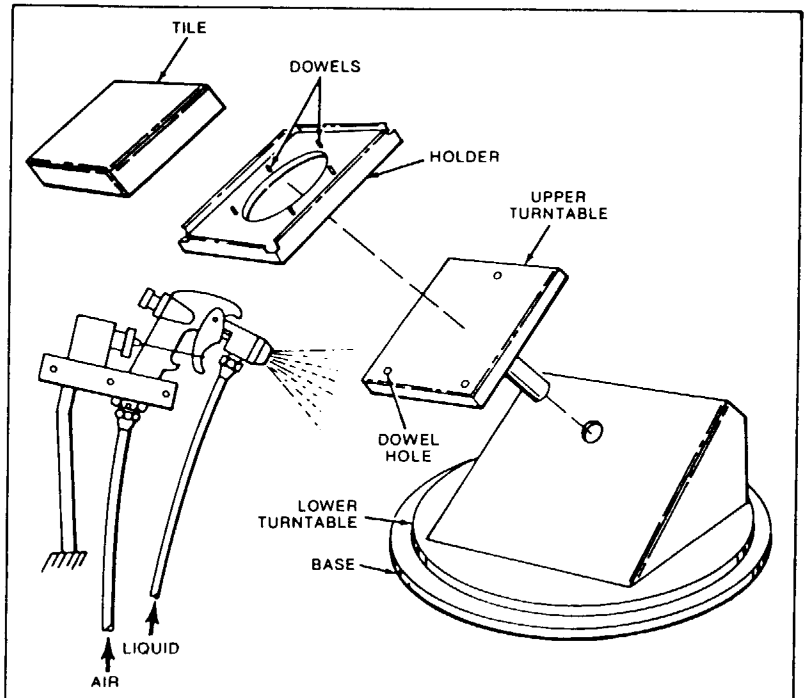
The cutting fixture presented in Figure 18, provided a square end cut of this flexible material in a minimal amount of time with safety and dimensional accuracy. This fixture could find commercial use in any industry such as the furniture upholstery industry, which requires cutting of felt type materials that are supplied in a strip configuration.

Figure 18. Fixture developed to accurately and quickly cut filler bars with square ends.

Rotatable Fixture for Spray Coating

Fixture that rotates about two axes ensures a uniform coating and minimizes handling of the coated workpiece.

Ames Research Center, Moffett Field, California



This **Rotating Fixture** supports a tile for spray coating. Dowel pins, which hold the tile on the holder and the holder on its turntable, could be replaced by a rotatable vacuum joint.

The rotatable fixture shown in the figure, developed for production spraying of Space Shuttle surface-insulation tiles, makes the coating process more efficient in several ways: (1) It rotates about two axes, presenting all sides of the tile to the spray; (2) it masks parts of the tile that are not to be coated; and (3) it allows the tile to be removed without handling the coated surfaces. Sprayed coatings are applied faster than with nonrotating or single-axis rotating fixtures, and the coatings are more uniform and less subject to handling damage. With some modifications, the fixture could be adapted for spraying parts other than tiles.

The tile holder is supported by a turntable canted 45° to the horizontal. Upwardly projecting lips along the edges of the holder shield part of the sides of the tile from the spray. A tile is installed by placing it on dowel pins extending up from the holder. Other dowels, extending down from the holder (not seen in the figure), engage holes in the top face of the turntable. A second turntable, which rotates in the horizontal plane at the base of the fixture, supports the 45° surface on which the tile turntable rests.

The 6- by 6- by 2-in. (15- by 15- by 5-cm) Shuttle tiles are held in the holder by pins 1/4 in. (0.62 cm) in length. With the tile holder in place, a glass slurry is sprayed from a gun roughly 6 to 8 in. (15 to 20 cm) from the surface to be coated. Each side of the tile is coated in sequence by moving the turntables until the surface is perpendicular to the spray. The front face is moved to face the sprayer by rotating the lower turntable. Approximately 0.0012 to 0.0015 in. (0.003 to 0.004 cm) of coating is applied on each pass, and the process is repeated until the desired thickness has built up.

To remove the tile, it and the holder are removed from the fixture and placed on a round pedestal that projects through the central aperture of the holder. The holder is lowered, leaving the tile on the pedestal.

This work was done by Victor Katvala, Ernest Porter, and Marnell Smith of **Ames Research Center.** *For further information, Circle 71 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Ames Research Center [see page A8]. Refer to ARC-11110.

Installation and Inspection

PRESTRESSED THERMAL PROTECTION PANELS

MSC-20254

Many concepts were initially proposed to thermally protect the aluminum structure of the Space Shuttle, particularly underneath the craft. One of the most interesting concepts was the one that used hexagonal panels made of high temperature resistant composites, such as carbon-carbon composites, which are mounted on a pre-stressed condition to prevent vibration and distortion under load.

This technique could be adaptable not only for thermal protection systems, but also could be used to assemble panels in aircraft, building walls or wherever large surfaces must be covered with stiff, flat sheets or panels which need to be easily removed for maintenance. As illustrated on the following pages, a brief description of this concept has already appeared in a NASA Tech Briefs issue.

Prestressed Thermal-Protection Panels

Panels are held securely with a minimum of mounting hardware.

NASA Tech Briefs, Summer 1984

Lyndon B. Johnson Space Center, Houston, Texas

Hexagonal panels of high-temperature-resistant composite material can be mounted in a prestressed condition to prevent vibration or distortion under load. Although originally developed for the Space Shuttle thermal-protection system, the technique may be adaptable to industrial thermal-protection systems as well. Furthermore, the panel shape and mounting arrangement are not limited to thermal-protection systems but can also be used on aircraft, building walls, or wherever large surfaces must be covered with stiff, flat sheets that can be easily removed for maintenance.

The panels in this case are made of an advanced 12-ply carbon/carbon composite with a thickness of 0.14 in. (3.6 mm). Each panel has stepped edges (see Figure 1). In the assembly, each stepped edge underlies an unstepped abutting edge of an adjacent panel. The panels therefore nest together, presenting a smooth outer surface interrupted only by a hexagonal pattern of cracks. There are no unobstructed gaps between the panels that would expose the substructure directly to high temperatures.

Each panel is fastened in a manner reminiscent of that used to mount the covers of automobile air filters. The panel as fabricated has a convex dish shape (see Figure 2). A depression or hub in the center has a hole for a high-temperature-resistant mounting screw.

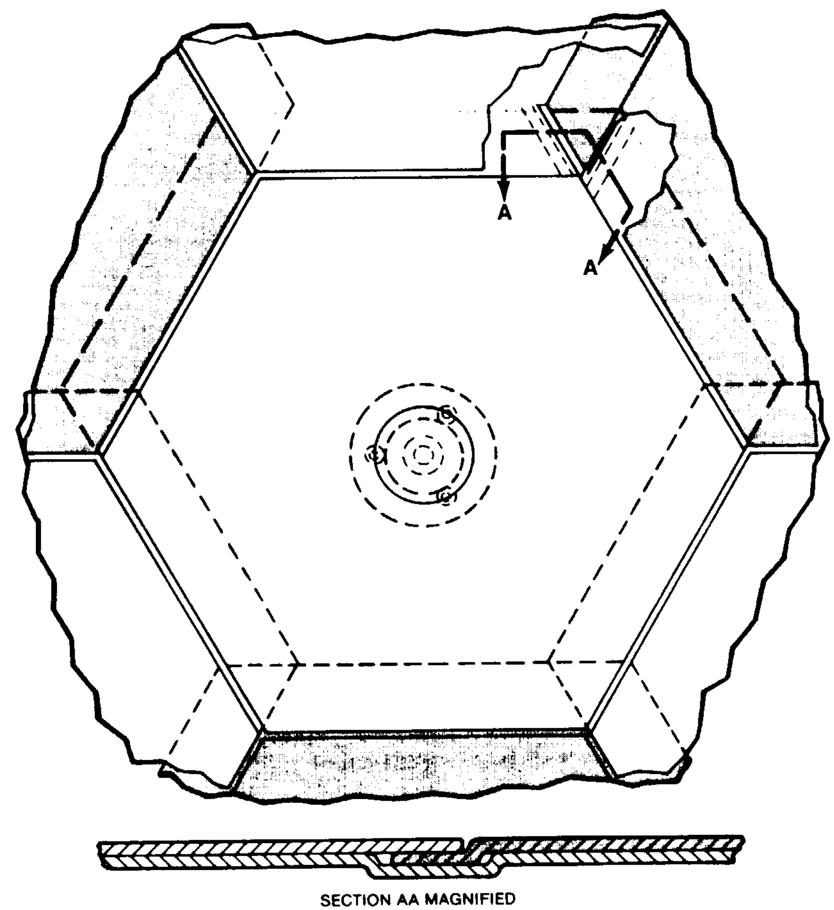


Figure 1. **Adjacent Panels Have Stepped Edges** so that they nest together in a hexagonal array and present a smooth outer surface.

C-2

Thermally-insulating spacer bars support the stepped edge of the panel in question or of an adjacent overlapping panel. A low-density batt insulation (for example, alumina/silica fibers in nickel foil) fills the space between the panel and the substructure.

When the screw is tightened in the threaded receptacle in the central insulating stanchion, the panel is flattened from its dished configuration. The panel is thus placed in bending stress; the spacer bars, in compression; and the insulating stanchion, in tension. The screw is tightened until the panel becomes flat and flush with the specified outer surface. The prestress is sufficient to resist distortion of the panel under normal operating loads.

After tightening, the screwhead is given some protection from oxidation by covering it with a carbon/carbon plug, the top of which is flush with the outer surface. The plug is held in place with a ceramic cement in a toroidal cavity around the edge of the hole.

This work was done by Thomas J. Dunn of **Johnson Space Center.** *For further information, Circle 129 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquires concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Johnson Space Center [see page A5]. Refer to MSC-20254.

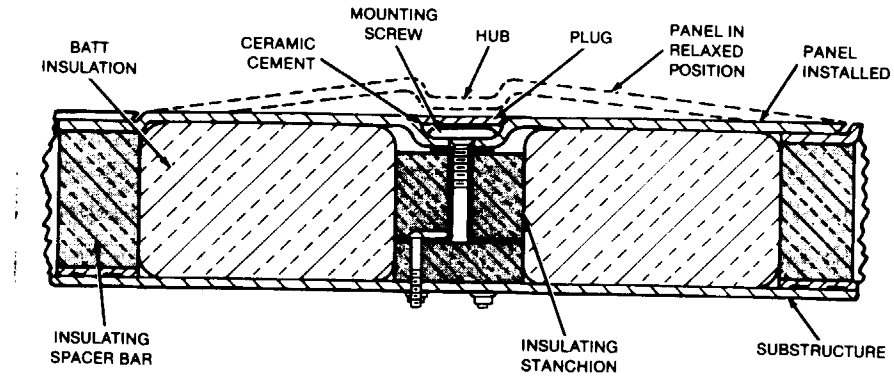
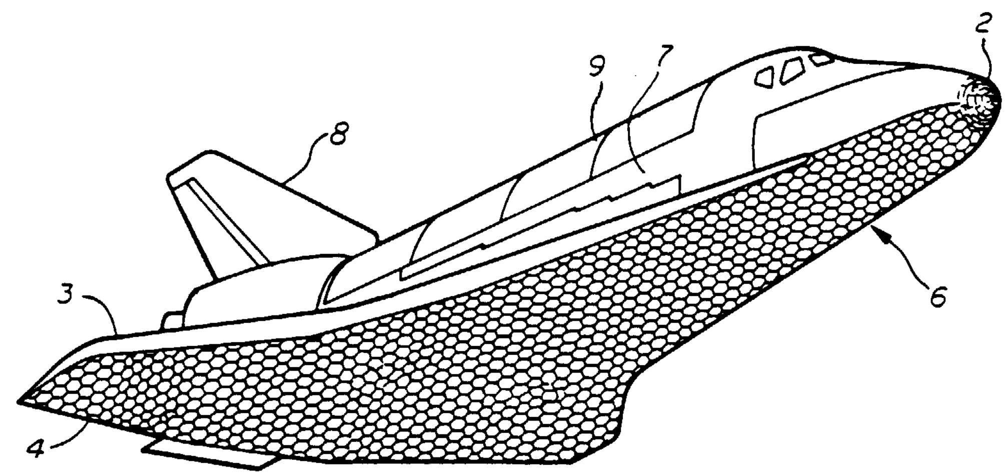


Figure 2. **Each Panel Is Held in Place** by a single screw that pulls it into a flat shape from its original shallow-dish shape. This shape and the prestressing make the panel stiff so that it resists vibration and withstands large mechanical loads.



FASTENING CONCEPTS DEVELOPED FOR THE TPS PROGRAM (RSISS)

The Orbiter TPS design requirements dictated that a positive tile-to-structure attaching system be developed which also isolated the delicate tiles from structure-induced loads. Many adhesive and mechanical types of fastening methods were developed and patented by NASA for this purpose. Of these concepts, only the adhesive fastening method was selected to attach the tiles to the skin of the Orbiter.

In the adhesive method, shown in Figure 19, (MSC-12619, U.S. Patent No. 4,124,732), the tile is bonded to the strain isolation pad (SIP) and this SIP is bonded to the aluminum structure using RTV-560 silicone adhesive. This adhesive material was selected for its low glass transition temperature (-170 ^{o}F) and its ability to remain flexible between this temperature and 500 ^{o}F for 100 Shuttle missions. An important element of this fastening concept is that the tile surface bonded to the SIP has been densified to improve the overall bond strength performance of the Tile-RTV-SIP system. This densification process has been performed using a colloidal silica slurry, U.S. Patent No. 4,338,368, or hydrolyzed tetra-ethyl-ortho-silicate (TEOS) solution, U.S. Patent No. 4,358,486.

Tiles in areas that required frequent access to the structure, such as penetrations, were also bonded to these surfaces using the adhesive method. The tiles at the perimeter of these surfaces, however, were made with a hole in the center through the thickness of the tile. This hole allowed the placement of a metal fastener from outside the Orbiter to affix the removable surface or penetration to the aluminum structure. A densified ceramic plug of refractory material similar to that of the tile was then used to fill the remaining gap between the top of the metal fastener and the outer surface of the tile. Tiles with ceramic plugs and inserts have been identified with a small arrow in Figures 6 and 20.

One of the mechanical fastening concepts developed for the TPS program is shown on Figure 19B and is described in MSC-20537, "Attaching Metal Fasteners to Silica Tiles." In this fastening concept, an internal threaded receptacle is placed inside an oversized hole of a densified ceramic plug made of refractory material similar to that of the tile. A close-cut disk is then inserted on top of the plug and this assembly bonded to a mating hole in the tile using silica cement. The receptacle floats in the tile and can accommodate lateral misalignment of $\pm$ 0.03 in. and azimuthal misalignment of a few degrees.

Although the bond strength and thermal expansion characteristics of mechanical fastening concepts , such as the auger attachment method (Figure 19D) were adequate for this application, they were unsuccessful in maintaining the integrity of the tiles during the high vibrational environment of the acoustic qualification tests. Tiles attached to test panels using these methods would actually disintegrate from the vibration induced, drill-like, action of the metal fastener inside the tile.

Fastening Concepts Developed for the TPS Program (RSISS)

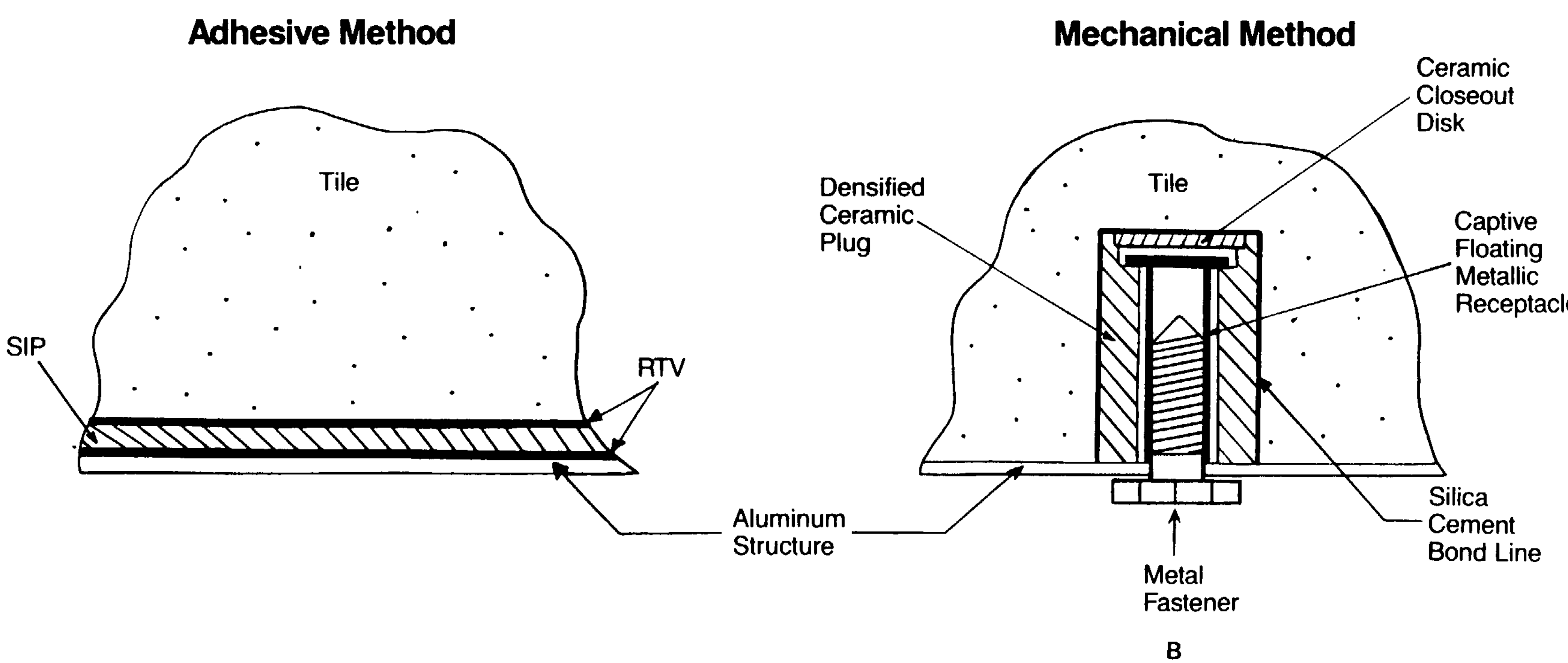


Figure 19. Fastening concepts developed for the rigid RSISS. The adhesive fastening system, Figures 19A and 19C, is the only method used to attach the tiles to the aluminum structure of the Orbiter. Mechanical fastening systems, such as the floating receptacle concept, Figure 19B, and the auger attachment method, Figure 19D, could be useful in applications which do not experience the aggressive acoustic induced vibration environment of the Shuttle.

THERMAL PROTECTION SYSTEM (TPS)

MATERIALS CONFIGURATION

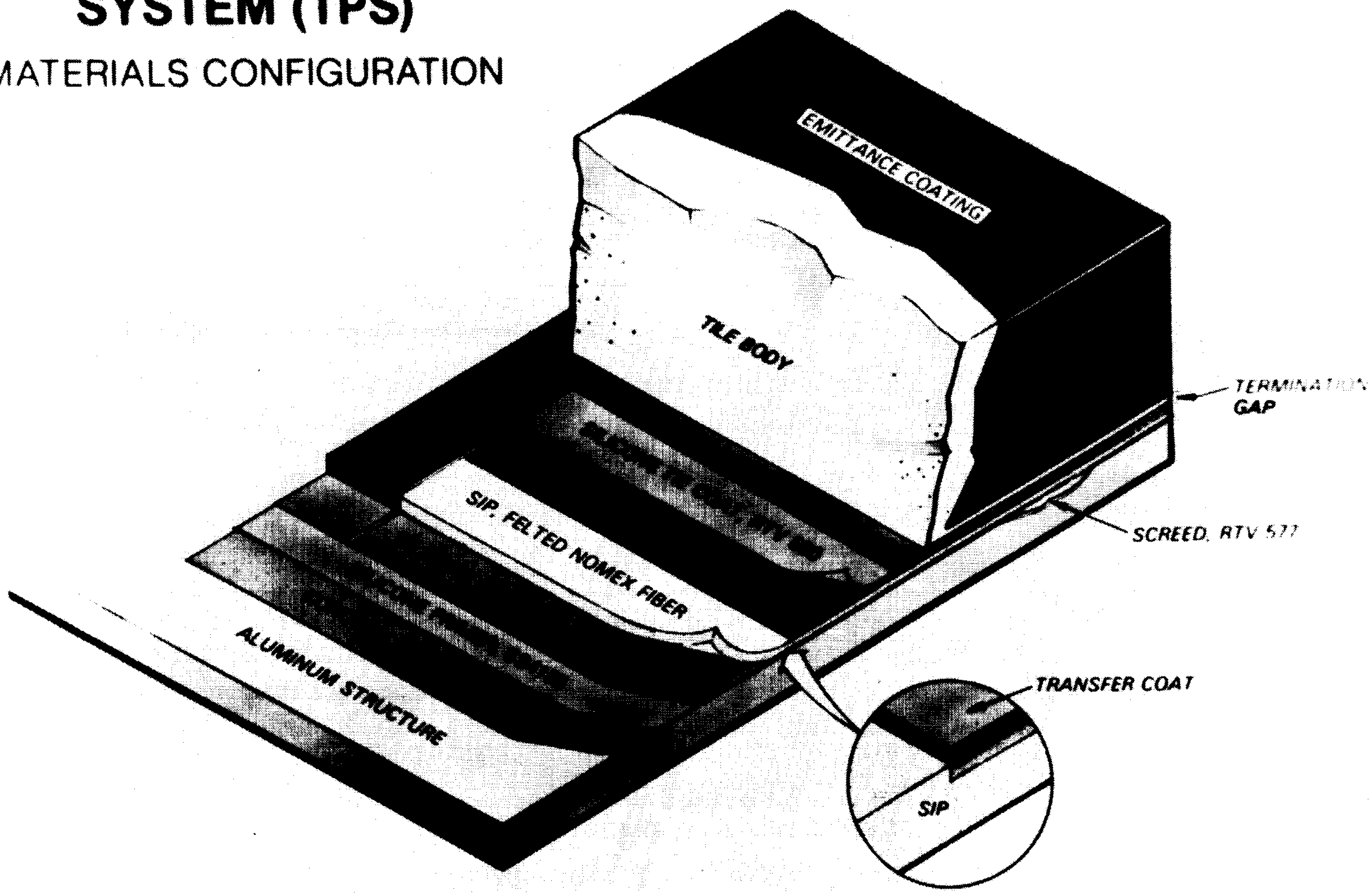


Figure 19C. Detailed view of the rigid RSI (tile) assembly.

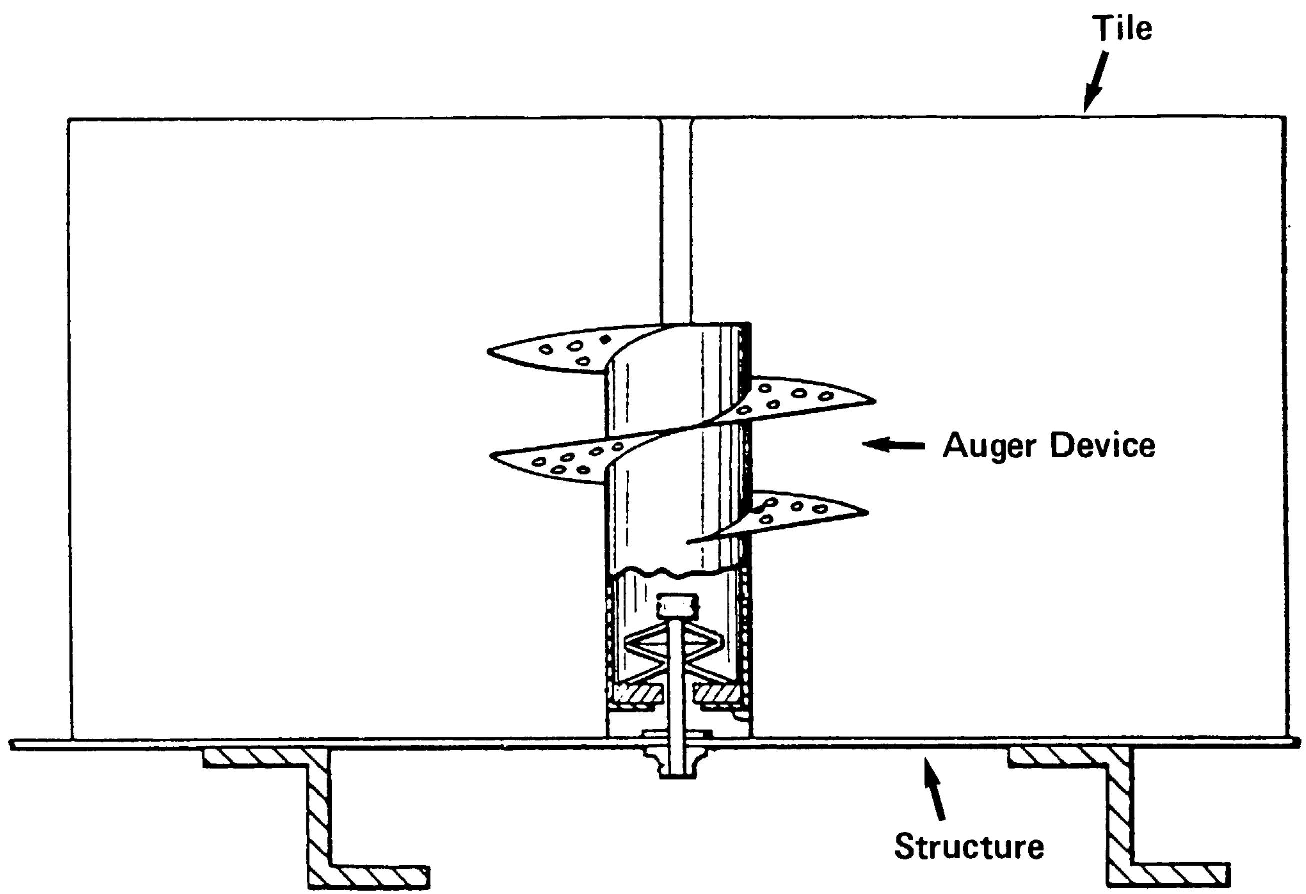


Figure 19D. The auger attachment method is a NASA invention (U.S. Patent No. 3,936,927). Few tiles attached with this mechanical fastening method were flown in one of the initial Columbia flights.

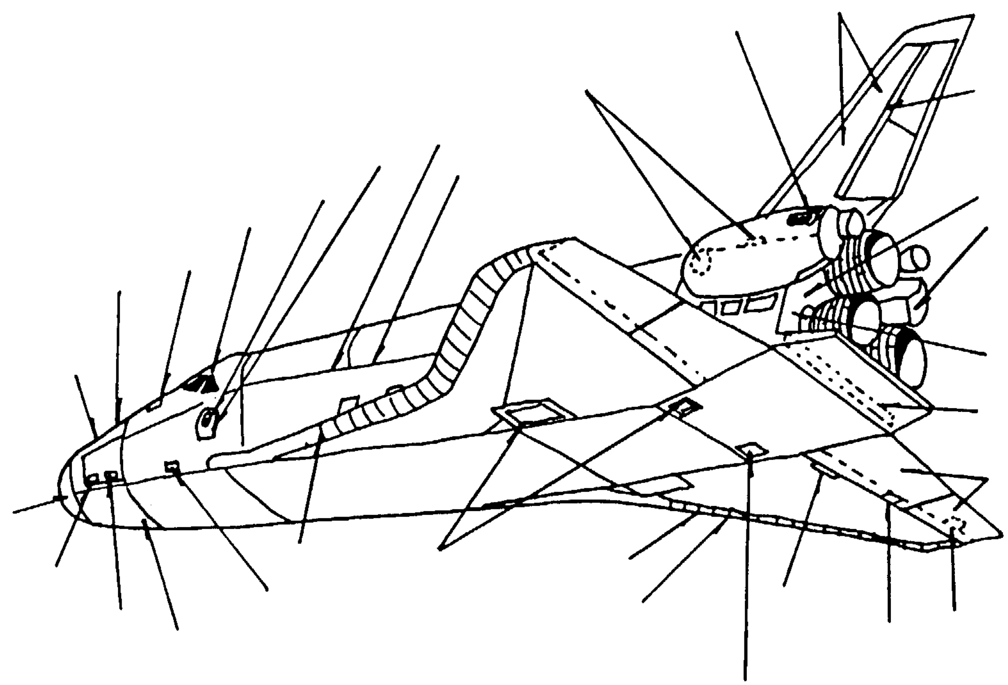


Figure 20. Silica inserts and plug location on Orbiter TPS. These plugs and inserts are used to cover the metal fasteners employed to hold the removable access panels around the perimeter of these penetrations. Some of these are bonded to the side of the tile in which they fit using special ceramic adhesives developed for this purpose.

"Densified" Tiles Form Stronger Bonds

A one-step application of colloidal silica more than doubles the bond strength.

Lyndon B. Johnson Space Center, Houston, Texas

The technique selected for strengthening the Space Shuttle surface-tile/substrate bond could be applied to similar ceramic-tile-attachment problems in other applications. The "densification" process strengthens the surface where the tile attaches to a felt strain-isolator pad, redistributing stresses and preventing failures at that point. Those interested in the technology of bonding delicate ceramics to nonrigid materials should also consider the Shuttle backup tile-strengthening technique, described in the following article, "Tile Densification With TEOS" (MSC-18737).

The reusable surface-insulation tiles that protect the orbiter from repeated high-temperature plunges through the atmosphere are susceptible to failure where they are bonded to felt pads that separate them from the Shuttle frame. In densification, a binder mixture impregnates the bottom of the fibrous ceramic tiles to form a high-strength skin. The platelike layer more than doubles the strength of the tile-to-felt-pad bond. In tests, tiles that were densified did not fracture until they were stressed to the limit of the bulk material (see figure). Undensified tiles, in comparison, fractured at the bond interface and at much lower stresses.

The process uses an aqueous mixture of colloidal silica as a cement and ball-milled silica particles, which — like sand in concrete — act as a reinforcement. Brushing forces the mixture into the interstices of the tile. The depth of densification is 0.06 to 0.11 inch (1.5 to 2.8 mm), depending on the original tile density. The outer 0.035 inch (0.9 mm) of densified tile are three to six times as strong as the bulk material. The densified layer is still porous and allows volatile materials to escape as the Shuttle heats up in the atmosphere.

Densified and Undensified Tiles fail at different locations and at different stresses. The undensified specimen (left) failed at about 11 psi (7.5×10^4 N/m^2) at its interface with the stress-isolation pad. The densified tile (right) failed at a much higher stress, 23 psi (1.6×10^5 N/m^2), and the break occurred in the bulk material.

The first step in densification is an application of isopropyl alcohol to the bottom (unglazed) surface of the tile. The alcohol allows the densifying agents to wet and penetrate the water-repellent treatment on the tile. The slurry of colloidal silica and silica powder, with a small amount of pigment added, is painted on the tile. The pigment, an inert ceramic (tetraboron silicide), makes the slurry easier to see and aids in an even application.

After air-drying for 24 hours and oven-drying at 150° F (66° C) for 2 hours, the tile is weighed to determine the amount of material that has been added. Usually the amount is about 0.5 g/in.^{2} (0.078 g/cm^2) of surface area. Finally, the tile is rewaterproofed by exposure to vapors or methyltrimethoxysilane and acetic acid (a catalyst) at 350° F (1.77° C). The tile is then ready for installation.

This work was done by Robert L. Dotts of **Johnson Space Center** *and Jack W. Holt of Rockwell International Corp. For further information, Circle 69 on the TSP Request Card.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Johnson Space Center [see page A5]. Refer to MSC-18741.

ARRAY FRAME ASSEMBLIES
(RSISS)
MSC-18571

An initial impression of the tile arrangement on the surface of the Space Shuttle Orbiter could suggest that they are installed in a random manner, see Figure 21A. In fact, the tiles are arranged in a very organized fashion using elemental units or arrays composed of a group of tiles. There were approximately 1,100 different arrays used in Orbiters Columbia and Challenger, less in subsequent Orbiters. Each array interlocks with one another, as shown in Figure 21B, to provide the maximum overall strength to the entire RSISS.

The arrays of tiles were installed using a device known as an "Array Frame Assembly" (AFA), Figure 21C. This AFA was designed to perform not only as an installation device, but also as a shipping container and a nesting fixture to hold the tiles while machining the inner mold line (IML). The IML represents the inner surface shape of each tile. As an installation device, the AFA was used to hold, to precisely locate using reference pins (arrows in Figure 21C), and to bond an array of tiles to the Orbiter skin. As a shipping container, the AFA protected the delicate tiles from possible transportation damage induced by impact, vibration, or handling. As a nesting fixture, the AFA had to hold between 6 and 50 unique tiles. Each tile had its own "vacuum nest" for holding the tile within the array for proper machining of the IML. Rigid peripheral and gap tolerances also had to be held.

An additional time saving benefit that resulted from the use of this device was that the tile dimensions installed using an AFA were checked by their ability to fit into the AFA with the required tile-to-tile gaps rather than each tile checked individually. The development and implementation of this multipurpose device was a significant new technology advancement and an extremely cost effective achievement for the TPS program.

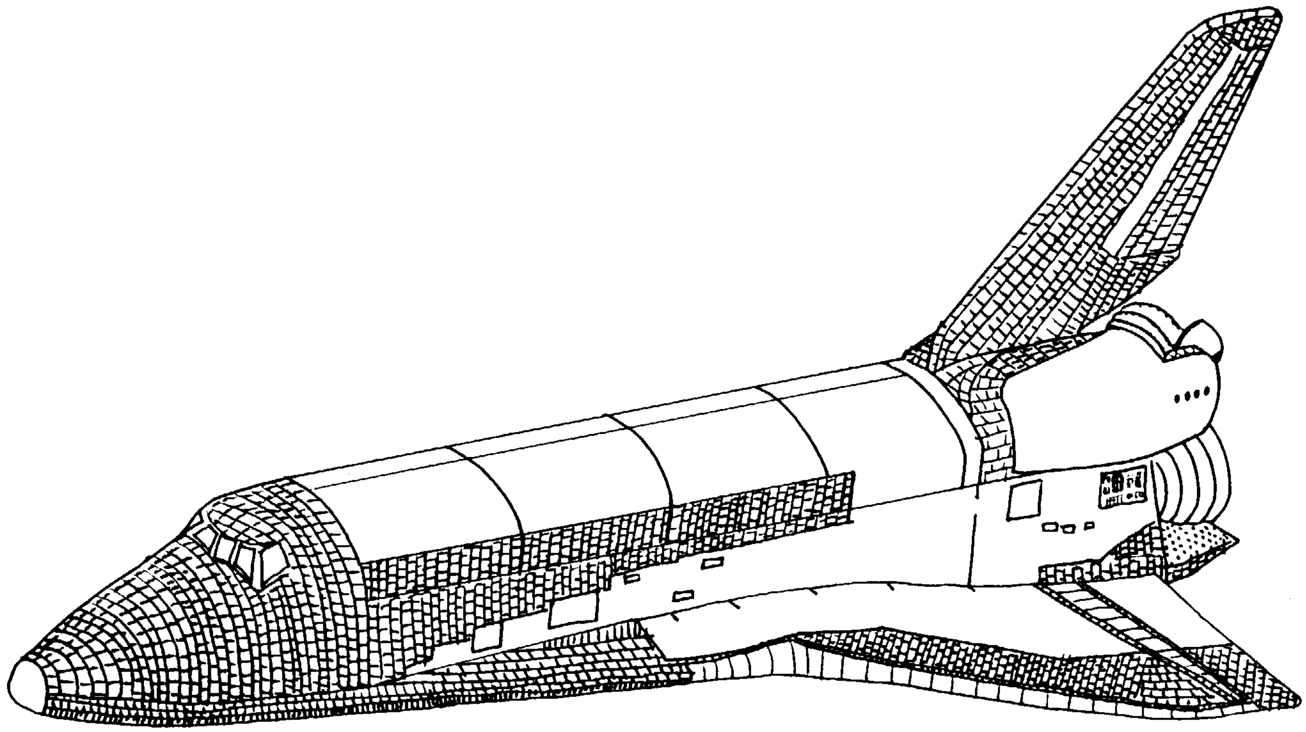


Figure 21. Tile configuration. Although at first glance the tile arrangement on the surface of the Orbiter may appear to be random in nature, Figure 21A, they are actually present in a very organized architecture as shown in Figure 21B.

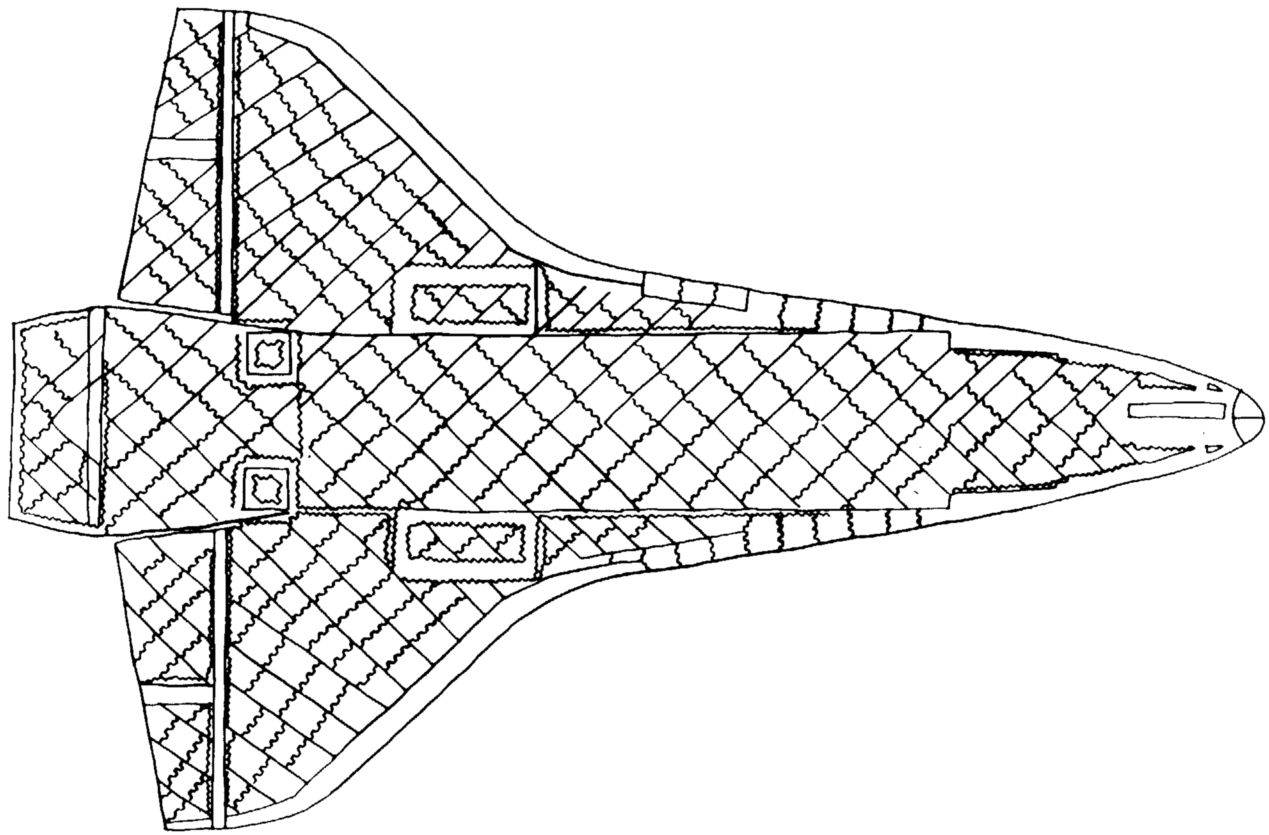


Figure 21B. Array frame assembly (AFA). Tiles are arranged along the surface of the Orbiter in very precise groups using an elemental unit known as an array frame assembly. There were approximately 1,100 AFA's in Orbiters Columbia and Challenger, less in subsequent Orbiters.

Figure 21C. A view is given of the AFA. This fixture was used as a machining bed, shipping container, and installation device. Arrows indicate reference pins used to accurately place each AFA on the Orbiter surface.

USE OF NETTING TO CONTROL THICKNESS OF ADHESIVE COATING

(RSISS)

MSC-19462

Coating the lightweight tiles with the highly viscous RTV adhesive created many problems, particularly since the thickness requirement for this adhesive layer was 0.0075 + 0.001 in. Coating by spraying the adhesive provided a non-uniform distribution of the adhesive, particularly around the edges. Coating methods such as brushing and direct rolling were also unsuccessful because the lightweight, fragile tiles and flexible SIP would tend to stick to the object used to apply the coating. Additionally, tiles coated with the RTV adhesive using these techniques vary in film thickness from 0.005 to 0.009 in.

A solution to this problem was to hold down the lightweight, fragile tile, and flexible SIP with a metal wire grid as illustrated in Figure 22, or a nylon net of specific thickness, and then coat them with the RTV adhesive using a short nap roller. This technique provided a uniform coating without edge smearing in a relatively short time. Cleanup and handling problems encountered early during the adhesive application to the tile-SIP assembly were also reduced. Tiles coated with RTV using this technique could be rapidly applied within the 0.001 in. thickness tolerance.

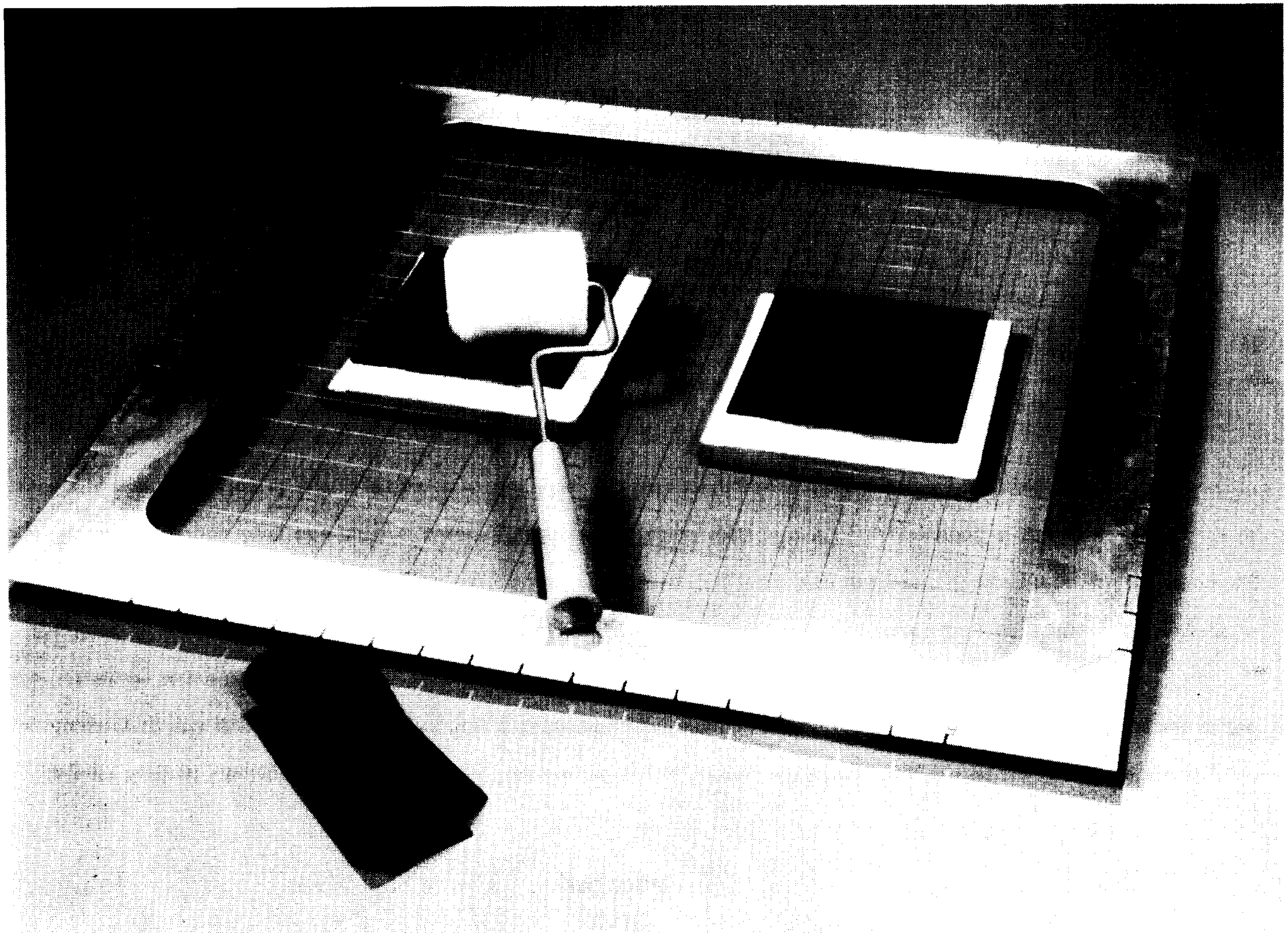


Figure 22. The most effective technique developed to apply RTV adhesive to the tile-SIP assembly.

ULTRASONIC DEVICE FOR THE DETERMINATION OF TILE FIBER ORIENTATION

(RSISS)

MSC-20229

The best thermal insulation properties of the tiles are along the through thickness direction where the fibers are aligned perpendicularly to this direction. Because of this, tiles installed with incorrect fiber directions could cause overheating of the Orbiter aluminum structure.

The fiber direction could only be determined visually prior to coating. Therefore, there was a need for an instrument to check the orientation of the fibers in installed coated tiles quickly and reliably. The ultrasonic device presented in Figure 23, can verify fiber direction from one surface reading on installed coated tiles. Possible applications of this device could be to establish the grain direction of rolling plates or stock. Also, to establish the fiber direction in similar fibrous materials.

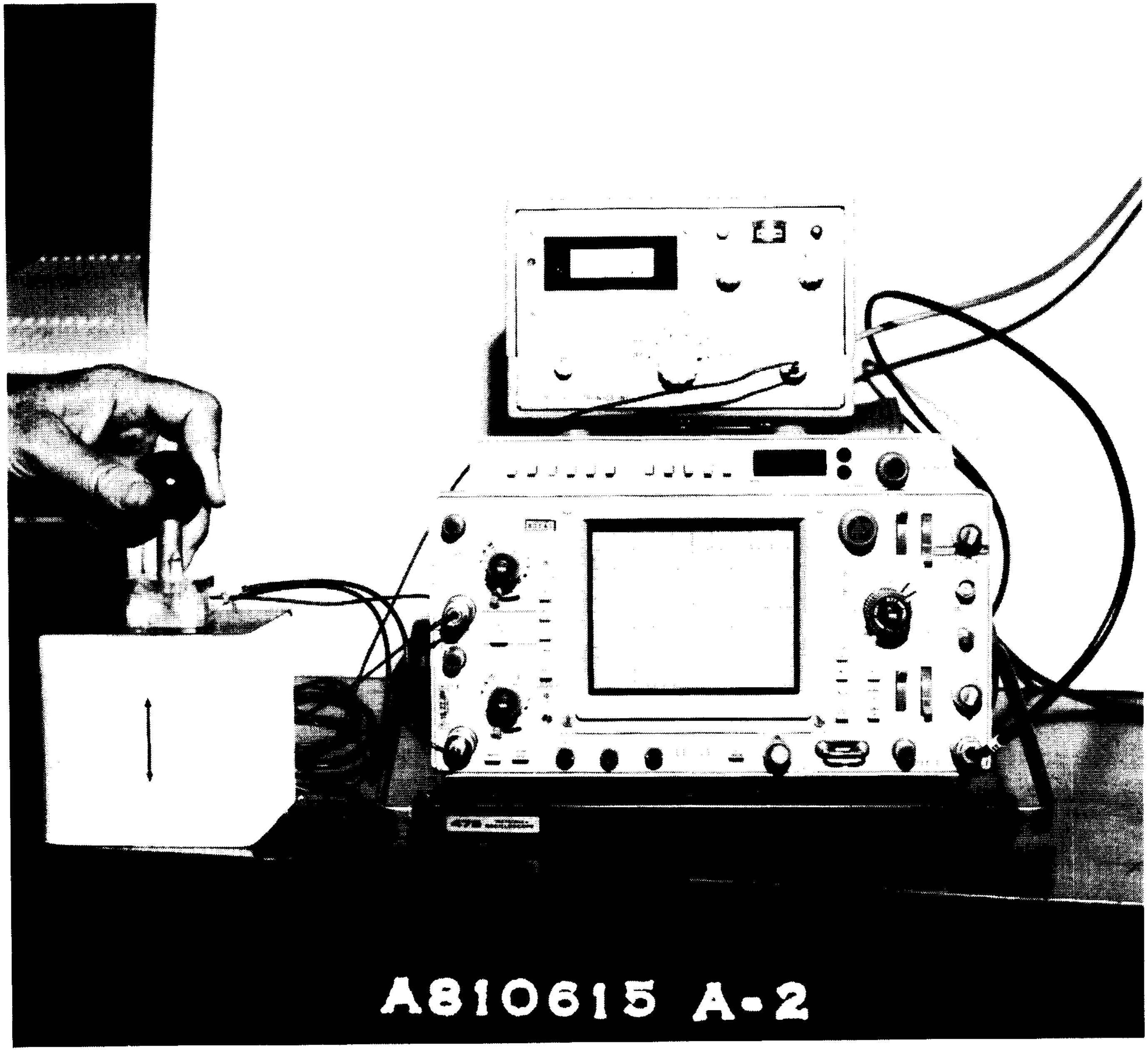


Figure 23. This instrument was developed to determine fiber orientation of installed, coated tiles.

THERMAL PROTECTION SUBSYSTEM TILE PROFILOMETER

(RSISS)

MSC-20389

The device shown in Figure 24 was originally used to measure the contour of the tile surface of the Orbiter. There was no previous method available for automatic, hands-off scanning of the Orbiter HRSI tiles. The profilometer, in a single scan of 50 in., could provide a print of the variation in the tile surface, defects, edge radii, as well as steps between tiles. Gaps between tiles were also indicated. Figure 4E gives a view of the mold line step criteria for the initial TPS of Orbiter Columbia. An average of ten tiles were scanned in one pass of the profilometer.

The profilometer was made with a captive steel ball stylus and a shaped stylus guide to interface with the surface being measured. Three styli with linear variable differential transformer sensors are used with a three-pen analog recorder to trace three scans simultaneously. The three styli are fixed in-line and are 0.025 in. apart on the tile surface. This tool and technique are applicable to measure any type of large, contoured delicate or nondelicate surface in which the detection of small defects or smoothness of variation is critical.

The current device used to perform the measurement of steps and gaps during maintenance operations is described in a NASA Tech Brief article titled " Measuring and Plotting Surface-Contour Deviations," Vol. 11, No. 9, pp. 83-84, October 1987.

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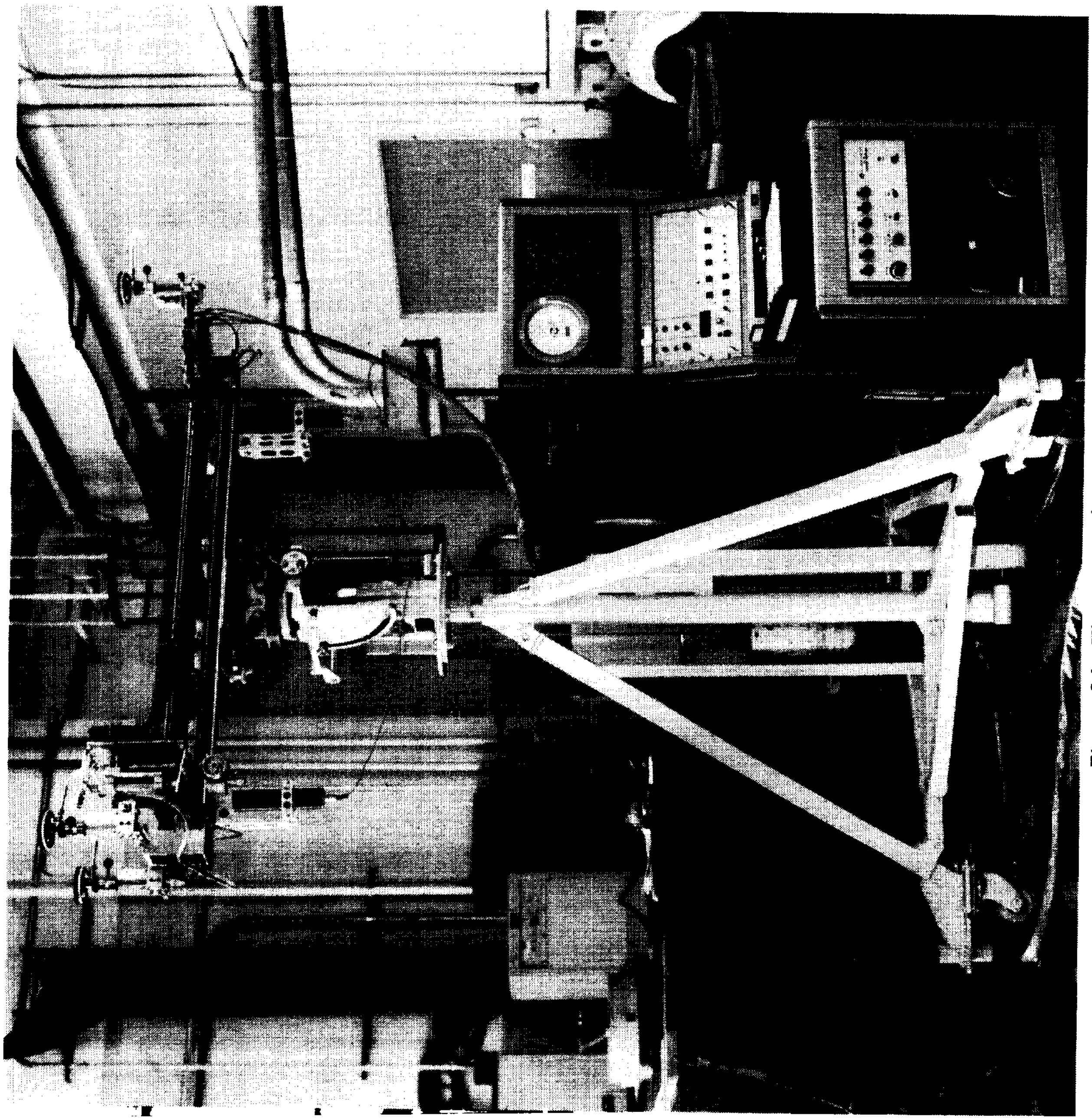

Figure 24. The tile profilometer.

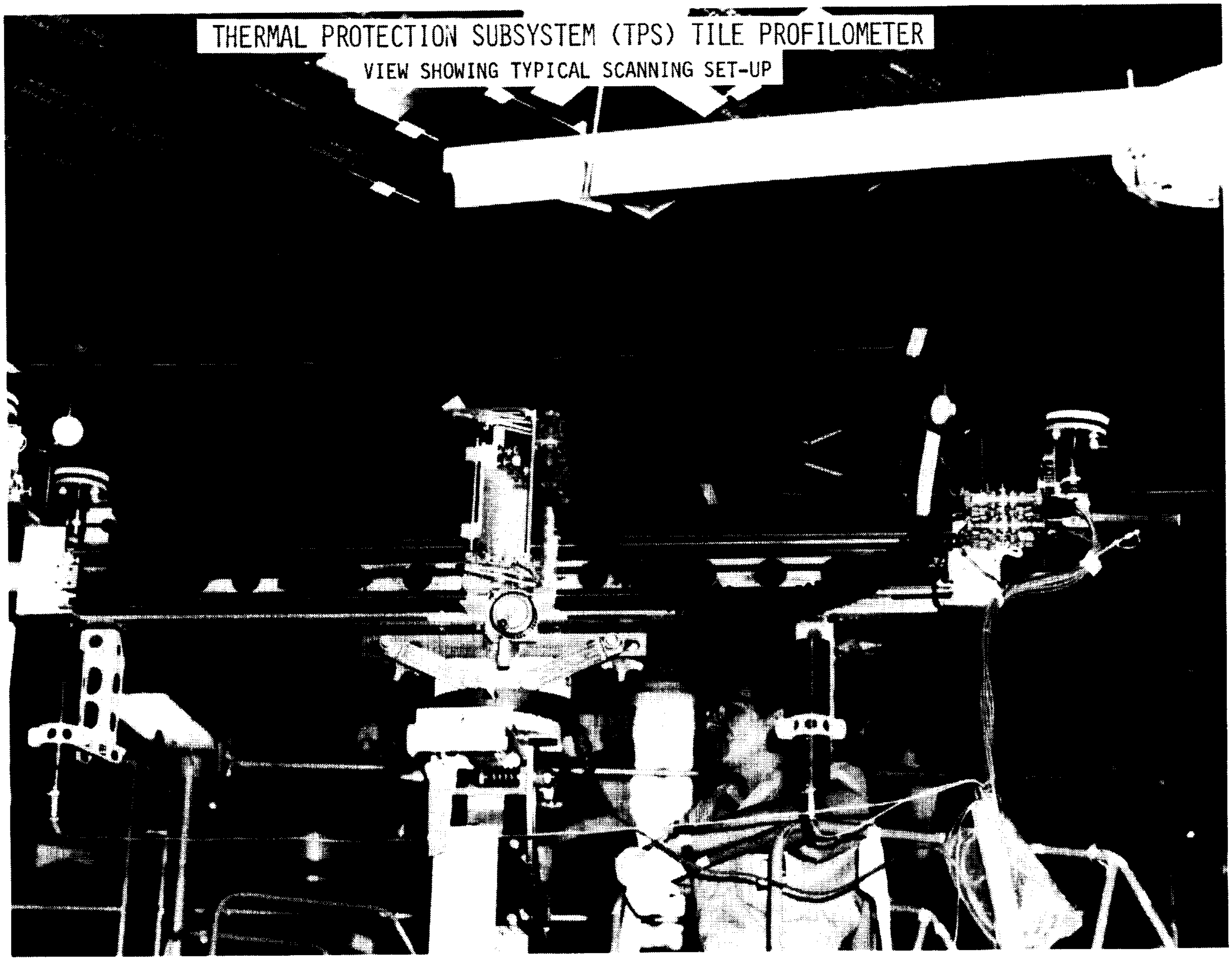

Figure 25. The tile profilometer is used to ensure that the tile outer mold line (OML) conforms to the designed aerodynamic contours.

Measuring and Plotting Surface-Contour Deviations

An electromechanical apparatus provides along-track and across-track displacement information to a plotter.

Lyndon B. Johnson Space Center, Houston, Texas

A hand-held device measures the deviation of the contour of a surface from a desired contour and provides an output to an x-y plotter. A carriage on the device is rolled along a track that represents the desired contour, while a spring-loaded stylus on the device deflects perpendicularly to the track to follow the surface (see Figure 1).

The stylus is connected by a pivot-arm mechanism to a linear-voltage displacement transducer (see Figure 2). Displacements of the stylus as it follows the contour are thus transformed into an electrical signal that is applied to the y-axis input of the plotter.

Two nylon rollers ride on the track, which is 5 ft (1.5 m) long and made of aluminum. The track is maintained at a 0.75-in. (1.9-cm) offset from the desired surface. The track is equipped with pads and handholds for manual placement and adjustment. The carriage also has a handgrip so that the operator can move it easily.

A rubber wheel protrudes through a cut- the wheel turns a potentiometer shaft. The potentiometer resistance thus changes continuously as the stylus moves along the contour, providing a signal for the x input of the x-y plotter. The wheel and the two rollers provide three-point support for the carriage.

This work was done by Lino A. Aragon, Thomas Shuck, and Leroy K. Crockett of Rockwell International Corp. for **Johnson Space Center**. *For further information, Circle 33 on the TSP Request Card.*
MSC-21163

Figure 1. An **Operator Moves the Carriage** of the contour-measuring device on a beamlike track. A stylus on the carriage traces the contour of the surface above it.

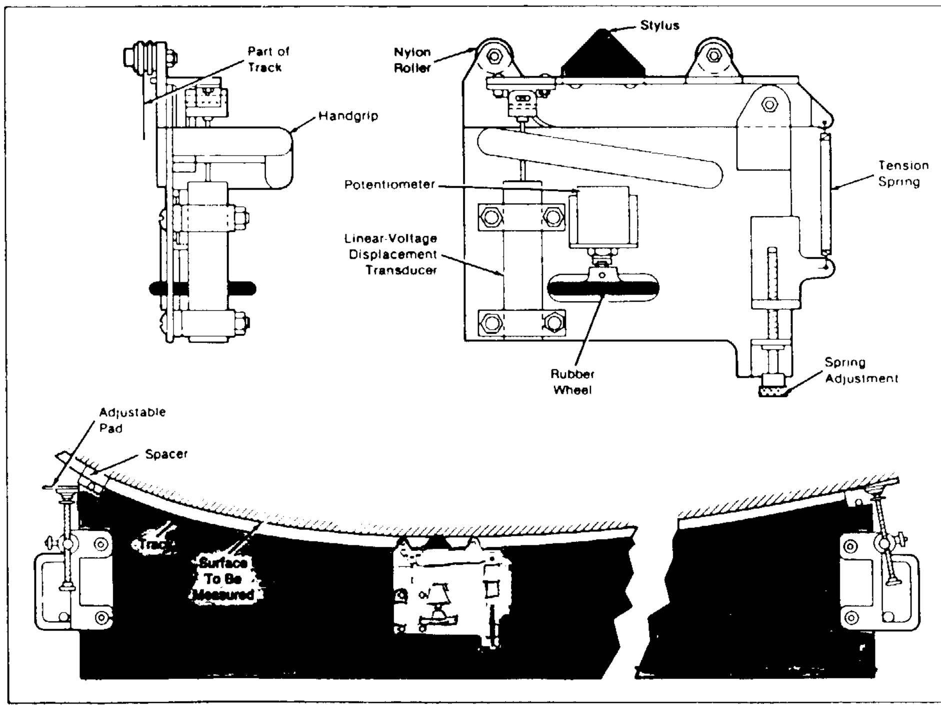

Figure 2. The **Carriage of the Measuring Device** holds a transducer that measures the cross-track displacement of the surface from the desired contour, and a multiple-turn potentiometer that measures the position along the track.

FILLER BAR TO TILE GAP GAGE

(RSISS)

MSC-20781

Ideally, tiles should rest on the filler bars to provide the seal required to prevent water and hot plasma gas from reaching the tile-RTV-SIP interface. Although the installation process is highly successful in achieving this seal, sometimes situations exist where there is an excessive gap condition as illustrated in Figure 26. Additionally, this gap condition creates unwanted "surface steps" or abrupt changes in surface profile, the nature of which is unknown to an outside observer.

With approximately 120,000 filler bars-to-tile interfaces in the OV-102, Orbiter Vehicle Columbia, spacing between tiles as small as 0.025 in., combined with depth of the slot of up to 5 inches, the task of examining these interfaces was formidable. For this reason, there was a need for a reliable method to quickly and qualitatively screen all the tiles-to-filler bar interfaces. Interfaces which failed the qualitative screening would then be examined using more rigorous quantitative inspection methods.

The device presented in Figure 28 was designed to detect gaps between the tile IML and filler bars quickly and reliably. It consists of a blade element equipped with two fiber optic bundles for sending and receiving light from a narrow mirror fixed to the end of the blade. The received reflected light decreases in proportion to the gap size, and thus, a means of indicating out of tolerance interfaces is obtained. This device could have applications in the inspection of inaccessible areas, such as the sides of deep narrow slots for finish irregularities and defects or bond delamination between a thick surface and a substrate.

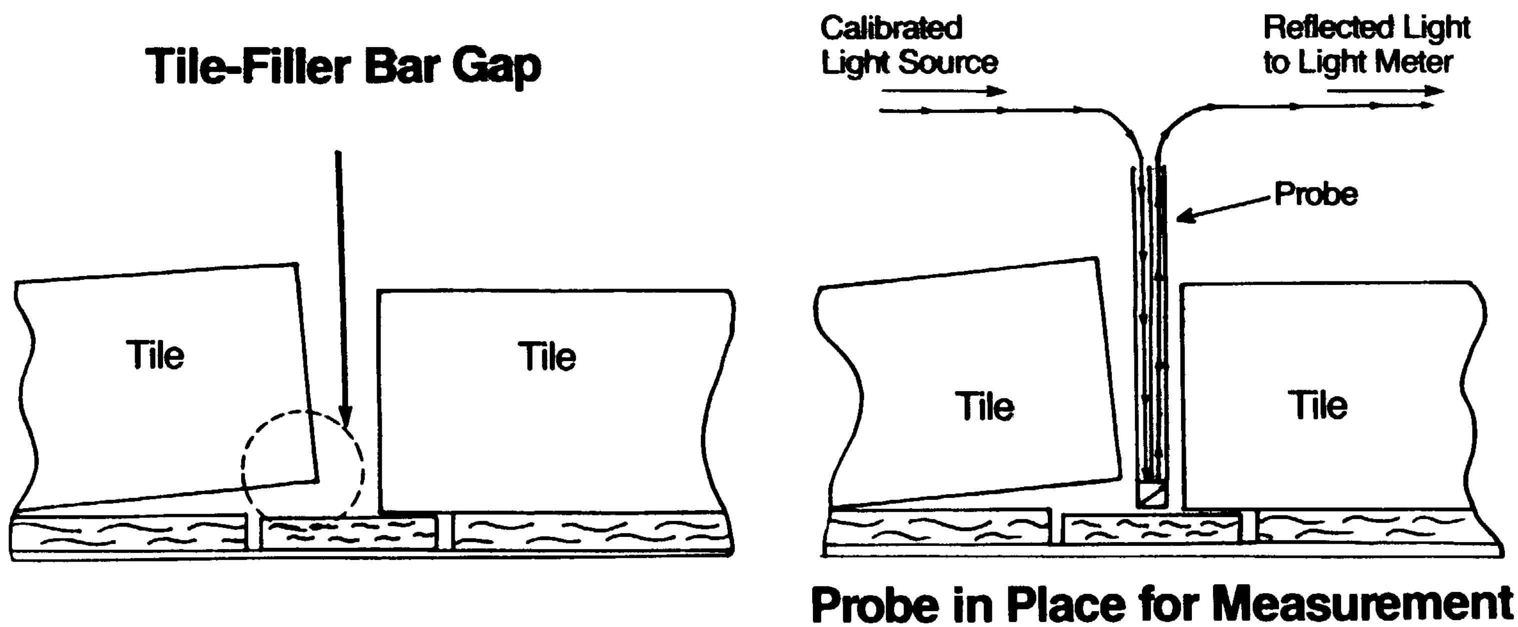

Figure 26. Fiber optic instrument developed to qualitatively measure tile to filler bar gap conditions.

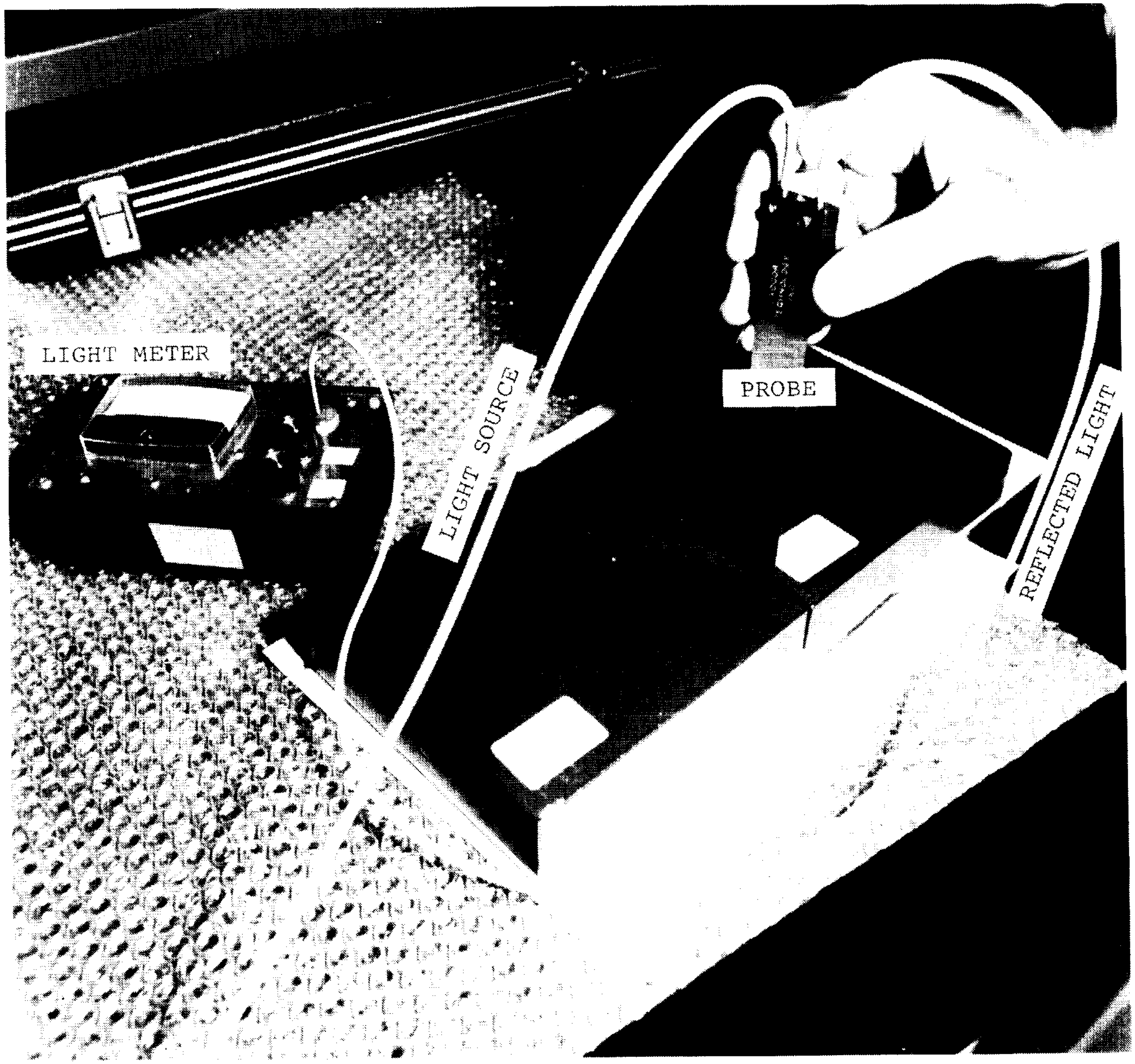

Figure 27. Tile to filler bar gap measuring instrument during a simulated use condition.

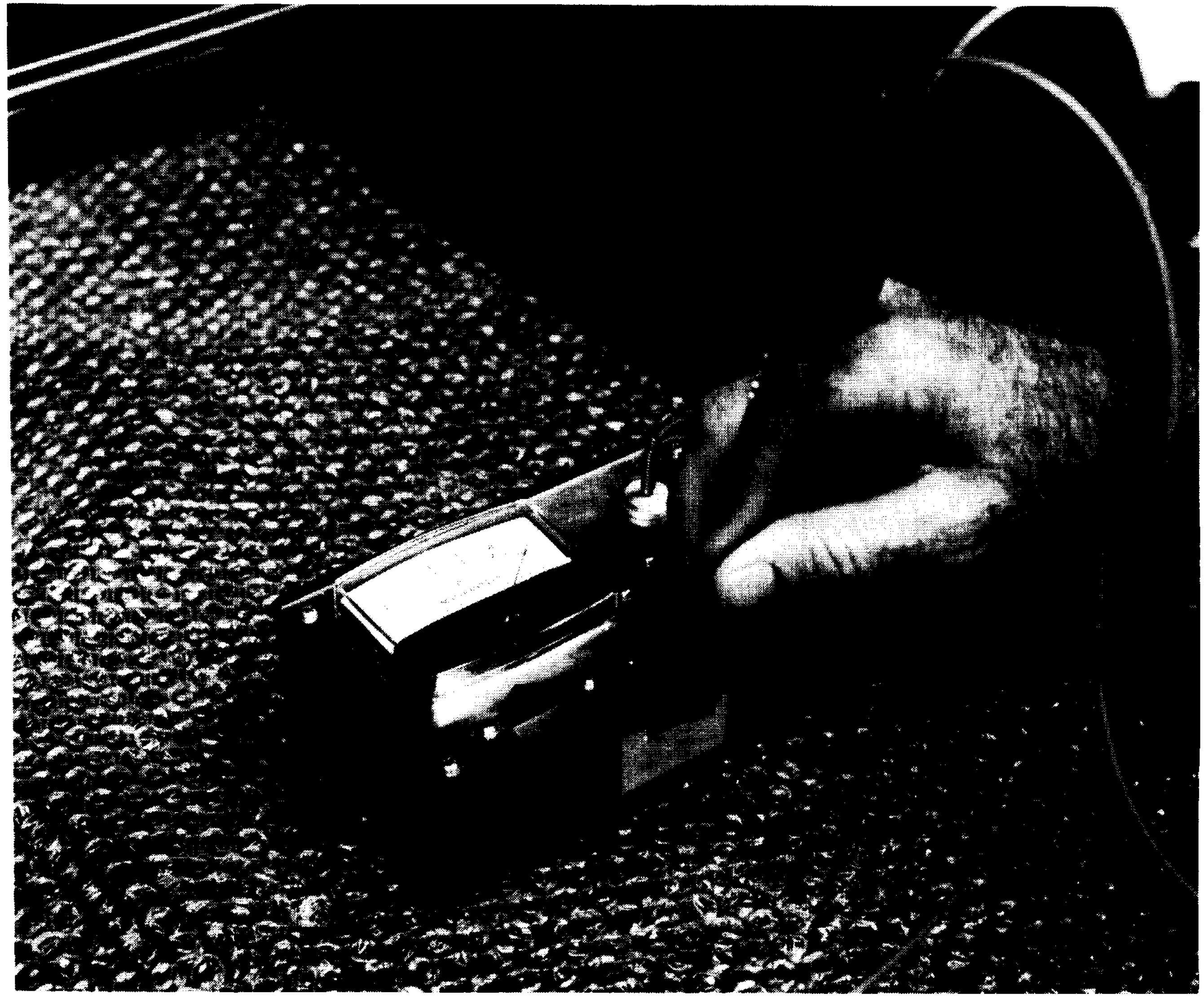

Figure 28. Tile-filler bar gap measuring instrument.

TILE GAP MEASUREMENT INSTRUMENT - SLIDING WEDGE TYPE

(RSISS)

MSC-19637

The gap between tiles, particularly those around the high pressure areas of the Orbiter, had to be within the specified limits of 0.050 ± 0.020 in. This gap requirement was necessary to prevent the infiltration of hot plasma gas into the tile-RTV-SIP interface which would result in debonding of the tile and consequently protection loss of the aluminum structure. Because of the many tile-to-tile gaps needing inspection, an instrument was required that would make these measurements easily, quickly, and reliably. Commercially available measuring devices proved unsuitable for mass measurement of these gaps, particularly where such surfaces were in the form of negative or positive tapers as well as parallel, Figure 29A. If tile-to-tile gaps had been few in number with only parallel sides, commercially available wire gages could have been used. Other instruments were also designed for this purpose using electronic type feeler gages. However, these were not easy to use and required expensive electronic hardware.

The measuring device illustrated in Figure 29 is entirely mechanical, lightweight, and shaped for easy manipulation at any hand attitude. It is comprised of a fixed and a sliding member with a contact wedge. The sliding member is affixed to a dial indicator, and is thumb actuated by a plunger via a spring-loaded override mechanism to avoid undue loading on the delicate tiles. This probe is designed to measure negative and positive tapered gaps as well as parallel gaps.

Although the sliding wedge action of this gage is believed to be a new feature, its commercial use would probably be limited to special applications such as quantitative and instantaneous measurement of a large number of gaps. Also it could be used to measure shaft end play and horizontal die alignment.

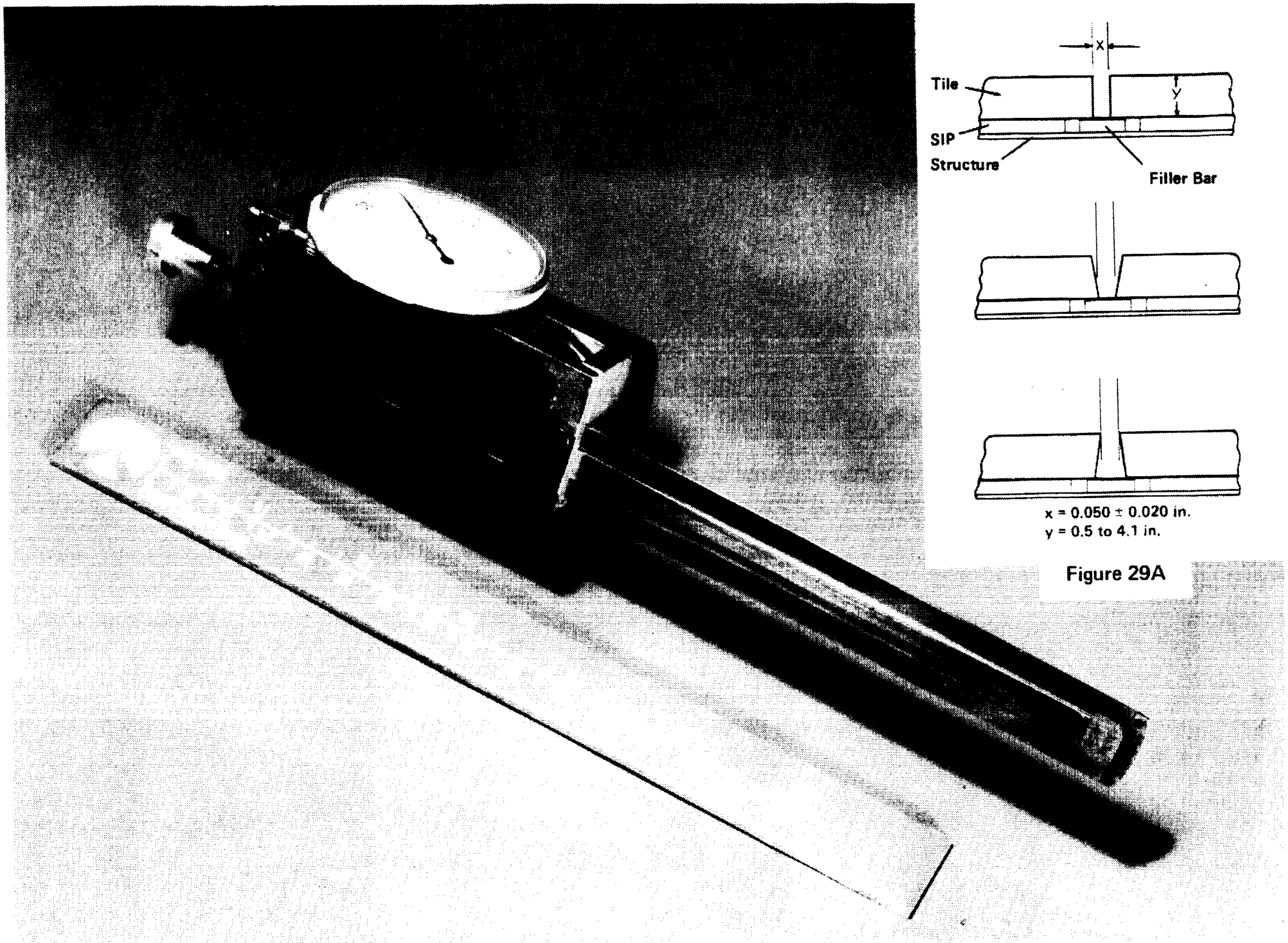

Figure 29. Sliding wedge-type gap measuring instrument. Figure 29A is a diagram of the various tile gap configurations.

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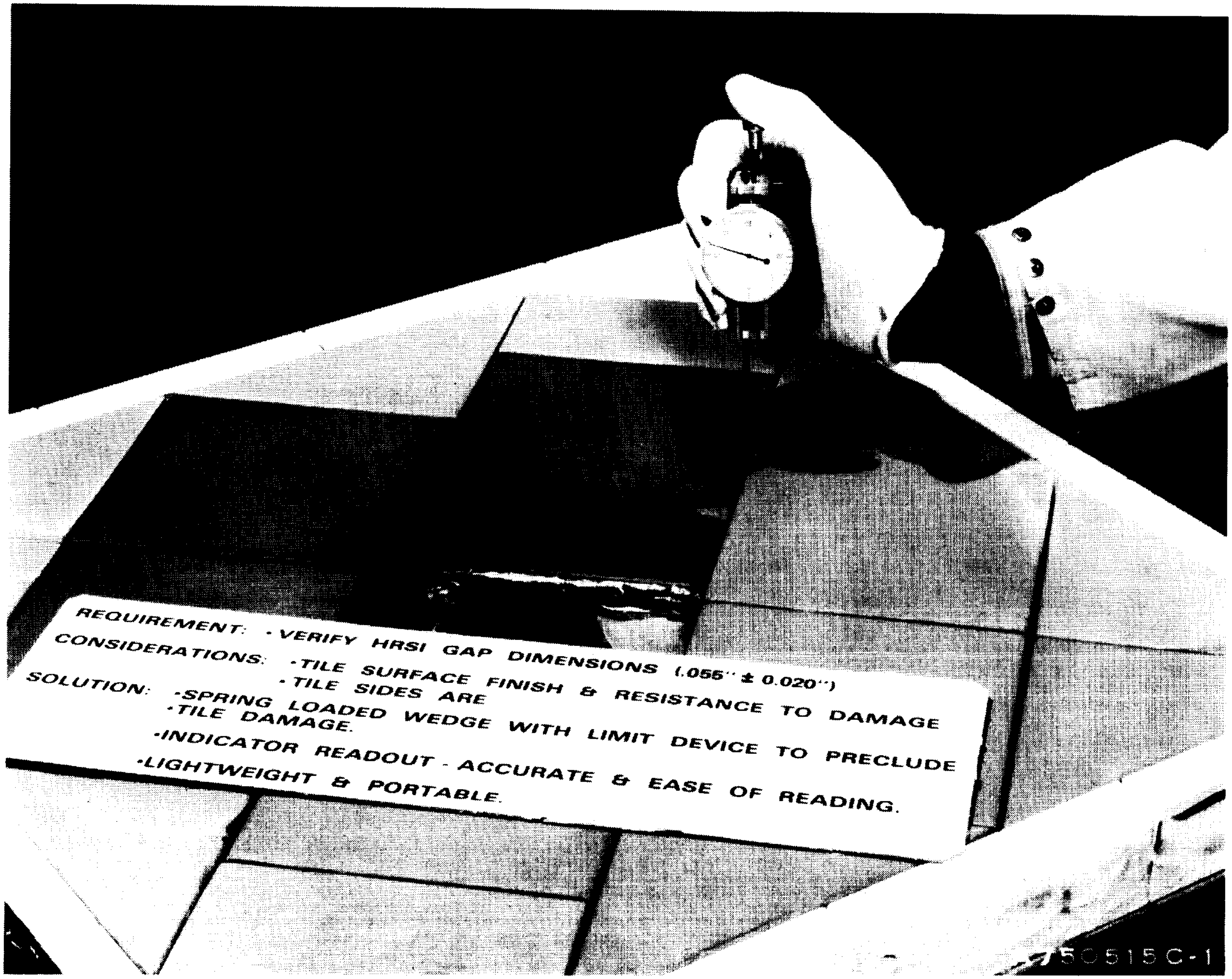


Figure 30. Sliding wedge-type gap measuring instrument during simulated use condition.

Gage Measures Recessed Gaps

Tool permits fast and easy measurements.

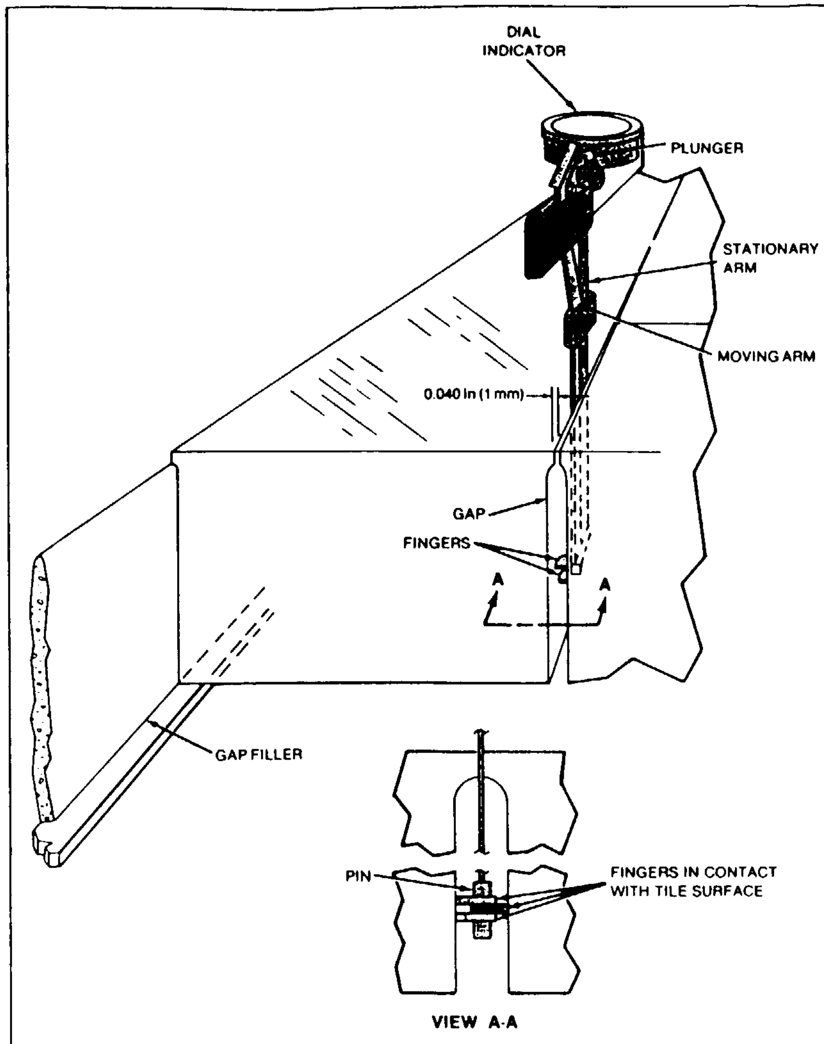
Lyndon B. Johnson Space Center, Houston, Texas

A new tool measures the separation between recessed parallel surfaces. The tool was developed for measuring the gap between adjacent tiles on the Space Shuttle. Because the tiles have overhanging edges, the tool is designed to slip into the gap from the end so that it extends through a 0.040-inch (1-millimeter) crack (see figure). It measures gaps between 0.200 and 0.400 inch (5.1 and 10.2 millimeters) so that gap fillers of the proper thickness can be selected.

The tool consists of two arms 0.030 inch (0.76 millimeter) thick having a three-finger gaging mechanism at the bottom and a dial indicator at the top. A pin attached to the moving arm passes through a slot in each finger (see figure). The pin-arm motion causes the fingers to fan out until they touch the sides of the gap. The moving arm causes movement of the plunger on the dial indicator that is attached to the other arm.

The new tool makes it unnecessary to use feeler gages to measure the gap. The feeler gages had to be used with great care to avoid damaging the tile; the new tool is both faster and gentler and should prove useful in numerous industrial situations involving gap measurements in inaccessible places.

This work was done by Jose L. Zepeda of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 57 on the TSP Request Card.*
MSC-20230



The **Fingers Spread,** the scissors arms open, the fingers rock, the dial plunger slides, and the gap at the base of adjacent tiles is measured.

TILE LOAD VERIFICATION, PULL STRIP METHOD

(RSISS)

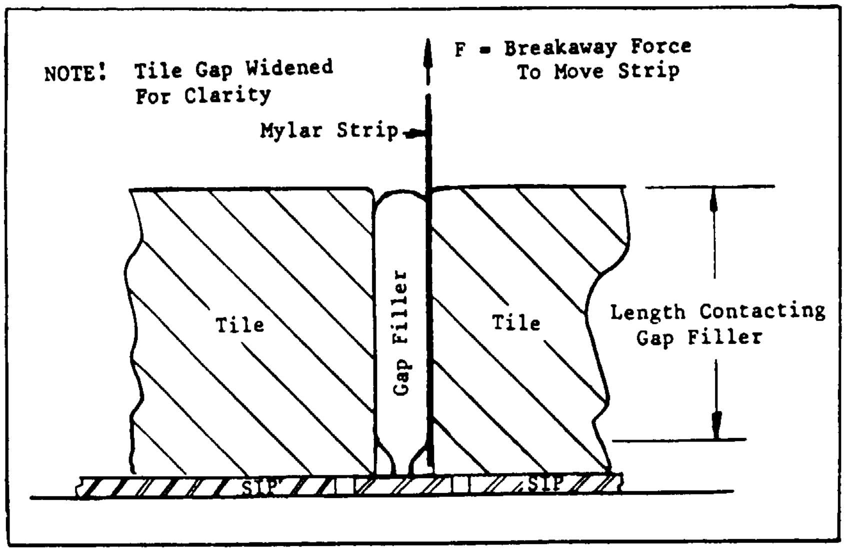
MSC-20231

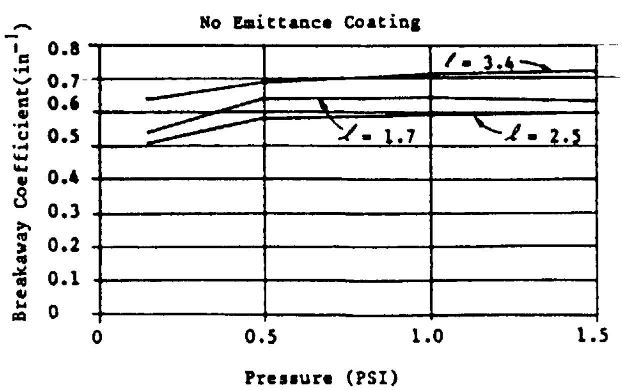
The device presented in Figure 31 is used to verify the size of pillow-type gap fillers between tiles. It consists of 3 in. wide Mylar strip sandwiched between the tile and the gap filler. Pulling on the Mylar strip discloses the breakaway force needed to move the strip.

Initially, the gap fillers were tested in compression prior to installation in the vehicle to make sure the compressed size of the filler would firmly fit into the gaps. This method was very time consuming and created a high degree of scrappage.

The pull-strip method eliminated the need for tile gap measurements (6 per tile interface), reducing handling, improving flow time, and accuracy. Acceptable gap fillers were ready for immediate bonding without additional processing improving the yield to 95%.

This concept can be used to determine interface pressure or loads where conventional load measuring equipment cannot be used. It is well suited for low cost, high volume quality control.





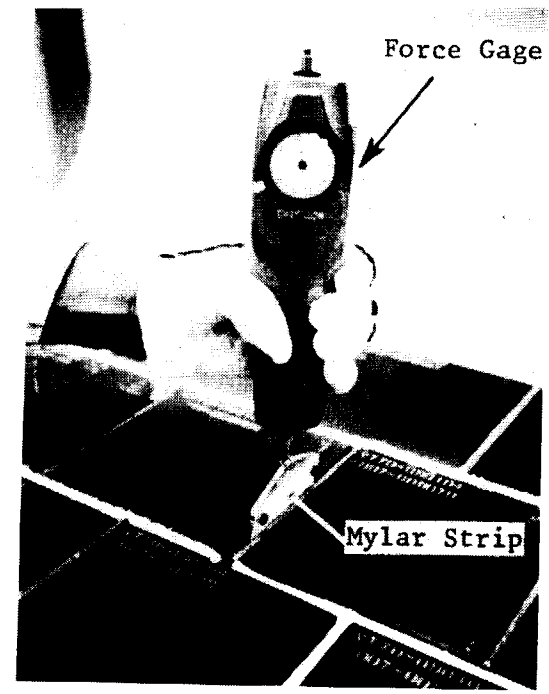


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Figure 31. Pull strip method developed to test for proper size of the pillow-type gap fillers.

INFLATABLE PROOF TESTER FOR TILES

(RSISS)

MSC-18910

The device shown in Figure 32 was designed to test the bond integrity of installed tiles to the SIP in areas of the Orbiter which were flat or have a curvature of 28 in. or more in radius. This tile pull test device consists of a vacuum chuck and an inflatable ring used not only to provide the required lifting force of 2 lb/in. but also to protect adjacent tiles from mechanical damage during the test. Although originally employed to proof test the tiles in the OV-102 Orbiter, its use was discontinued because of the difficulties encountered in maintaining the required vacuum during the test. However, the concept could be adapted for any commercial application where incremental movement between two units is desired for alignment, adjustment or positioning with finite control and low reaction force.

For tiles that deviate from a relatively flat OML, other devices were developed such as the "Articulated Vacuum Chuck" and the "Universal Vacuum Chuck" (MSC-18933 and MSC-20666, respectively). Both devices apply the required tension load using the same technique. The chuck is attached to the tile surface using a vacuum and a mechanical tension load is applied to the tile through the chuck. The difference between these instruments is that the articulated vacuum chuck device required the fabrication of a custom chuck for each tile while the universal vacuum chuck device could be adapted to fit most special tile surfaces with one chuck. Originally, the individual chuck for the articulated vacuum chuck device were to be shaped by a milling machine from the same computer-numerical-control tapes used to generate the tile surface. However, the development of a splash molding technique (MSC-20795) made this expensive and time-consuming procedure unnecessary.

Both of these devices, however, exhibited the same problems encountered with the inflatable proof tester. Mainly that it was very difficult to maintain and sometimes achieve the required vacuum due to imperfections in the coating such as pinholes and cracks. Also, improper alignment between the chuck and tile center of gravity did not provide accurate proof loads.

The method currently employed for testing the integrity of the Tile-RTV-SIP assembly is the hot melt adhesive chuck device (MSC-20672 and MSC-20685). The chuck is attached to the tile using a hot melt adhesive. After testing, the adhesive is heated and re-melted and the chuck is removed. This technique has provided the best testing of the tile-RTV-SIP assembly regardless of surface shape and condition of coating.

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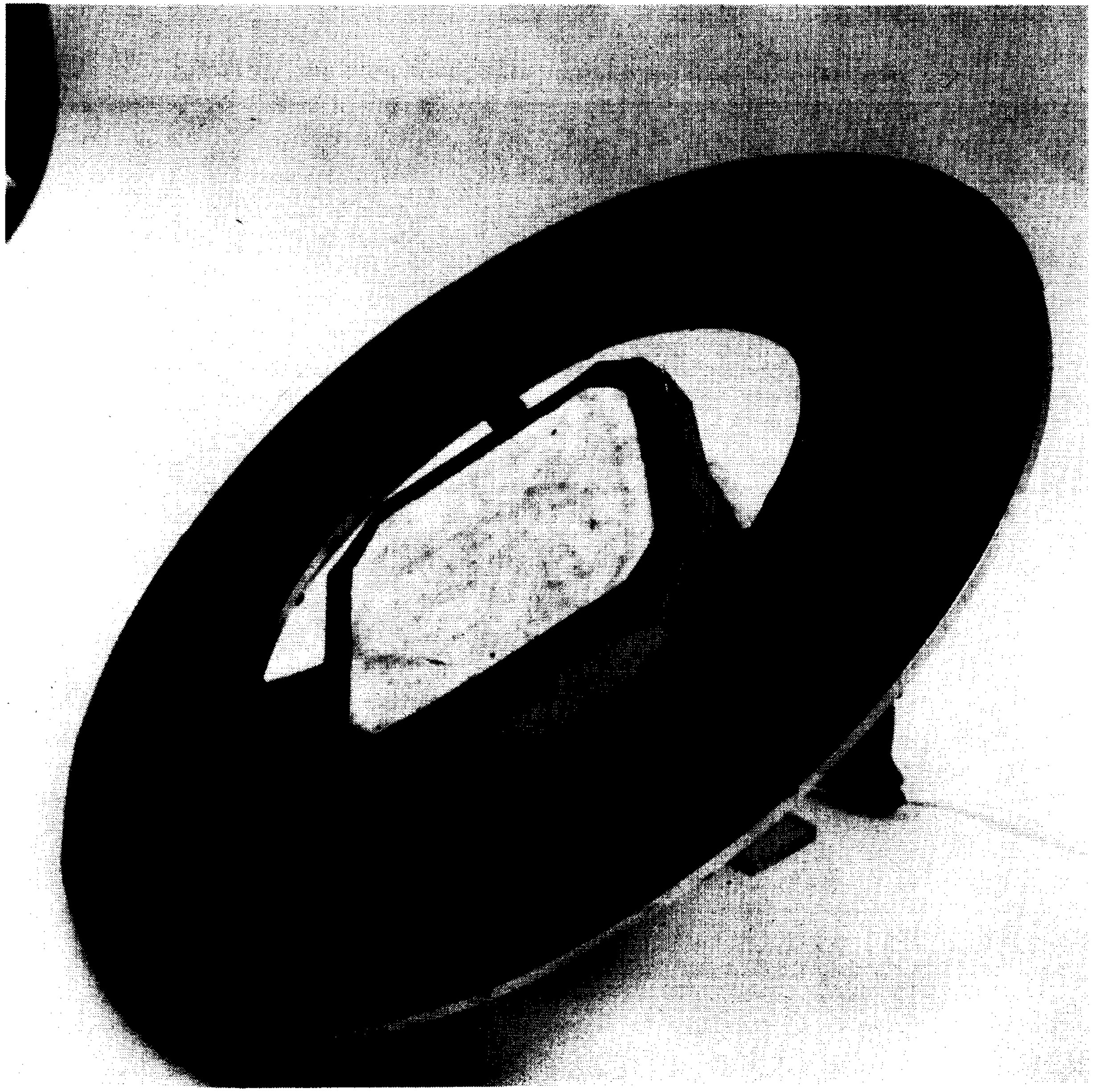


Figure 32. Inflatable proof tester developed to test the tile-RTV-SIP assembly strength after installation on the Orbiter surface.

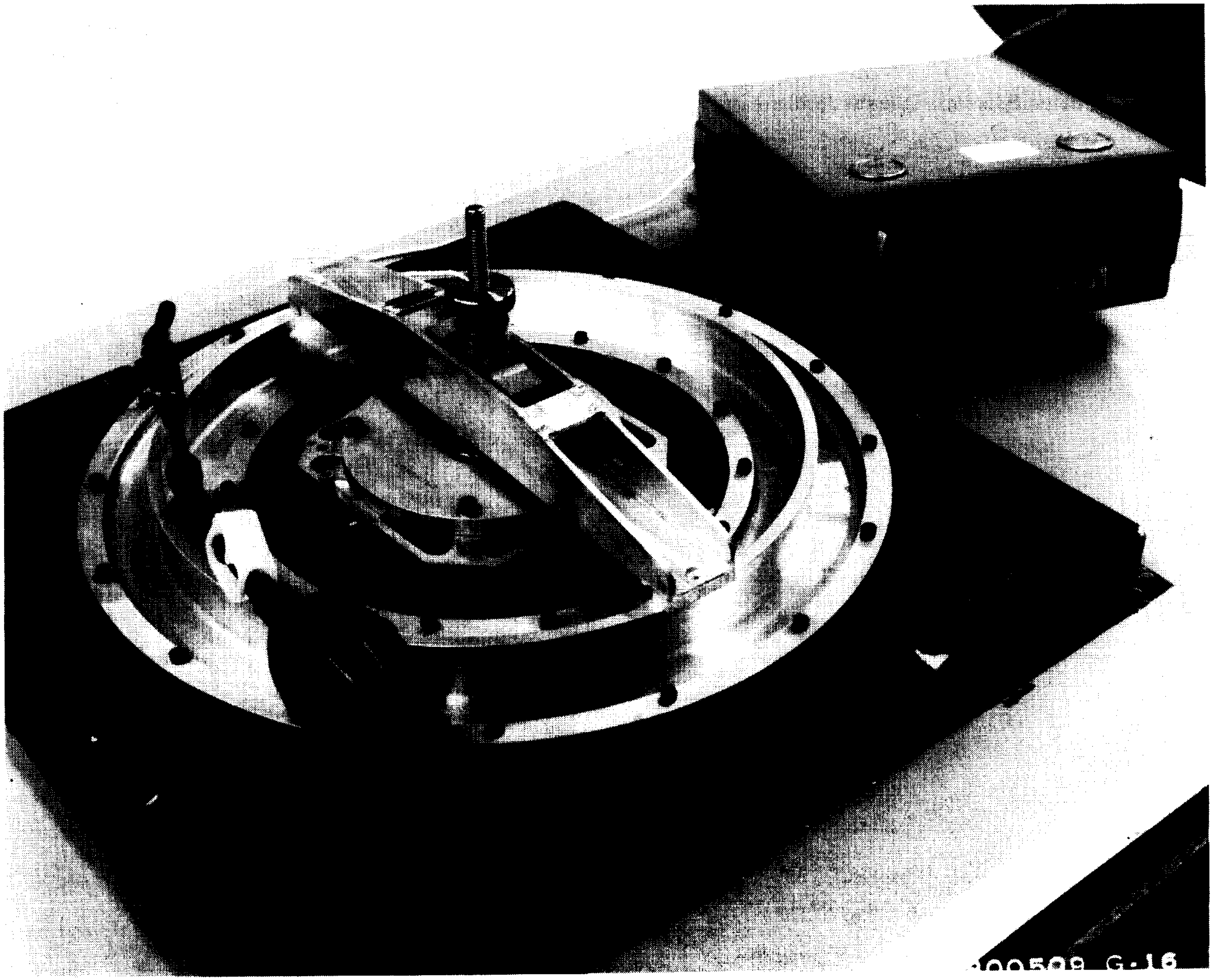


Figure 33. Inflatable proof tester during a simulated use condition.

Articulated Vacuum Chuck

A pull tool conforms to the work surface.

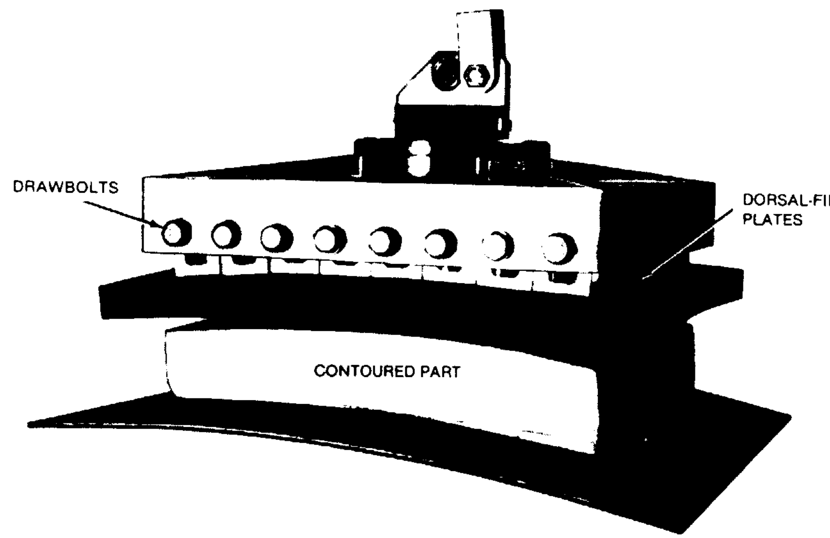
NASA Tech Briefs, Fall/Winter 1981

Lyndon B. Johnson Space Center, Houston, Texas

A vacuum chuck originally developed to pull-test Space Shuttle surface tiles conforms to complex surface contours. Its gripping surface is a polyurethane panel embedded with links of roller chain. The panel flexes under vacuum to adjust to the surface contour, and then bolts are tightened to lock the configuration. Possible applications of the new chuck are in pull-testing contoured surfaces, holding assemblies together for repairs, or for handling unusually-shaped parts.

As shown in the photograph, eight hexagonal-cap bolts can be tightened to lock an array of slotted plates after vacuum is applied. (The vacuum line is not visible in the photo.) In all, there are 112 plates arranged in 8 groups of 14 each. The bottoms of the plates are linked by lengths of roller chain embedded in the polyurethane. A load can be applied to the clevis at the top of the chuck.

This work was done by Scott A. Peterson of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 49 on the TSP Request Card.*
MSC-18933



Articulated Vacuum Chuck conforms to the contoured part. The vacuum is applied, and then the drawbolts are tightened, freezing the chuck shape with that of the work.

FLEXIBLE MATERIAL STEP DIFFERENTIAL MEASUREMENT DEVICE

(RSISS)

MSC-20736

The use of Advanced Flexible Reusable Surface Insulation (AFRSI) as insulation material on the external surface of the Orbiter produced several thermodynamic and aerodynamic challenges. Abrupt changes in surface features, such as steps or thickness, (arrows in Figures 34 and 35) produced aerodynamic turbulence which tended to abrade the fabric and increase local heating during flight.

In order to prevent these surface irregularities, a method had to be developed to measure any difference in the thickness of the soft, fluffy material at any interface butt joint of two panels. The device designed to perform this task was a self-contained, hand held, electro-mechanical, and battery operated instrument, Figure 34. The instrument employs two spring loaded "feet" suspended from a bridge support assembly used to measure the differential levels between two planes on the surface of the AFRSI blanket. Potential uses of this principle are to measure the thickness, step, and slope of any soft, highly compliant material such as insulation, foam rubber, etc.

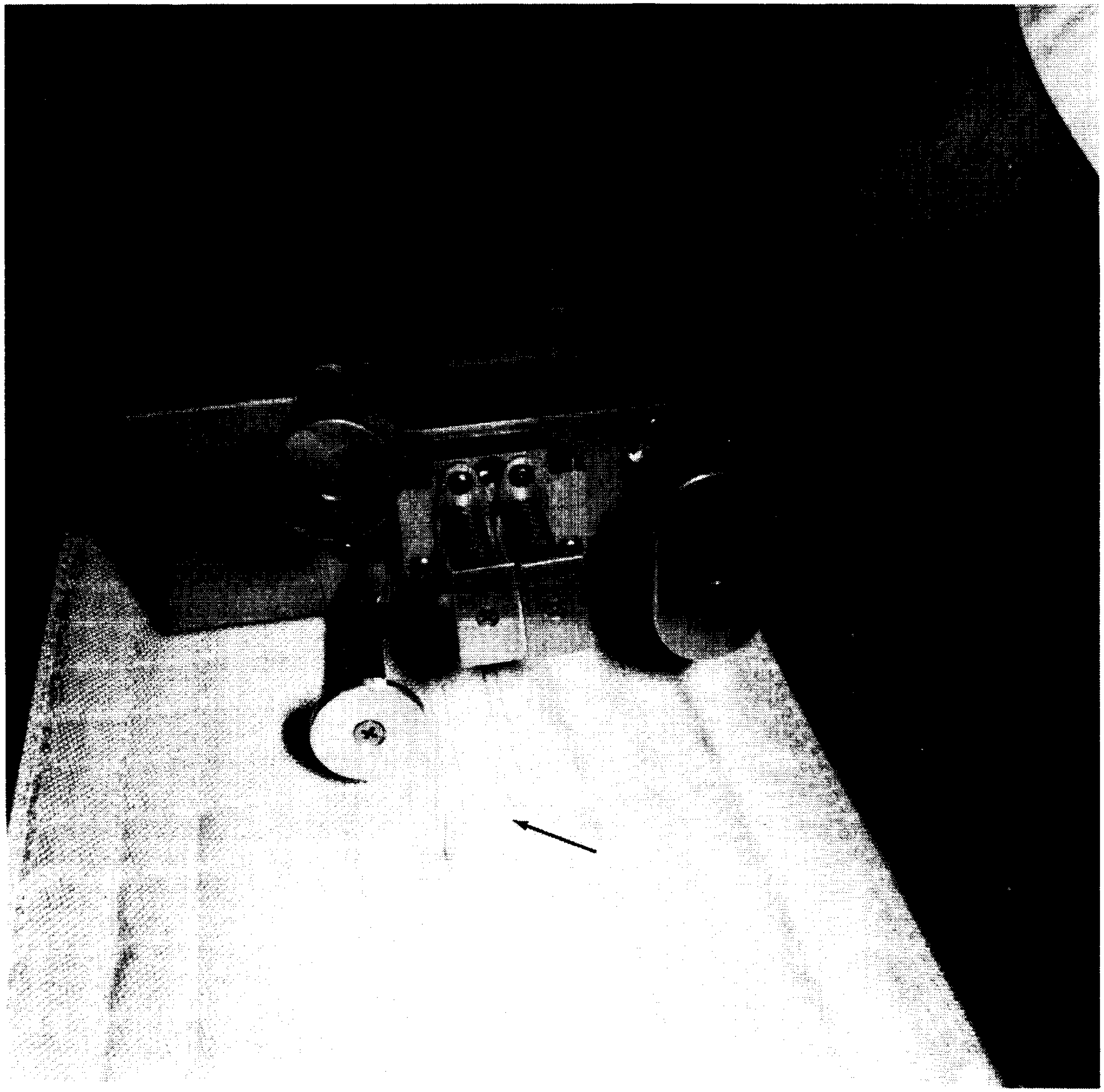


Figure 34. Instrument developed to make step differential measurements on AFRSI.

Figure 35. Step differential measurement device during use on AFRSI thermal blankets.

Applications

APPLICATIONS OF MATERIALS DEVELOPED FOR THE RSISS

Some of the potential applications for the RSISS materials are presented in the following page. Obviously, these materials are excellent thermal insulators and thus they should have many applications in this field. However, their use in this area will probably be limited due to their high cost. For example, a finished coated LI-900 tile (6 in. by 6 in. by 2 to 3 in.) costs around $2,000. An unmachined billet of tile material 13 in. by 13 in. by 5 in. costs about $1000. A square foot of a typical insulation board of about the same thickness and with equivalent effective thermal conductivity (K_{eff}) in air can be obtained for 5% of the billet cost. The cost of the AFRSI blanket is about $100 for a 12 in. by 12 in. by 0.5 in. section. The cost of the FRSI blanket is about $150 for a 12 in. by 12 in. by 1.0 in. section. All cost figures are based on 1988 dollars.

For tile materials, another limitation is their low ultimate use temperature (2000°F) under continuous use conditions as compared to the high-temperatures encountered in industrial processes such as steel forming. This should not be a problem; however, if the tile material can be used as a back-up insulation to a hot face insulation material which can reduce the temperature to the required limit for the tile.

The high purity of tile materials makes them excellent candidates for applications requiring a high degree of purity control such as in microelectronics, biotechnology, or other manufacture of very pure materials. One interesting potential application is tile use as a high temperature filter for liquid metals.

Another possible application for the tile materials is to allow manipulation of laboratory ware at high temperature. A hot object encapsulated with a holder made of a tile material would cool so quickly upon removal from a furnace that, if desired, it could be held with bare hands almost immediately. The operator could then use thinner insulation gloves with better dexterity which would improve the handling control of the hot object.

Finally, the best applications for the tile materials as they are currently produced are as thermal conductivity and black body radiometric standards (see NASA Tech Briefs, PO-13830, Vol. 2, No. 2, p. 208). These materials are truly the state-of-the-art in high temperature ceramic fiber thermal insulation.

Applications of RSISS Materials

Thermal Insulation

- Transportation systems
  - Spacecraft
  - Aerospace plane
  - Terrestrial systems
- Furnaces
  - Microelectronics
  - Manufacture of high purity materials
  - Rapid heating and cooling applications
- Miscellaneous
  - Holder for hand protection from hot laboratory ware

Filtration

- Liquid metal filters

Calibration Standard

- Thermal conductivity standard
- Blackbody radiometric standard

SOME UNIQUE PROPERTIES OF TILES AND COATINGS (RSISS)

Qualities of Tiles

- Tile materials are made from extremely high purity ingredients. This fact makes them the highest purity thermal insulation materials available.

- The physical and chemical properties of the tiles are remarkably homogeneous throughout the material. For example, the strength of an undensified tile material remains fairly constant through-the-transverse direction regardless of its thickness size. Very few commercial insulating materials exhibit this characteristic. This characteristic is due to their high degree of purity, their manufacturing procedure, and the rigorous quality control standards they must surpass during manufacture.

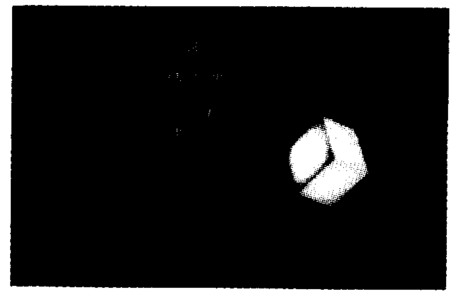
- The Shuttle rigid RSI has strength properties approaching those of an insulation brick. At the same time, it has thermal insulation properties as good as those of refractory fiber board or insulating blankets.

- LI-900, LI-2200, and FRCI-20-12 tiles have the best thermal shock resistance of any available ceramic, rigid insulation. For example, FRCI tiles at 2400 ^{o}F can be introduced into liquid nitrogen (-320 ^{o}F) successively for 100 cycles without failure.

- The tiles have the lowest effective thermal conductivity of any rigid thermal insulation materials.

- The tiles have the lowest effective thermal conductivity of any available thermal insulation material operating in near-vacuum environments and where heat transfer is almost entirely by thermal radiation. In air, tiles insulate similarly to the best commercially available non-rigid insulators.

- The tiles are the lowest density rigid material exhibiting anything like their thermal insulating properties. The low density contributes to their very high specific strength and low heat capacity, or storage. Consequently, the tiles heat up and cool down quickly. For example, a tile heated to 2000 ^{o}F can be grasped in the bare hand a few seconds after removal from the furnace even though it is still glowing from internal heat. This is possible because heat is dissipated so rapidly at the surface, that this surface behaves as its own insulator to the remaining internal heat.



Limitations of Tile Materials

- The maximum continuous use temperature of tiles is about 1850 ^{o}F; the maximum intermittent use temperature is 2370 ^{o}F. Tiles creep under low loads at these temperatures.
- The tiles are delicate, and do not have high impact strength.
- The properties of the tiles will be adversely affected in the presence of very contaminated industrial atmospheres. Thermal properties of this material are dependent on its high purity.
- These materials have, for the most part, been tested continuously for only a limited number of hours (e.g., 16 to 24 hours) at high temperatures (2350 ^{o}F).

Qualities of Tile Coating Materials

- The NASA tile coating reaction cured glass (RCG) or Class 2 coating is the most physically and optically stable ceramic coating in aeroconvective (e.g., plasma arc) environments up to 2700 ^{o}F.

- RCG or Class 2 (black borosilicate glass) coating is cured at a lower temperature (2200 ^{o}F) than its continuous use temperature (>2350 ^{o}F). This means that a RCG coated substrate can be first cured at a lower temperature where shrinkage is less and dimensional tolerances can be better maintained and then used at a higher temperature.

- No devitrification has occurred on RCG specimens after 800 hours of exposure to 2300 ^{o}F in air. Devitrification is the change of silica from an amorphous to a crystalline structure, with a corresponding large increase in thermal expansion coefficient. This change in thermal expansion coefficient greatly reduces thermal shock resistance.

- The Class 2 or white borosilicate glass coating invented by LMSC, is equally stable at temperatures up to 1200 ^{o}F. Class 1 coating has an emittance of 0.7 at 1200 ^{o}F. While at room temperature it achieves a ratio of solar absorptance to emittance of less than 0.4.

Limitations of Tile Coatings

- Tile coatings exhibit low impact strengths. This is due in part to the small thickness requirement imposed on the RSI. Coating thickness requirements vary according to the type and RSI location on the Orbiter. For HRSI (black coated tiles) thicknesses range between 0.009 to 0.015 in. while for LRSI (white coated tiles) thicknesses range between 0.006 to 0.015 in.

Properties of TPS Materials (RSISS)

	LI-900	FRCI	HTP
Density, lb/ft^3	9	12	12
Maximum use temperature, °F	2400	2500	2600
Room temperature compressive strength (TTT), lb/in^2	28	132	141
Thermal conductivity (TTT), $Btu\text{-}in/ft^2\text{-}hr\text{-}°F$ 1 atm; R.T. to 2000 °F	0.3 - 2.0	0.4 - 2.2	0.4 - 2.2

Tiles Can Be Considered as Materials with the Thermal Properties of Fiber Board and the Strength of Insulation Brick

	Fiber Board*	FRCI-12	Insulation Brick**
Density, lb/ft^3	13	12	31
Room temperature compression strength, lb/in^2	30 @ 10% deformation	132	145
Thermal conductivity (TTT), Btu-in/ft^2-hr - °F 1 atm; R.T. to −2000 °F	0.3 -1.8	0.4 - 2.2	0.9 -1.9
Use temperature, °F	2600	2500	2300

*Carborondum, Fiberfrax Duraboard 2600
**Harbison-Walker, H-W 23 insulating fire brick

APPLICATIONS OF TECHNIQUES DEVELOPED FOR THE RSISS

The materials concepts of FRCI, HTP, AETB, FRSI, AFRSI, and TABI should aid the design of other high performance composites. The agents and methods used to render these materials waterproof, some of them which must withstand repeated exposures to a maximum temperature of 1050°F, should have potential for commercialization. Definitely, when bonding a porous, low density material to a fibrous substrate, densification of the bonding surface should be considered to increase the bonding strength.

Any fibrous, high temperature transparent material should benefit from the opacification techniques developed to decrease the effective thermal conductivity (K_{eff}) of the tile at high temperatures. These techniques increase the number of thermal radiation scatter sites by an increase in the number of fibers and fiber joints, and also, by the addition of a small, high emittance particle such as SiC throughout the material.

The number of fibers and fiber joints are increased with the use of the high dry-density concept and by the reduction of the average diameter of the fiber. The high dry-density concept is the technique which increases the density of the unfired billet prior to sintering to as high a value as it is practically possible. The use of this technique decreases the density change that occurs when the unfired billet is sintered. Its use results in a billet with more, smaller diameter fibers per unit volume.

The reduction of the average diameter of fiber is achieved in the FRCI tile material, by a decrease in the content of the larger diameter Nextel 312 fiber. Even though all of these techniques decrease the size of the pores, the overall density of the material is not increased.

The addition of up to 3% of fine SiC particles has also resulted in noticeable improvements in the reduction of the K_{eff}. As expected, the greater the number of these particles or the smaller the size of the particle for a given particle content, the higher the reduction in the K_{eff}.

In addition to the current technique selected to install tiles to the Orbiter, other methods developed for this purpose should have potential for industrial applications. For example, the mechanical method using the auger device (U.S. Patent No. 3,936,927) developed by NASA provided the mechanical bonding strength and thermal expansion characteristics required to bond the delicate, low thermal expansion tile to the Orbiter aluminum skin. Its disadvantage over the selected SIP-adhesive method (U.S. Patent No. 4,124,732) also developed by NASA, was its inability to maintain the integrity of the tile during the extreme vibration conditions experienced by the Orbiter in the take-off phase of the mission. Sound pressure levels obtained during this phase of the mission can reach a maximum of 166 dB, a value which is much greater than that encountered in typical industrial processes. Of course, when bonding two materials which are very dissimilar in strength and thermal expansion characteristics in a high vibration environment, strain isolation should be provided using a material with similar characteristics as the SIP.

Other possible applications of the techniques developed for the RSISS include:

- Design of testing equipment for fragile materials;
- Manufacture of precision parts made of fragile materials and textiles;
- Large installation and inspection of fragile parts with very close tolerance; and
- Manufacture of ceramic fibrous thermal insulation.

APPLICATIONS OF CARBON-CARBON COMPOSITES
(LESS)

One of the most reliable materials of the Orbiter TPS is the carbon-carbon composite of the leading edge structural subsystem (LESS). This application probably represents the largest external reusable hot structure ever made for a flying vehicle with temperature capabilities of up to 2750 ^{o}F.

Carbon-carbon composites exhibit good thermal shock properties in addition to having the highest specific strength of any existing material above 2000 ^{o}F. Although the prevention of failure by oxidation is the major concern with the use of these materials, coating systems have been developed which provide reliable protection within the Shuttle design life cycle. Proof of the reliability of these coating systems is given by the fact that all original carbon-carbon parts installed in the Shuttle Orbiters are still flying.

Some potential applications of carbon-carbon composites are presented in the following pages of this document. One of these applications relates to the use of carbon-carbon materials in making pistons for internal combustion engines. The many advantages gained using carbon-carbon pistons have been described in an article from the NASA Tech Briefs publication included in this report.

Applications of LESS
(Reinforced Carbon-Carbon)

Friction

- Brakes
- Clutches

Reentry

- Thermal protection
- Planetary probes
- Hot structures

Industrial

- Heat exchangers
- Furnace elements

Engines

- Rocket
  - Exit cones
  - Throats
  - Thrust vectoring
  - Combustion liners
- Turbine
  - Nozzles
  - Augmentors
  - Combustors
  - Rotors
- Piston
  - Pistons
  - Cylinders
  - Valves

Applications of LESS
(Uncoated Carbon-Carbon)

Vacuum Furnaces	Biomedical
• Heating elements	• Bone prostheses
• Thermal insulation	• Space radiators
• Hot pressing dies	

Carbon/Carbon Pistons for Internal Combustion Engines

Carbon/carbon material reduces piston weight and eliminates piston rings.

NASA Tech Briefs, Winter 1985

Langley Research Center, Hampton, Virginia

The carbon/carbon piston performs the same function as aluminum pistons in reciprocating internal combustion engines while reducing weight and increasing the mechanical and thermal efficiencies of the engine. The carbon/carbon piston concept (see figure) features a low piston-to-cylinder wall clearance — so low, in fact, that piston rings and skirts are unnecessary. These advantages are made possible by the negligible coefficient of thermal expansion of carbon/carbon [0.3×10^{-6} in./in./° F (0.54×10^{-6} cm/cm/° C)]. Aluminum, in comparison, has a coefficient of thermal expansion over 40 times that of carbon/carbon.

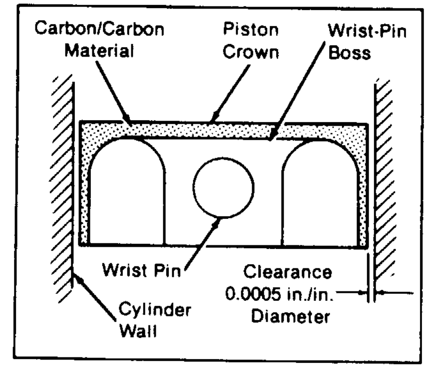
The carbon/carbon material maintains its strength at elevated temperatures, allowing the piston to operate at higher temperatures and pressures than a comparable metal piston. The high emittance and low thermal conductivity of the carbon/carbon piston will improve the thermal efficiency of the engine because less heat energy is lost to the piston and cooling system. The elimination of rings reduces friction, thus improving mechanical efficiency.

Besides being lighter than an aluminum piston for the same application, the carbon/carbon piston will produce cascading effects that will reduce the weight of other reciprocating components such as the crankshaft, connecting rods, flywheel, and balances, thus improving specific engine performance. The engine can run at higher speeds than an aluminum piston engine, again improving specific engine performance.

An alternate carbon/carbon piston concept includes the use of a carbon/carbon cylinder wall provided by a carbon/carbon sleeve in a cylinder block of a more conventional material. This modification further increases the specific power and thermal efficiency by lowering heat losses to the coolant.

This work was done by Allan H. Taylor of **Langley Research Center.** *No further documentation is available.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Langley Research Center [see page 29]. Refer to LAR-13150.



A **Carbon/Carbon Piston Eliminates Piston Rings** and allows a minimal piston skirt length.

Heat Radiators for Electromagnetic Pumps

Carbon/carbon composite radiators withstand high temperatures.

A report proposes the use of carbon/carbon composite radiators in the electromagnetic coolant pumps of nuclear reactors on spacecraft. Carbon/carbon composite materials can function well at temperatures in excess of 2,200 K. Aluminum, which would normally be used, has a melting temperature of only 880 K.

Two configurations have been proposed. In the first case, the composite radiator structure would be bolted and clamped to the three outer ends of the stator-core laminations of a polyphase, annular linear induction pump. The operating temperature would be about 600 K. The radiator structure would conduct heat from the stator core and windings to an outside shield, from which the heat would be radiated into space.

Heat from the portion of the core away from the radiator would be conducted circumferentially around the core in the copper stator windings and into the part of the core near the radiator. Heat that leaks into the stator from the heat-transfer fluid being pumped, including heat induced by the oscillating magnetic field of the pump, joins the thermal flux conducted from the stator to the radiator. In the second case, a composite structure would serve as the waste-heat radiator of an integral thermoelectric generator that supplies power to a thermoelectromagnetic coolant pump. The maximum operating temperature would be 825 K. In both cases, the radiators would be made large enough, not only to radiate heat adequately, but also to shield the equipment from meteorites and other debris.

This work was done by Robert J. Campana of GA Technologies, Inc., for **NASA's Jet Propulsion Laboratory**. *To obtain a copy of the report, "Electromagnetic Pump Heat Radiators," Circle 81 on the TSP Request Card.*
NPO-16458

Applications of PSS

Transportation Systems	Furnaces
• Aerospace vehicles • Space craft • Terrestrial vehicles – Thermal barrier – Seals	• Thermal insulation – Thermal barrier – Seals

POSSIBLE APPLICATION OF FLEXIBLE PILE THERMAL BARRIER SEAL (PSS) MSC-19568

One interesting thermal seal concept developed for the PSS is the flexible pile thermal barrier seal illustrated in Figure 36. This brush-like material is made with fiberglass strands and is used to seal areas between the payload bay and payload bay doors (arrow in Figure 37). The unique feature of this form of thermal seal is that it can be used to insulate large gaps required for the movement between assemblies or parts as illustrated in the kiln car application, Figure 38. The machining of this brush seal is performed using a freeze-cut technique. A description of this technique can be found in the article entitled "Contouring Pile-Brush Seals," NASA Tech Briefs Vol. 3, No. 4, Winter 1978 p. 588 (MSC-16231).

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

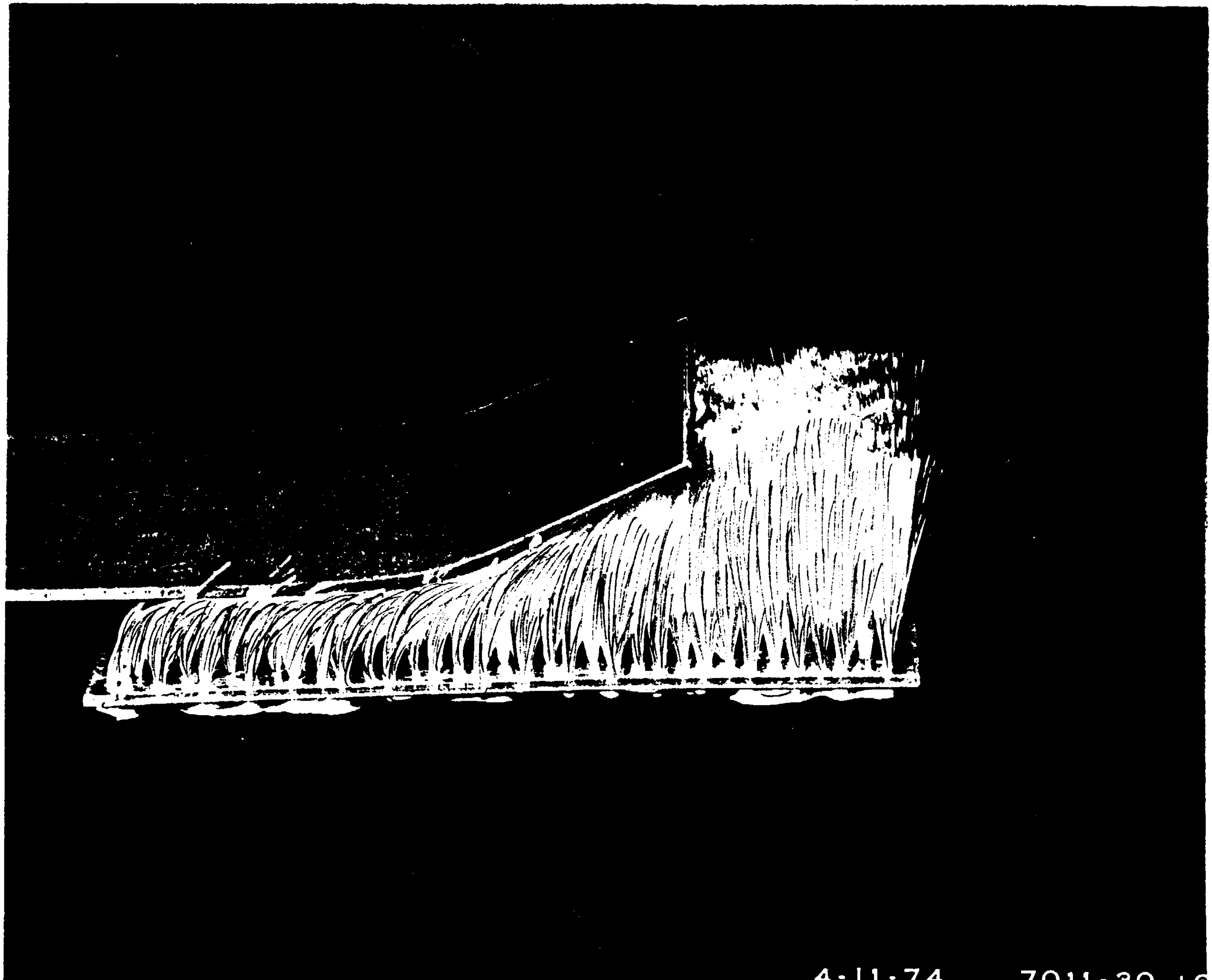


Figure 36. Flexible pile thermal barrier seal of the PSS.

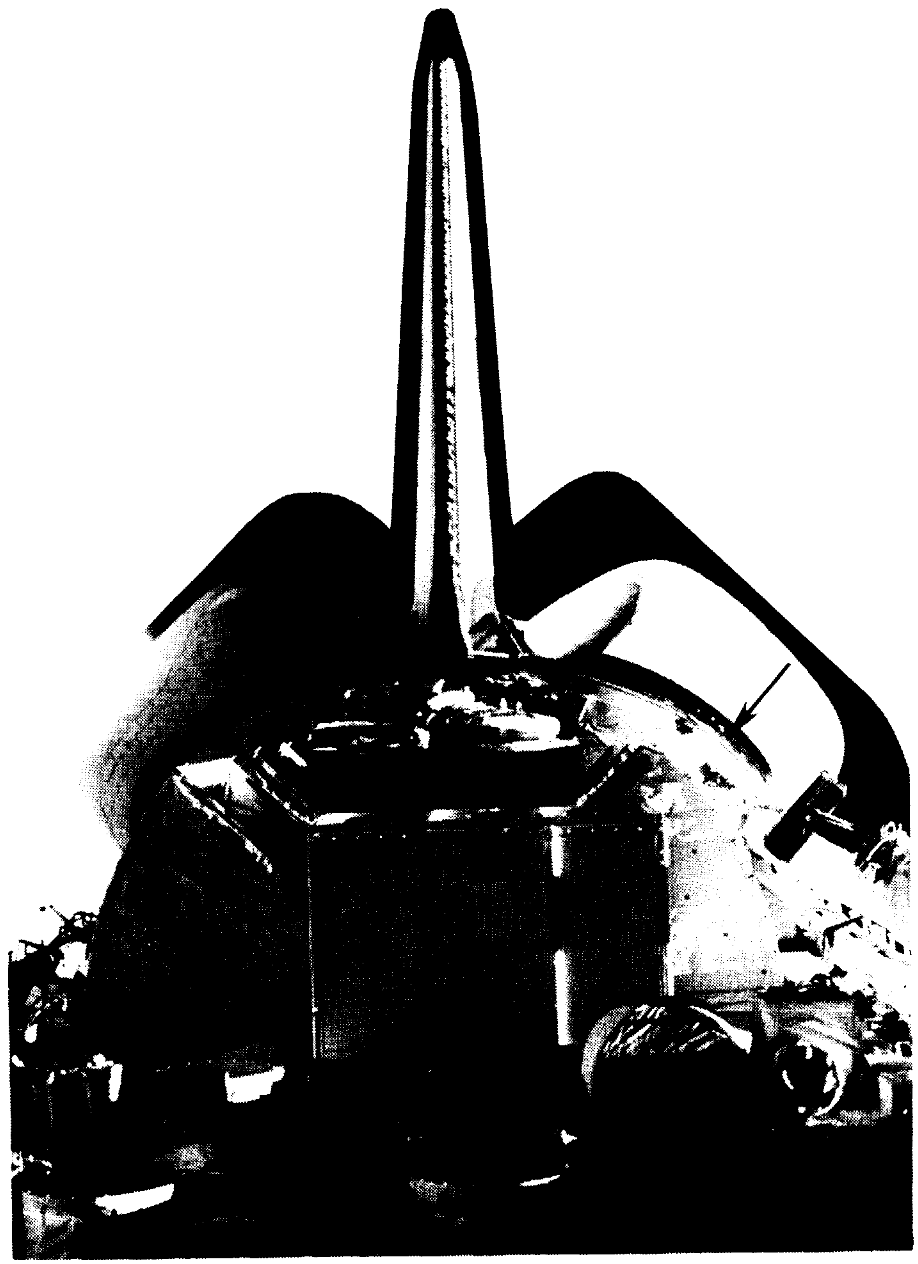
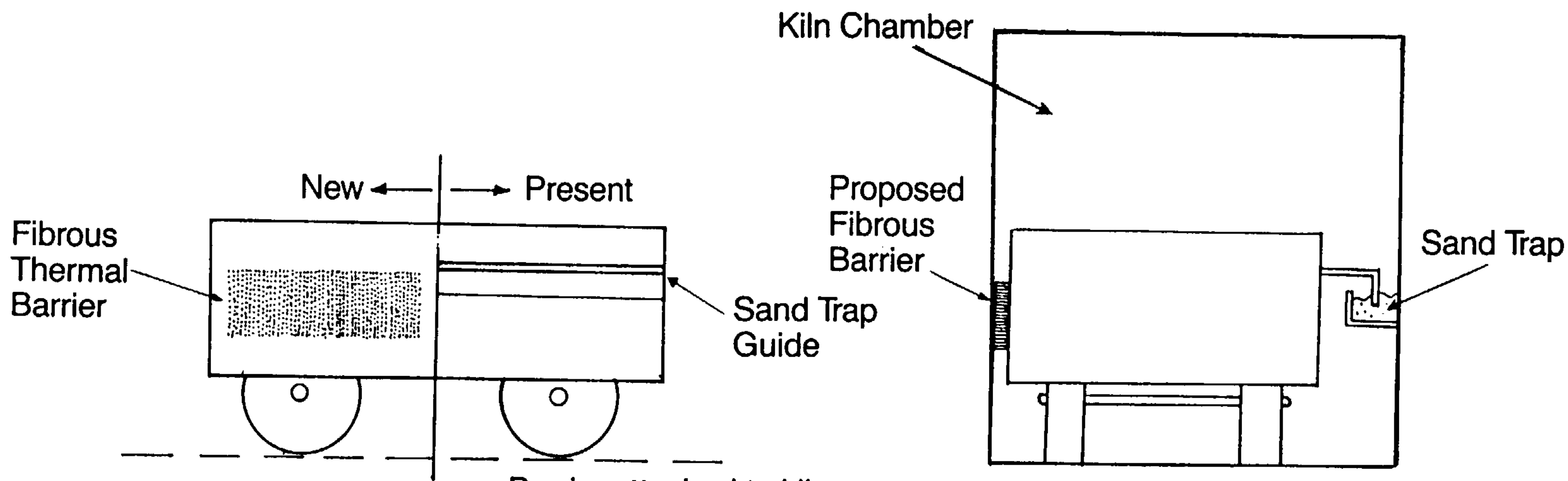


Figure 37. The flexible pile thermal barrier seal is used between the payload bay and payload bay door (arrow).

Example of a Possible Industrial Application



- Barrier attached to kiln car to permit inspection after each pass through kiln, if desired
- Barrier much smaller than sand trap permitting use of larger kiln cars

Figure 38. Possible industrial application of the flexible pile thermal barrier seal.

Factors Affecting Commercialization of TPS Materials and Techniques

	RSISS (Tiles)	LESS (Carbon-Carbon)	PSS
Major factors	• High cost – Labor – Materials	• High cost – Labor – Materials	• High cost – Labor – Materials
Other factors	• Low temper-ature capability	• Oxidation of materials • Long process-ing time	• Unknown to public

Considerations for the Commercialization of TPS Materials and Techniques

RSISS (Tiles)	LESS (Carbon-Carbon)	PSS (Thermal Barriers)
• Increase production • Injection molding of tiles • Relaxation of quality control measures	• Increase production • Automatic lay-up of precursor fabric • Reduction of resin to carbon transform-ation time • More effective oxidation resistant coating	• Increase production • New applications

GLOSSARY

GLOSSARY

Ablation	The removal of heat generated by aerodynamic heating, from a vital part by providing for its absorption in a nonvital part which melts or vaporizes and then passes away taking the heat with it.
Aeroconvective Heating	Also, aerodynamic heating. An increase in the temperature of a fluid, and the solid surface over which it flows, caused by viscous stresses in the fluid boundary layer doing shearing work in the fluid at high velocity. Aeroconvective heating rates increase roughly proportionally to flight velocity.
AETB	Alumina Enhanced Thermal Barrier. A NASA-ARC developed tile material. Its chemical composition differs from that of the HTP tile material developed at LMSC in that it also contains Nextel 312® fibers. Information on AETB can be obtained from the Thermal Protection Branch at NASA-ARC.
AFRSI	Advanced Flexible Reusable Surface Insulation. A waterproofed, quilt-like blanket made with amorphous silica (Q-fiber) felt between two layers of glass cloth and sewn together with Teflon coated silica thread forming a 1 in. quilt pattern. It is coated with a silica base material, has an average density of 11 lb/ft^3, and thermal conductivity equivalent to the LI-900 and FRSI materials. AFRSI is a NASA-ARC developed material based on a commercial thermal blanket registered under the trade name Min-K, made by the Manville Products Corporation. AFRSI is also manufactured by this corporation under the tradename of Quilite®. AFRSI was developed later in the TPS program and is easier to install, simpler to maintain, and more durable than FRSI and LRSI. Because of these advantages, it has replaced these insulations on the original Orbiter TPS. This blanket is usually made in sections 3 ft by 3 ft and in thicknesses from 0.45 to 0.95 in. Optical properties of AFRSI include an "α/ϵ" ratio lower than 0.4 to achieve a low temperature while in orbit and an "ϵ" of 0.8 to allow for maximum re-radiation of the convective heating energy during reentry.

AB 312	See Nextel 312®.
AFA	See ATA.
ARC	NASA Ames Research Center, Moffett Field, CA. Also, NASA-ARC.
Ames Gap Filler	See Gap Fillers.
Arc-Jet Testing	See Plasma Arc Jet Testing.
Arrowhead	RCC panel used around the external tank (ET) disconnect. It is located directly behind the nose landing gear door and is shaped like an arrowhead. This component was a retrofit to the original tile design because RCC is better able to withstand the explosive forces witnessed upon ET separation from the Orbiter during launch.
Atlantis	Fourth Operational Orbiter Vehicle (OV-104).
ATA	Array tile assembly. It consists of a polyurethane frame loaded with 1 to 40 tiles (averaging 22 tiles per assembly). Tile IML's are cut in the frame. The ATA is also used as a shipping container and for the simultaneous bonding of these tiles on the Orbiter. This manner of bonding takes place when a new vehicle is constructed. It is important to know if a tile is an array tile or a close-out tile when a replacement tile is to be manufactured. Array tiles can be made per drawing while close-out tiles require use of a pattern for manufacture. Also called array frame assembly (AFA).
Billet	A fired block or piece of LI-900, LI-2200 or FRCI-12 from which several tiles are manufactured. All tile billets are manufactured by LMSC. The dimensions of the billets are typically 13 in. by 13 in. by 5 in. thick.
Breather Area	The uncoated area on the sides of the tile that starts at the coating terminator line and extends to the tile IML. The breather area allows air to vent out during ascent to preclude a loss of coating due to pressure equalization. Also known as terminator gap.

CADAM	Lockheed Computer Aided Design and Manufacturing System; Interactive graphics system used for the design of tile geometries and the control of machines which cut the tiles to the desired shapes.
CATIA	A Computer-Graphics Aided Three-Dimensional Interactive Application system developed by Dassault Aircraft in France. It is used similarly to CADAM.
Carrier Panel	A removable section having one or more tiles or blankets bonded to it that permits access to the internal structure of the vehicle without requiring TPS removal. These panels are attached to the primary structure with fasteners passing through the tile or blanket thickness.
Cerachrome®	A ceramic fiber thermal insulation material manufactured by the Manville Products Corporation, previously known as Dynaflex. It is used in the internal insulation package of LESS, and also in some of the pillow gap fillers.
Challenger	Second Operational Orbiter (OV-099).
Chin Panel	An RCC panel contoured to the Orbiter's curvature behind the nose cap seals and in front of the nose landing gear doors. This panel has replaced the tiles in this area due to the large movement experienced by these tiles after each flight from their original positions.
Class 1 Coating	See LRSI.
Class 2 Coating	See HRSI.
Close-Out RSI	The last of the RSI (tiles or blankets) to be installed on the Orbiter, usually around penetrations. Since the dimensions of these last spaces cannot be determined theoretically (because of many unknown variables of the installation process) close-out RSI must be manufactured to fit unique spaces. Close-out tiles are manufactured by RI from LMSC billets. Close-out blankets are

manufactured by RI from blanket stock provided by Globe-Albany, Inc. (FRSI) and Manville Products Corporation (AFRSI). Along the perimeter of removable panels or penetrations, close-out RSI is made with holes to allow easy access to the mechanical fasteners which hold these penetrations. These holes are then capped using removable ceramic plugs or inserts. See Carrier Panel.

Columbia First operational Orbiter Vehicle (OV-102). Used for the initial Orbiter Flight Tests (OFT) program. Its initial TPS employed the greatest number of tiles.

Cristobalite The crystalline form of silica stable above 2,680 °F. It crystallizes in the tetragonal system at low temperature (α-phase) and the isometric system at high temperature (β-phase). The thermal expansion coefficients of these phases are very different which explains the poor thermal shock resistance exhibited by this form of silica. Also see silica glass fiber.

Discovery Third operational Orbiter Vehicle (OV-103).

Dry Density The density of an LI-900, LI-2200 or FRCI-20-12 billet prior to sintering.

Enterprise First Orbiter Vehicle assembled (OV-101). Non-operational, used only for subsonic test flights. TPS assembly methods and integrity of its materials were evaluated on this Orbiter.

Filler Bar The filler bar is a strip of the same waterproofed Nomex® felt used to make the SIP and FRSI which is coated with RTV-560® silicon adhesive. The filler bar is part of the assembly method used for tiles. It is placed at the bottom of the tile-to-tile gaps and is about 10% thicker than the SIP so that it can behave as a seal to these gaps. Filler bars minimize the possibility of water and hot plasma gas from reaching the bondline between the SIP and tile. Filler bars are bonded to the Orbiter structure, although sometimes, they are also bonded to tiles. Their thickness varies between 0.090 and 0.160 in. (depending on SIP used) and are 0.750 in. wide. Their length depends on tile dimensions. Maximum Shuttle re-use temperature is 800 °F for 100 missions or 1,000 °F for one mission.

FRCI Fibrous Refractory Composite Insulation. A NASA-ARC developed material, U.S. Patent No. 4,148,962. See FRCI-20-12.

FRCI-20-12 Also known as FRCI-12. A rigid composite fiber insulation made of 78% amorphous silica, Q-fibers, 22% Nextel 312® fibers with 3% by weight silicon carbide powder as an emittance agent. The first number represents the average content of Nextel 312 fiber and the second number represents the average density of the material. It exhibits an average density of 12.5 lb/ft^3, a thermal conductivity close to that of the LI-900 tile with similar thermal shock resistance. Billets, ATA tiles, and complex tiles are manufactured by LMSC. Close-out FRCI-20-12 tiles are manufactured by RI from LMSC billets. FRCI-20-12 tiles are comparable in strength to the LI-2200 but less dense. Similarly to the LI-2200 tiles, they are used in areas that experience high mechanical loading environments, mostly around penetrations and leading edge regions. Developed later in the TPS program, the FRCI-20-12 tile has replaced most of the originally used LI-2200 tile.

FRSI Felt Reusable Surface Insulation. A NASA material U.S. Patent No. 4,151,800, developed at NASA-JSC. It is composed of a waterproofed Nomex® fibrous pad similar to the SIP, coated with a white RTV silicone (Dow Corning DC 92-007). This coating has vent holes approximately 0.32 in. in diameter on 6 in. centers to allow venting of the felt. FRSI is manufactured by Globe-Albany, Inc. at an average density of 5.4 lb/ft^3. FRSI maximum reusable temperature for 100 Shuttle missions is 700 ^oF. Maximum continuous use temperature of this material is 550 ^oF. It has a thermal conductivity similar to that of the LI-900, and thus, it exhibits the lowest thermal conductivity to density ratio of all the TPS materials. FRSI is made in blankets as large as 3 ft. by 6 ft. and in thicknesses between 0.16 and 0.4 in. and as thick as 0.64 in. Optical properties of FRSI are similar to those of AFRSI. Most of the FRSI present in the original TPS of the first two operational Orbiters (Columbia and Challenger) has been replaced with AFRSI.

Gap Fillers — Gap fillers are seals used in gaps between tiles located in regions of the Orbiter which experience the highest aerodynamic pressures. These seals protect the aluminum structure beneath these gaps by preventing the influx of hot plasma gas. There are many types of gap fillers but these can be classified into two categories: pillow and layer, both of which bond to the surface of the filler bar. The pillow type is resilient enough to keep the gap sealed as the tile gap opens or closes in response to structural and thermal environments. These gap fillers consist of an envelope of ceramic fabric stuffed with a resilient ceramic fiber batt, and sometimes with a metal foil which is added for reinforcement. The fabric which at times is coated with an emittance layer, is sewn together at the ends and bottom with quartz thread. Pillow gap fillers are usually installed in tile-to-tile gaps with a width greater than 0.1 in. and can accommodate a total excursion of 0.04 in. The layer gap fillers, also called "Ames Gap Filler," is a NASA-developed material, U.S. Patent No. 4,308,309, which was adopted for tile gaps with a width of less than 0.1 in., and usually in the range between 0.03 and 0.06 in. One layer of Ames Gap filler is 0.017 in. thick. There are various types of Ames gap filler but basically they are single or multiple plies of ceramic cloth impregnated with one or two RTV silicon-type coatings depending on the ultimate use temperature. Ames gap fillers do not follow the gap movement as do the pillow type gap fillers. Since the RTV in this gap filler tends to char upon reentry, they have a life of only a limited number of flights. Up to four Ames gap fillers can be used on the bottom surface, and up to six on the upper surface. Ames and pillow type gap fillers are also combined in what is known as a composite gap filler. This is done to increase the thickness of the pillow gap filler.

Glass Transition — See Tile-RTV-SIP Assembly and RTV-560®.

HTP — High Thermal Performance tile material. A third generation of rigid RSI material developed by LMSC for NASA. HTP uses a combination of amorphous silica (Q-fiber) and alumina fibers with boron nitride and silicon carbide powders. Boron nitride powder is used to generate the bond between the silica and alumina fibers while silicon carbide particles are used as an emittance or opacifying agent. HTP can be tailored to achieve specific properties. The denomination used to represent HTP material is similar to that of FRCI material. HTP-22-12 represents a material with 22% alumina fibers at a density of 12 lb/ft^3.

HRSI High Temperature Reusable Surface Insulation. The HRSI is the black coated tile used to protect those areas of the Orbiter which experience maximum surface temperatures between 1,200 and 2,400 ^oF. The coating for this rigid insulation is also known as Class 2 coating. Most HRSI tiles are made in a 6 in. by 6 in. square planform with typical thicknesses between 0.5 and 3.5 in., and as thick as 5 in. There are many areas of the Orbiter that require more complex geometry with curved rather than flat tiles, usually of varying thickness and width instead of uniform size and cross section. Most of the HRSI tiles use the LI-900 as the base tile material. However, a limited number of more dense HRSI tiles are employed in those areas of the Orbiter that require a material with additional strength and thermal stability, mainly at the interface between HRSI and the other TPS subsystems. These denser HRSI tiles include the LI-2200 and the FRCI-20-12. The HRSI or Class 2 coating is a borosilicate glass referred to as reaction cured glass (RCG). Its optical properties are an "α/ϵ" ratio greater than or equal to 0.8 at 1200^oF and "ϵ" greater than 0.8 at any temperature to allow for maximum re-radiation of the convective heating energy during reentry.

IML Inner Mold Line. The boundary line between the inner side of the TPS and the Orbiter aluminum structure.

In-plane The direction which is perpendicular to the TTT or tile pressing direction.

JSC NASA Lyndon B. Johnson Space Center, Houston, TX. Also, NASA-JSC.

K_{eff}	Effective thermal conductivity. The quality of heat transmitted through a material in unit time, per unit temperature gradient along the direction of flow, and per unit of cross-sectional area. It includes the combined effect of thermal conductivities due to solid conduction, radiation transmission, and gas conduction.
KSC	NASA Kennedy Space Center, Kennedy Space Center, FL. Also, NASA-KSC.
LaRC	NASA Langley Research Center, Hampton, VA. Also, NASA-LaRC.
Layer Gap Filler	See Gap Fillers.
LESS	Leading Edge Structural Subsystem. LESS is composed of RCC as the leading surface for the nose cap and wing leading edge regions. It also includes the internal thermal insulation, thermal barriers, seals, and fastening concepts required for this leading surface. LESS is exposed to the most aggressive thermal-mechanical environment of all the TPS. Other RCC surfaces which have been retrofitted to the original tile design are the arrowhead and the chin panel.
LI-900	A LMSC rigid, all-silica, fibrous insulation developed for the Space Shuttle with an average density of 8.75 lb/ft^3. Over 90% of the tiles used on the Orbiter are made of this material. It exhibits the lowest effective thermal conductivity of all rigid RSI materials. It is made with the amorphous silica fiber known as the Q-fiber and a small amount of colloidal silica used as a binding agent for these fibers.
LI 2200	A rigid, all-silica, fibrous insulation made with the Q-fiber and with about 2% by weight silicon carbide powder for emittance purposes. Fiber bonding is achieved solely by sintering. Developed by NASA-ARC, U.S. Patent No. 3,952,083, it is manufactured at an average density of 22 lb/ft^3. The LI-2200 is being replaced with the stronger, less dense, and more thermal shock resistance FRCI-20-12 tile. Billets, ATA, tiles, and complex tiles are manufactured by LMSC while close-out LI-2200 are made by RI from billets provided by LMSC.

LMSC	Lockheed Missiles and Space Company, Inc., Sunnyvale, CA. Manufactures all of the billets used to make the tiles, and most of the tiles.
LRSI	Low Temperature Reusable Surface Insulation. The LRSI is a white coated tile used to protect those areas of the Orbiter which experience maximum temperatures between 700 and 1,200 ^{o}F. The coating for this rigid insulation is also known as Class 1 coating. The tile material used in the LRSI is the LI-900. In general, LRSI tiles are made in an 8 in. by 8 in. square planform with a thickness between 0.2 and 1.0 in. Some of the LRSI used on Columbia during initial test flights were cut or diced into smaller sections, each about 2 in. by 2 in. This technique allowed buckle to occur in those particular areas of the structure without fear of losing an entire 8 in. by 8 in. planform tile. Optical properties of the LRSI tile coating include an "α/ϵ" ratio between 0.2 and 0.4 to achieve low temperature while in orbit and "α" of 0.7 at 1,200 ^{o}F to allow maximum re-radiation of the convective heating energy during reentry.
Ludox AS®	A colloidal silica solution containing 28% silica which is a product of the DuPont Company. It is one of the ingredients of the AFRSI coating. Also used as an ingredient of the slurry developed for one of the tile surface densification processes (U.S. Patent No. 4,338,368). See Tile Surface Densification.
MSC	Manned Spaced Flight Center. Also, NASA-JSC.
NASA-ARC	See ARC.
NASA-JSC	See JSC.
NASA-LaRC	See LaRC.
Nested tile	A tile that is individually measured for planform dimensions and has its IML cut while being held in a polyurethane nest. Nested tiles usually fit on the outside of an ATA or AFA.

Nextel 312®	An alumina-boria-silica fiber made by the Minnesota Mining and Manufacturing (3M) Co. Previously known as AB-312. The fiber is used in the manufacture of FRCI tile. Fabrics made with this fiber are used in the manufacture of some of the gap fillers and the thermal insulation package for the internal components of LESS and PSS.
Nomex®	A poly (1,3-phenylene iso-phthalamide) which is a product of the DuPont Company. This fiber is used in the manufacture of the felt employed in the SIP, the filler bar and FRSI. Its selection was based on its ability to remain flexible and stable in the required temperature range between -250 and 500 °F.
OV	Orbiter Vehicle: OV-099 (see Challenger); OV-101 (see Enterprise; OV-102 (see Columbia); OV-103 (see Discovery); and OV-104 (see Atlantis).
OML	Outer Mold Line. Outer, aerodynamic surface of the TPS which experiences maximum aerodynamic heating during ascent and reentry. It includes both the upper and lower surfaces of the Orbiter.
Penetration	Areas for the operation and service of the Shuttle Orbiter Vehicle such as service access panels, windows, and doors.
Pillow Gap Fillers	See Gap Fillers.
Plasma Arc Jet Testing	Testing method or condition which most closely represents the actual flight reentry environment. It can simulate transient heating and gas flow rates, show pressure gradient effects and aerodynamic loading, determine tile gap heating and can be used to evaluate the catalytic effects of the tile coatings. Since these conditions can be applied simultaneously, interaction effects between these variables can be observed.

Plasma Flow	Also, Plasma Gas. Heating medium or condition generated at surface of TPS during reentry of Orbiter. A highly or completely ionized gas composed of equal numbers of positive ions and electrons. Usually produced from a neutral gas by applying very high heat flux.
Q-fiber®	See Silica Glass Fiber and Cristobalite.
Quilite®	Trade name of the Manville Products Corporation for AFRSI.
PSS	The Penetration Subsystem. The many insulation concepts, thermal barriers and seals, required to protect the penetrations. It is usually the last of the TPS subsystems to be installed. This subsystem provided some of the most challenging problems of the entire TPS.
RI	Rockwell International Corporation, Space Transportation and System Division, Downey, CA. Principal Contractor for the Space Shuttle.
RCC	Reinforced Carbon Carbon. A carbon fiber, carbon matrix composite coated with a combination of silicon carbide and silica glass for protection against oxidation. RCC exhibits the highest temperature capability (3,000 ^{o}F) and the highest strength at temperature of any of the TPS materials.
RCG	Reaction Cured Glass. A NASA-ARC developed material (U.S. Patent No. 4,093,771). It is the black coating used on HRSI, a borosilicate glass with silicon tetraboride as an emittance agent. RCG was the first HRSI coating to survive and remain stable during the stringent Shuttle requirements of 100 simulated missions in an aeroconvective or plasma-arc environment.
RSI	Reusable Surface Insulation. RSI includes all the other surface insulation used in the RSISS. It is classified into rigid, as in the tiles (HRSI and LRSI), or flexible, as the blankets (AFRSI and FRSI).

RSISS — Reusable Surface Insulation Subsystem. The RSISS is composed of rigid RSI (tiles) or flexible RSI (blankets) and the assembly concepts and materials used for these insulations. The elements of the tile system or rigid RSISS are: the coated tile (LRSI or HRSI), the densification of the tile bonding surface, the RTV adhesive, and the SIP. The tile system also includes the seals used between tile-to-tile gaps which are the filler bar and the gap filler. The elements of the blanket system include the thermal blanket system (AFRSI or FRSI), the RTV adhesive and the methods used to seal or stiffen the edges of these blankets. See also Tile-SIP-RTV Assembly.

RTV-560® — Room Temperature Vulcanizing Silicon Adhesive. A methylphenyl silicone manufactured by the General Electric Company. It is the adhesive used to bond RSI to the structure, that is, bond the tile to the SIP, bond this tile/SIP assembly to the structure, and bond the blankets or flexible RSI to the structure. This adhesive is also used to bond the filler bars to the structure and the gap fillers to the filler bar. Additionally, it is employed as a coating for the filler bars and some of the gap fillers. Its selection was based on its low glass transition temperature (-170 ^{o}F) and its ability to remain useful up to 500 ^{o}F. Also see Tile-RTV-SIP.

Saffil® — A ceramic fibrous blanket used in the manufacture of thermal insulation for internal components of LESS. Also used in the fabrication of some of the pillow-type gap fillers. It is a product of ICI Americas, Inc.

Silica Fiber — The silica fiber, known as the Q-fiber, is the most used ingredient in all of the ceramic fibrous RSI (rigid and flexible). The development of this fiber was obtained from the efforts of Mr. R. Beasley of LMSC and John Koch of the Manville Products Corporation. The silica Q-fiber is amorphous, has a high degree of purity (99.7% silica), a length of about 0.25 in. and a diameter of 1.4 microns. This fiber allows all of the tile materials to exhibit the required thermal shock resistance due to its stability to high temperature devitrification. One of the most significant limitations in the use of silica or high-silica glasses at high temperature is their stability with respect to devitrification to cristobalite. This structural change from a glass or amorphous to a crystalline solid normally results in a loss of material integrity (thermal shock resistance) because of the β-to α-cristobalite phase transformation, especially if it is subsequently cooled to room temperature in use. This loss in thermal shock resistance is due to the large difference in thermal expansion coefficients between these phases of cristobalite. The development of the silica Q-fiber was responsible for the selection of the LI-900 as the main RSI tile material for the Orbiter in 1973. It is made from a source of silica sand of very high purity by the Manville Products Corporation previously known as the Johns-Manville Corporation. Also see cristobalite.

SIP Strain Isolation Pad. The SIP is located between the tiles and the Orbiter structure and isolates the tile from the thermal and mechanical strains of this structure. It is made of the same waterproofed Nomex® felt used in the manufacture of FRSI and filler bars. All of the Nomex® felts are made by Globe-Albany, Inc. The fibers in the SIP are closely arranged and randomly oriented whereby each filament fiber supports the tile independently of the other fibers in the isolation pad. The Nomex® fiber was selected for the SIP because it remains flexible and stable in the temperature range between -250 and 500 ^{o}F. The SIP is made in three thicknesses: 0.160, 0.116, and 0,090 in. Typically, the 0.160 in. SIP, which has the lowest strength, is used with the LI-900 tile and also in areas of the substructure where there are protruding rivets. The 0.090 in. SIP is used with the heavier tiles, LI-2200 and FRCI-20-12, since it is two to three times as strong as the 0.160 in. SIP. The 0.115 in. SIP is limited in use to tiles that are bonded to the removable plates (carrier plates) found mainly around penetrations. The SIP is sized 1.3 cm (0.5 in.) smaller than the tile planform on each side so that the edges of the SIP and RTV do not degrade from the higher temperatures encountered in the tile-to-tile gap.

TABI	Tailorable Advanced Blanket Insulation. A NASA-ARC developed thermal blanket (ARC-11697) which is structural design advancement over the AFRSI material concept. The layers of batt insulation made with the amorphous silica Q-fiber are held in place in an integrally woven core of ceramic yarns made of fibers such as Nextel 312®. The integrally woven cores of TABI provide higher strength, more reliability and can sustain greater heat flux than can the older, sewn sections of the AFRSI blanket. TABI is planned for use in future reentry vehicles. See AFRSI and silica glass fiber.
Termination Line	The line that is placed on the sides of most tiles to define the extent of the coating down the sides.
Tile	Name used for rigidized ceramic fiber thermal insulation material employed on the Orbiter. Three types of materials have been used on the Orbiter. These are: the LI-900, the LI-2200 and FRCI-20-12. Other tile materials with better specific strengths are higher temperature stability such as FRCI-20-8 and HTP-20-12 have been developed to replace the LI-900 and FRCI-20-12 material, respectively. However, these newer materials have not yet been certified for flight since current Shuttle requirements are satisfied with presently used materials. None of the tiles have exactly the same dimensions.
Tile-RTV-SIP Assembly	RSISS assembly method used to install all the tiles. A NASA invention, U.S. Patent No. 4,124,732. The key element of this assembly method is the SIP structure which allows the use a thin adhesive layer (0.0075 in.) on each side of the SIP. Although the layers of adhesive will undergo glass transition in outer space below -170 ^{o}F, that is, they become rigid below this temperature, this transition is not detrimental to the tile. Due to the thinness of the rigid RTV bond layers, the combined RTV-SIP structure does not alter the flexibility of the SIP. Any strains transferred from the Orbiter structure through the rigid bond layer during the on-orbits phase of a mission can still be absorbed by the SIP.
Tile Step	Difference between the OML of two adjacent tiles. Similarly to gaps between tiles, these imperfections must be kept to a minimum and closely controlled.
Tile Surface Densification	Process used to increase the surface density of the side of the tile bonded to the SIP, that is, along the IML of the tile. This process which is a NASA invention (U.S. Patent Nos. 4,338,368 and 4,358,486),

increases the local strength of the tile from three to six times. Since the thickness of densified material is less than 0.2 in., overall weight increase is typically 3-10%. Surface densification of tiles is achieved with silica-type solutions.

The Tile System — See RSISS and Tile-RTV-SIP Assembly.

Tile-to-Tile Gap — Distance between tiles required to allow for thermal expansion differences between the tile and the aluminum structure and for structural deformations induced by vibration, aerodynamic, and thermal loads. Since these gaps are imperfections of the TPS and also of the aerodynamic surface, their size must be kept to a minimum with very close tolerances (0.045 ± 0.016 in. for the lower wings and fuselage and 0.055 ± 0.016 in. for the upper wings, fuselage, and vertical fin). The structure underneath these gaps is protected with a seal known as the filler bar. Also, the structure located in regions of the Orbiter which experience high pressure gradients, such as the nose cap, is additionally protected with a seal between these gaps known as the gap filler. See Filler Bar and Gap Fillers.

TPS — Thermal Protection System of the Orbiter. It is composed of the RSISS, LESS, and PSS.

TTT — Through-the-Thickness direction. Also, the pressing direction during the casting operation.

Witness Line — Same as Termination Line.

GREEK SYMBOLS

ϵ Spectral emittance.

α/ϵ Ratio of solar absorptance to total hemispherical emittance.

® Registered trade name.

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APPENDIX I

APPENDIX I

TPS DISCLOSURES

(KEY TO ALPHA & NUMERICAL CODES)

PHASE

ED	=	EARLY DEVELOPMENT
NT	=	NON-TILE DESIGN/DEVELOPMENT
B	=	BASIC TILE DESIGN/DEVELOPMENT
MF	=	MANUFACTURING/FABRICATION
IT	=	INSPECTION/TEST
A	=	ATTACHMENT/INSTALLATION
MR	=	MAINTENANCE/REPAIR
S	=	SPECULATION/ADVANCED APPLICATIONS

ORIGINATING AGENCY/COMPANY

LMSC	=	LOCKHEED MISSILES & SPACE CO., SUNNYVALE, CA
MDAC	=	MCDONNELL DOUGLAS
SL	=	ST. LOUIS, MO
LB	=	LONG BEACH, CA
GE	=	GENERAL ELECTRIC
PP	=	PHILADELPHIA, PA
B-H	=	BOEING, HOUSTON, TX
MM	=	MARTIN MARIETTA
GD/CONT	=	GENERAL DYNAMICS/CONVAIR
RI	=	ROCKWELL INTERNATIONAL, ST&SG, DOWNEY, CA
LAD	=	LOS ANGELES DIVISION (NOW NAAO)
SC	=	SCIENCE CENTER, THOUSAND OAKS, CA
LaRC	=	LANGLEY RESEARCH CENTER
JSC	=	JOHNSON SPACE CENTER
ARC	=	AMES RESEARCH CENTER
STU	=	STANFORD UNIVERSITY, PALO ALTO, CA
MIT	=	MASSACHUSETTS INSTITUTE OF TECHNOLOGY

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DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-12615 -71	Auger Attach Method for Insulation (A)	NASA
MSC-12619 -72	Thermal Insulation Attachment Method (A)	NASA
MSC-12644 11-73	TPS Mechanical-Type Fastener (A)	LMSC
MSC-14182 —	Strain Arrestor Plate for Fused Silica Tile (A)	LMSC
MSC-14287 —	High Temperature Heating Array (A)	MDAC
MSC-14346 10/31/72	Adhesive for Attaching RSI on SSV's (A)	GD/CON
MSC-14541 5/73	High Temperature Ceramic Bond (A)	GE PP
MSC-14551 6/73	Mechanically Attached RSI (A)	MDAC SL
MSC-14576 7/73	Increased Thermal Stability of Trifunctional Silicone Resins in Reduced Pressure Environments (A)	GE PP
MSC-19427 8/73	Flexible Locator Tooling for Tile Layup on Contoured Surfaces (A)	RI
MSC-19462 5/74	Use of Netting to Control Thickness of Adhesive Coating (A)	RI
MSC-16338 7/76	High Temperature Resilient Thermal Barrier (A)	RI

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-16366 8/76	Diced Tile Thermal Protection for S/C (A)	NASA
MSC-16419 6/76	RTV-to-RTV Silicon Bonding (A)	RI
MSC-16483 9/76	Fused Silica Threaded Plug & Locking Device (A)	RI
MSC-16548 10/76	Combination Tile Array Tool A/MF)	RI
MSC-16634 11/76	Central Vacuum Monitor for Vacuum Bag (A)	RI
MSC-16875 6/77	Reuseable Rubber Bags for Annealing Rigid Polyurethene Foam (A/MF)	LMSC
MSC-16892 3/77	Velcro Attach Scheme for Nose Cap Closeout Tile Array (A)	RI
MSC-16929 5/77	New Thermal Barrier for TPS Tile Gaps (A)	RI
MSC-16930 6/77	Room Temperature Bonding Material for High Temperature Service (A)	RI
MSC-18039 11/77	Size Transport System -- Space Shuttle Closure Tile (A)	LMSC
MSC-18141 12/77	High Purity High Temperature Cement (Fused Silica/Colloidal Silica Adh) (A/MF)	RI
MSC-18253 9/79	Fastener System for Thermal Insulation Blankets (A)	RI

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-18345 7/78	Tile Array Positioning Fixture (A)	RI
MSC-18571 5/79	Array Frame Assembly (A)	LMSC
MSC-18791 2/80	Apparatus for Accurately Preloading Auger Attachment Means for Frangible Protective Materials (A)	RI
MSC-18998 10/80	A Method of Improving Applicator/Mixer Performance (A)	MM
MSC-18999 10/80	A Method to Avoid Gas Leakage in the TPS Applicator/Mixer (A)	MM
MSC-20060 11/80	Hard TPS Gap Filler (A)	RI
MSC-20194 6/81	Installation Device for Inserting an Auger (A)	NASA JSC
KSC-11053 —	Tile Bonding Tool (A)	RI
KSC-11182 —	Air Bag Applies Uniform Bonding Pressure (A)	RI
MSC-20537, 20538, 20539	Metal Insert and Ceramic Plug in Lightweight Ceramic Material for Mechanical Attachment (A)	RI
MSC-20480 —	Insulation Blanket Transfer Coating (A)	RI
MSC-20540 —	Bonding Process for Quartz Fabric Materials to Silica Materials with Ceramic Glazed Coatings (A)	RI

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DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-20541 —	Bonding Process for Quartz Fabric Material to Uncoated Lightweight Silica Ceramic Material (A)	RI
MSC-12638 1/74	All Polyimide Syntactic Foam LRSI (B)	NASA JSC
MSC-12664 6/25/74	Axysymetric Modeling of HRSI Tiles (B)	LMSC
MSC-12671 7/23/74	HRSI Twin Positionable Radiant Heat Sting (B)	LMSC
MSC-12678 8/28/74	HRSI Material Binder Purification (B)	LMSC
MSC-14583 7/73	Designing TPS for Buckles Structures (B)	MDAC SL
MSC-19379 6/73	Tile Geometry Design Demonstrator (B)	RI
ARC-10721 —	Silica Reuseable Surface Insulation (B)	LMSC
ARC-11169 —	Fibrous Refractory Composite Insulation (B)	LMSC
MSC-20601 —	Use of Smaller Particle Size Silicon Carbide in Fibrous Refractory Composite Insulation Reduces Effective Thermal Conductivity (B)	—
MSC-20654 —	Improved Fibrous Composite Structure (B)	LMSC
MSC-20655 —	Improved High Temperature Fibrous Insulation (B)	—

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-20656 —	Improved High Temperature Composite Insulation (B)	—
MSC-20714 —	Addition of Silicon Carbide Power to LI-900 Insulation Material (B)	—
MSC-20678 —	Increased Class 2 Coating Thickness (B)	LMSC
MSC-12526 11/71	Ablative Heat Shield Structure (ED)	NASA JSC
MSC-14243 5/72	Impregnated HCF-RSI (ED)	MDAC SL
MSC-14322 —	Rigidization of Pile Fabricator (ED)	GE PP
MSC-14443 3/73	High Temperature Leading Edge Heating Array (ED)	MDAC SL
MSC-14553 6/73	One-step Inorganic Coatings for HCF (ED)	MDAC SL
MSC-14187 —	Method & Device for Detection of Surface Discontinuities (IT)	NASA
MSC-14549 6/73	Testing of RSI Joints (Variable Width Gaps) in Arc Tunnel Nozzle (IT)	MDAC SL
MSC-14550 6/73	RSI Mechanical Property Test Methods (IT)	MDAC SL
MSC-14552 6/73	Mission Environmental Testing of RSI-TPS Panel (IT)	MDAC SL

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-14726 6/74	Instrumentation of RSI Tiles (IT)	MDAC SL
MSC-19363 6/73	NDE Technique for Determining Adhesive Bond Voids (IT)	RI
MSC-19462 2/74	Electronic Feeler Gage (IT)	RI
MSC-19554 1/75	Semi-Automatic Tile Dimensional Verification Machine (IT)	RI
MSC-19637 5/75	Tile Gap Measuring Instrument-Sliding Wedge Type (IT)	RI
MSC-19645 6/75	Technique for Testing Brittle Ceramic Materials (IT)	RI
MSC-19671 8/75	Electro-Dynamic Transducers for Inspecting Porous Materials (IT)	RI (IDWA- RI/SC)
MSC-19672 8/75	Damped Array Transducers for Inspecting Porous Materials (IT)	RI (IDWA- RI/SC)
MSC-19710 10/75	Faying Surface Mismatch Verification Tool (IT)	RI
MSC-19714 10/75	Multi-Pin Contour Gage (IT)	RI
MSC-19718 10/75	Optical Gap Width Gage (IT)	RI
MSC-16063 2/76	Defect Recording Technique (IT)	B-H
MSC-16105 4/76	Quartz Tie-Rods for High Temperature Extensometry (IT)	LMSC

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-16156 4/76	Photo Detection/Recording Technique for TPS Test Specimens (IT)	RI
MSC-16481 2/76	Mismatch Comparator (IT)	RI
MSC-16544 10/76	Some Consideration of the Dynamics of the Space Shuttle Thermal Protection System (IT)	RI
MSC-16576 12/76	HRSI X-Ray Inspection-Computerized Density Variation Analysis Technique (IT)	LMSC
MSC-16580 12/76	Attaching Strain Transducers to Fragile Material (IT)	LMSC
MSC-16659 2/77	NDT Thickness Measurement of LI-900 Coatings by Beta-Backscatter Method (IT)	LMSC
MSC-16718 3/77	Delineation of Silica-based Coating Sub-Coat by Characteristic X-Ray Analysis (IT)	LmSC
MSC-16897 6/77	Prediction of Thermal Insulation Performance (IT)	RI
MSC-18181 12/77	Jet Abrasive Gun/Dolly-Constant Tip Distance (IT)	RI
MSC-18263 5/78	Coating Residual Stresses (IT)	LMSC
MSC-18397 10/78	Dimensional Measuring Technique for HRSI Tiles (IT)	LMSC

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-18423 11/78	Vacuum Bag Material Defect Detection Technique (IT)	LMSC
MSC-18424 11/78	Breather Area Check Gages for High Temperature RSI Tile (IT)	LMSC
MSC-18910 6/80	Inflatable Proof Tester for Tiles (IT)	RI
MSC-18933 6/80	Articulated Vacuum Chuck (IT)	RI
MSC-20055 9/80	Low Cost Surface Waviness Measurement Device (IT)	RI
MSC-20057 9/80	Gap Measurement Tool (IT)	RI
MSC-20105 3/81	Pull Test for Mechanical Property Characterization of Material (IT)	LMSC
MSC-20139 9/80	Tile Failure Probability Model (TFPM) (IT)	RI
MSC-20229 11/80	Ultrasonic Device for Determination of the Fiber Orientation (IT)	RI
MSC-20230 3/81	Gap Measurement Gage (IT0	RI
MSC-20231 3/81	Tile Load Verification Pull-Strip Method (IT)	RI
MSC-20376 1/82	Water Repellancy Testing of Thermal Barrier Fabrics (IT)	RI

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-20389 12/81	TPS Tile Profilometer (IT)	RI
MSC-20672 —	Bond Verification Using Hot Melt Adhesive (IT)	—
MSC-20685 —	Hot Melt Glue Chuck for Pull Testing Space Shuttle Tiles (IT)	—
MSC-20731 —	Development of Fixturing & Extensometry for Elevated and Cryogenic Temperature Testing of Brittle Silica Insulation Materials (IT)	—
MSC-20781 —	Filler Bar to Tile Gap Gage (IT)	—
MSC-20673 —	Thermocouple Installation for Fabric Blanket Insulation (RI)	RI
MSC-20736 —	Flexible Material Step Differential Measurement Device (IT)	RI
MSC-20755 —	Peel Test Fixture for Insulation Blankets (IT)	RI
MSC-20834 —	Unique Heating and Temperature Measuring Technique (IT)	RI
MSC-12682 9/5/74	HRSI Tile High Temperature Surface Marking (MF)	LMSC
MSC-12716 2/75	Vacuum Fixturing for HRSI Tile Machining (MF)	LMSC
MSC-14270 —	2-Component Ceramic Coating for Silica Insulation (MF)	LMSC

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-14270 —	3-Component Ceramic Coating for Silica Insulation (MF)	LMSC
MSC-14461 2/73	Machining Lightweight Ceramic Insulation (MF)	MM
MSC-14918 7/75	HRSI Tile Waterproofing Material (MF)	LMSC
MSC-19271 11/72	Process for Applying Permanent Location Grid on Contoured Carrier Panels (MF)	RI
MSC-19340 4/73	Technique for Coating Tile Arrestor Isolator Assemblies (MF)	RI
MSC-19666 7/75	Adjustable Securing Base (MF)	RI
MSC-19737 11/75	Top Coating of Silica Foam Tile Insulation (MF)	RI
MSC-16102 4/76	HRSI Material Vacuum Casting/Forming System (MF)	LMSC
MSC-16104 4/76	Servo-Controlled Extensometer Pressure Probe (MF)	LMSC
MSC-16270 5/76	Multi-Spindle Variable Contour Vacuum Holding Fixture (MF)	RI
MSC-16350 8/76	HRSI Tile Fabrication System - Computerized Numerical Control Machining Technology (MF)	LMSC
MSC-16775 4/77	Permanent High Temperature Marking for Application on Rigidized Lightweight Fibrous Material Surface (wet or dry) (MF)	LMSC

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-16787 5/77	High Solids Controlled Rheology High Temperature Ceramic Coating Systems (MF)	LMSC
MSC-16793 5/77	Accelerated Purification of Colloidal Silica Solutions for Use in High Thermal Environments (MF)	LMSC
MSC-16825 7/77	Fiber Cleaning by Hydrocyclone Technique (MF)	LMSC
MSC-16891 7/77	Low Expansion Glass Coating (MF)	LMSC
MSC-16927 5/77	Magnetic Spray Mask Tooling (MF)	RI
MSC-16947 8/77	Thermal Control Coating (MF)	LMSC
MSC-18211 3/78	Space Shuttle Heat Shield Material (LI-900) Modified for Heat Container Utilization (MF)	LMSC
MSC-18239 4/78	Automated Conveyer/Metering System for Multi-Lot Fiber Blending (MF)	LMSC
MSC-18248 5/77	Modified Baloney Slicer for Orbiter Mini-Tile Strain Isolation Pad Slicing (MF)	RI
MSC-18276 6/78	Sintering of High Temperature Ceramic Coating System Thru Controlled Glass Frit Particle Size Distribution (MF)	LMSC

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-18322 7/78	Technique for Solids Content Optimization of Ceramic Coating Slurry Mix (MF)	LMSC
MSC-18328 7/78	Template Masking System for High Temperature Ceramics Application (MF)	LMSC
MSC-18346 6/78	High Temperature Gap Filler Rigidizing (MF)	RI
MSC-18368 7/78	Rework Procedure for Large Defects in Tiles (MF)	RI
MSC-18383 10/78	Preparation of LI-900 Tiles for Ceramic Coating/Optical Properties Considerations (MF)	LMSC
MSC-18392 9/79	Patching Tile Insulation/LI-900 (MF/MR)	RI
MSC-18393 9/78	Resilient Thermal Barrier Fabrication Process (MF)	RI
MSC-18461 1/79	Patching Material Commonality for All HRSI Ceramic Insulation Coatings (MF)	LMSC
MSC-18517 4/79	High Temperature Inorganic Silica Adhesive System (MF)	LMSC
MSC-18539 4/79	Auto Mirror/Image Capability for HRSI DNC System (MF)	LMSC
MSC-18540 4/79	Stripping and Recoating of Ceramic Coated HRSI Tiles (MF)	LMSC

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-18542 5/79	Fluid Energy Milling of High Temperature Ceramic (MF)	LMSC
MSC-18543 5/79	Stability of Ceramic Coating Slurry System by Chemical Treatment (MF)	LMSC
MSC-18544 5/79	Dispersion of Methocel Solution into Alcohol (MF)	LMSC
MSC-18545	Blending of Fiber Lots for Production LI-900 Tiles (MF)	LMSC
MSC-18570 5/79	Blending of Fibers for Heat Treatment (MF)	LMSC
MSC-18752 1/80	Technique for Reducing Voids in LI-900 Using Stirring and Vibration	LMSC
MSC-18785 9/79	Use of Reflectors to Reduce Insulation Weight (MF)	RI
MSC-18832 4/80	High Temperature Silicon Carbide Impregnated Insulation Material (MF)	NASA JSC
MSC-20011 9/80	Felt Trimming Process (MF)	RI
MSC-20063 12/80	RSI Tile Coating Strength Enhancement (MF)	RI
MSC-20178 4/81	Guillotine-Type Cutting Fixture (MF)	RI
MSC-20372 12/81	Rapid Fabrication of On-Site Gap Filler (MF/MR)	RI

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-20378 1/82	Fibrous Strengthening of Reaction Cured Glass (MF)	RI
ARC-11051 —	Reaction Cured Glass & Glass Coatings (MF)	NASA ARC & STU
ARC-11110 —	Spray Coating Apparatus Having a Rotatable Workpiece Holder (MF)	NASA ARC
MSC-20612 —	Enhance Densification Process for Low Density Silica Materials (MF)	RI
MSC-20660 —	Cleaning of Johns Manville Silica Fibers (MF)	LMSC
MSC-20679 —	Development of Conical Cutter Tool (MF)	LMSC
MSC-20680 —	Prediction of Sintering Soak Time Based on Silica Fiber Compact Volume Shrinkage Results (MF)	LMSC
MSC-20681 —	Double Firing of Tiles to Achieve Dimensional Requirements (MF)	LMSC
MSC-20684 —	HRSI Tile Automated Offset Application (MF)	LMSC
MSC-20708 —	Thread Coating Process (MF)	RI
MSC-20784 —	Royalite Molds for Foaming Applications (MF)	RI
MSC-20795 —	Curved Chuck-Tile Contour Matching (MF)	RI

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-20824 —	Gap Filler Fabrication Tool (MF)	RI
MSC-19549 1/75	High Temperature Fused Silica Insulation Repair (MR)	RI
MSC-19574 2/75	TPS Tile Access Plug Design (MR)	RI
MSC-16773 12/75	High Temperature Waterproofing Material (MR)	LMSC
MSC-16861 5/77	Pneumatic Tile Removal (MR)	RI
MSC-18344 7/78	Single Mission Waterproofing of High Temperature RSI (MR)	RI
MSC-18796 3/80	Silicone Ablative Materials (MR)	MDAC LB
MSC-18851 4/80	High Temperature Emittance Coatings (MR)	—
MSC-18852 4/80	Spray Applicator for Spraying Coatings and Other Fluids in Space (MR)	—
MSC-18856 4/80	Hardening Method for Repair (MR)	RI
MSC-18916 7/80	Adhesive Mixer/Applicator (MR)	GE
MSC-20117	Non-Fired Coating for RSI Rile	RI
MSC-20137 1/81	High Temperature Reusable Gap Fillers (MR)	RI

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-20198 4/81	Gap Filler Removal Tool (MR)	RI
MSC-20255 8/81	A Method to Improve Mixing and Expulsion Applicator/Mixer (MR)	MM
MSC-20336 10/81	Repair of RSI Thermal Test Tiles (MR)	RI
MSC-20340 12/81	Waterproofing Rigid Porous Structures (MR)	—
MSC-20364 2/82	Waterproofing Material Refurbishment System for STS Orbiter Tiles (MR)	LMSC
KSC-11097 —	Method for Repair of Thin Glass Coatings (MR)	RI
KSC-11171 —	Mobile Glazing Unit (MR)	RI
MSC-20651 —	Rework of Coating Defective Tiles (MR)	LMSC
MSC-20695 —	TPS Re-Waterproofing Injection Gun (MR)	—
MSC-20696 —	Portable Fluid Injection Gun (MR)	—
MSC-20851 —	Radius Rework for Gap Fillers (MR)	RI
MSC-20899 —	Ceramic Bonded Tile Gap Repair Process (MR)	RI

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-12737 -75	Thermal Insulation Protection Means (NT)	NASA JSC
MSC-14229 5/5/72	Boron-Silicon Diffusion Coating for RCC (NT)	Vought
MSC-16152 4/76	Flutter Analysis of Membrane on Elastic Foundation (NT)	RI
MSC-16382 8/76	Edge Sealing High Temperature Felt Insulation (NT)	RI
MSC-16641 1/77	Metallic Foil Flow Restrictor (NT)	RI
MSC-16644 1/77	Process for Forming Metal Foil Flexible Band Seals (NT)	RI
MSC-16967 6/77	Subliminating Basting Thread for High Temperature Fabrics (NT)	RI
MSC-18390 7/78	Thermal Barrier Seal (NT)	RI
MSC-18916 8/80	Adhesive Mixer/Applicator (NT)	RI
LAR-12547 —	Graphite/Polyimide Structural Applications (NT)	—
LAR-12719 —	Variable Anodic Thermal Control Coating (NT)	—
LAR-12858 —	Solvent Resistant, Thermoplastic Aromatic Polyimidesulfine and Process for Preparing Same (NT)	LaRC & MIT

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-20584 —	Silicon Carbide Fabric Gap Fillers (NT)	—
MSC-20799 —	High Temperature AFRSI Protective Coating (NT)	—
MSC-20829 —	High Impact Ceramic Coating (NT)	—
MSC-20452 —	Lightweight Insulation Blanket (NT)	RI
MSC-20671 —	Flexible Insulation Blanket Coating (NT)	RI
MSC-20254 8/81	Pre-stressed TPS (S)	—
LAR-12620 —	Metallic Panels Insulate to 2700^{o}F (S)	NASA LaRC
LAR-12862 —	Shell Tile TPS (S)	—
MSC-20782 —	Fabricating Rigid, Lightweight, High Temperature Insulation - A Concept (S)	—
MSC-20831 —	Ultra Lightweight Rigidized Insulation (S)	—
MSC-20975 —	Modified Cooling Procedure to Prevent Cracking of the Class 2 (RCG) Coating on Compositions (S)	—

DISCLOSURE NO. DATE	TITLE (Phase)	ORIGINATOR
MSC-20976 —	High Temperature Flash Firing of Class 2 Coating on HTP to Minimize Stress Build-up Between Coating and Substrate (S)	—
MSC-21005 —	TPS (Training Planning System) Display Module (S)	—

APPENDIX II

APPENDIX II

PROPERTIES OF TPS MATERIALS

The properties of TPS materials presented in this document have been obtained mostly from two literature sources. These are:

1. "Materials Properties Manual, Volume 3, Thermal Protection System Materials Data," December 1982, PUB 2543-W Rev 5-79, Rockwell International Space Transportation and Systems Group; and

2. The product literature on rigid RSI materials (LI-900, LI-2200, FRCI-12, and HTP) published by Lockheed Missiles and Space Company, Sunnyvale, California.

The figures and tables presented on pages II-4 through II-11 and from page II-9 to II-10, respectively, are reproduced from the Rockwell International publication (1). In these figures, emphasis has been given to the properties of the FRCI-20-12 tile material because:

1. It is a NASA-developed material;

2. It exhibits one of the best thermal shock property of any rigid ceramic thermal insulation material available;

3. It exhibits the best specific strength of any of the tile materials currently used on the Orbiter.

The figures presented on pages II-12 through II-23 give a graphical representation of the behavior of these properties, mainly as a function of temperature. These graphs were performed by the University of Washington under the direction of Mr. Alan D. Miller.

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II-2

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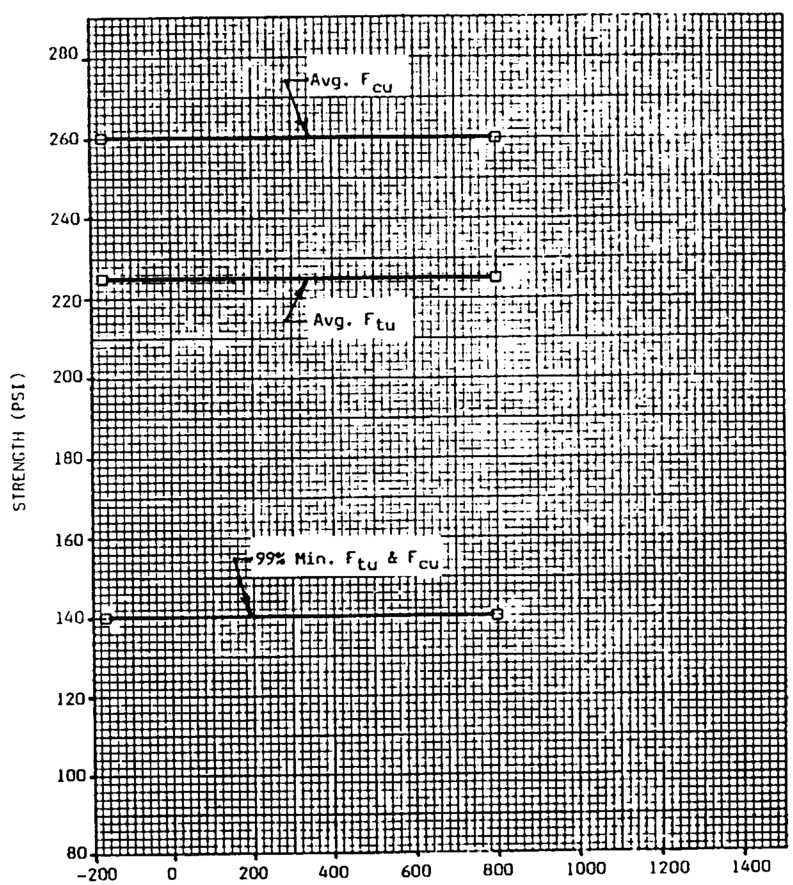
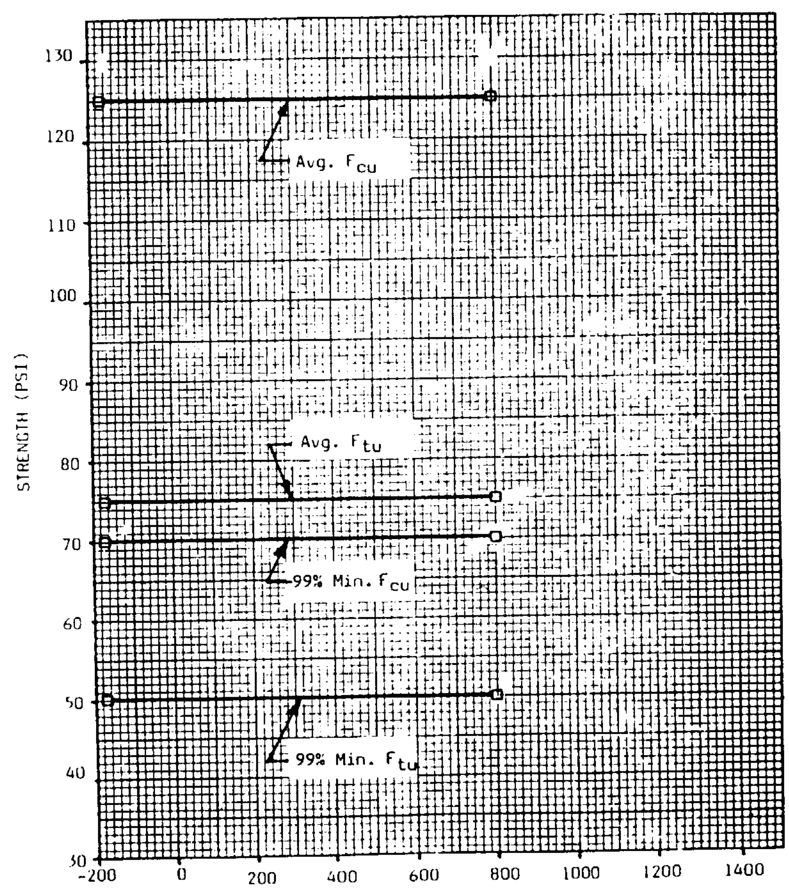
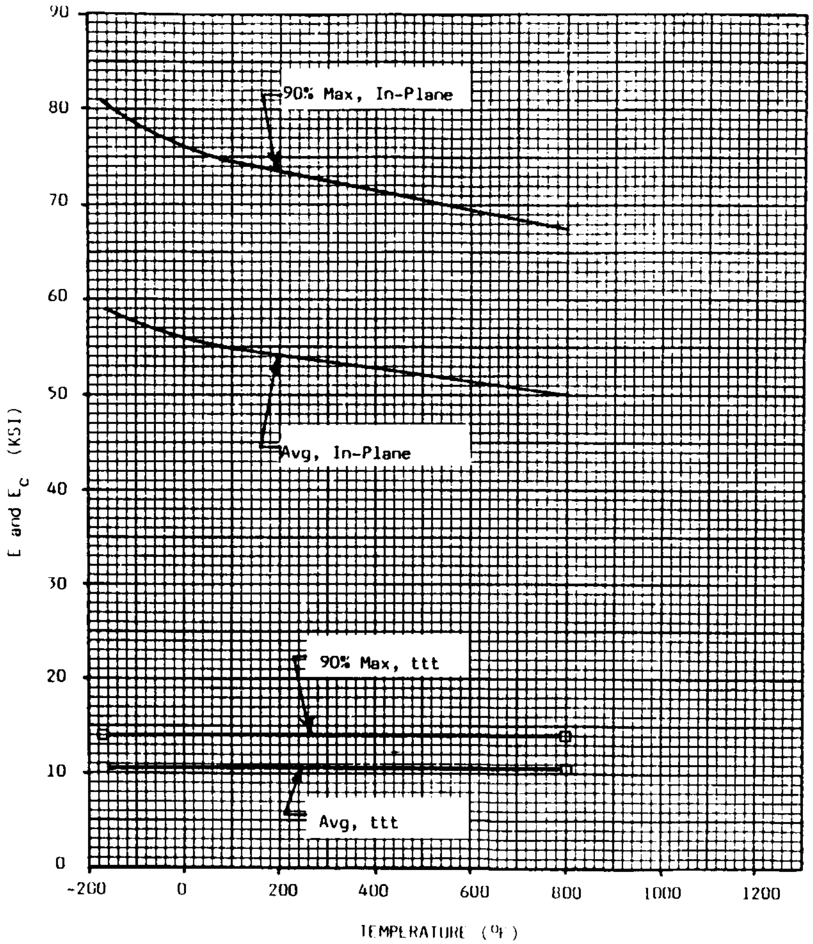
FRCI-12 FIBROUS INSULATION PER MB0115-032
TENSILE AND COMPRESSIVE STRENGTH VS. TEMPERATURE
IN-PLANE DIRECTION



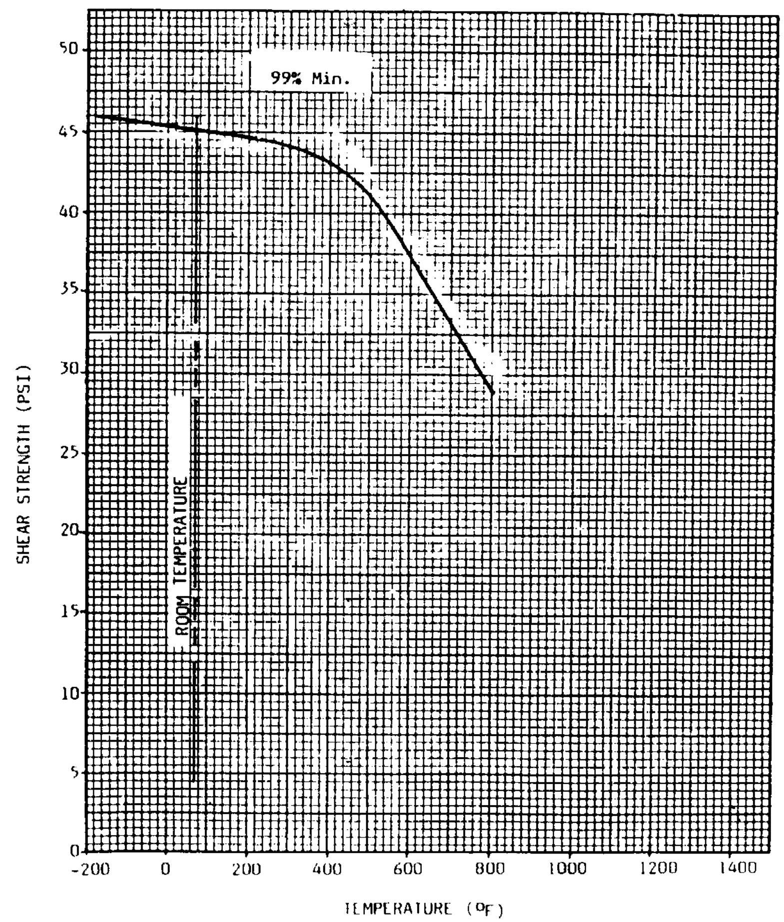
FRCI-12 FIBROUS INSULATION PER MB0115-032
TENSILE AND COMPRESSIVE STRENGTH VS. TEMPERATURE
TRANSVERSE (ttt) DIRECTION



FRCI-12 FIBROUS INSULATION PER MB0115-032
TENSILE AND COMPRESSIVE MODULUS VS. TEMPERATURE

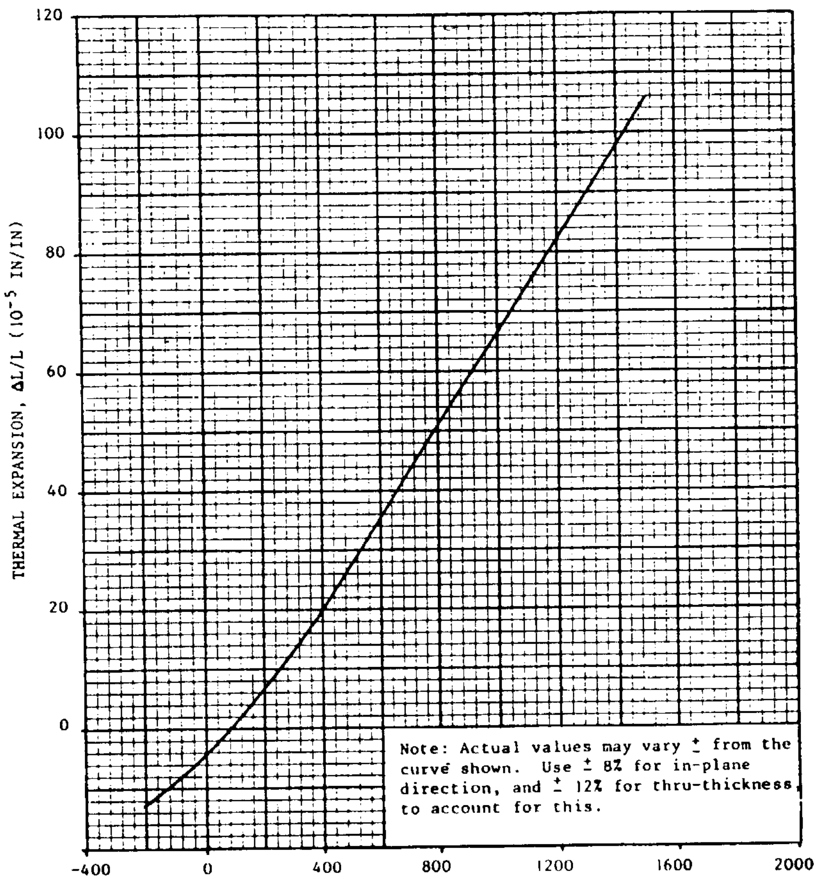


FRCI-12 FIBROUS INSULATION PER MB0115-032
ULTIMATE SHEAR STRENGTH VERSUS TEMPERATURE
ALL DIRECTIONS

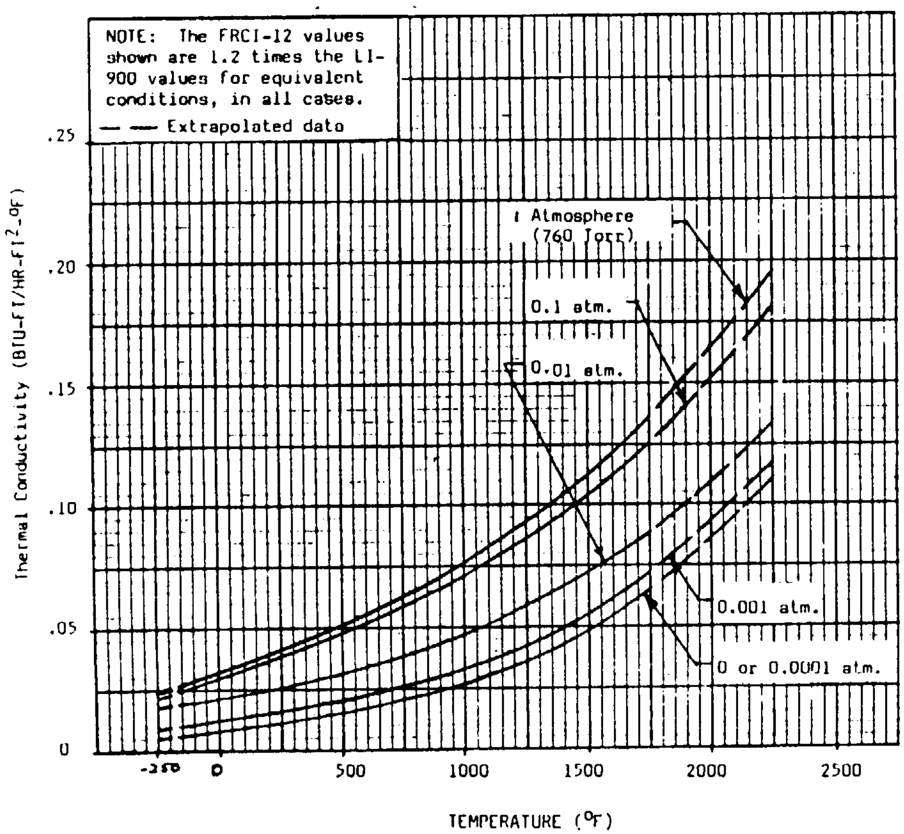


FRCI-12 FIBROUS INSULATION PER MB0115-032

THERMAL EXPANSION, IN-PLANE AND THRU-THICKNESS

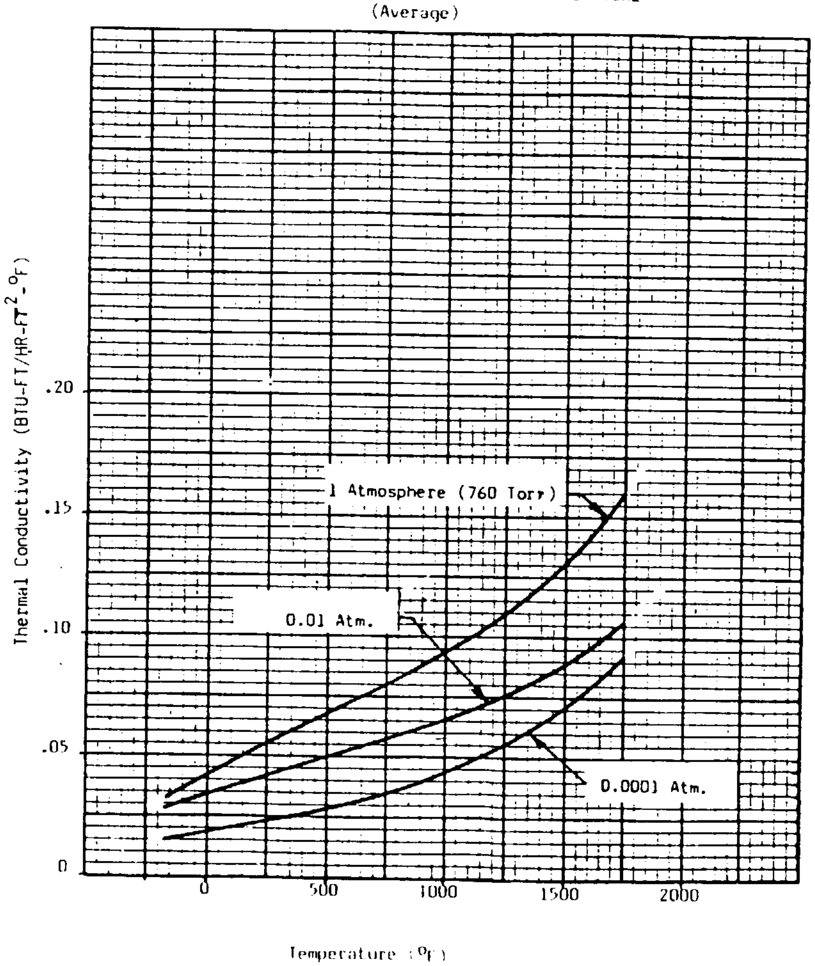


FRCI-12 FIBROUS INSULATION PER MB0115-032
TRANSVERSE THERMAL CONDUCTIVITY VS. TEMPERATURE
(Average)

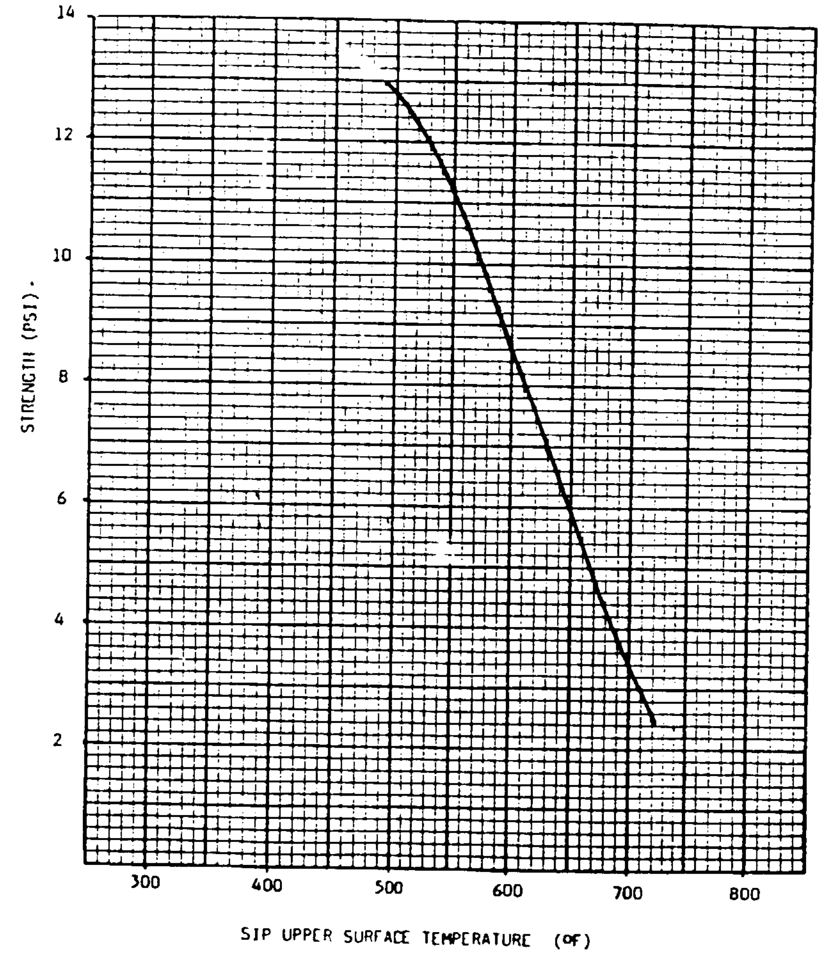


FRCI-12 FIBROUS INSULATION PER MB0115-032

IN PLANE THERMAL CONDUCTIVITY VS. TEMPERATURE
(Average)



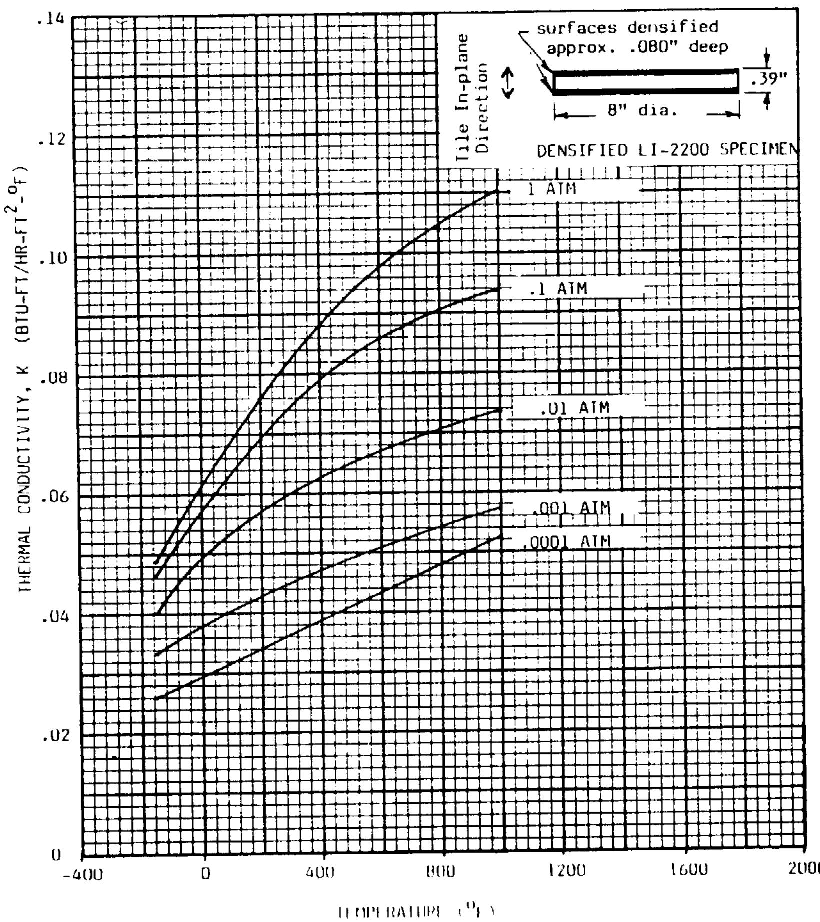
HRSI TILE BONDED TO .160 SIP, PER MA0106-319
FLATWISE STRENGTH VS. TEMPERATURE OF SIP UPPER
SURFACE (TILE/SIP INTERFACE); LOWER SIP SURFACE AT 360F
NOTE: FOR TEMP. ≥ 600F APPLIES TO A SINGLE MISSION CYCLE ONLY



DENSIFIED LI 2200 TILE (Densified per MA0609-303)

IN-PLANE THERMAL CONDUCTIVITY VS TEMPERATURE

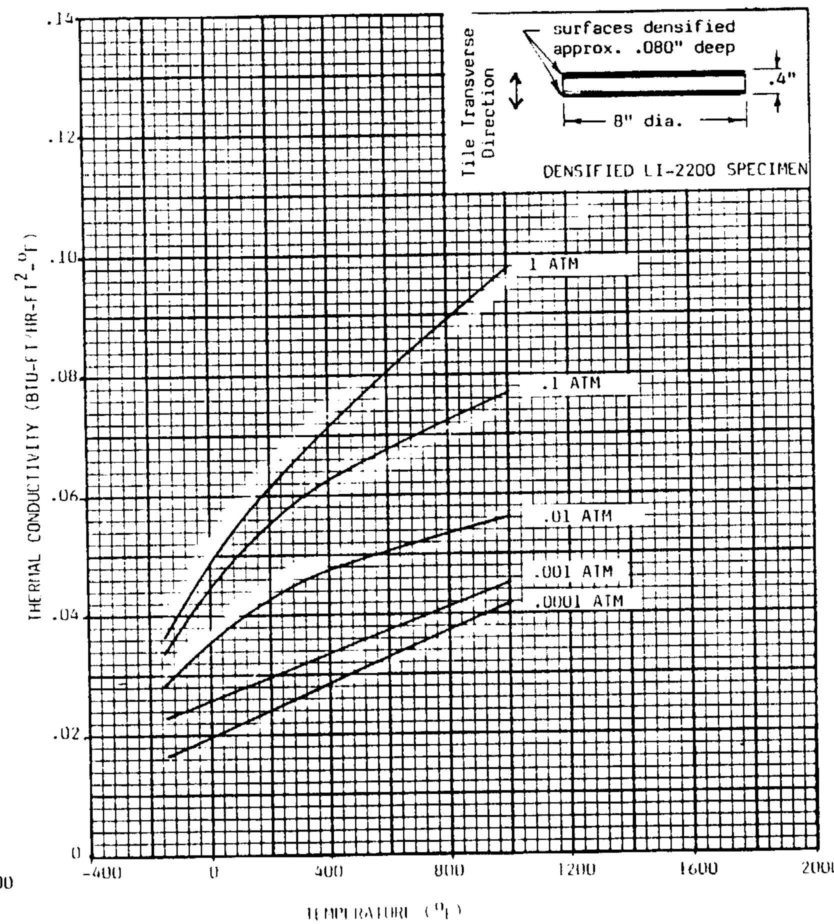
(Average)



DENSIFIED LI 2200 TILE (Densified per MA0609-303)

TRANSVERSE THERMAL CONDUCTIVITY VS TEMPERATURE

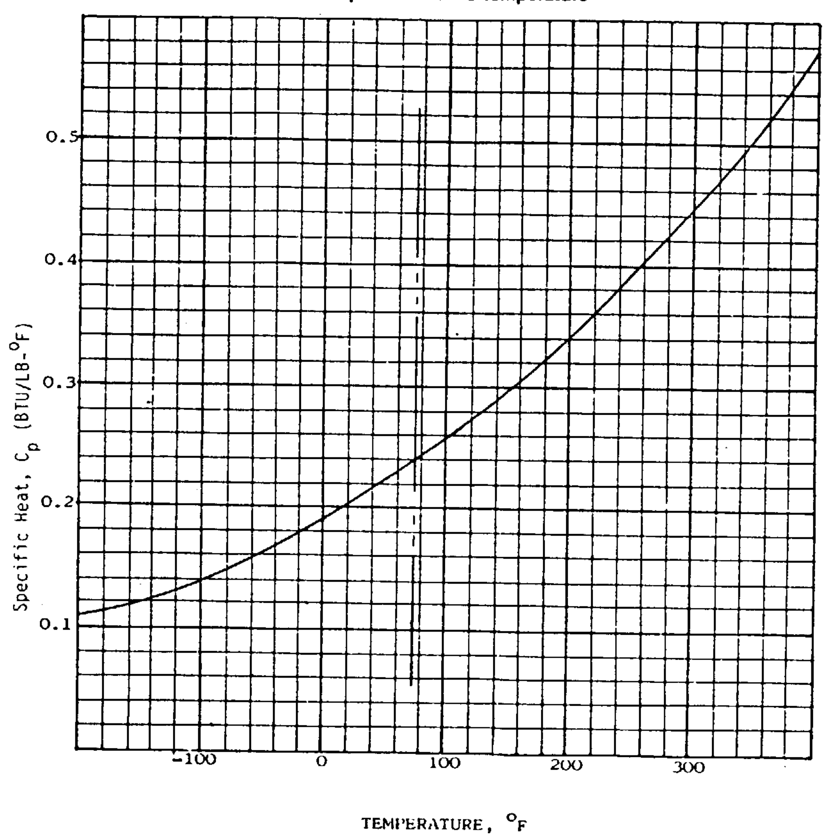
(Average)



.090, .115, and .160 SIP per MBO135-051

Filler Bar per MBO135-058, and
FRSI per MBO135-056

Specific Heat vs Temperature



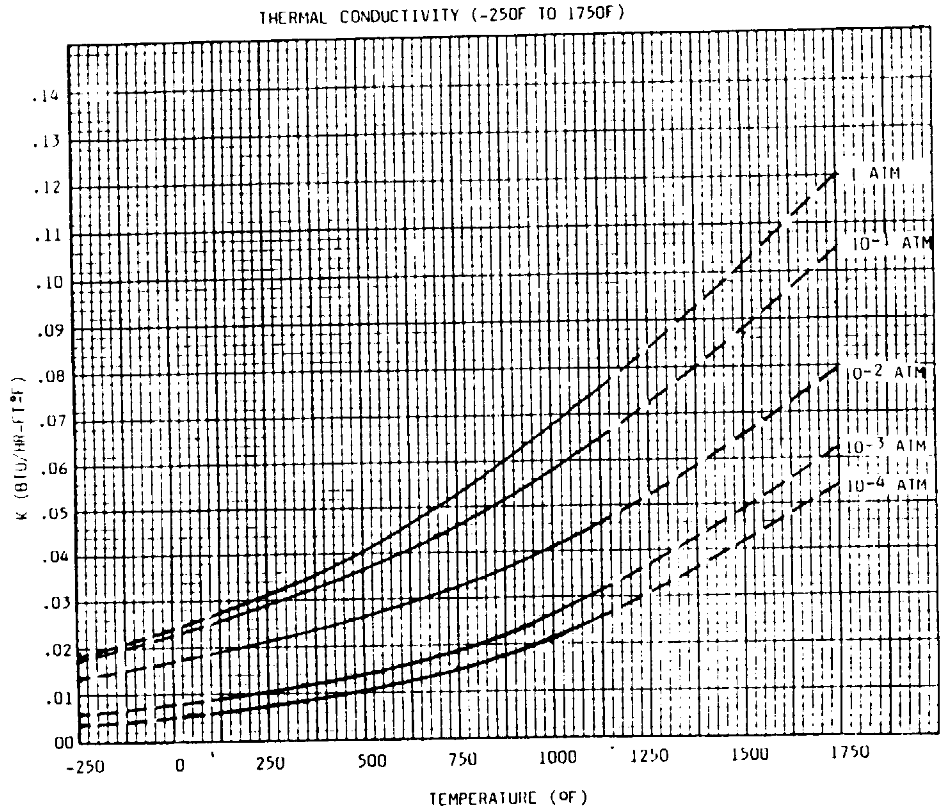
FRSI per MBO135-056

Thermal Conductivity, BTU-Ft/Hr-Ft^2-°F, vs Temperature and Pressure

PRESSURE LB/FT^2	0.021	0.212	2.116	21.16	211.6	2116
ATMOSPHERE	10^{-5}	10^{-4}	10^{-3}	10^{-2}	10^{-1}	1
TEMPERATURE °F						
-250	.0065	.0070	.0080	.0092	.0102	.0110
0	.0080	.0105	.0140	.0171	.0198	.0206
100	.0086	.0120	.0166	.0205	.0238	.0250
200	.0095	.0138	.0194	.0240	.0275	.0290
300	.0102	.0155	.0222	.0275	.0322	.0335
400	.0110	.0170	.0250	.0316	.0370	.0382
600	.0130	.0207	.0315	.0407	.0475	.0489
800	.0150	.0250	.0380	.0500	.0608	.0620

MB0135-085 AFRSI BLANKETS

THERMAL CONDUCTIVITY (-250F TO 1750F)



LRSI and HRSI Coating Data

I. Mechanical Properties (Preliminary)

Property	Temp	Direction	Average Value
E	R.T.	In Plane	4×10^6 psi
F_{tu}	R.T.	In Plane	4×10^3 psi
F_{cu}	R.T.	In Plane	10×10^3 psi

II. Coating Thickness

Class 1 (LRSI)	0.007 - 0.011
Class 2 (HRSI)	0.009 - 0.015

III. Thermal Expansion Coefficient

Temperature Range (°F)	α (10^{-6} in/in-°F)
70 to 1500	0.57
70 to -100	0.56
70 to -200	0.46
70 to -250	0.41

IV. Density

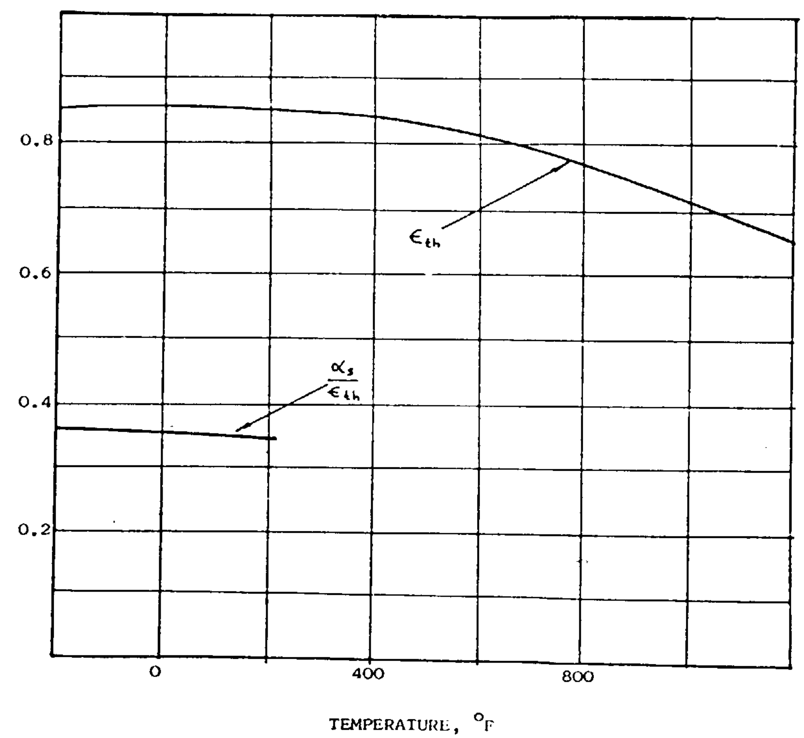
.061 lb/in^3

Optical Properties
LSRI–Class 1 Coating

TOTAL HEMISPHERICAL EMITTANCE (ϵ_{th})

AND

$\frac{\alpha_s}{\epsilon_{th}}$ (SOLAR ABSORPTANCE)

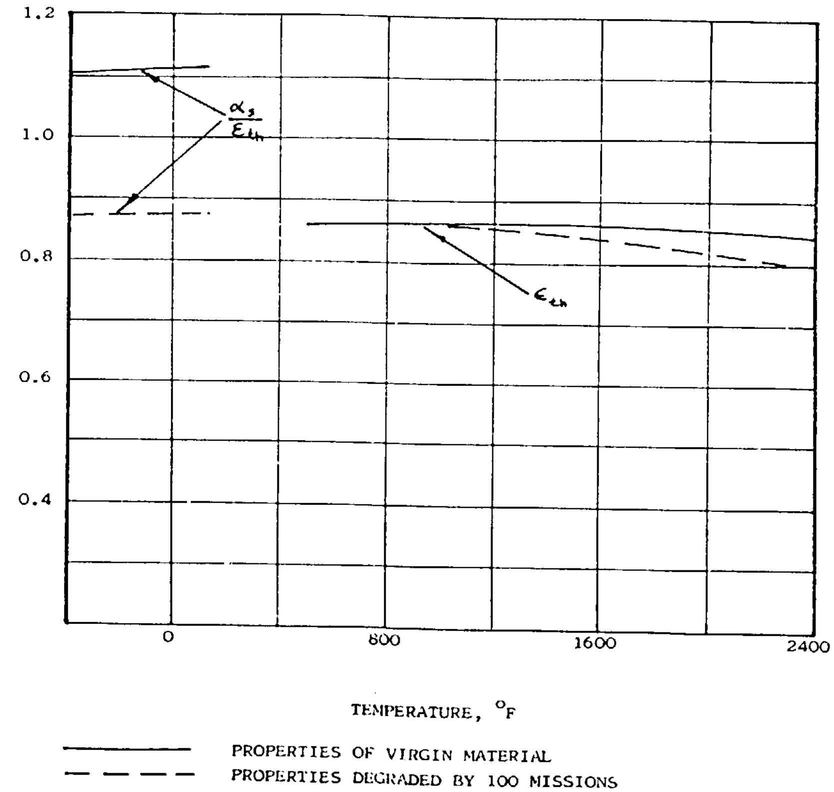


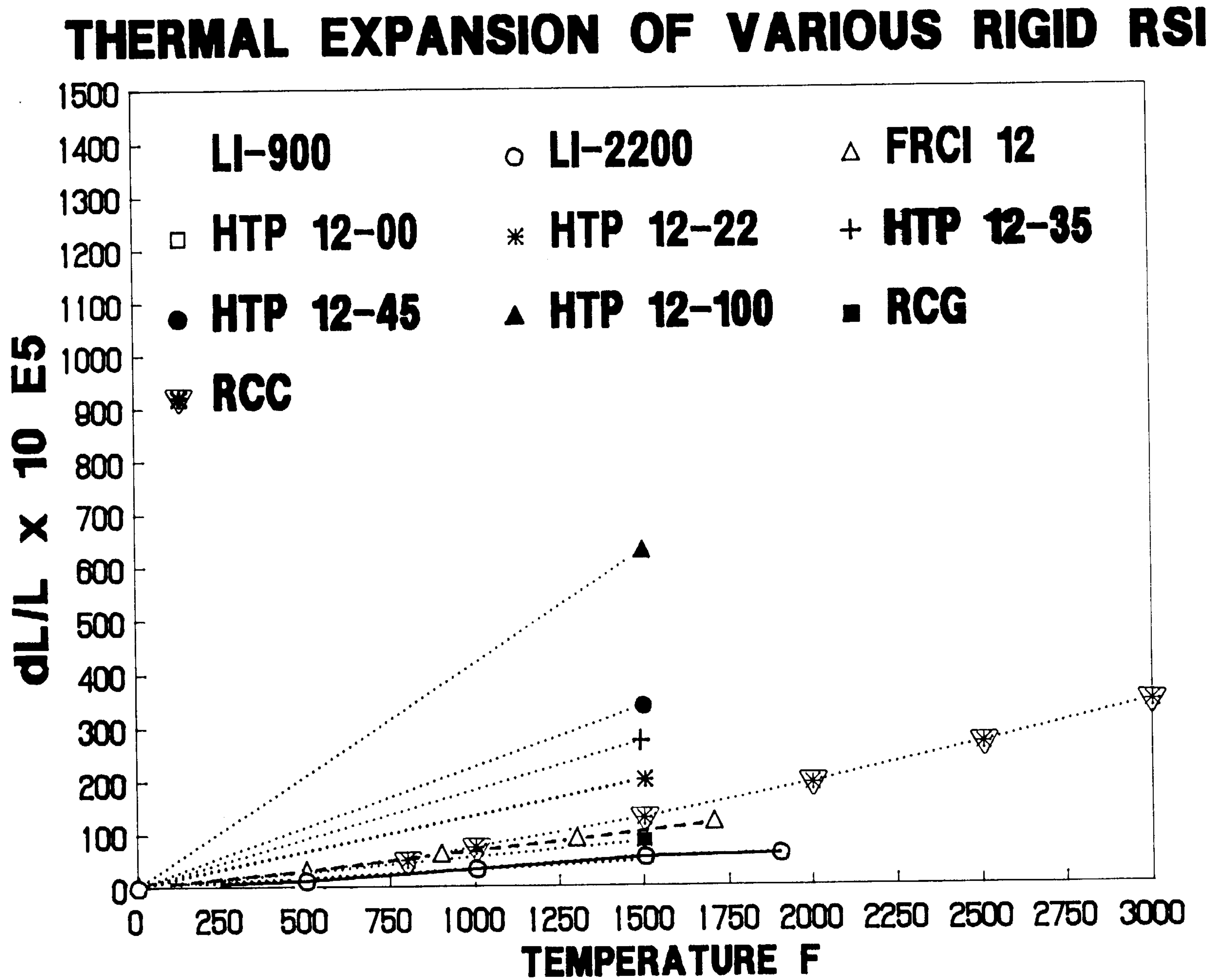
Optical Properties
HRSI–Class 2 Coating

TOTAL HEMISPHERICAL EMITTANCE (ϵ_{th})

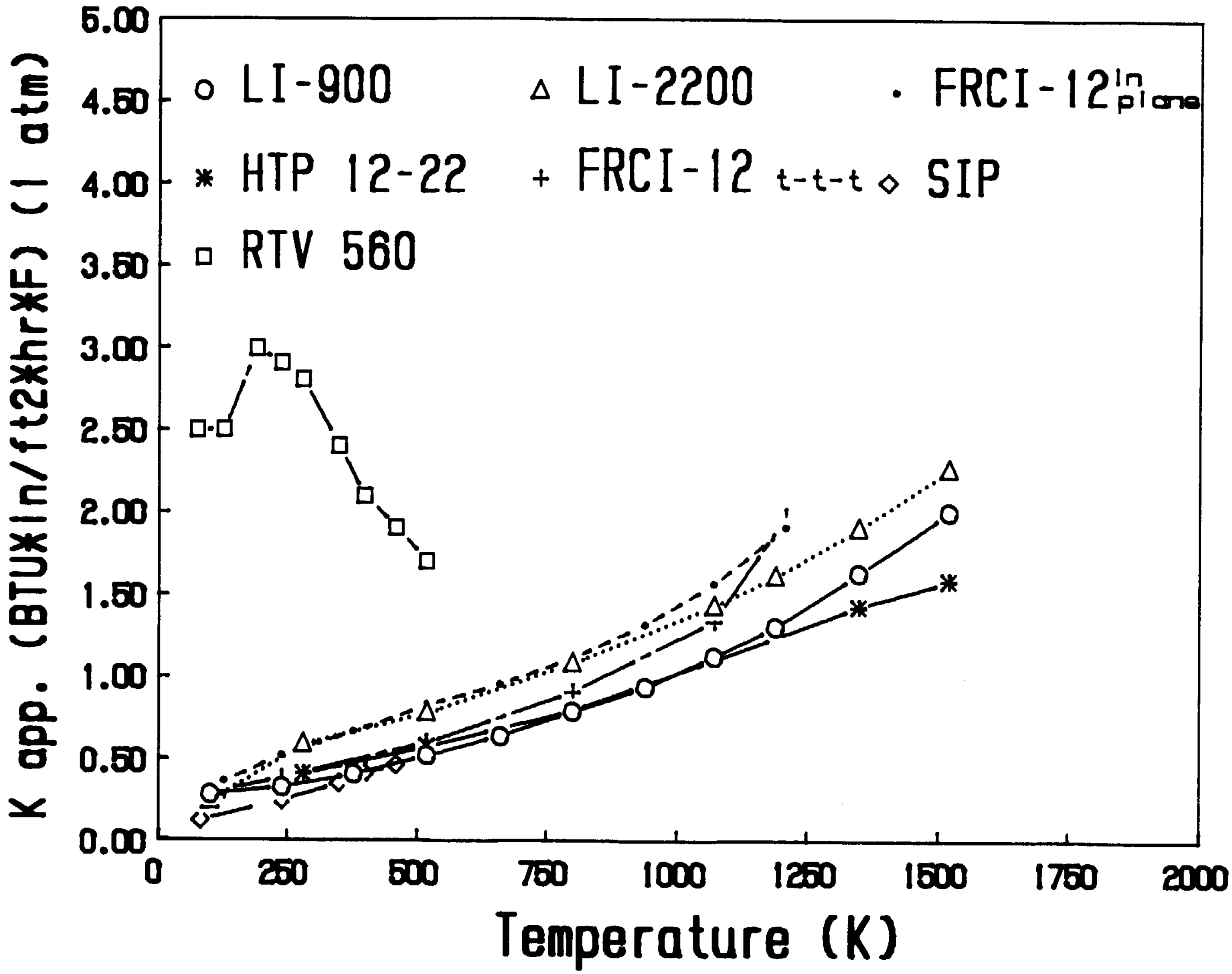
AND

$\frac{\alpha_s \text{ (SOLAR ABSORPTANCE)}}{\epsilon_{th}}$



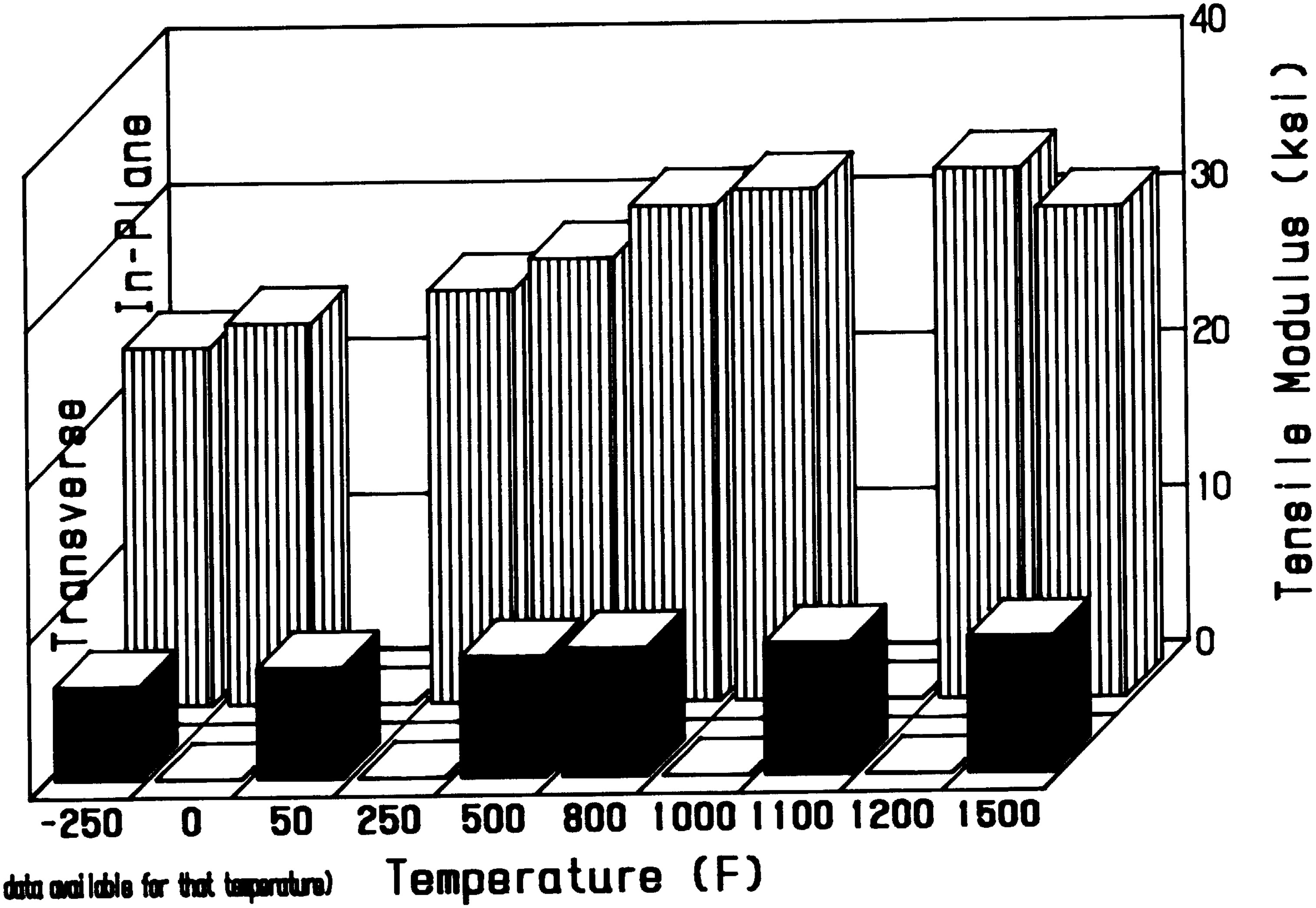

THERMAL EXPANSION OF VARIOUS RIGID RSI
LI-900
LI-2200
FRCI 12
HTP 12-00
HTP 12-22
HTP 12-35
HTP 12-45
HTP 12-100
RCG
RCC
dL/L x 10 E5
TEMPERATURE F
0
100
200
300
400
500
600
700
800
900
1000
1100
1200
1300
1400
1500
250
500
750
1000
1250
1500
1750
2000
2250
2500
2750
3000

THERMAL CONDUCTIVITY

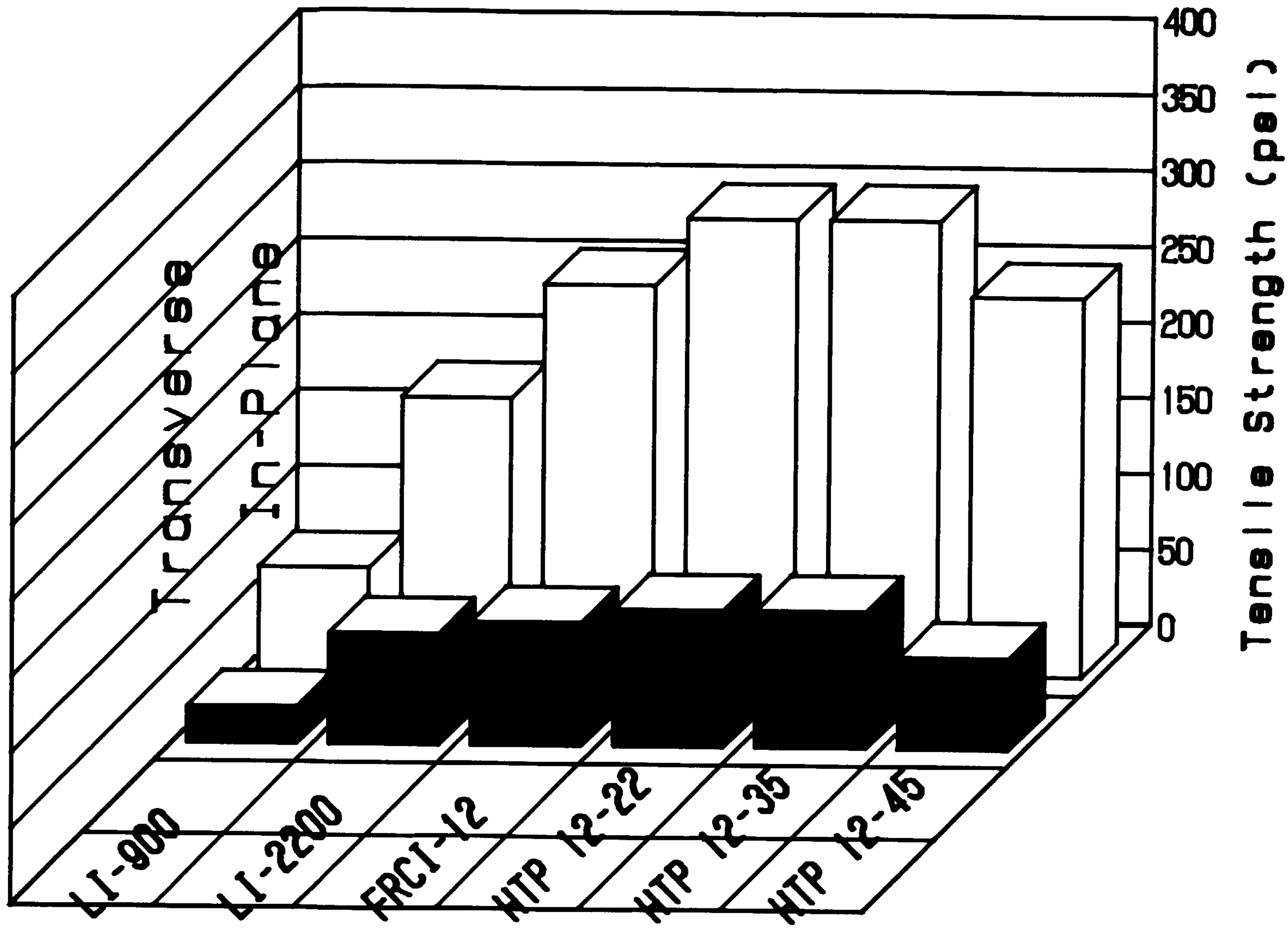


LI-900

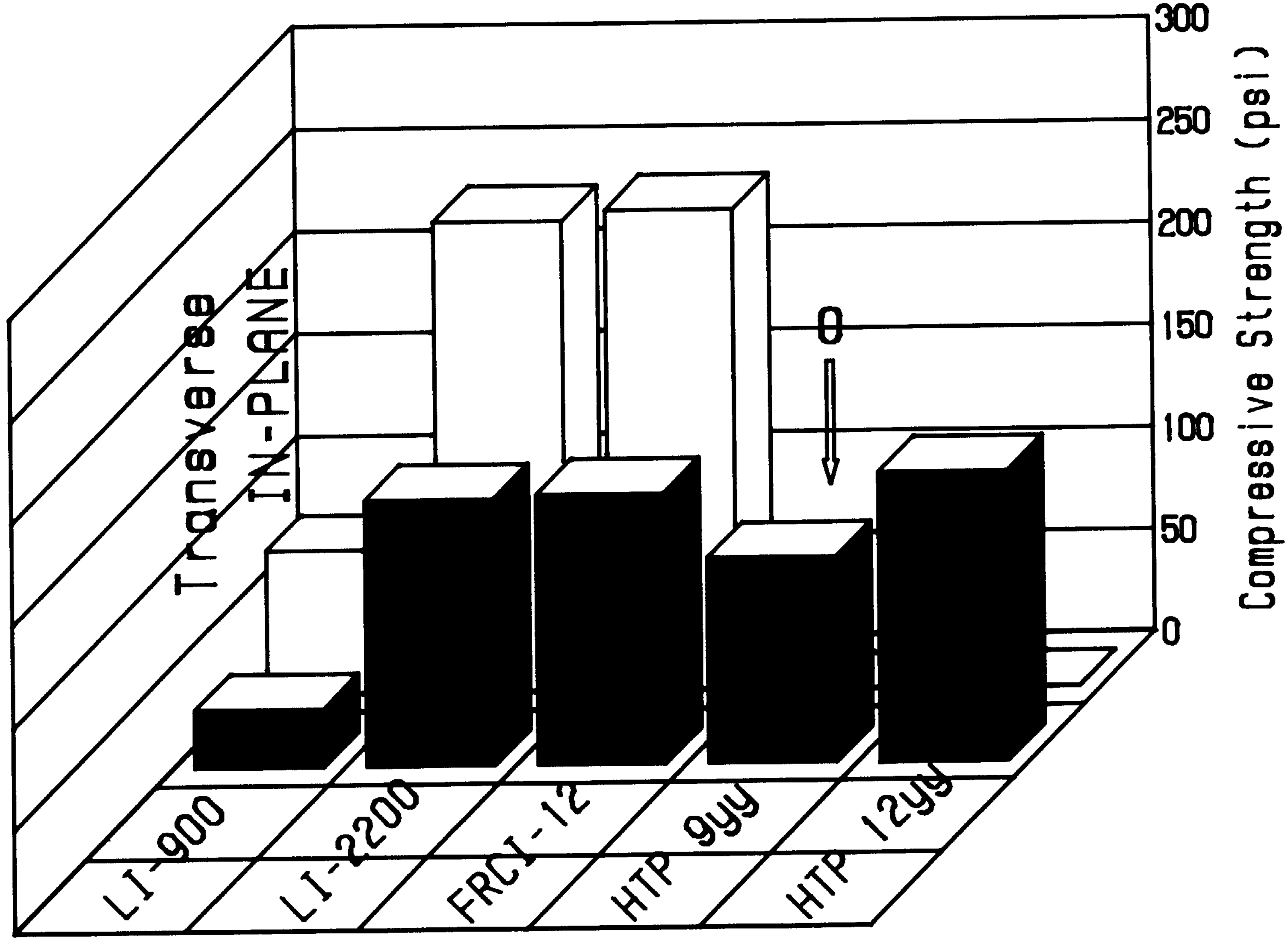
In-Plane and Transverse Tensile Moduli



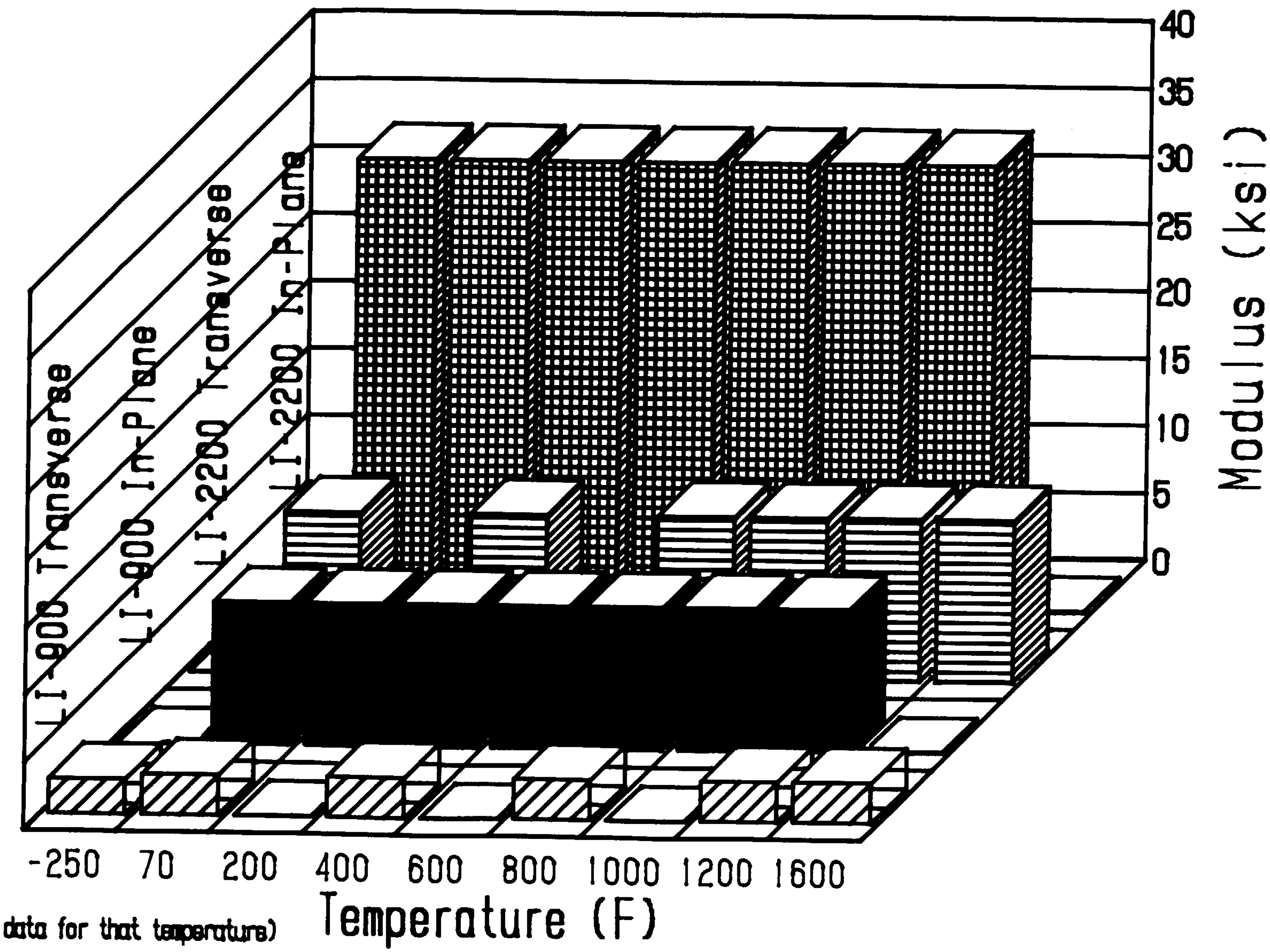
Tensile Strength (R.T.) of Various Rigid RSI



COMPRESSIVE STRENGTH (R.T.) OF VARIOUS RIGID RSI

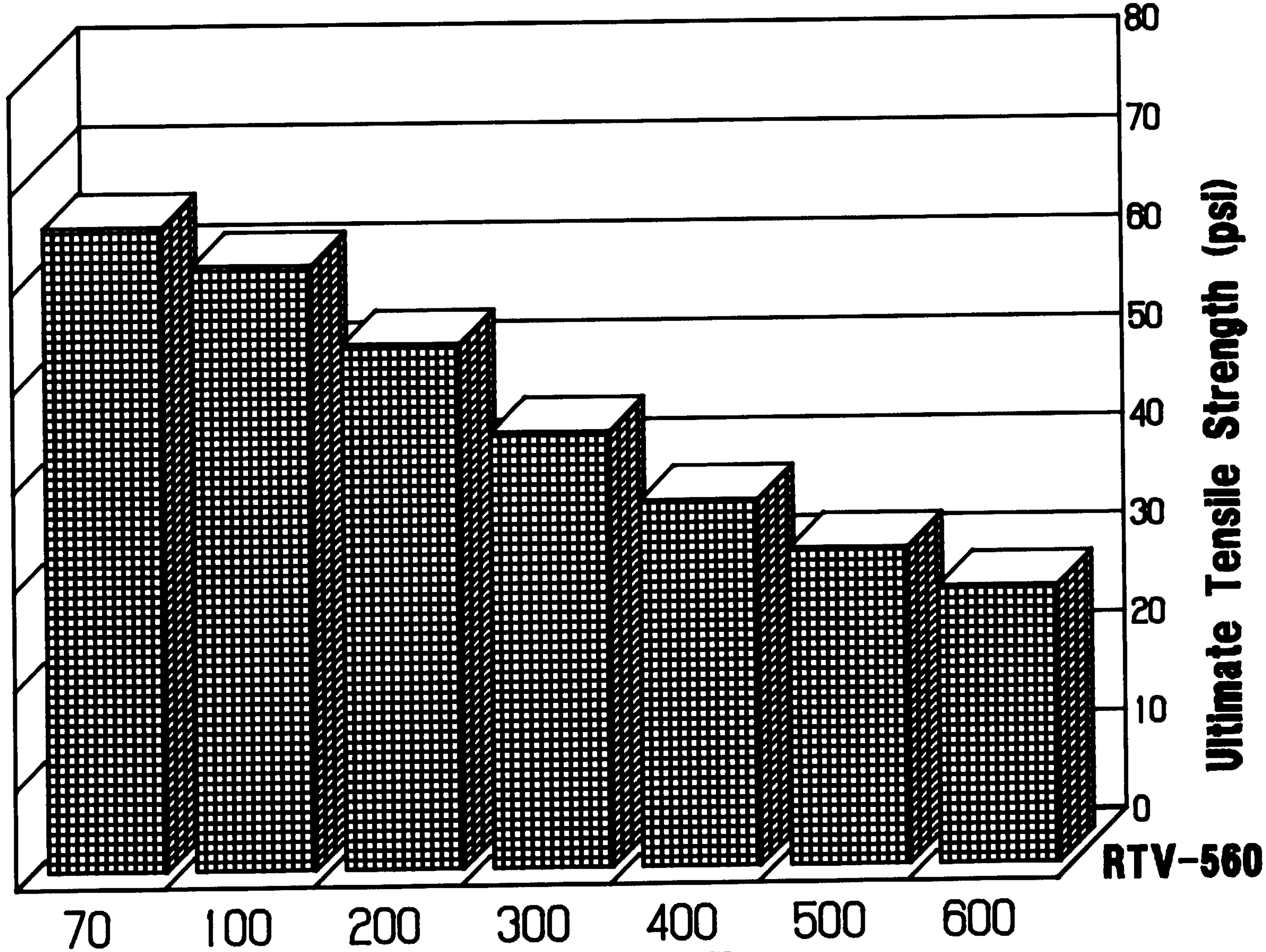


Various Shear Moduli of LI-900 AND LI-2200

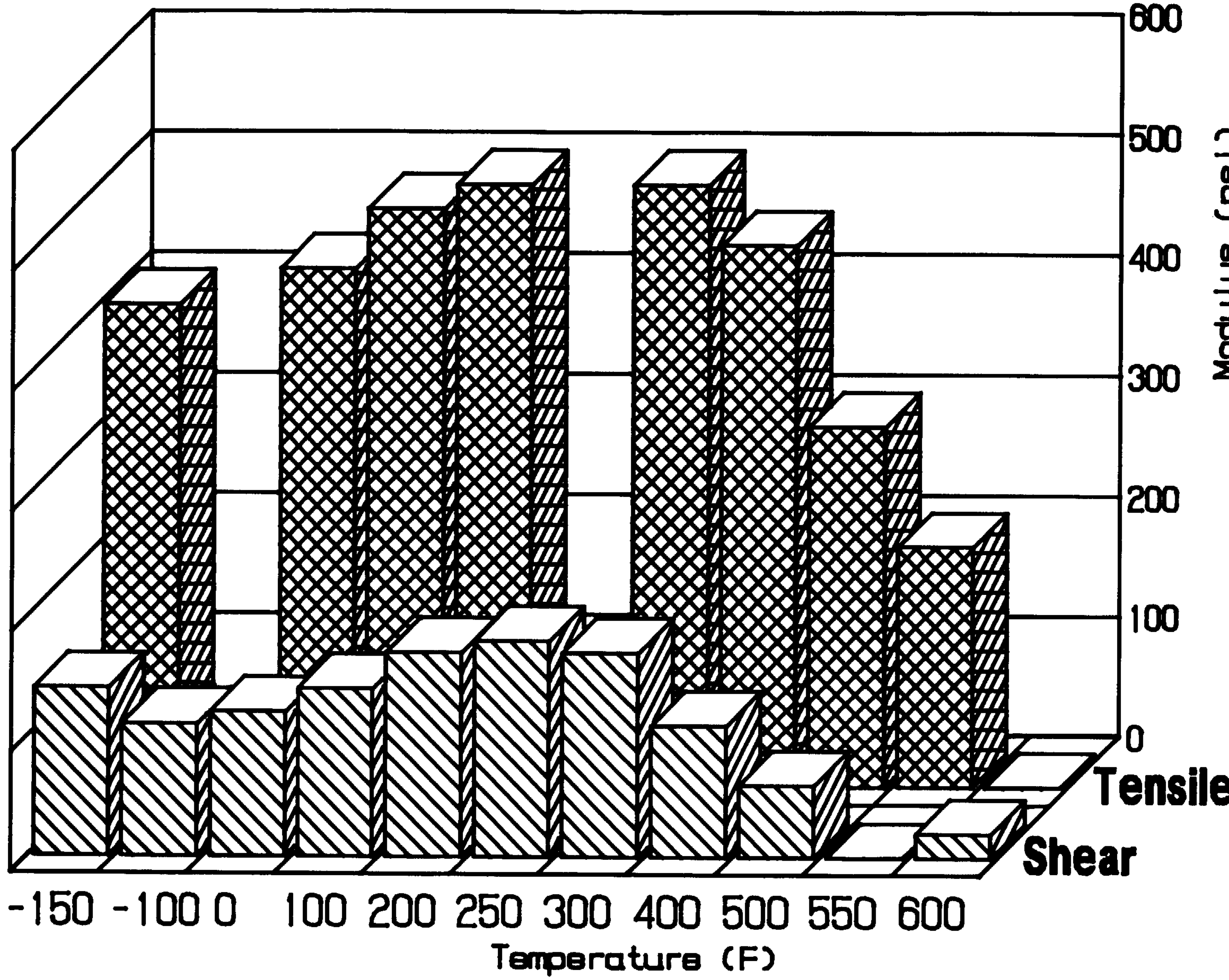


(0 means no data for that temperature)

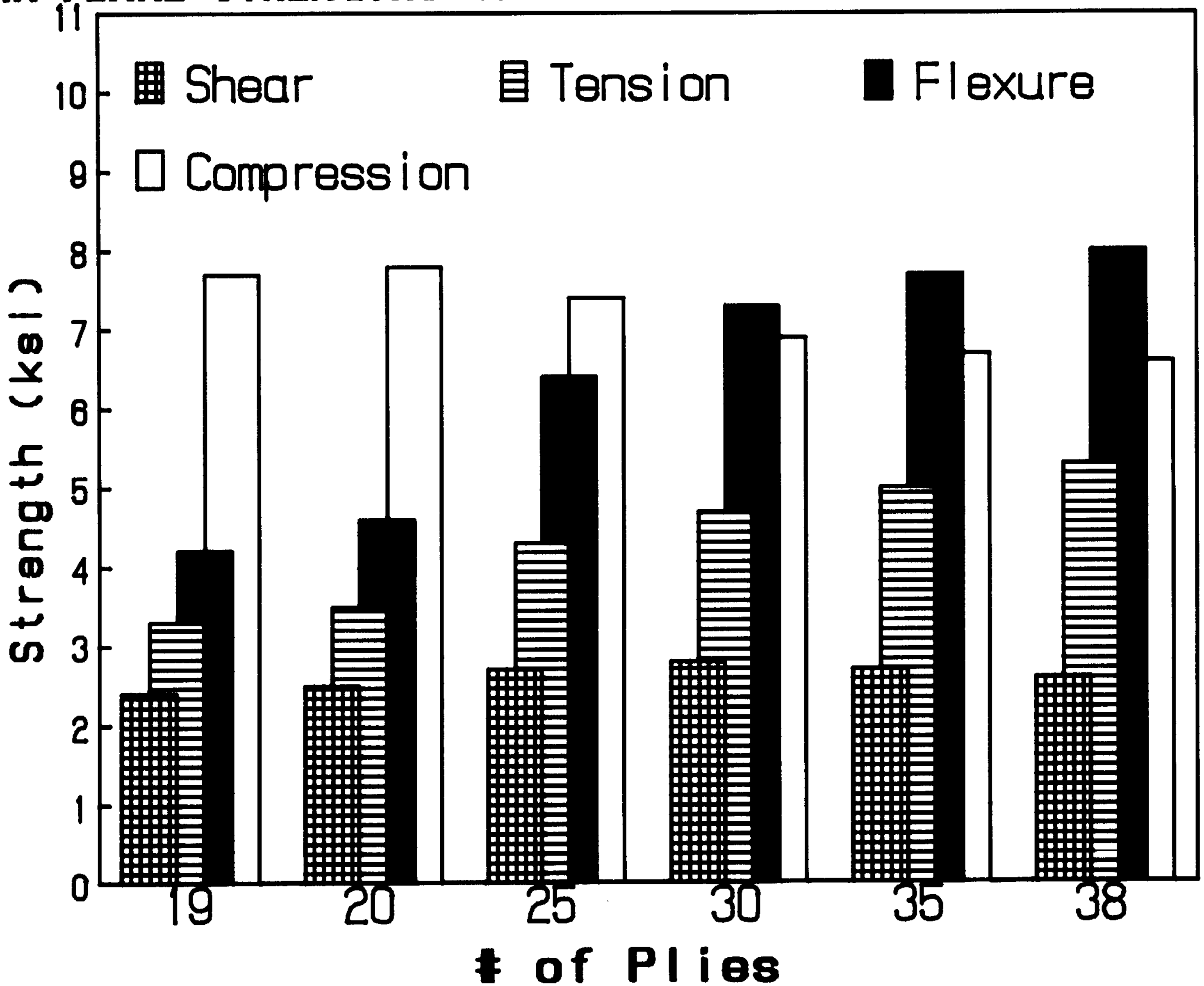
Flatwise Tensile Strength of RTV-560



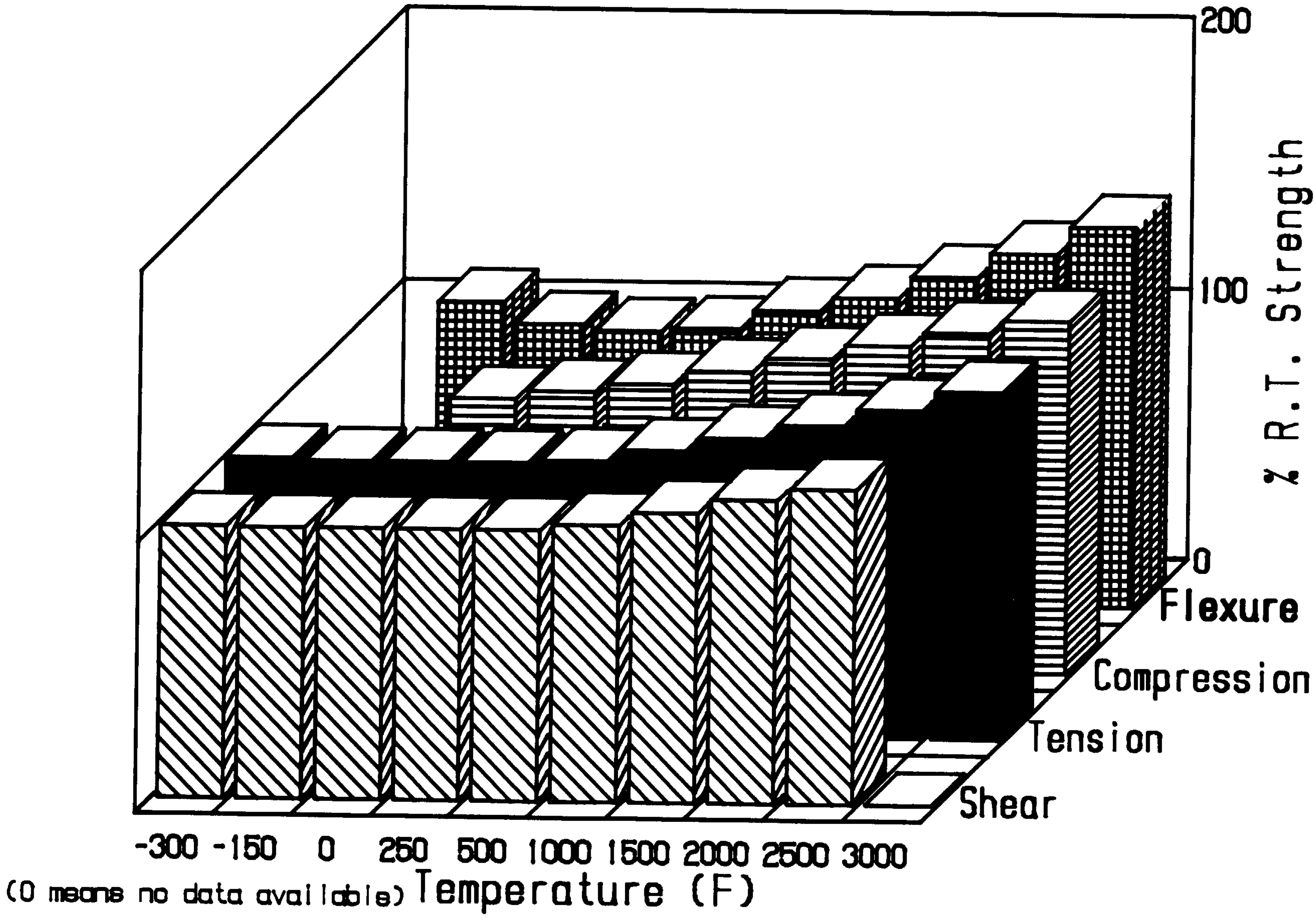
Tensile And Shear Moduli of RTV-560



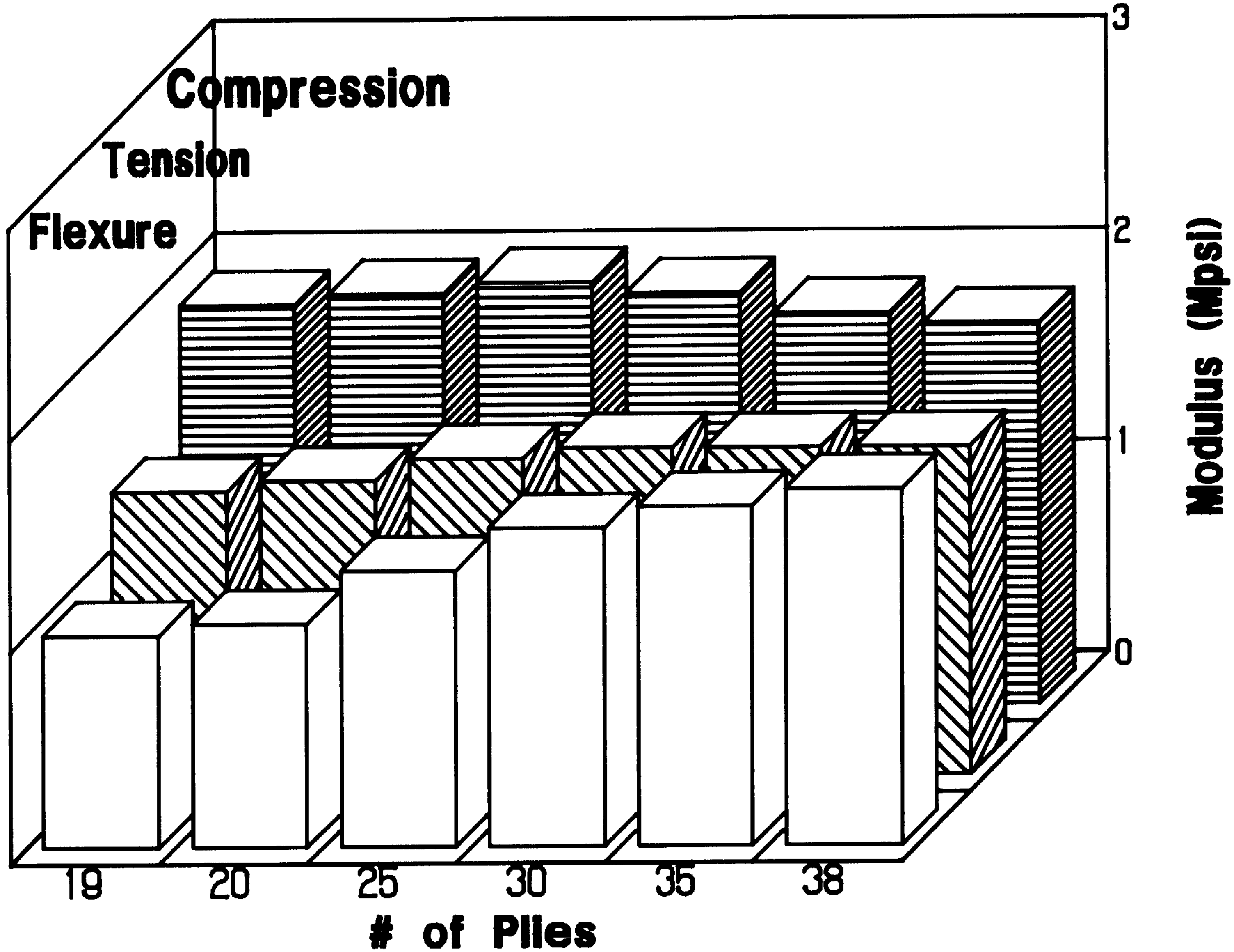
IN-PLANE STRENGTHS OF REINFORCED CARBON CARBON (RCC)



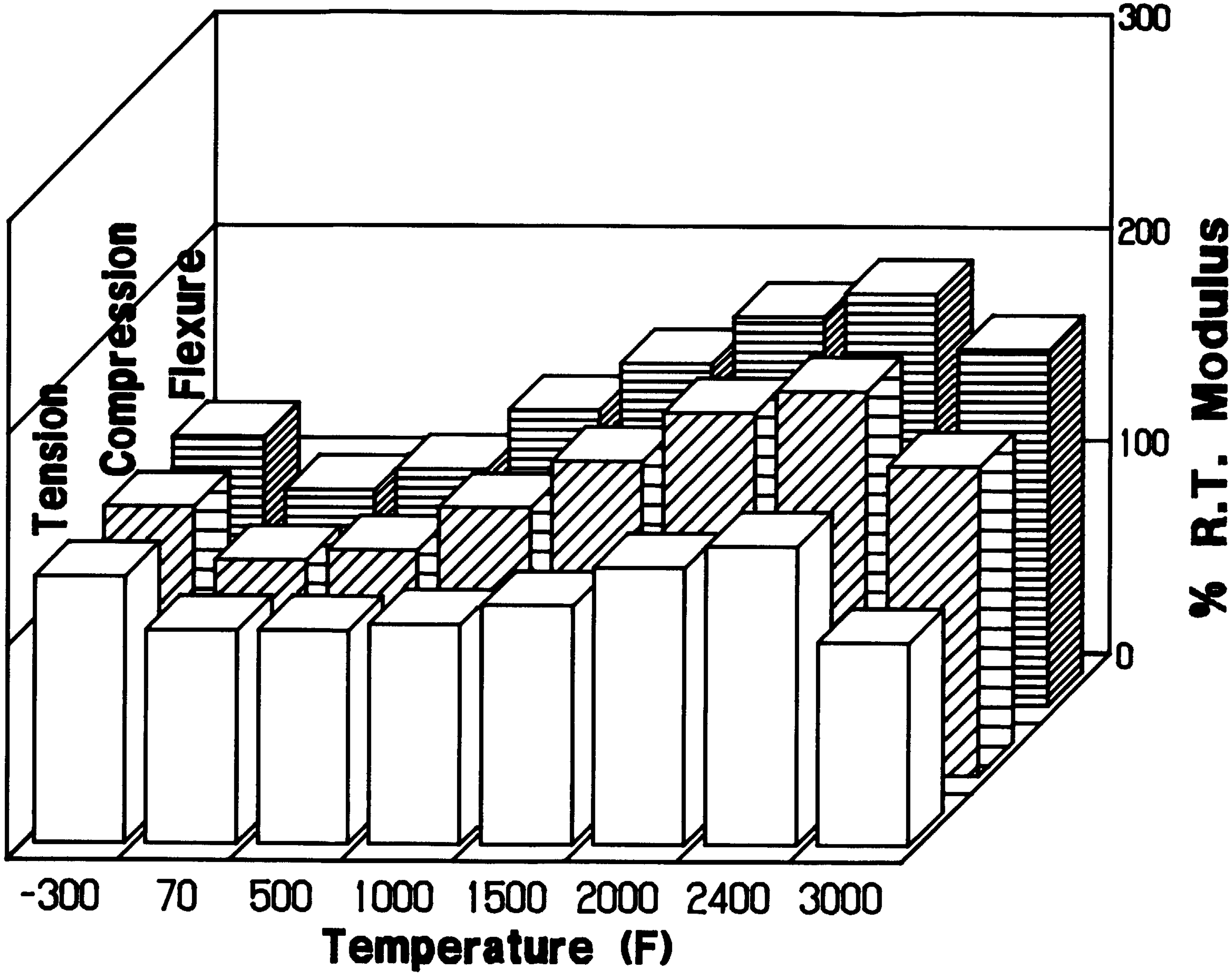
%R.T. In-Plane Strengths of RCC



Various Moduli of RCC



% R.T. Moduli of RCC



APPENDIX III

Fibrous Refractory Composite Insulation

An insulating material with improved mechanical and thermal properties

Ames Research Center, Moffett Field, California

A new family of high-temperature, low-density refractory composite insulations is made from aluminoborosilicate and silica fibers in a weight ratio ranging between 1:19 to 19:1. This ratio determines the properties of the resultant composite insulation. For example, the temperature capability of the insulation can be increased by increasing the ratio of aluminoborosilicate fiber to silica fiber, but at the expense of a higher thermal conductivity and thermal expansion coefficient. The reverse is also true. Improved resistance to silica fiber devitrification if required can be obtained by adding 0.5 to 30 percent of the total fiber weight boron oxide. Fibrous refractory composite insulation has a strain to failure that is greater than 0.5 percent because it does not contain any nonfibrous binder, which would inherently limit its strain to failure. Other properties include a modulus of rupture of above 6.4×10^6 N/m^2 at 0.32 g/cm^3 density and a temperature capability of 1,540° C or higher in transient applications.

The insulation is prepared using high-purity silica fibers that are first washed and dispersed in a hydrochloric acid solution and/or deionized water. About 30 to 150 parts liquid (pH 3 if acid is used) are used for 1 part fiber. The fibers are washed for 2 to 4 hours to remove the surface chemical contamination and nonfibrous material. After washing, the fibers are rinsed three times for 10 to 15 minutes with deionized water using the same liquid-to-fiber ratio. Excess water is drained off, leaving a ratio of 5 to 10 parts of water to 1 part fiber.

The aluminoborosilicate fibers are prepared by dispersing them in deionized water. About 10 to 40 parts of water are mixed with 1 part fiber in a V-blender for 2-1/2 to 5 minutes. In general, as the diameter and length of aluminoborosilicate fibers are increased, sufficient dispersion is obtained with more mixing time.

The dispersed silica and aluminoborosilicate fibers are then combined, and the pH is adjusted to basic with ammonium hydroxide. The slurry, containing 12 to 25 parts water to 1 part fiber, is mixed to a uniform consistency in a V-blender for 5 to 20 minutes.

The mixed slurry is poured into a mold for pressing into the desired shape (tiles in this case). The water is withdrawn rapidly, and the resulting felt is compressed at 10 to 20 psi (6.9×10^4 to 13.8×10^4 N/m^2). Rapid removal of the water prevents the two fibers from separating. The final density of the finished tile is determined in part by the amount of compression placed on the felt, varying the wet molded dimension in relation to the fiber content. Typical densities used have ranged from about 0.08 to 0.48 g/cm^3; however, higher densities can be prepared.

Following molding, the insulation tile is oven-dried for 18 hours and fired. The initial drying temperature of 38° C is raised at 11° C per hour to 104° C and is held there for 4 hours. From that level, the temperature is raised again at 11° C per hour to 150° C and is held there for 4 hours.

The dried tile is taken directly from the oven and placed into the firing furnace. There it is fired at a temperature rise rate of 220° C per hour (or less) to prevent warping or cracking the tile. The maximum firing temperature may vary from 1,260° to 1,360° C, depending on the fiber ratio used and the final density of the insualtion tile desired. After the firing, the tiles are machined to final dimensions.

The process has been studied using various proportions of the described constituents and process chemicals and temperatures. A very-thermally-shock-resistant, low-density insulation has been obtained with a ratio of about 4:1 silica fibers to aluminoborosilicate fibers, the latter containing about 14 percent of boron oxide. This composition has been found appropriate for use as a reusable heat-shield material.

This work was done by Howard E. Goldstein and Marnell Smith of **Ames Research Center** *and Daniel B. Leiser of Stanford University. For further information, Circle 48 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Ames Research Center [see page A8]. Refer to ARC-11169.

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Lightweight Ceramic Insulation

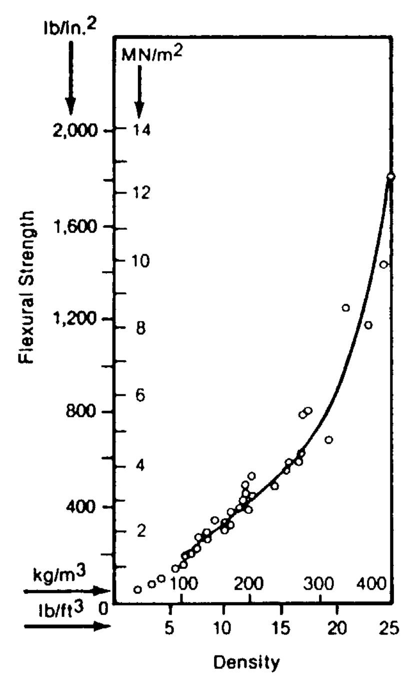
A fiber burnout process
yields low densities.

NASA Tech Briefs, May/June 1986

Lyndon B. Johnson Space Center, Houston, Texas

A rigid ceramic insulation has a density of 2 to 6 lb/ft^3 (32 to 96 kg/m^3) — low enough that it can replace flexible insulation in applications where low weight is essential. In contrast with the loose fibers or blankets of flexible insulation, the new insulation does not pack together or shift. It is thermally stable, and, with a flexural strength of 25 to 75 $lb/in.^2$ (170 to 500 kN/m^2), it retains its shape under light loading (see figure). Originally developed for use on the Space Shuttle and other spacecraft, the rigid insulation can be machined to the requisite shape and bonded in place.

The low density is attained by a process of sacrificial burnout. Graphite or carbon fibers are mixed into a slurry of silica, alumina, and boron-compound fibers in amounts ranging from 25 to 75 percent of total fiber content by weight. The mixture is formed into blocks and dried. The blocks are placed in a kiln and heated to 1,600° F (870° C) for several hours. During this time, the graphite or carbon fibers slowly oxidize away, leaving voids and thereby reducing the block density. Finally, the blocks are heated to 2,350° F (1,290° C) for 90 minutes to bond the remaining ceramic fibers together.



The **Flexural Strength** of lightweight rigid insulation (solid dots) follows the curve for higher density ceramic tiles of the same composition (small circles).

Before the graphite or carbon fibers are added to the casting slurry, they are chopped and dispersed in deionized water. Graphite and carbon were chosen as the materials for the burnout after experiments with cotton and nylon fibers. The graphite and carbon fibers proved to be more compatible with the rigid-insulation manufacturing process.

This work was done by William H. Wheeler and John F. Creedon of Lockheed Missiles & Space Co. Inc. for **Johnson Space Center.** *For further information, Circle 56 on the TSP Request Card.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Johnson Space Center [see page 29]. Refer to MSC-20831.

Improving Emittance of High-Temperature Insulating Tile

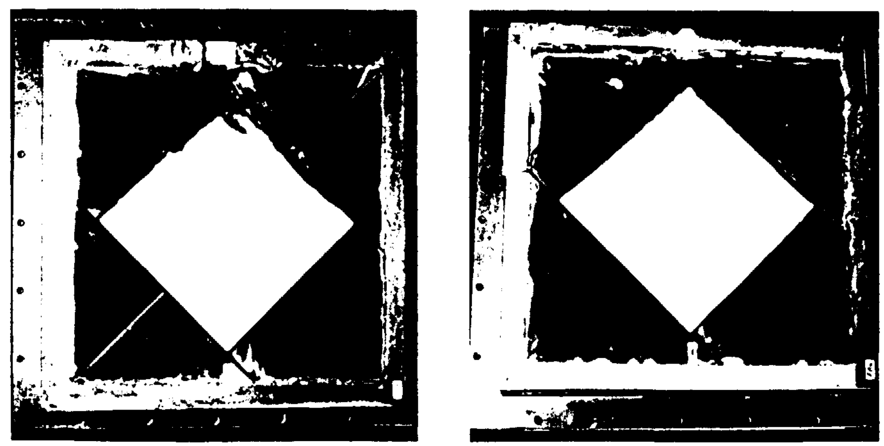
An additive protects the tile even
when the surface coating fails.

Lyndon B. Johnson Space Center, Houston, Texas

A simple addition to ceramic insulating tiles provides backup properties that minimize the transfer of heat through the tiles when their surfaces become damaged. Previously, such tiles have been coated with reaction-cured glass to provide the surface with the high emittance and other optical properties necessary to prevent the heating of the tile surface to temperatures above 2,300° F (1,260° C). The coating also makes the tiles abrasion-resistant and impervious to water. However, if a tile becomes damaged and loses part of its coating, it no longer protects the underlying structure, because the emittance of the white interior ceramic is less than that of the coating.

The addition of 3 percent by weight of 320- or 600-grit silicon carbide powder to the ceramic during production results in an impregnated tile material that resists overheating. The silicon carbide increases the emittance and decreases the transmittance of the ceramic. As a result, increased radiation outward from the surface reduces the surface temperature, and less radiant energy is transmitted from the tile surface through the ceramic insulation to the aluminum structure.

Tiles with and without the coating were subjected to arc-plasma tests (see figure). The coated tiles withstood a 2,300° F (1,260° C) surface temperature with no degradation. Uncoated tiles eroded and partially melted, and their aluminum substructure melted locally. But uncoated tiles with silicon carbide survived the tests with only minor shrinkage.



An **Insulating Tile** without reaction-cured glass coating is damaged after arc-plasma testing (left), and the aluminum supporting it is partly melted. A tile with silicon carbide additive (right) is slightly shrunken after the test but is otherwise in good condition, even though it is uncoated. There is also less damage to the aluminum structure.

This work was done by Edward R. Gzowski of Lockheed Missiles & Space Company, Inc., for **Johnson Space Center.** *No further documentation is available.*

Inquires concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Johnson Space Center [see page 29]. Refer to MSC-20714.

ORIGINAL PAGE IS
OF POOR QUALITY

Accelerated Purification of Colloidal Silica Sols

A heat/deionization scheme sharply reduces the time required to purify silica sols.

NASA Tech Briefs, Winter 1978

Lyndon B. Johnson Space Center, Houston, Texas

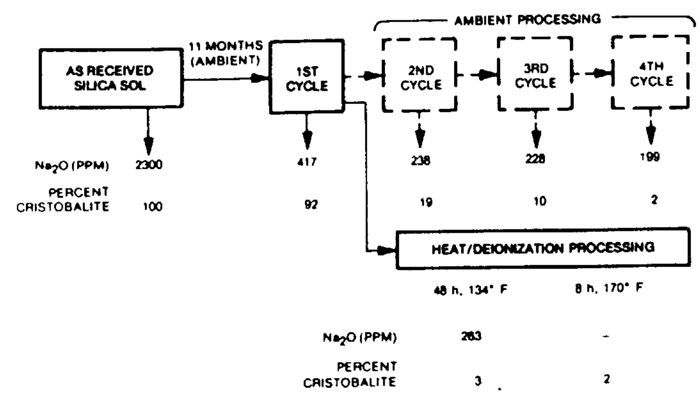
In an accelerated purification process for colloidal silica sols, the waiting time between deionization cycles is reduced from several months to a few days. The process produces the same high-purity silica sol as the previous, more time-consuming method. The key to the new technique is the use of elevated temperatures to accelerate the removal of Na^+ ions following the first deionization.

High-purity silica sol is a binder for pure silica fibers in reusable surface insulation. The sol must retain its amorphous structure if the insulation is to be stable at high temperatures. A major contaminant and destabilizing factor is sodium, which is used in the processing of colloidol sols. It is derived either from sodium silicate or from high-pressure/high-temperature processing in which NaOH prevents gelation. At high temperatures, the sodium devitrifies the structure into crystalline phases, such as cristobalite, degrading the cohesiveness of the binder.

The objective of the purification process is to reduce the Na_2O content from around 2,300 ppm to a purified level of less than 300 ppm. In the conventional approach, a quantity of sol is treated by column deionization to remove Na^+ by ion exchange. This is followed by a 30-day storage at ambient temperatures to allow residual Na^+ to migrate to the surface of the silica particles. The process is repeated in three additional ionization cycles requiring 30-day intervals between each.

The new process was developed after a series of experiments. The first approach — heating the product prior to the first deionization — showed no improvements. Significant improvement, however, did occur when heat was applied following the first deionization cycle, and the particle diameters were about 15 μm. The purified sol was obtained after about 48 hours at 134° F (57° C) and in 8 hours at 170° F (77° C). Moreover, ion-exchange resin deionization with the sol at elevated temperatures further reduced purification times to 3 hours at 170° F. The new process and results are compared with the previous method in the figure.

Other alternatives have also been examined. One, for example, involves growing the colloidal particles from parent silica acid stabilized with an amine rather than the NaOH. This process, however, is limited to particle sizes of only about 4 μm. Another ion-exchange process, using a recycling, pressurized heating system to build up sols with 15- to 150-μm particle sizes, does not meet the high-purity standard but is of interest in particle-growing systems evolving colloidal silica solutions.



A Comparison of Conventional and New Purification Processes for Colloidal Silica Sols is shown here. The particle diameter of silica is about 15 μm. The Na_2O-ppm content is based on ignited weight, and the percent of cristobalite formed after 4 hours at 2,300° F (1,260° C) indicates the stability of the product at elevated temperatures.

This work was done by E. Bahnsen, S. Garofalini, and A. Pechman of Lockheed Missiles & Space Co., Inc., for **Johnson Space Center.** *No further documentation is available.*

MSC-16793

Coated-Felt Thermal Insulation

Thin, flexible, lightweight insulation tile for temperatures below 700° F

NASA Tech Briefs, Winter 1978

Lyndon B. Johnson Space Center, Houston, Texas

A thin coated-felt insulation tile has been shown to be a lighter and easier-to-install replacement for silica tiles. During the design of NASA's Space Shuttle Orbiter, the felt insulation was developed as a potential replacement for low-temperature reusable surface insulation (LRSI) made of silica.

At temperatures under 700° F (370° C), silica tiles of the minimum durable thickness (about 0.5 cm) would overprotect the Orbiter, thereby adding unnecessary weight. The felt tiles could not only be made thinner and lighter but could be applied in larger sheets (1 by 2 meters) to save time, and they are flexible to accommodate buckling and conform to curved surfaces.

The felt is a commercially-available modified nylon-fiber material. For use as insulation, it is pretreated and coated. The pretreatment is exposure to 750° F (400° C) to prevent adverse shrinkage during service. The felt is heated in five incremental steps to drive off volatiles that might otherwise burn in the presence of oxygen.

For application as a reusable insulation tile, the felt must be coated to prevent charring, to waterproof it, and to give it the needed optical (thermal) properties. An elastomeric silicone, pigmented with TiO_2, meets these requirements, giving the tile a solar absorptance-to-emittance ratio less than 0.4, and also imparts a relatively-smooth aerodynamic surface. The coating is applied and allowed to cure at room temperature (with a 15-minute postcure at 650° F/345° C).

The felt thickness can be from 0.16 to 0.4 in. (0.4 to 1 cm) as dictated by the local structural heat sink, the expected maximum heat load, and other considerations. It can be applied to the surface to be insulated with a room-temperature-vulcanizing adhesive.

This work was done by Robert L. Dotts, Robert J. Maraia, James A. Smith, Ivan K. Spiker, and George Strouhal of **Johnson Space Center.** *For further information, Circle 50 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Johnson Space Center [see page A8]. Refer to MSC-12737.

Insulation Blankets for High-Temperature Use

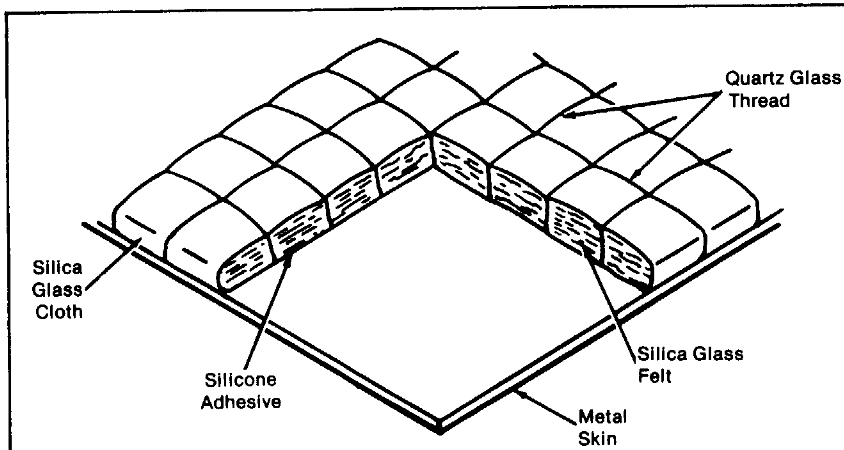
Lightweight, flexible material resists intense heat.

Ames Research Center, Moffett Field, California

An insulating blanket resists temperatures up to 1,500° F (815° C). It is useful where high-temperature resistance, flexibility, and ease of installation are important — for example, as insulation for odd-shaped furnaces and high-temperature ducts, as curtains for furnace openings and for fire control, and as conveyor belts in hot processes.

The blanket is a quilted composite consisting of two face sheets: the outer one of silica, the inner one of silica or other glass cloth with a center filling of pure silica glass felt sewn together with silica glass threads (see figure). The felt fibers have diameters ranging from 1 to 3 μm. The fibers are amorphous silica rather than crystalline; thermal expansion is therefore low, and crystalline inversions causing sudden changes in volume do not occur. A major advantage of the blanket insulation is its low mass density — about 8 lb/ft^3 (128 kg/m^3). It is lighter than the ceramic tile insulation used on the Space Shuttle, for example.

The new insulating blanket is manufactured in various thicknesses from 1/8 to 2 inches (3.2 to 50.8 mm) in a standard 33- by 33-inch (83.8- by 83.8-cm) size. Small pieces can be cut from stock for application to sharp corners or areas of complex curvature. It is quilted in squares approximately 1 inch (2.54 cm) on a side. The outside layer of the blanket can be coated with a moisture-resistant material. The inside layer can be bonded directly to the metal or other skin of the object to be insulated by a room-temperature-vulcanizing adhesive (RTV-560 or equivalent). This adhesive, applied at a thickness of 0.010 inch (0.25 mm), keeps weight low and minimizes thermal-expansion stress during temperature changes.



Flexible, Reusable Insulation for surfaces is composed of quilted silica felt covered on the top by silica and on the bottom by silica or other glass cloth. The insulation can withstand temperatures of 1,500° F (816° C).

This work was done by Howard Goldstein, Daniel Leiser, Paul M. Sawko, Howard K. Larson, Carlos Estrella, and Marnell Smith of **Ames Research Center** *and Frank J. Pitoniak of the the U.S. Air Force. No further documentation is available.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Ames Research Center [*see page 29*]*. Refer to ARC-11453.*

Silicone-Rubber Stitching Seal

Placement of sealant along seams
prevents thread loosening.

Lyndon B. Johnson Space Center, Houston, Texas

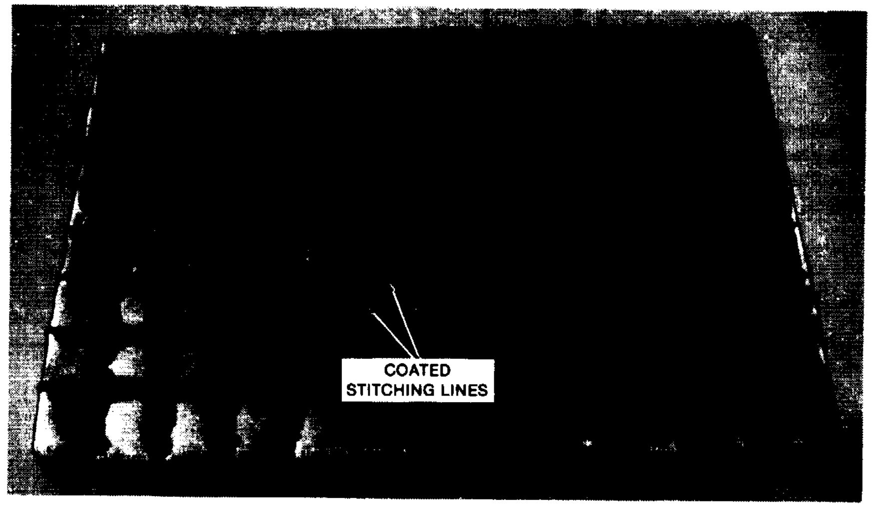


Figure 1. The **Stitches of an Insulating Quilt** are coated with silicone rubber. The rubber anchors the threads, preventing "runs."

Fabric products can be protected from raveling by coating the threads and filling the stitching holes with silicone rubber. The rubber also waterproofs the seam and helps to prevent gas leakage.

The technique is designed to coat a flexible insulating quilt along the stitches on one surface only (see Figure 1). For other products, the technique can be varied as needed, provided that care is taken to keep the silicone rubber in the stitching holes.

Uncured silicone rubber is applied to the stitching lines with an air-pressurized sealant gun (see Figure 2). Next, a plastic release film is placed on the coated side, and the blanket is flipped over so that the release film lies underneath. This allows gravitation to pull the liquid rubber downward, away from the surface not to be coated; and the plastic film prevents the liquid rubber from leaking out of the stitching holes. The blanket may then be bagged and the adhesive cured under a partial vacuum of about 3.5 psi (24 kN/m^2) or under pressure.

Possible applications include balloons, parachutes, ultralight aircraft, sails, rescue harnesses, tents, or other fabric products that are highly stressed in use.

This work was done by David S. Wang of Rockwell International Corp. for **Johnson Space Center.** *No further documentation is available.*
MSC-20708

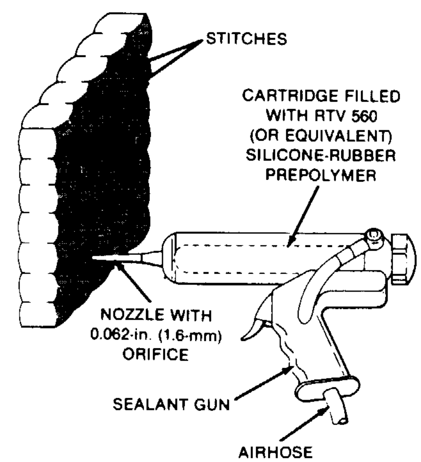


Figure 2. A **Sealant Gun** applies uncured silicone rubber to a row of stitches.

NASA Tech Briefs, Summer 1985

Bonded Lockstitch for Insulating Blankets

An adhesive prevents unraveling at high temperatures.

NASA Tech Briefs, Winter 1983

Lyndon B. Johnson Space Center, Houston, Texas

An improved sewing technique for high-temperature [750°-to-1,200° F (400°-to-650° C)] insulating blankets prevents stitch failure in hot, turbulent environments. In the new technique, the standard lockstitch is modified to isolate single-stitch failures.

The insulating blanket consists of a fibrous batting sandwiched between a quartz-fabric outer facing and a glass-fabric inner facing, with nominal total thickness ranging from 0.45 to 0.95 in (1.1 to 2.4 cm). The inner and outer facings are stitched with an outer (needle) thread of polytetrafluoroethylene-coated quartz and an inner (bobbin) thread of glass, with a stitch density of 4 ± 1 per in. (1.6 ± 0.4 per cm).

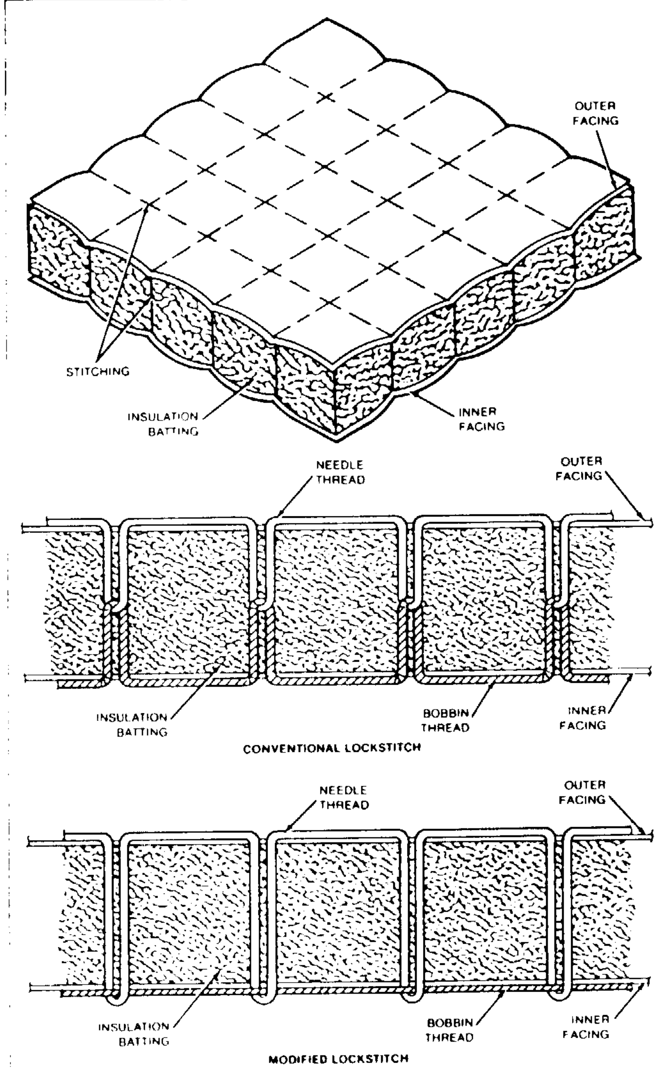
In conventional lockstitching (shown in the middle part of figure), a loop of the needle thread is passed through the two facings and caught by the bobbin thread. The needle thread is then pulled back until the interlacing of both threads is midway between the facings. If the thread fails at one stitch, the thread can loosen along its entire length, thus unraveling an entire row of stitches.

In the improved technique (lower part of figure), the bobbin thread is kept at the blanket surface. A silicone adhesive is applied to all the bobbin/needle intersections, so that a failure at one point will not propagate along the thread.

The improved blanket has been tested in experiments and is scheduled for use on the Space Shuttle. It is suitable for use in aerodynamic and other applications where there is turbulence. Fabrication is simple, since the modified lockstitch can be produced by a simple adjustment of thread tension and sewing-machine timing.

This work was done by Jonathan M. Rivin, Charles A. Morant, and Richard M. Ehret of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 64 on the TSP Request Card.*

MSC-20283



An Insulating Blanket (top) is sewn with conventional (middle) or modified (bottom) lockstitches. The threads are cemented together at their intersections in the modified stitching method to prevent unraveling.

Abrasion-Resistant Coating for Flexible Insulation

A two-step process increases the ruggedness of fragile quartz fabric.

Lyndon B. Johnson Space Center, Houston, Texas

A ceramic coating increases the durability and heat resistance of flexible high-temperature insulation. The coating is compatible with the quartz-fabric insulation and allows it to remain flexible during and after repeated exposures to temperatures of 1,800 °F (982 °C). It prevents the fabric from becoming brittle while increasing its resistance to aerodynamic abrasion and loading.

The coating consists of a penetrating precoat and a topcoat. The major ingredients are high-purity colloidal silica binder and ground silica filler, which ensure stability and compatibility with the fabric at high temperatures. The ratio of binder to filler must be carefully controlled to ensure erosion resistance. Also essential to erosion resistance is adhesion to the fabric; this is provided by the precoat, which penetrates the waterproofing on the fabric and interlocks with the fibers.

The precoat consists of a solution of 80 parts by volume of colloidal silica in 20 parts isopropyl alcohol. The ingredients are mixed by adding the alcohol to the silica while stirring.

The topcoat consists of a mixture of 47 percent by weight colloidal silica and 53 percent ground silica. The ingredients are mixed on a rolling mill in a porcelain grinding jar containing Alundum (or equivalent) aluminum-oxide pebbles. Both the precoat and the topcoat should be stored in sealed polyethylene containers. The materials have long shelf lives, provided that the precoat is agitated once per week and again just before use.

The precoat is brushed or sprayed on the fabric so as to coat it completely and uniformly. About 0.12 lb/ft^2 (0.59 kg/m^2) of precoat should be applied per 100 ft^2. It should be allowed to dry for at least 4 hours before the topcoat is applied.

The topcoat is similarly brushed or sprayed, with special care taken that the slurry is worked into the spaces between fibers. About 12 lb should be applied per 100 ft^2. The topcoat should be dried for 24 hours before the fabric is used. Both the precoat and topcoat can be cured at room temperature.

This work was done by Daniel Mui and Ronald E. Headding of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 50 on the TSP Request Card.* *MSC-20799*

Fitting Flexible Coverings to Contoured Surfaces

Contour-transferring technique simplifies
precise fitting of flexible insulation.

NASA Tech Briefs, Winter 1983

Lyndon B. Johnson Space Center, Houston, Texas

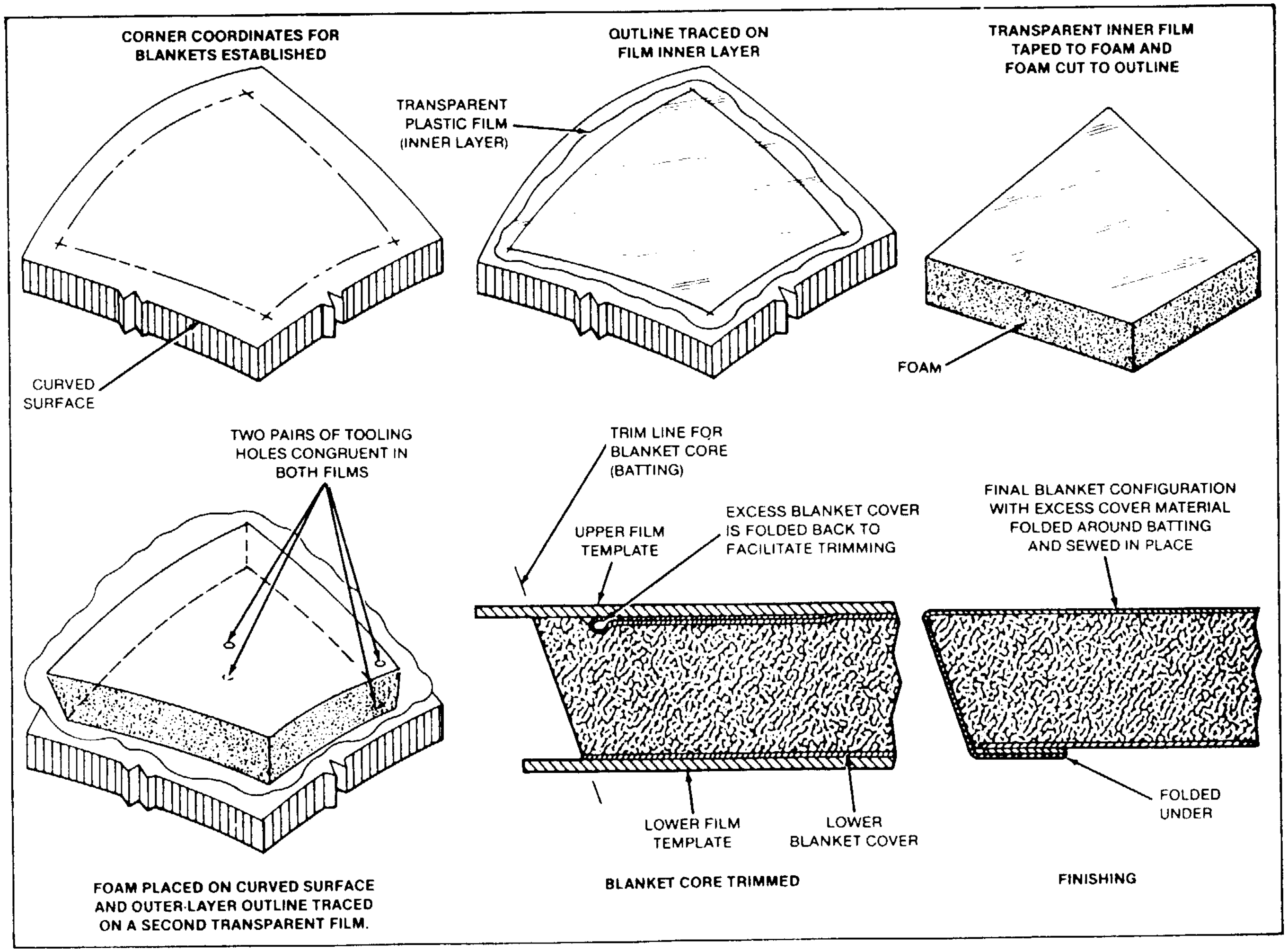


Figure 1. **Steps in the Contour-Transferring Procedure** are straightforward. With the procedure, many pieces of blanket insulation can be cut so that they fit snugly together on a contoured surface.

A method using two transparent plastic sheets and a polyethylene foam spacer produces flat templates from contoured surfaces. Once prepared, the templates are laid flat, and insulation inserted between the two templates is cut to shape to fit the contoured surface. The method was developed for fitting flexible insulation blankets over contoured surfaces aboard the Space Shuttle.

Figure 1 shows how the templates are prepared. The contoured surface is marked with corner coordinates where the actual insulation will be placed. A transparent plastic film is laid over the surface, and the outline is traced on the film layer. Polyethylene foam of the same thickness as the insulation is next taped on the outlined film and cut along the outline. This completes one template that defines the inner mold line.

The shaped foam and film are placed again on the marked surface, film down. Another sheet of film is placed on the foam, and the outline of the outer surface of the foam is traced on it. Alinement holes are punched in both films, and the upper film is trimmed along the foam perimeter. The second template is complete and defines the outer mold line.

The assembly is removed and the foam spacer detached. Insulation of the same thickness as the spacer is inserted between the two alined templates and trimmed along the template perimeters. When this insulation is installed, it will conform to the contoured surface.

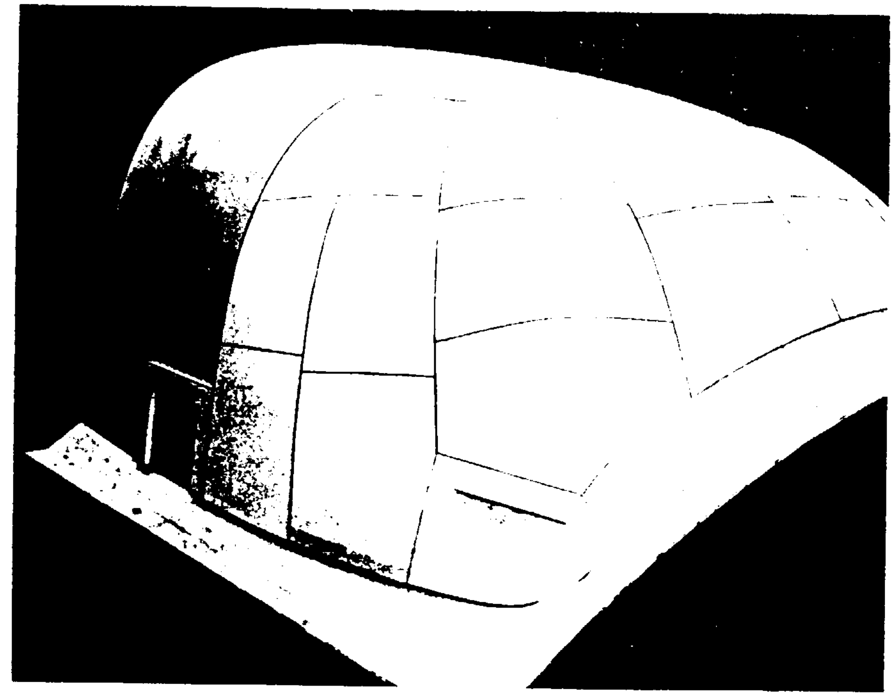


Figure 2. Sections of Insulation prepared by the contour-transferring technique fit the contoured surface precisely.

The procedure can also be used for tailoring protective covers or for the installation of vibration-absorbing material over contoured surfaces. Figure 2 shows a contoured surface with insulation sections in place.

This work was done by Donald D. Helman, Stanley Y. Yoshino, and David S. Wang of Rockwell International Corp. for **Johnson Space Center.** *No further documentation is available.*
MSC-20503

Tailorable Advanced Blanket Insulation (TABI)

NASA Tech Briefs, September 1987

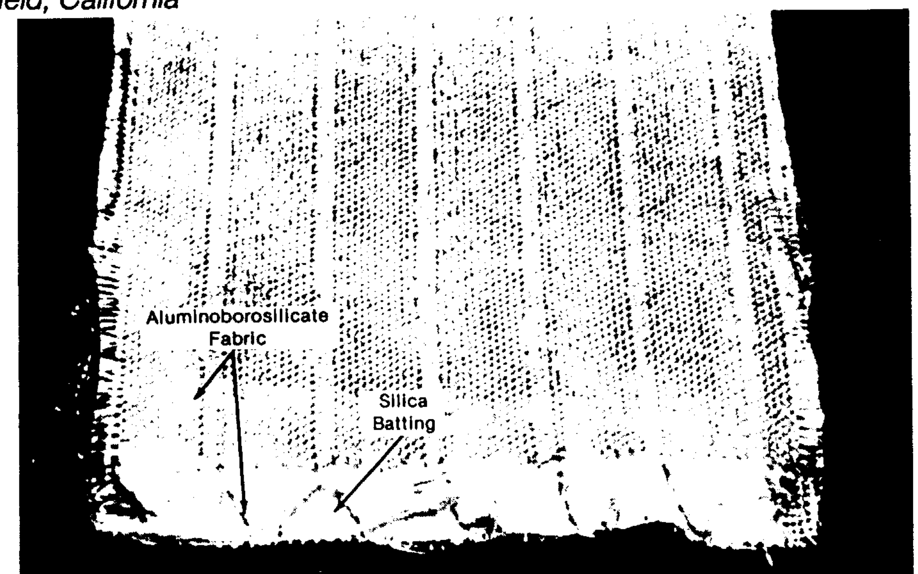
An integral woven core of ceramic yarns holds silica batting for thermal protection of space vehicles.

Ames Research Center, Moffett Field, California

Both single layer and multilayer insulating blankets for high-temperature service can now be fabricated without sewing. The new blankets, called TABI, can be used at higher temperatures because the heat resistance of the sewing thread is no longer a limiting factor. Moreover, the blankets are no longer subject to damage and weakening by the sewing process. The blanket is smoother and its thickness more uniform, because it is no longer quilted by stitches.

The new TABI blanket consists of an integrally woven, filled core between two fabric faces. The faces and core can be woven from any of three ceramic yarns: silica, aluminoborosilicate, or silicon carbide. The core may be woven as hollow rectangular or triangular flutes and filled with silica-fiber batting (see figure). Alternate batting materials such as aluminoborosilicate or alumina could also be utilized. A double-layer core has also been woven, one layer filled with the silica batting and the other with rigid ceramic tile.

The woven-core blankets can withstand substantially greater heat flux than can the older sewn silica blankets. For example, the woven aluminoborosilicate blanket can withstand as much as 7.5 Btu/ft^{2}s (85 kW/m^2), and the woven silicon carbide blanket can handle up to 30 Btu/ft^{2}s (340 kW/m^2). In contrast, the quilted silica-fiber fabric is limited to 4.4 Btu/ft^{2}s (50 kW/m^2).



The Tailorable Advanced Blanket Insulation (TABI) woven fabric is made of aluminoborosilicate. The triangular-cross-section flutes of the core are filled with silica batting. The flexible blanket can be formed into curved shapes, providing high-temperature and high-heat-flux insulation.

This work was done by Paul M. Sawko and Howard E. Goldstein of **Ames Research Center**. *For further information, Circle 10 on the TSP Request Card.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Ames Research Center [see page 14]. *Refer to ARC-11697.*

Reflecting Layers Reduce Weight of Insulation

The density of insulation blankets can be halved if metalized films are placed between layers.

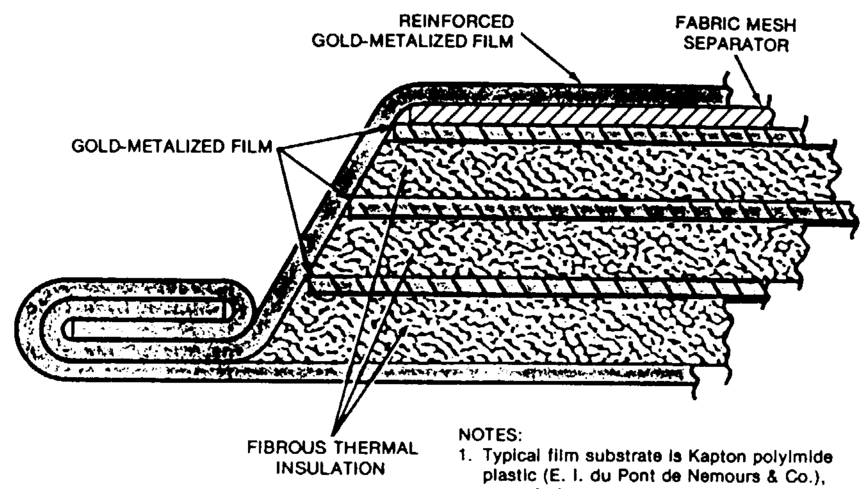
NASA Tech Briefs, Winter 1980

Lyndon B. Johnson Space Center, Houston, Texas

The weight of insulating blankets is reduced 40 percent by interleaving gold-metalized films of plastic between the layers of fibrous material. Use of the lighter weight material with the "goldized" film in the Space Shuttle orbiter has saved approximately 710 pounds (320 kg). It should also be useful in commercial aircraft to reduce weight and increase cargo space.

Previously, thermal conductivity was reduced in fibrous-batting insulation blankets by increasing their density (and thus increasing their weight). However, the new insulation design (see figure) reduces density and weight while maintaining equivalent thermal conductivity. Development tests indicate that insulation with a density of 1 lb/ft^3 (17 kg/m^3), plus reflectors of goldized film spaced between the layers of insulation, has a thermal conductivity comparable to that of 2 lb/ft^3 (34 kg/m^3) insulation without the films.

This work was done by James D. Cole, Edward D. Schlessinger, and Harley J. Rockoff of Rockwell International Corp. for **Johnson Space Center.** *No further documentation is available.*
MSC-18785



NOTES:
1. Typical film substrate is Kapton polyimide plastic (E. I. du Pont de Nemours & Co.), or equivalent.
2. Typical film substrate thickness is 0.0003 in.
3. Film is gold-metalized on both sides.

Lightweight Insulation batting is formed by interleaving gold-metalized films between layers of fibrous material. Because only half the density of fibrous material is required for a given thermal protection when the film is used, this construction saves 40 percent in weight.

Coatings for Mullite Insulation

A family of ceramic coatings waterproofs mullite insulation and protects it from physical damage.

Langley Research Center, Hampton, Virginia

Panels of fibrous mullite insulation are porous and friable. These properties make the mullite insulation vulnerable to moisture absorption, dust penetration, and damage from careless handling; anyone of which could seriously impair its thermal efficiency. A series of coatings has been developed which provides the fibrous mullite panels with a hard, impermeable, waterproof layer. In addition, the inclusion of selected color oxides in the coating composition imparts high emittance to the surface.

The coating is water repellent and thermally compatible with the mullite. In selecting the coating, various kinds of glass frit, refractory fillers, and colorants were screened. Refractory fillers investigated included TiO_2, $BaO \cdot ZrO_2$, $SrO \cdot TiO_2$, zircon, spodumene, petalite, and kryptonite. Colorants included Cr_2O_3, NiO, and CoO.

This preliminary screening produced combinations of kyanite, petalite, NiO, Cr_2O_3, and various kinds of glass frit. Selected dry mixtures of these constitutents were ball milled. A small amount of each mixture was then slurried with a 3-percent aqueous solution of polyvinyl alcohol and applied to mullite panels for evaluation. The coated mullite was dried at 230° F (110° C) then fired at 2,500° F (1,370° C) for at least ten minutes. Several very good coatings resulted from this work. Fibrous mullite panels, ranging in size from one-inch cubes to panels 13 × 11 × 2 in. (33 × 28 × 5 cm) have been coated with these mixtures and have then been tested for thermal endurance and thermal shock resistance. The mixtures were found to be waterproof and thermally compatible with mullite even after extended exposure to at least 2,100° F (1,150° C).

This work was done by Phillip N. Bolinger and Harry W. Rauch, Sr., of General Electric Co. for **Langley Research Center.** *For further information, Circle 48 on the TSP Request Card.*
LAR-11150

NASA

Characterizing Glass Frits for Slurries

Solids content of ceramic-coating mixture can be optimized as the result of one simple measurement.

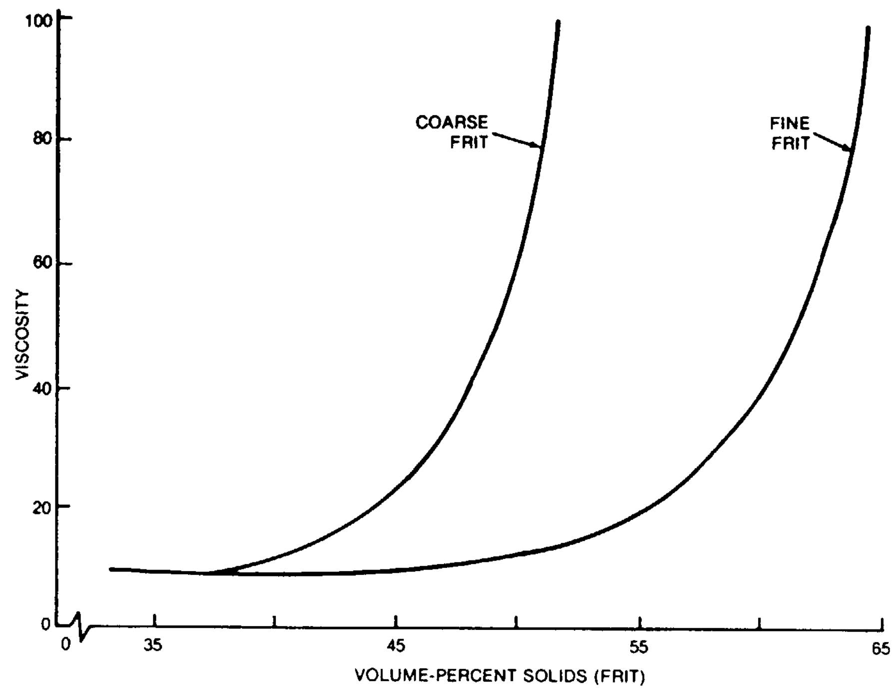
NASA Tech Briefs, Spring 1979

Lyndon B. Johnson Space Center, Houston, Texas

Glass-frit slurries can now be mixed with consistently reproducible properties, even from different batches of glass frit, using a new technique for characterizing the frit. When a slurry is used to make high-temperature ceramic coatings, glazes, and the like, the application rate, coating uniformity, and freedom from defects depend on having the proper viscosity. However, various batches of frit may have quite different particle-size distributions and therefore require different quantities of liquid for a given viscosity (see figure). In the absence of a simple characterization of the frit, trial-and-error mixing of each slurry is necessary.

The new test technique measures one quantity that determines the integrated properties of the frit for combination with a given liquid. This quantity, the "liquid absorption," is the amount of the given liquid that must be added to 5 grams of frit powder to make a flowable mix. The flow point is readily distinguishable and definitive.

The liquid absorption determines the particle packing factor or volume percent of particles present in the mix at the flow point. Curves can be prepared that show the viscosity of a slurry as a function of the percent solids for frits with various liquid absorptions; thus one measurement of a frit sample allows reliable preparation of a slurry with any desired viscosity.



The **Viscosity of a Slurry** depends on its glass-frit content and on the particle-size distribution of the frit. A given curve on this graph is characterized by the "liquid absorption" measurement of the frit.

This work was done by Harry N. Nakano of Lockheed Missiles & Space Co., Inc., for **Johnson Space Center.** *For further information, Circle 80 on the TSP Request Card.*
MSC-18322

Longer Shelf Life for Ceramic Slurries

A simple additive stabilizes the viscosity of water-based glass-frit slurries.

NASA Tech Briefs, Winter 1979

Lyndon B. Johnson Space Center, Houston, Texas

The viscosity of ceramic-coating slurries containing an organic acrylate viscosity-control agent can be stabilized by the addition of ammonium hydroxide. With the additive, slurries can be stored for over 2 months without significant change in viscosity, whereas without the additive they become unacceptably thin in only 1 day (see table).

The ammonium hydroxide treatment was developed for glass-frit water-base slurries that form high-temperature ceramic coatings for the Space Shuttle. The treatment could benefit other ceramic-coating slurries as well. Since treated slurries can be stored, frequent mixing of fresh batches is unnecessary; and, conceivably, less pure (and therefore less expensive) materials could be used in the slurry mix.

The glass-frit water-base slurry in the Shuttle formulation contains an organic acrylate as a viscosity-control agent. Iron contamination in the frit reacts with ammonia ions in the acrylate (as do ions of other metallic elements). When it is deprived of ammonia, the acrylate becomes insoluble in the water base, and the slurry viscosity decreases. Even an iron content of less than 0.02 percent of frit weight sharply lowers the slurry viscosity.

The addition of ammonium hydroxide to the mix gives the metal-ion contaminants an alternative source of ammonia. During the slurry preparation, 2.5 percent (by weight of the glass frit) of 29 percent concentrated ammonium hydroxide is added. The slurry is otherwise unchanged, consisting of the glass frit in water with 5.5 percent (by weight of glass frit) of acrylate. The ammonium hydroxide additive does not affect the firing of the ceramic coating.

This work was done by Y. D. Izu and T. M. Tanabe of Lockheed Missiles & Space Co., Inc., for **Johnson Space Center.** *No further documentation is available.*
MSC-18543

Coating Slurry	Viscosity, Ford Cup (Seconds)		
	As Prepared	After Storage	
		1 Week	2 Months
Treated	23.3	22.7	23.1
Untreated	17.0	12.0*	

*Unsatisfactory coating slurry application (occurs after 1 day storage)

The **Viscosity of an Untreated Slurry** quickly deteriorates, and the slurry becomes unusable. A slurry treated with ammonium hydroxide, in contrast, retains its viscosity for at least 2 months.

Repairing High-Temperature Glazed Tiles

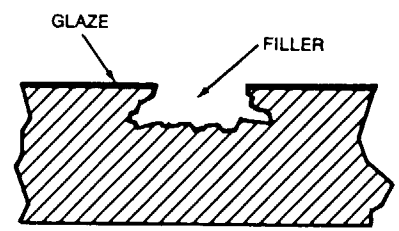
Chips and cracks in glaze are fixed quickly and dependably with a mixture containing tetraethyl orthosilicate.

Lyndon B. Johnson Space Center, Houston, Texas

Surface-damaged insulation tiles (see preceding articles) survive exposure to hostile sonic and temperature environments when they are repaired according to a new procedure. The new method consists of filling damaged areas on the glazed surface of the tile after proper preparation. The primary ingredient in the filler material is tetraethyl orthosilicate (TEOS).

The filler is prepared by mixing hydrolyzed TEOS, silicon tetraboride powder, and pulverized tile material. The silicon tetraboride gives the filler a black color; for white tiles it can be omitted.

To repair a tile, the damaged area is first cleaned. With the aid of dental tools, the edges of the damaged area are undercut (see figure), and any residue is removed. The area is coated with liquid TEOS and dried with heat lamps or a heat gun. After another coat of liquid TEOS is applied to the damaged area, the filler paste is applied to the crack with a spatula. The paste is carefully worked into the damaged area, including the undercut edges. Packed gently but firmly so as not to crack the surrounding coating, the pase is built up to the surface of the tile. The surface then is heated to dry and cure the filler. Finally, excess filler is sanded off the tile surface, and a final coat of liquid TEOS is brushed on.



Undercutting the Edge of a Surface Crack, as shown in this cross section, helps to anchor filler material securely. The filler is carefully built up to the level of the tile glaze. The procedure can be performed in less than 1-1/2 hours.

Tiles repaired in this manner survived tests in which they were subjected to intense acoustic emissions, arc jets, and intense heat radiation. The new method is not only more reliable but also is less time-consuming than previously used methods. It can be performed in less than 1-1/2 hours with the tile in any orientation — even facing downward; the tile does not have to be removed for repair.

This work was done by Glenn M. Ecord and Calvin Schomburg of **Johnson Space Center.** *For further information, Circle 71 on the TSP Request Card.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Johnson Space Center [see page A5]. Refer to MSC-18736.

Mobile Glazing Unit

The unit provides a programed thermal cycle for in situ repair of ceramic coatings.

John F. Kennedy Space Center, Florida

A mobile glazing unit programs a thermal cycle from 100° to 2,300° F (40° to 1,260° C) for firing ceramic glaze coatings on refractory surfaces. The prototype can be moved and used in any attitude and position to apply high radiant heat to a surface.

In contrast to many conventional radiant heating units, which are stationary, this unit can be brought to the work. A manipulator boom places the heater next to the surface to be fired.

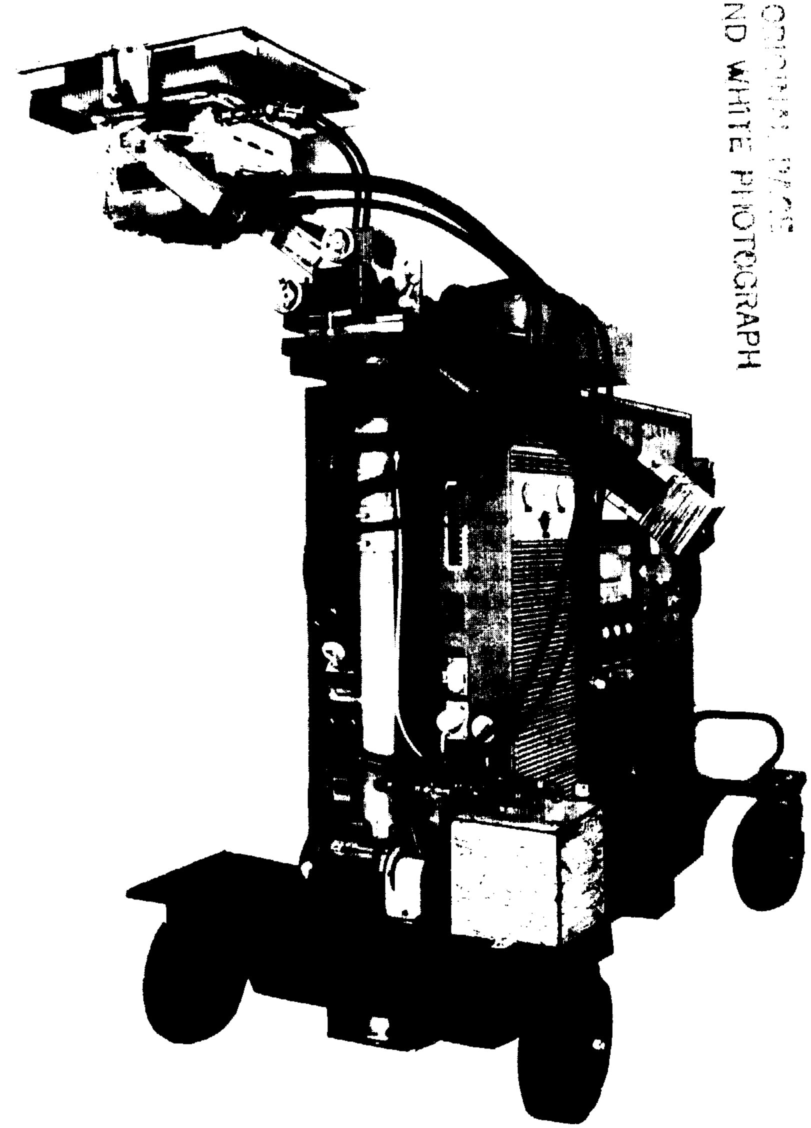
The mobile glazing unit (see figure) consists of a control console, heater assembly, protective cover, and manipulator boom. The manipulator boom supports the heater and protective cover. The heater assembly, shown facing upward at the top of the manipulator boom arm, has 18 quartz radiant-heat lamps with a water-cooled reflector. A blower air-cools the lamp end terminals. The protective cover between the heater assembly and the surface being heated prevents impact damage and protects the surrounding area from the radiant heat.

The equipment racks mounted directly on the dolly contain the control console, which includes power and temperature controllers, a recorder, a programer, and the water-cooling system. Fail-safe interlocks prevent damage due to equipment malfunctions. The unit requires a 480-V power source.

The unit was built for firing a ceramic repair coating on damaged thermal-insulation tiles on the Space Shuttle. It was built from off-the-shelf components. The unit should be useful industrially for in situ repair applications of glazing on ceramics or for curing individual refractory blocks during the maintenance of furnaces.

This work was done by Jack W. Holt of Rockwell International Corp. for **Kennedy Space Center.** *Further information may be found in NASA N81-70850/NSP, "Model 1023/8. Mobile Tile Glazing High Density Radiant Heating System—Preliminary Manual" and "Instruction Manual" [$20]. A copy may be purchased [prepayment required] from the National Technical Information Service, Springfield, Virginia 22161.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Kennedy Space Center [see page A5]. Refer to KSC-11171.



The **Mobile Glazing Unit** consists of a control console, heater assembly, protective cover, and manipulator boom. The heater assembly has 18 quartz radiant-heat lamps and a water-cooled reflector that provide thermal cycling in the temperature range of 100° to 2,300° F.

Waterproofing Agents for Silica Tiles

A new silylating agent is easy to apply
and withstands high temperatures.

NASA Tech Briefs, Spring 1985

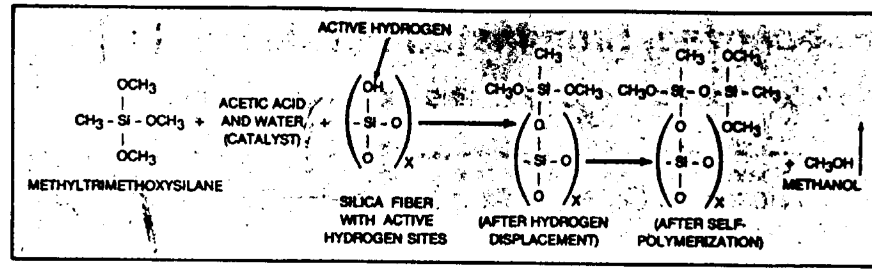
Lyndon B. Johnson Space Center, Houston, Texas

The waterproofing agent methyltrimethoxysilane is applied to silica thermal-insulation tiles in a simple vapor-deposition process. Other waterproofing agents in the same series include methylsiloxane (sprayed with a solvent) and hexamethyldisilazane (deposited from vapor). Of the three materials, methyltrimethoxysilane imparts the best waterproofing and thermal properties. Originally developed for insulating tiles for spacecraft, one or more of these agents may also find uses in roofing tiles, insulation for buildings or solar-energy systems, or solar reflectors.

The vapor deposition can be done under a variety of conditions, including open air at room temperature, bagged or otherwise enclosed at room temperature and pressure, assisted by controlled or directed gas or airflow, the injection of vapor directly into the tile, and other variations of flow or pressure that facilitate the permeation of the tile. The curing reactions take place at room temperature, but can be speeded by heating. Radiant heating can be used, or the flowing gas can be heated during or after the vapor deposition.

The wettability of the tiles is caused by active hydrogen sites that remain on the silica fibers from the tile-manufacturing process. The methyltrimethoxysilane or other agent reacts with these sites (see figure). The required hydrophobic property is imparted to the silica with only a small increase in weight — typically 0.8 percent.

Tiles treated with methyltrimethoxysilane retain their water repellency after exposure to 427° C in air for 15 hours, whereas tiles treated with the other agents do not. The hydrophobic coatings of all three agents decompose at 649° C, but the decomposition products of the methyltrimethoxysilane and hexamethyldisilazane coatings do not cause darkening or otherwise affect the solar absorption or emissivity of the tiles.



Methyltrimethoxysilane Reacts With Active Hydrogen Sites on silica fibers. The hydrogen is removed and replaced by hydrophobic molecular fragments.

This work was done by Harry N. Nakano, Y. Douglas Izu, and Ernest N. Yoshioka of Lockheed Missiles & Space Co., Inc., for **Johnson Space Center.** *For further information, Circle 83 on the TSP Request Card.*

Inquires concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Johnson Space Center [see page 29]. Refer to MSC-20364.

Curing of Furfuryl Alcohol-Impregnated Parts

A longer cure and improved quality control prevent delaminations.

NASA Tech Briefs, Winter 1982

Lyndon B. Johnson Space Center, Houston, Texas

A delamination problem in reinforced carbon/carbon parts impregnated with oxalic acid-catalyzed furfuryl alcohol is overcome by instituting two additional quality-control tests on the alcohol and by changing the curing conditions. The delamination had occurred following autoclave curing of the catalyzed furfuryl alcohol.

It had been assumed that monitoring alcohol viscosity would suffice as a quality-control check, but from time to time, the impregnating solution developed a froth that interfered with the impregnating process, even though viscosity was proper.

Differential scanning calorimetric (DSC) tests show a rapid, sharp exotherm at 293° F (145° C) as well as a broad exotherm extending beyond 400° F (204° C). It was concluded that the previous 1-hour cure at 290° F (143° C) was insufficient and that postcuring to 400° F at ambient pressure could also release moisture (as a byproduct of the curing reacton), causing delamination in thick regions of the parts being fabricated.

As a result of the investigation, the added alcohol quality-control tests are as follows:

- Alcohol in production use is tested weekly for water content. Water content must remain below 9.5 percent.
- DSC tests used to check the exothermic reaction characteristics of the impregnating solution. The exotherm must be strong.

The revised curing cycle calls for at least 2 hours at 300° ± 10° F (149° ± 6° C) followed by cooling to 175° F (80° C) under autoclave pressure. The extended cure drives the slower reaction nearer to completion. Since instituting these changes, no part delaminations have occurred.

This work was done by James W. Lawton and Thomas H. Brayden of Vought Corp. for **Johnson Space Center.** *No further documentation is available.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Johnson Space Center [see page A5]. Refer to MSC-20224.

Detecting Pores in SiC Coatings

A nondestructive testing method produces images of inadequately covered areas.

Lyndon B. Johnson Space Center, Houston, Texas

A liquid-penetrant/fluorescence technique reveals cracks and pinholes in protective coatings. The technique was developed for checking the quality of overcoatings on silicon carbide layers on advanced carbon/carbon substrates. The difference in thermal expansion rates between the substrate and the silicon carbide tends to crack the silicon carbide layer. The silicon carbide coating prevents the exposed substrate from oxidizing, but the effectiveness of the coating is greatly reduced if it is cracked or porous.

The new technique is similar to other liquid-penetrant/fluorescence techniques used to make pores visible. If porosity is excessive, the material can be recoated or perhaps subjected to a healing process that would close the pores.

The penetrant is a suspension of organic fluorescent particles, 3 to 50 μm in diameter, in isopropyl alcohol. The part can be dipped in the suspension, or the suspension can be brushed or poured on the part. After about 5 min, the part is viewed in ultraviolet light. Porous areas absorb more of the suspension and therefore accumulate more fluorescent particles. They therefore fluoresce more brightly than their surroundings.

The absorption does not adversely affect the protective characteristics of the coating or the ability of the part to accept a recoat. The technique could be automated by using spray nozzles to apply the suspension and a scanning laser and photodetectors to find the pores.

This work was done by Anthony B. Hamilton, Kenneth L. Tummons, and James W. Lawton of LTV Aerospace and Defense for **Johnson Space Center.** *No further documentation is available.*
MSC-21041

NASA Tech Briefs, September/October 1986

Ensuring the Consistency of Silicide Coatings

The optimum fusion time is determined from a simple diagram.

NASA Tech Briefs, Spring/Summer 1982

Lyndon B. Johnson Space Center, Houston, Texas

A diagram specifies the optimum fusion time for given thicknesses of refractory metal-silicide coatings on columbium C-103 substrates. Adherence to the indicated fusion times ensures consistent coatings and avoids underdiffusion and overdiffusion.

The diagram was developed as a result of X-ray tests on chamber-nozzle assemblies for the Space Shuttle reaction-control-system thrustors. For oxidation protection, the columbium parts of the assemblies are coated with silicide. The coating is applied as a slurry of silicon, chromium, and titanium powders then vacuum-fused at 2,580° F (1,415° C). X-ray diffraction inspection revealed structural anomalies in the coatings on some of the parts. The anomalies were traced to a metal-trisilicide hexagonal phase on the coating surface. This phase rapidly oxidizes, then spalls when it is heated.

Excessive time at the fusion temperature causes a loss of silicon by vaporization and promotes formation of the hexagonal phase after the normal metal-disilicide phase is formed. Figure 1 shows the development of the hexagonal phase as the time increases, for one coating thickness.

The phase diagram that relates coating thickness to fusion time is seen in Figure 2. It shows the allowable combinations of coating thickness and fusion time that ensure that only the normal phase is present. The accuracy of the diagram has been confirmed by tests.

This work was done by Ven Ramani and Francis K. Lampson of The Marquardt Co. for **Johnson Space Center.** *No further documentation is available.*
MSC-18900

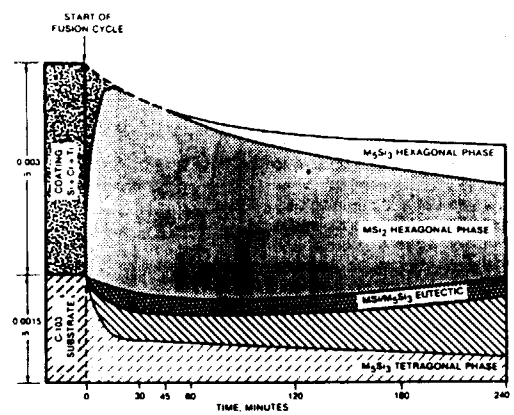


Figure 1. The **Development of the M_5Si_3 Hexagonal Phase** on silicide coatings is seen in this graph. For this coating thickness [0.003 in. (0.076 mm)], the optimum cycle time is between about 20 and 45 minutes.

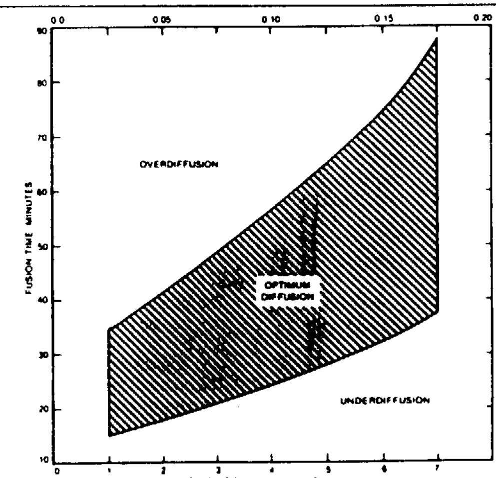


Figure 2. The Time-vs.-Thickness Diagram for refractory metal-silicide coatings shows an area of acceptable combinations of fusion time and coating thickness. Within the area, the normal metal-disilicide phase results. Outside the area, overdiffusion, with production of the trisilicide hexagonal phase, or underdiffusion results.

User Chooses Coating Properties

Anodizing technique allows the independent selection of coating thermal emittance and solar absorption.

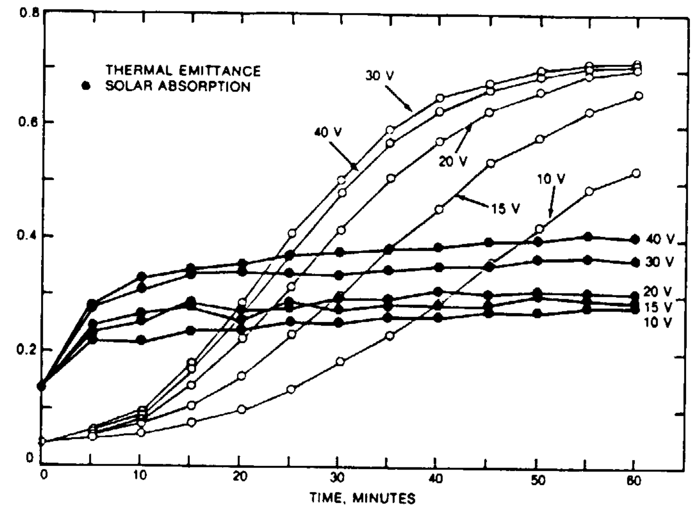
Langley Research Center, Hampton, Virginia

A versatile thermal-control coating is prepared by modified chromic acid anodizing of aluminum. The coating, with ranges of 0.10 to 0.72 for thermal emittance (ϵ_T) and 0.2 to 0.4 for solar absorptance (α_S), allows the selection of any value of ϵ_T or α_S in the specified range to within an accuracy of ± 0.02.

Other thermal-control coatings include conversion coating (Alodine), other anodic coatings, and dielectric films and paints. These, however, have limited ranges of ϵ_T and α_S and do not allow the independent selection of these properties.

The variable anodic thermal-control coating process has three phases: initial material processing, anodizing, and material postprocessing. The initial processing prepares the material for anodizing and establishes the initial values of ϵ_T and α_S. The aluminum is immersed in a metal-cleaning bath, rinsed, immersed in a deoxidizer solution, and rinsed by physical agitation to remove all particulates from the aluminum surface.

The aluminum is anodized by immersing it in a chromic acid solution containing 3 to 10 percent CrO_3 (by weight) balanced with water. The voltage is applied between the aluminum and the chromic acid solution at a predetermined rate up to a selected voltage and is maintained for a selected period of time. The rate, voltage, and time, along with the initial values of ϵ_T and α_S, temperature of the chromic acid solution, acid concentration, and material to be anodized, determine the final values of ϵ_T and α_S. Material postprocessing involves rinsing with water, placing in a sealing bath of clear water at 180° F (82° C), and drying with filtered forced air.



Thermal Emittance and Solar Absorptance of the anodic thermal-control coating are shown for various anodizing times and voltages. The cell voltage is increased to the operating voltage in 30 seconds in these examples. Solution temperature is 95° F (35° C), and the pH is 0.5.

Typical results are shown in the figure, where ϵ_T and α_S are plotted for various anodizing voltages and times. The anodic process has been applied on 6061, 1145, and 2024 aluminum with thicknesses as low as 0.001 inch (0.003 cm). Preliminary stability tests in vacuum have shown less than 15 percent degradation over a 2,000-hour solar exposure. The technique is sensitive to voltage, the rate of voltage application, time, temperature, acid concentration, material pretreatment, and sealing. However, consistent results are obtained if the processing parameters remain constant.

This work was done by Charles S. Gilliland and Roy J. Duckett of **Langley Research Center.** *For further information, Circle 49 on the TSP Request Card.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Langley Research Center [see page A5]. Refer to LAR-12719.

Improved Silicone-Rubber-to-Silicone-Rubber Bonding

Strongest bonds result when the precured rubber is lightly abraded and left unprimed.

NASA Tech Briefs, Winter 1977

Lyndon B. Johnson Space Center, Houston, Texas

The standard procedure for bonding room-temperature-vulcanizing silicon rubber to itself is to apply a freshly-mixed silicone rubber onto a precured rubber, with a layer of silicone primer applied first. Although this has appeared to give a satisfactory bond, a new investigation indicates that the bond strength can be increased if the primer is eliminated and the surface is lightly abraded.

In a test of this procedure, an aluminum flatwise tensile block was primed and silicone rubber was applied and allowed to cure at room temperature for a minimum of 72 hours. Silicone primer was then applied to the cured silicone rubber and to the surface of a matching aluminum flatwise tensile block. Finally, freshly-prepared silicone rubber was applied to both surfaces of matching blocks. Test specimens without the primer between the rubber-to-rubber interface were fabricated by applying silicone rubber directly to the precured surface.

As the Test I data in the table indicate, the unprimed specimens had considerably higher bond strengths than the primed specimens. The unprimed samples exhibited approximately 30 to 40 percent adhesive failure at the bond interface as evidenced by glassy unbonded areas.

Test I - Silicone-Rubber Surface Not Abraded			
Cured, Not Primed		Cured and Primed	
Specimen	Load (PSI)	Specimen	Load (PSI)
1	325	1	137
2	355	2	196
3	336	3	176
4	136	4	181
	339 (Average)		173 (Average)
30% to 40% Unbonded Area in Rubber-to-Rubber Joint		All Failures in Primed Surface	

Test II - Silicone-Rubber Surface Abraded			
Cured, Not Primed		Cured and Primed	
Specimen	Load (PSI)	Specimen	Load (PSI)
1	375	1	195
2	379	2	190
3	363	3	220
4	348	4	207
	366 (Average)		203 (Average)
90% to 100% Cohesive Failure in Rubber-to-Rubber Joint 0% to 10% Adhesive Failure in Rubber-to-Primed Aluminum Interface		All Failures in Primed Surfac·	

Flatwise Tensile Data indicate that stronger silicone-rubber-to-silicone-rubber bonds result if the cured rubber is left unprimed (Tests I and II). Bond strength is further enhanced if the surface is lightly abraded (Test II).

A second series of tests, identical to the first except that the precured surfaces were abraded lightly to remove the gloss, produced similar results, as shown in the Test II data. Furthermore, bond strengths for the abraded specimens were consistently higher than for the unabraded specimens, as evidenced by a reduction in the adhesive failure rate to between zero and 10 percent.

The tests indicate that the silicone primer between the rubber-to-rubber bond hinders the bond joint, resulting in a substantially lower bond strength.

This work was done by Kuniyoshi Teramura of Rockwell International Corp. for **Johnson Space Center.** *No further documentation is available.*
MSC-16419

Room-Temperature Adhesive for High-Temperature Use

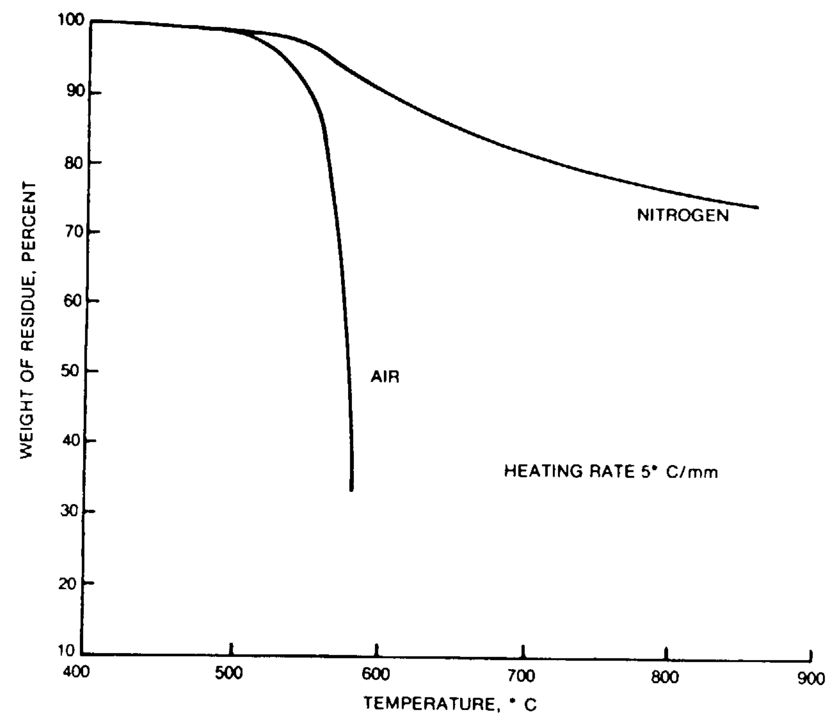
Room-temperature-curing adhesive retains strong bond in extreme temperature environments.

Lyndon B. Johnson Space Center, Houston, Texas

Adhesives used in bonding reusable surface-insulation tiles to metallic substrates must withstand extreme temperature environments that may be encountered by the tile surface. Typical adhesives used in this application are the thermosetting type, which are cured at temperatures at least as high as those expected in service, and adhesives cured by chemical reaction in controlled humidity. One chemically curing adhesive requires careful weighing and thorough mixing of catalyst for proper cure. Its service temperature range is between -170° F (-113° C) to 500° F (262° C).

A significant improvement is obtained with polyphenylquinoxaline (PPQ). PPQ cures at room temperatures and withstands temperature extremes between -300° F (-186° C) and 750° F (402° C). Laboratory studies have shown that PPQ used with chloroform as solvent produces the best bond capable of withstanding both mechanical and thermal stresses (see figure).

Basically the PPQ-chloroform solution is sprayed or brushed onto a precleaned metal substrate. The other adherent, typically a porous material to permit quick solvent evaporation, is brought into contact with the PPQ film. The solvent is then allowed to evaporate at room temperature, leaving a strong bond. Evaporation of the solvent may be speeded up by the application of vacuum and/or elevated temperature.



Excellent Thermal Stability of PPQ is demonstrated by thermogravimetric analysis as the polymer is heated. Decomposition in air or nitrogen does not begin until the temperature exceeds about 440° C.

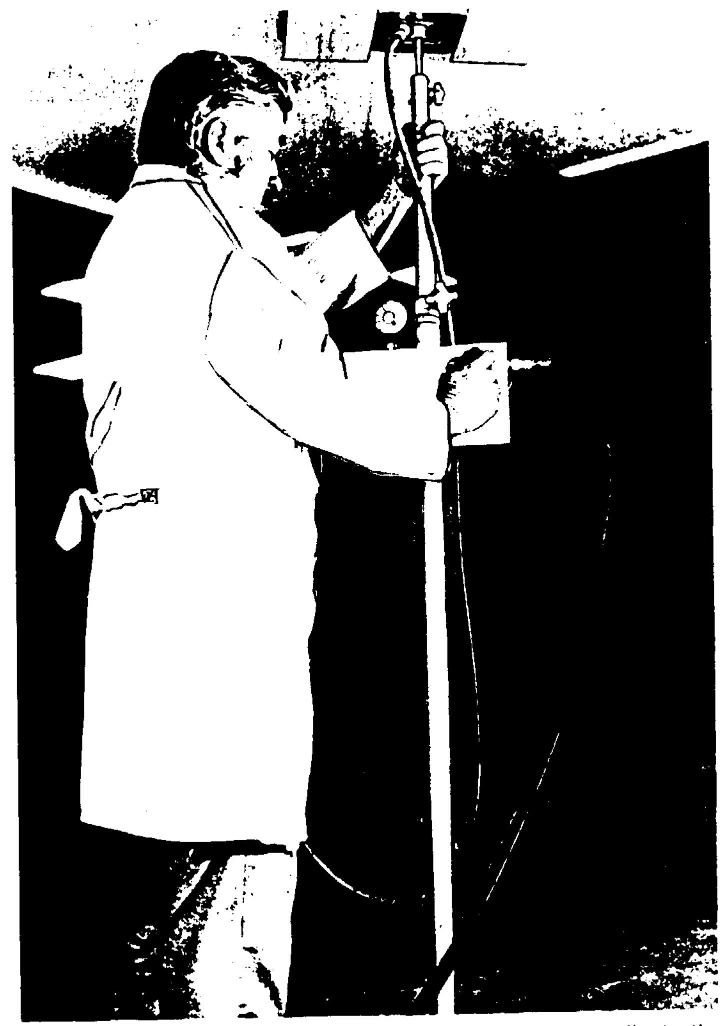
This work was done by Jon L. Brooks, William L. Hill, and Charles R. Rousseau of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 75 on the TSP Request Card.*
MSC-16930

Tile-Bonding Tool

Device applies uniform, constant, precise pressure to hold tiles in place during bonding.

NASA Tech Briefs, Spring 1978

John F. Kennedy Space Center, Florida



The **Tile-Bonding Tool** is being used by an operator who adjusts the pressure on a bladder mounted on an extended pole. The bladder applies a controlled uniform pressure over the surface of flat or contoured tiles. Only one bladder is shown here, but several can be used to bond many tiles simultaneously.

A simple tool holds tiles in place for bonding at precisely the right pressure for the tile material/adhesive combination. Designed for replacing the approximately 350 surface-insulation tiles that are expected to be damaged on a Space Shuttle during flight, the tool should be useful in other applications where holding or positioning fragile materials is necessary.

The tool consists essentially of pressure bladders supported by an adjustable pole. The pole, which has an antiskid foot, is placed on a floor or work surface and is extended until tiles on the uninflated bladders just touch the surface to which they are to be bonded. The user then opens a valve to a pressurized air supply to inflate the bladders to the desired pressure, which can be maintained within 1/2 psi (3.5×10^3 N/m^2). Each bladder applies a uniformly distributed force.

The pole can accommodate a single bladder to bond a single tile or multiple bladders to bond many tiles at once. Multiple bladders can be pressurized individually. Tiles can be flat or contoured. The pole can be used in any orientation as long as a solid reaction surface is available.

This work was done by Cyrus C. Haynie and Jack W. Holt of Rockwell International Corp. for **Kennedy Space Center.** *For further information, Circle 73 on the TSP Request Card.*
KSC-11053

Bonding Heat-Resistant Fabric to Tile

Acid etching, densification, and silica cement ensure a strong bond.

Lyndon B. Johnson Space Center, Houston, Texas

Procedures have been developed for bonding quartz fabric to silica tiles, both glazed and unglazed. The procedures use high-temperature materials exclusively and are therefore suitable for securing flexible seals and heat barriers around doors and viewing ports in furnaces and kilns.

The bond withstands temperatures in excess of 1,200° F (650° C). It provides a peel strength of 3 pounds per linear inch (500 N/m) on glazed tile and 7.4 pounds per linear inch (1.3 kN/m) on unglazed tile. Previously, the tile and fabric were joined with a room-temperature-vulcanizing adhesive that was limited to service temperatures below 600° F (316° C).

The key step in the preparation for bonding to glazed tile is etching the quartz fabric and the tile with acid: This increases the adhesion of the silica cement used to form the bond.

Major steps in preparing the quartz fabric for bonding are as follows:

- Application of a liberal coat of isopropyl alcohol to the contact surface,
- Application of three coats of 0.10 normal hydrochloric acid,
- Rinsing, and
- Coating with a wetting agent.

Glazed tile is prepared by the following steps:

- Masking with tape so that only the bond contact area is exposed,
- Application of a thick coating of paste imbued with hydrofluoric acid for 10 minutes,
- Rinsing, and
- Removal of the masking tape.

Bonding of the quartz fabric to the tile must be done within 4 hours after the tile glaze has been etched. Silica cement is applied to the prepared areas of the fabric and the tile. The cement should form a fillet along the edges of the fabric contact area. The fabric is applied to the tile and held in place with tape. The cement is allowed to dry in air for 48 hours, after which the bond is complete.

The key step in bonding to unglazed tile is densification of the tile to create a stronger joint. The tile is masked with tape to expose only the contact area. The exposed area is densified by repeated brushings of colloidal silica slurry, then is allowed to dry. The area is waterproofed by brushing on three coats of a silane solution. The surface is then lightly sanded.

The quartz fabric is prepared in the same way as for bonding to glazed tile. The bonding of fabric to tile with silica cement is also performed in the same way as for glazed tile.

This work was done by Jack W. Holt and Laurence W. Smiser of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 142 on the TSP Request Card. MSC-20540.*

Ceramic Adhesive for High Temperatures

The fused-silica/magnesium-phosphate adhesive resists high temperatures and vibrations.

Lyndon B. Johnson Space Center, Houston, Texas

Developed to bond gap fillers to low-density ceramic insulating tiles used in the thermal-protection system of the Space Shuttle, a ceramic adhesive can be used in aerospace, metallurgical, ceramic, electronic, and other applications where conventional adhesives cannot meet requirements for service at high temperatures. The adhesive consists of a base material of fused silica and a bonding agent of magnesium phosphate.

An older procedure for filling unacceptably large gaps between the tiles involved room-temperature-vulcanizing silicone-rubber adhesives, which could not tolerate temperatures above 550 °F (288 °C) and therefore had to be replaced after each mission. The new adhesive, which has a coefficient of expansion compatible with that of the tiles, is unaffected by extreme temperatures and by vibrations. Assuring direct bonding of the gap fillers to the tile sidewalls, the new adhesive obviates the expensive and time-consuming task of the removal, treatment, and replacement of tiles.

The process for filling tile gaps is as follows:

- Water and colloidal silica are added to the ceramic adhesive powder, and the mixture is stirred to a homogenous paste.
- The surfaces of the shim and the tile to be joined are wetted thoroughly.
- The ceramic adhesive is applied to the surface of the shim.
- The shim is installed against the tile sidewall, which has been diamond-abraded, and pressure is applied to assure full contact.
- The adhesive is cured for 24 hours in air.

This work was done by Everett G. Stevens of Rockwell International Corp. for **Johnson Space Center**. *For further information, Circle 25 on the TSP Request Card.*
MSC-21085

Preassembly of Insulating Tiles

A mesh backing carries ceramic tiles to ease installation and prevent damage.

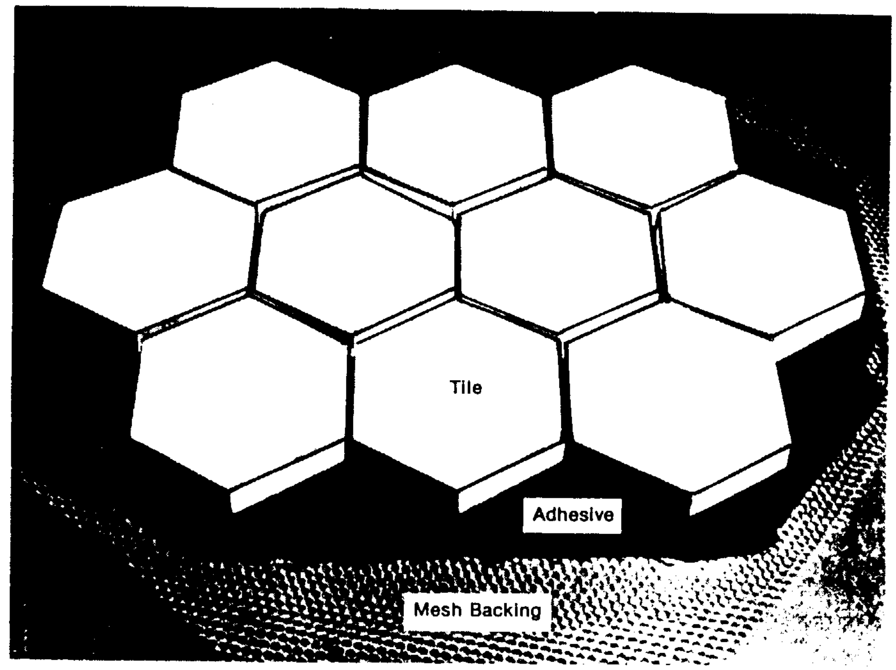
Lyndon B. Johnson Space Center, Houston, Texas

A concept for preassembling high-temperature insulating tiles speeds and simplifies installation and repair and reduces damage from handling. Moreover, the preassembly concept would facilitate the placement of tiles on gently contoured surfaces as well as on flat ones. The concept resembles that for preassembled kitchen and bathroom tiles. Applications might include boilers, kilns, and furnaces.

The tiles would be bonded with a silicone adhesive to a nylon mesh backing, chosen for light weight and high resistance to strain. The tiles would be hexagonal so that a sheet of them on the backing could nearly conform to a curved surface. For example, preassembled 2.5-in. (6.35-cm) hexagons of 0.5-in.- (1.27-cm)-thick ceramic would readily conform to a surface having a 3-ft (0.9-m) radius of curvature, and probably even to a surface of 1-ft (0.3-m) radius. Smaller radii may be accommodated by smaller hexagonal tiles.

The specific backing mesh would be chosen according to the requirements of specific applications. For the Space Shuttle service for which the concept was developed, the primary purpose of the backing was to maintain the assigned configuration of the preassembled tiles, with minimal influence on the supporting structure.

Once the tiles have been bonded to it, the backing carries them through such final processing steps as drying, sintering, coating by spraying or dipping, and computer-controlled machining to final thickness. Through all, preassembly on the backing will reduce the cost of processing as compared with that of the handling of individual tiles. Tile preassemblies can be emplaced in a "cut-and-paste" operation similar to that of commercially available tiles for the home.



Tiles Are Bonded to nylon mesh with a room-temperature-vulcanizing silicon rubber. The spacing between tiles is 0.03 in. (0.76 mm).

This work was done by Y. D. Izu, E. N. Yoshioka, and T. Rosario of Lockheed Missiles and Space Co. for **Johnson Space Center**. *For further information, Circle 67 on the TSP Request Card.*
MSC-21204

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Compensating for Shrinkage in Machined Ceramics

The machine tool automatically compensates for the shrinkage that occurs when the ceramic is baked.

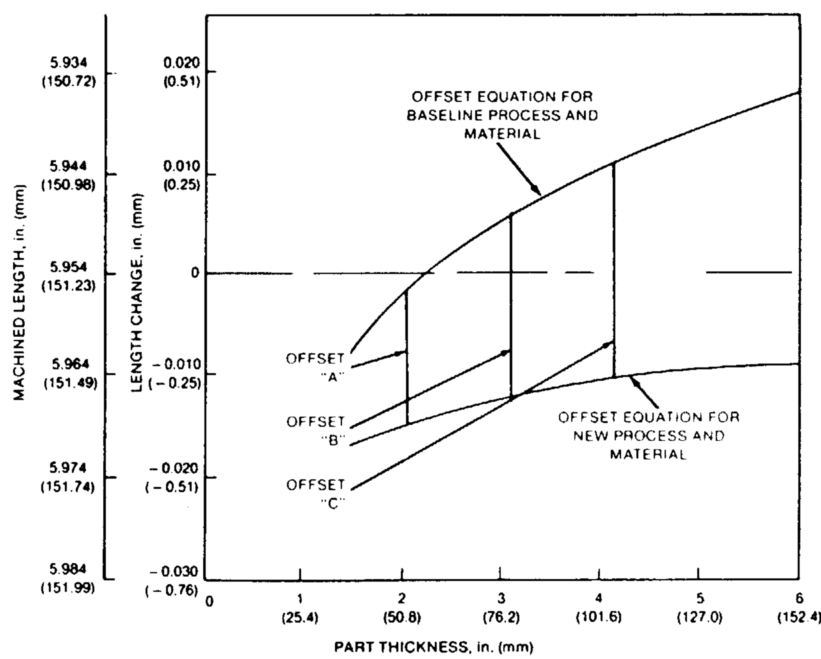
NASA Tech Briefs, Fall 1985

Lyndon B. Johnson Space Center, Houston, Texas

A technique originally developed for Space Shuttle surface-insulation tiles insures that machined ceramics shrink to the correct dimensions after they are baked in a kiln. The new method automatically compensates during machining for the shrinkage that will occur later, when the part is baked. It is applicable to numerically controlled machines that include a provision to adjust for variations in cutting-tool size, but do not provide for automatic verification of the dimensions of machined parts. Because of the lack of verification, the machine can be "deceived" into compensating for a fictitious tool size.

A typical numerical-control program includes a provision for entering a letter-code control signal to denote the tool size and an equation (see figure) that calculates the offsets that compensate for part shrinkage. This offset equation applies to only one baseline material and treatment schedule and is derived from prior measurements on parts of the same composition and process.

For parts of other compositions and processes, shrinkages must be measured anew. But rather than rewriting the program by revising the offset equation for each new material or process, the difference between the baseline and the new case is calculated for each part and assigned a letter code corresponding to an equivalent tool size.



Letter Codes denote the machine-tool offset to be used for a given blank thickness. The offset depends on the composition of the part and on the fabrication process. The dimensions shown here are for the example only.

Each part to be machined is usually accompanied by a computer card punched with such relevant information as the source, date of fabrication, method of fabrication, and material. The card also includes the applicable letter code. When the part is brought to the machine, the card is fed into the machine, which reads the letter code and makes the offset adjustment.

This work was done by L. Aguilar and B. T. Fitchett of Lockheed Missiles & Space Co., Inc., for **Johnson Space Center.** *No further documentation is available.*

MSC-20684

Graphical Method for Predicting Steady-State Temperature

The temperature that a heated or cooled passive system will reach is predicted from temperature-versus-time curves.

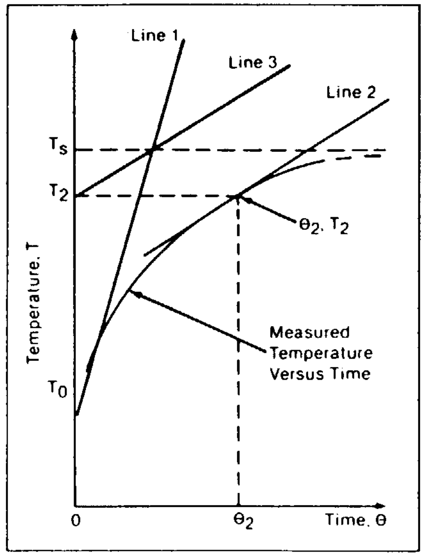
NASA Tech Briefs, January/February 1986

Lyndon B. Johnson Space Center, Houston, Texas

A simple graphical construction method, originally developed for Space Shuttle applications, predicts the equilibrium temperature of a system that is being heated or cooled. The method assumes that the temperature relaxes exponentially toward an equilibrium value. The results can be obtained in minutes without a computer or complex calculations.

A plot of the early temperature-versus-time history of the system must first be determined by measurement for a given set of constant thermal conditions. From this plot, the ultimate steady-state temperature that the system will attain can be constructed. The construction method for a system being heated is as follows:

- A tangent is drawn to the temperature-versus-time curve at the point at which heating or cooling starts (line 1 in the figure).
- A second tangent is drawn to the curve at any point θ_2, T_2 (line 2 in the figure). Line 2 must have a slope different from that of line 1.
- Line 3 is drawn parallel to line 2 through temperature T_2 on the vertical axis. The temperature T_s at the intersection of lines 1 and 2 is the steady-state temperature.



The **Intersection of Two Lines** in a graphical construction gives the asymptotic temperature of a system. Developed for analyzing thermal anomalies during flights of the Space Shuttle, the graphical method is also applicable to everyday heating and cooling problems.

The procedure for a system being cooled is identical except that the slopes are negative.

The time θ_s for the system to progress through 99 percent of the temperature difference between the initial and steady-state values is computed from the equation

$$\theta_s = [(T_s - T_0)/(\text{slope of line 1})] \times \ln(100)$$

where T_0 is the temperature of the system at time zero. The 99-percent point is used as a convenient approximation, since theoretically it would take an infinitely long time to reach the final temperature.

The method is applicable to a variety of problems. Examples include the temperatures of buildings, powerplant or refinery components, or chemical reactors, voltages on charging or discharging capacitors; and concentrations of reactants in some chemical processes.

This work was done by Robert L. Case, Jr., of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 36 on the TSP Request Card. MSC-20835*

Furnace for Tensile Testing of Flexible Ceramics

Ceramic cloth and thread are tested quickly at temperatures up to 1,250 °C.

NASA Tech Briefs, November/December 1986

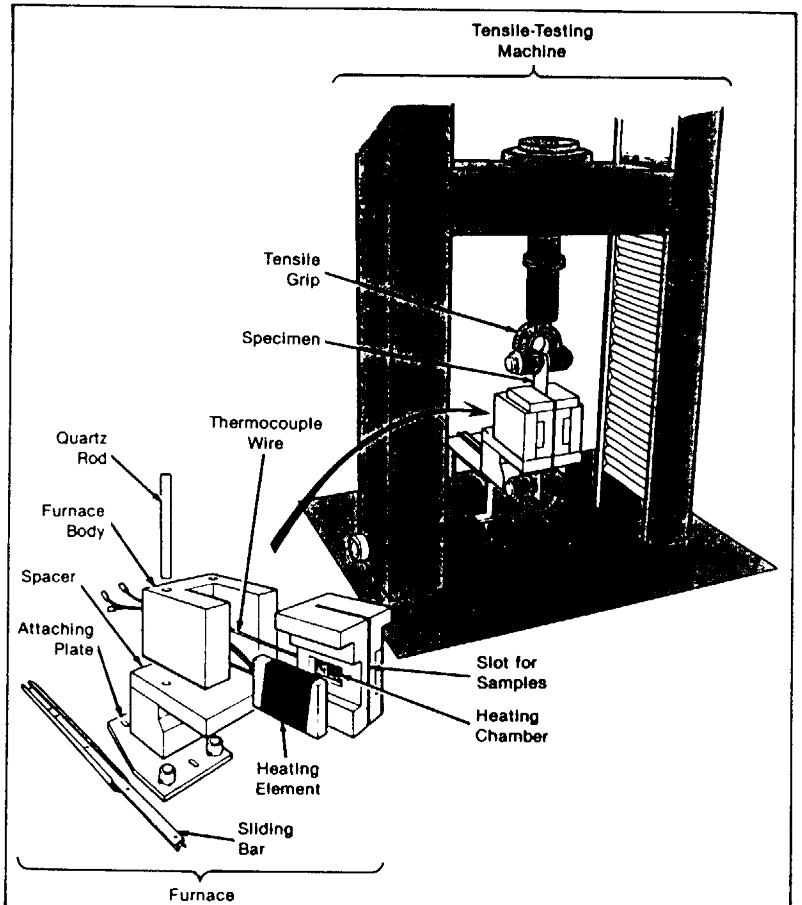
Ames Research Center, Moffett Field, California

Tensile strengths of ceramic cloths and threads can be measured conveniently in a new furnace at specified temperatures up to 1,250 °C, using an ordinary mechanical tester. The samples are heated along part of their lengths in the furnace slots. The interchangeable furnace chambers and the matching heating elements are sized to match the size of the tested ceramic material.

The furnace is made of foamed silica with temperature-controlled embedded resistant-wire heating elements powered by a 110-V, 25-A supply. Designed to heat strips about 1 in. (2.5 cm) wide or threads that are less than 0.06 in. (0.15 cm) thick, the furnace is 4 in. (10.2 cm) wide, 5 in. (12.7 cm) deep, and 3 in. (7.6 cm) high. The furnace test chambers are one-half in. long, one-half in. wide, and 1 in. high (1.27 by 1.27 by 2.54 cm) for thread, and 1-1/2 in. long, one-half in. wide, and 1 in. high (3.81 by 1.27 by 2.54 cm) for cloth.

There are two type "K" thermocouples — one for the temperature control, the other for temperature monitoring — mounted one-eighth in. (3 mm) from the tested material. The small furnace opening — a 1/16-in. (1.6-mm) slot — allows the furnace to be slid into place around, or removed from, the specimen, with insignificant loss of heat at the test temperature.

The tensile-test specimen is mounted in the mechanical tester with a 4-in. (10.2-cm) opening between the tensile grips (see figure). Mounted on a horizontal sliding bar, the preheated furnace is moved forward to enclose the tensile specimen, the strength of which is then measured in the standard way. After the specimen breaks, the furnace is moved away from the test specimen, a new specimen is mounted in the tensile grips, and the furnace is slid into place for the next test. Because the furnace remains hot and does not enclose the mechanical tester, no time is lost waiting for the furnace to reheat.



The **Modular Design of the Furnace** allows the tensile-strength testing of both ceramic thread and cloth.

This work was done by Marnell Smith, Carlos A. Estrella, and Victor W. Katvala of **Ames Research Center**. *No further documentation is available.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Ames Research Center *[see page 27]*. *Refer to ARC-11589.*

Attaching Strain Transducers to Fragile Materials

An A-shaped clamp prevents
damage to thin, brittle specimens.

NASA Tech Briefs, Spring 1979

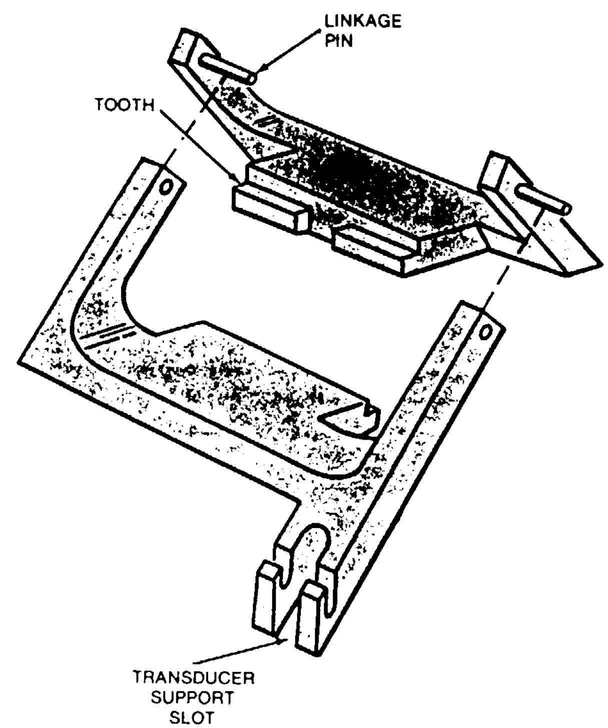
Lyndon B. Johnson Space Center, Houston, Texas

A simple clamp gently holds brittle or fragile strips during high-temperature stress/strain testing. The clamp also supports a displacement transducer away from the heated zone, and it defines reference points for the strain measurement on the specimen surface. Unlike rigidly attached supports, the new clamp prevents the specimen from cracking due to unequal thermal expansion between it and the holder. Moreover, the clamping force is constant over the wide range of temperatures encountered in the test.

The clamp is assembled from two components (see figure). The components form an A-bar that pivots about the linkage pins at its apex. The teeth on the two members rest against opposite sides of the specimen.

The clamping force is increased by the weight of the transducer, which hangs from slots on opposite sides of the A-bar. The geometry and dimensions of the clamp determine the force on the specimen. For stress/strain measurements, it is desirable to clamp the specimen firmly but with minimum stress. Thus the device illustrated has a rather large moment arm from the teeth to the pivot pin.

This work was done by Michael F. Duggan of Lockheed Missiles & Space Co., Inc. for **Johnson Space Center.** *No further documentation is available.*
MSC-16580



This **Clamp for Thin, Brittle Strips** consists of two parts that link together by pins. The teeth gently clamp the sides of a test specimen (not shown). A strain transducer can be supported from the slots in the clamp. The holder was originally developed to test ceramic glazes on samples of silica used for surface insulation on the Space Shuttle.

Comparative Thermal-Conductivity Test Technique

Approximate thermal conductivities are determined rapidly.

NASA Tech Briefs, March/April 1986

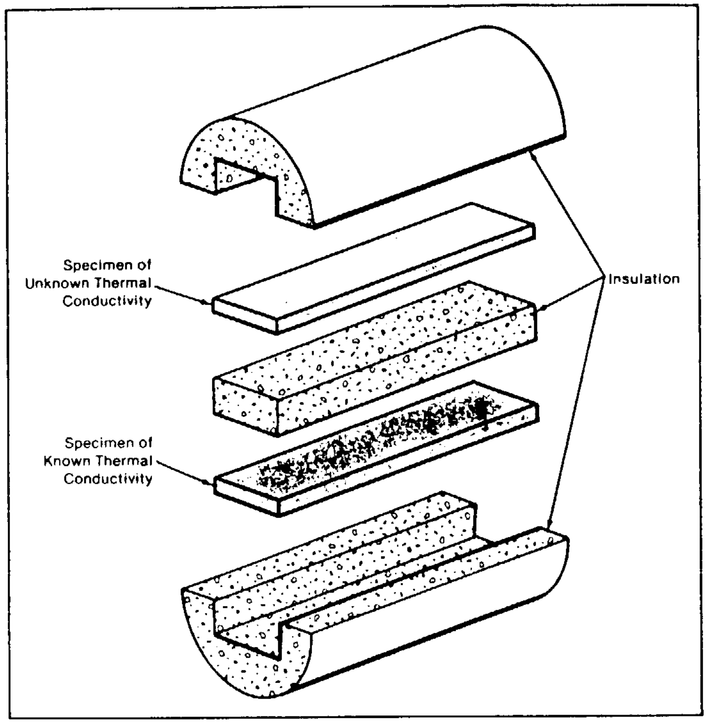
Lyndon B. Johnson Space Center, Houston, Texas

The thermal conductivity of a material can be estimated conveniently in a test procedure that compares it to a material of known conductivity. This procedure can be used to rate quickly the candidate materials for applications in which thermal conductivity is a prime consideration. Once the selection has been reduced to one or two candidates, formal thermal-conductivity test methods can be used to verify the results of the inexpensive comparative technique.

An exploded view of the test arrangement is shown in the figure. The two 1- by 6-in. (2.5- by 15-cm) test specimens, one of known thermal conductivity, are well separated by insulation and then wrapped in insulation. This assembly is then partially inserted into a tube furnace so that one end of each specimen is exposed to the furnace temperature, and the other end is open to the atmosphere. Thermocouples are attached to both ends and to the middle of each sample. The thermocouple readings are monitored for 15 to 30 min.

The steady-state conduction equations for each specimen are ratioed and solved for the conductivity ratio (candidate sample to known sample). The heat-flow ratio — needed for this calculation — is determined using the temperatures of the heated ends of the specimens to calculate the convective and radiant interchanges between the specimens and the furnace. The data-reduction procedure can be programed into a hand-held calculator. Originally applied to advanced carbon/carbon composite materials at temperatures above 3,000° F (1,600° C), this procedure should be applicable to a wide variety of materials and a wide range of temperatures.

This work was done by Charles N. Webster and James K. Willis of LTV Aerospace Corp. for **Johnson Space Center.** *No further documentation is available.*



The Two Specimens, the conductivities of which are to be compared, are placed in an assembly with insulation. One end of the assembly is then placed in a furnace. The temperature of the furnace, of each end, and of the center of each specimen are then recorded.

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Johnson Space Center *[see page 29].* *Refer to MSC-20980.*

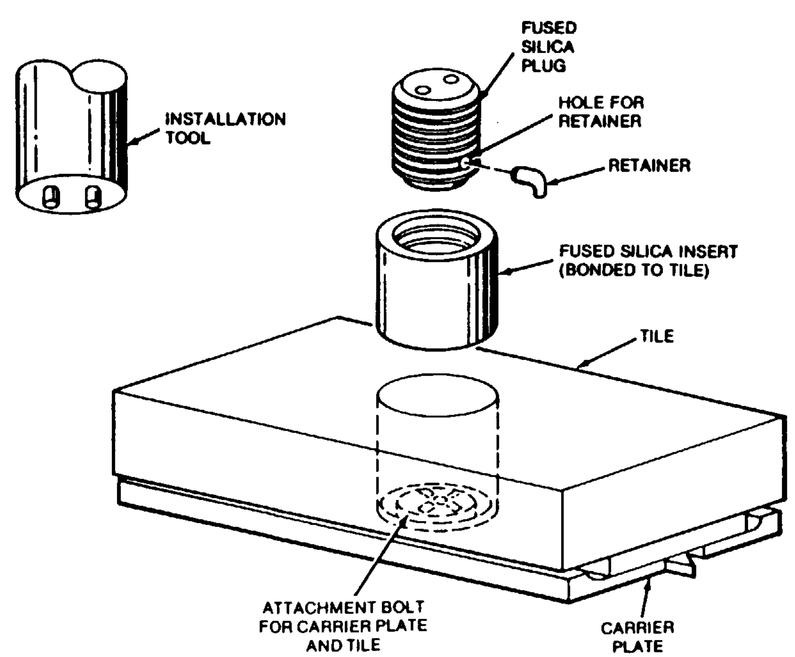
Removable Fastener for Insulating Tiles

Combination of contoured ceramic and rubber parts seals tightly and absorbs thermal expansion.

NASA Tech Briefs, Spring 1979

Lyndon B. Johnson Space Center, Houston, Texas

A fastening device seals holes in ceramic tiles securely over wide temperature excursions without cracking from thermal stresses. The fastener consists of an internally threaded silica insert, a threaded silica plug, and a molded rubber retainer (see figure). The insert is bonded in a hole in the tile, and the plug with the retainer is screwed into the insert by a pronged tool. The retainer, which has a cloverleaf cross section, is designed to absorb the expansion of the parts when the assembly is hot, and thus prevent cracking, and yet prevent the parts from loosening when the assembly is cool. The fastener can withstand temperatures up to 500° F (260° C).



Rubber Retainer with cloverleaf cross section holds plug securely but accommodates expansion at temperatures up to 500° F.

A single size of retainer fits a variety of plug sizes. The retainer is reusable, but can easily be replaced if necessary. The leg section of the retainer produces the required thread-area interference, and its body produces the required plughole interference. The locking load is the same for all plug sizes; however, the locking torque increases with plug size.

The fastener was designed to cover access holes in insulating tiles on the Space Shuttle. Such holes give access to bolts that hold tiles to the airframe, and they must be easy to seal after a tile has been attached and to unseal when a tile must be detached — hence the screwplug design of the fastener. It is expected to be useful in high-temperature industrial applications where ceramic parts must be removed and replaced.

Previous versions of the fastener used silica-topped metal pieces. These, however, tended to fail because of large internal stresses at high temperatures.

This work was done by James N. Brown, Dale H. Cade, and Harry A. Logston of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 93 on the TSP Request Card.*
MSC-16483

Fastener for Thermal Insulation Blankets

A new fastener allows rapid attachment and detachment of insulating blankets.

NASA Tech Briefs, Winter 1978

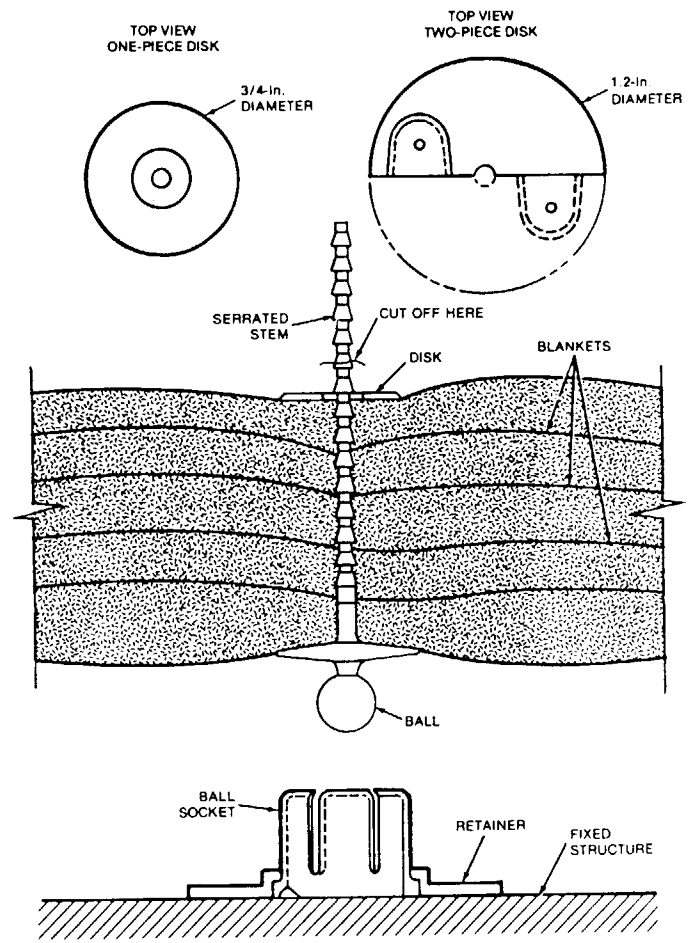
Lyndon B. Johnson Space Center, Houston, Texas

A serrated-stem fastener, similar to those that hold wire harnesses, has been adapted to a new application — attaching insulating blankets to supporting structures. The fastener (see figure) consists of a serrated stem and ball mount, a separable disk or washer, a ball socket, and a retainer. The stem and the disk hold the blankets, and the ball and socket allow the blankets to be attached to and detached from the support quickly.

The new fastener has advantages over several commercial fasteners that were originally considered for use on the Space Shuttle, including canvas snaps and string and tie buttons. The snaps were found to be difficult to install and aline; and blankets held by string and tie buttons took too long to remove and replace for area maintenance.

Installation of the fastener begins by mounting several retainers and ball sockets on the supporting structure, using adhesive. An alternate version of the retainer is riveted to the structure. Next, the ball ends of the serrated fasteners are inserted into the ball sockets. The stem, ball, and disk are made out of Torlon, or equivalent material.

The insulation blankets are mounted by pressing them against the protruding serrated stems until the blankets are pierced. The pierced blankets are held in place by flexible (metallic or nonmetallic) disks that are snapped onto the stems. For applications above 350° F (177° C), a two-piece Torlon disk was designed. (A one-piece Torlon disk was found to be too brittle and inflexible to be snapped onto the stem.) These disks are inserted on the stems, and the two interlocking halves are snapped together. Disks can be installed on both the top and the bottom portions of the stems. Stem sections protruding above the disks are snipped off.



This **Fastener for Insulation Blankets** has a serrated stem and a ball mount. The ball is engaged by a socket that is held on a support by a retainer. The insulation blankets are held in place by disks on the serrated stem. Blankets are easily installed or removed by engaging or detaching the ball mounts.

This work was done by James D. Cole and Michael L. Marke of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 80 on the TSP Request Card.*
MSC-18253

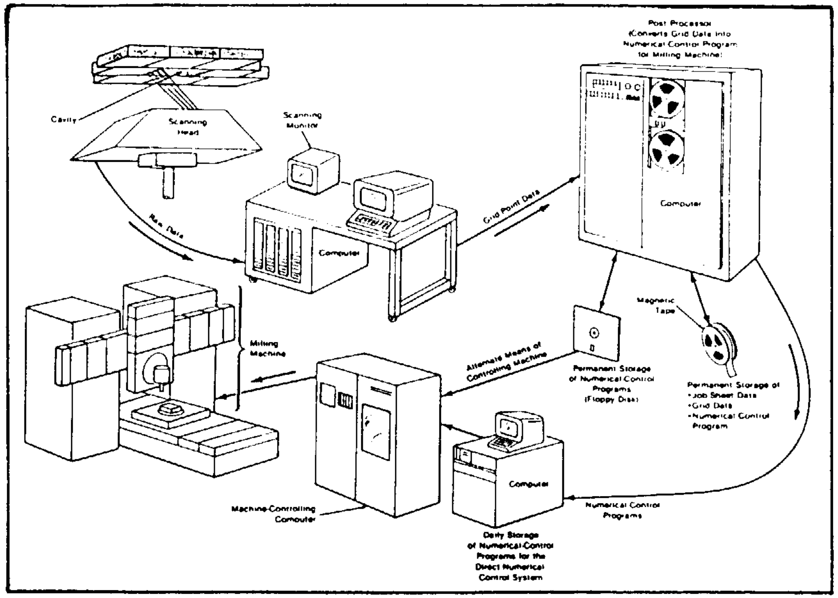
Laser Scanner for Tile-Cavity Measurement

NASA Tech Briefs, September 1987

Irregular surfaces are mapped and digitized for numerical-control machinery.

Lyndon B. Johnson Space Center, Houston, Texas

A fast, accurate laser scanning system measures the size and shape of a cavity without making any physical contact with the cavity and its walls. The measurements are processed into control signals for the numerically controlled machining of a tile



The Laser Scanning System generates a map of grid points representing a cavity and a portion of the outer surface surrounding the cavity. The map data are used to control a milling machine, which cuts a tile or block to fit in the cavity.

or block to fit the cavity. Other likely applications for this system include noncontact surface measurements, robotic vision systems, and inspection of machined and sheet-metal parts.

The system (see figure) includes a scanning head (consisting of five sets of laser/camera pairs), a scanning control computer, a television monitor, and an operator terminal. A 50-ft- (15-m-) long power, data, and control cable enables the scanning head to be repositioned for different measurements.

The laser beam is directed in front of the camera and sweeps across the cavity walls and the adjacent outer surface. The measurement data travel through the cable to a computer where they are translated into positional coordinates. Altogether, the system thus generates the coordinates of 7,000 points to map the surface in about 50 s. In addition, the system can map the missing portion of the outer surface at the cavity opening by mathematical extrapolation from the adjacent outer surface. The grid-point data are fed into a post-processor, where they are translated into the numerical control language understood by a milling machine.

This work was done by Stanley Y. Yoshino, Donald H. Wykes, George R. Hagen, Gene E. Lotgering, Michael B. Gaynor, Paul G. Westerlund, and Thomas A. Baal of Rockwell International Corp. for **Johnson Space Center**. *For further information, Circle 68 on the TSP Request Card. MSC-21136*

Duplicating Curved Tile Surfaces for Pull Testing

Vacuum chucks can be
custom-made at low cost.

Lyndon B. Johnson Space Center, Houston, Texas

The strength of adhesive bonds to fragile objects with complex shapes (for example, curved-contour ceramic tiles) can be tested easily with vacuum chucks. A fast splash-molding technique, originally developed for vacuum chucks used on the silica tiles attached to the outer surface of the Space Shuttle orbiter, is used to custom-fabricate a chuck for a specific tile.

In a bond-strength test, the contact surface of the chuck is pressed against a matching surface of the bonded tile. A vacuum line extending through a fine bore in the chuck creates a vacuum at the chuck-and-tile interface. Thus, when the chuck is pulled away, atmospheric pressure tends to push the tile away with the chuck. If the adhesive resists a predetermined pull exerted by the chuck on the tile, the bond is acceptable.

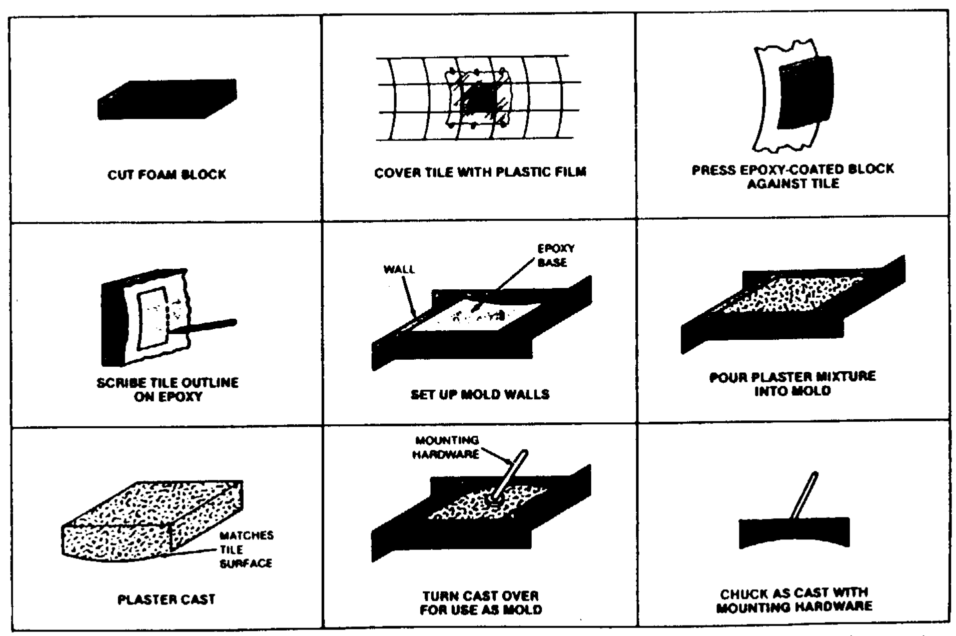
It is essential, of course, that the contour of the chuck be a close replica of the mating tile contour; otherwise, the vacuum will not hold. Original plans were for the chuck to be shaped by a milling machine from the same computer-numerical-control tapes used to generate the tile surface. Splash molding, however, made this expensive and time-consuming procedure unnecessary.

"Splashing" consists of pressing a quickly setting layer of epoxy resin against the tile with a block of flexible polyurethane foam. The epoxy layer hardens, retaining the contour of the tile. The "splashing" and subsequent steps of the technique are illustrated in the figure.

To prevent the epoxy from adhering to the tile, the tile to be splashed is first covered with a plastic film such as that used to wrap food. The film is smoothed to remove creases and air bubbles and is fastened at the edges with tape. The epoxy components are mixed and poured evenly on a foam block large enough to cover the tile. The epoxy-coated block is immediately pressed by hand against the film-covered tile. The block is removed, and, with the aid of a straightedge, the outline of the tile in the epoxy impression is scribed. The outlined perimeter of the epoxy block is cut with a bandsaw and its edges are sanded smooth.

Four walls are clamped to the plate, forming a mold. A release agent is applied to the mold, and a plaster mixture is poured into it. After hardening for 30 minutes, the plaster cast is removed. The surface of this cast duplicates that of the tile. The plaster cast is turned over and the mold walls placed around it again. The vacuum chuck and its integral mounting hardware are then cast on the plaster duplicate.

This work was done by Richard H. Sebenick of Rockwell International Corp. for **Johnson Space Center.** *No further documentation is available.*
MSC-20795



A Step-by-Step Procedure for splash molding results in a pull-test chuck contoured for a specific tile. Although the chucks are custom-made, the fabrication process is less expensive than machining.

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OF POOR QUALITY

Hot-Wire Tile Removal Tool

Tiles held in place by heat-activated bonding agents can be released without damage to the tile or to the agent.

NASA Tech Briefs, Fall 1976

John F. Kennedy Space Center, Florida

One means of bonding flat, planar objects (such as heat-resistant ceramic tiles, flooring tiles, and the like) to a surface is to use a thermosetting material. The material simplifies tile installation; however, it is often very difficult to remove the tiles for repair or replacement without damaging or destroying them and creating unwanted debris in the work area.

A tool, which works and looks like a wire-blade cheesecutter, has been developed to aid in tile removal without damaging the tile or harming the thermosetting embedment. In use, the tool is inserted into the material (between the tiles), is pushed down beyond the tile depth, and is drawn across the tile and up out from the material. The tile is then removed.

The cheesecutterlike tool uses an electrically heated wire to slice through the embedment. Two beryllium/copper arms [0.030 in. (0.076 cm) thick], which hold the wire, fit between the tiles to pull the wire through the thermosetting compound. The heated wire is Nichrome 0.025 in. (0.064 cm) in diameter and, as illustrated, is attached to the arms for removal or adjustment as required. The wire is powered via an external voltage regulator which sources it with 6.5 volts at 9 amperes.

The **Hot-Wire Tile Removal Tool** rapidly removes flat, planar objects from the bonding agent without adjustment and without producing dust or debris.

This work was done by Jack W. Holt of Rockwell International Corp. for **Kennedy Space Center.** *For further information, Circle 99 on the TSP Request Card.*
KSC-11043

Metallic Thermal Seal

An easy-to-install metal barrier prevents hot gas from leaking through small gaps.

NASA Tech Briefs, Winter 1978

Lyndon B. Johnson Space Center, Houston, Texas

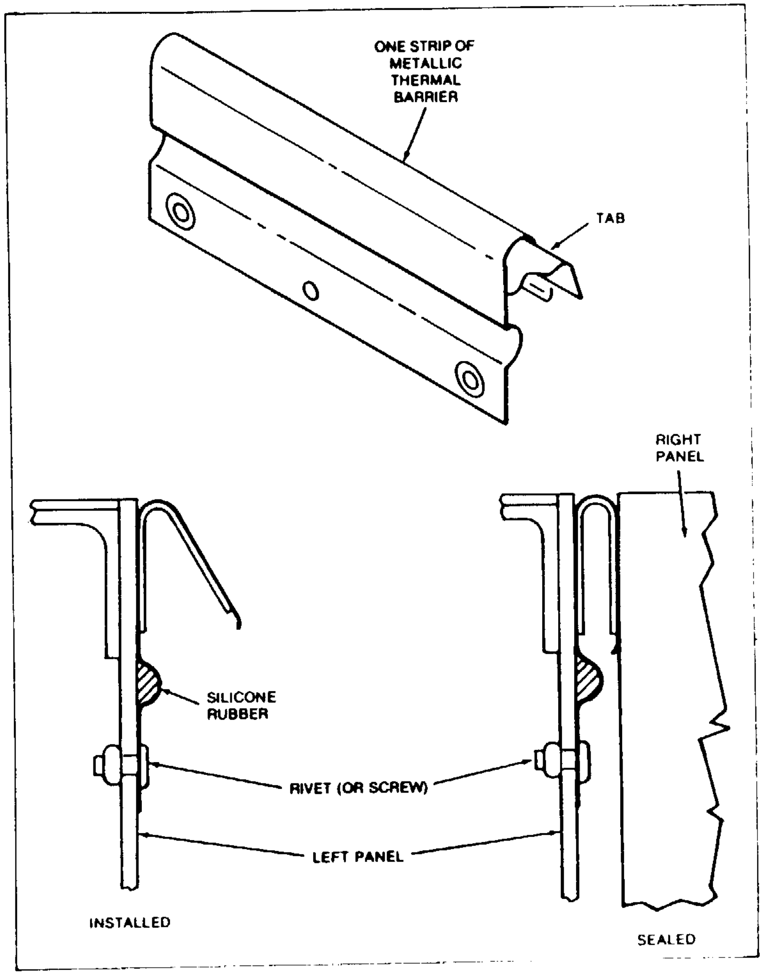
A simple metallic seal, originally designed to prevent hot gas from leaking through gaps between the split rudder and speed brake on the Space Shuttle, could be adapted for home and industrial use. It could, for example, be used to prevent heat from escaping through gaps around doors, windows, partitions, and similar structures.

The barrier consists of a series of U-shaped metal strips that are installed along the entire length of the gap. Each strip (see figure) is secured by rivets or screws to one side of the gap. This side also includes a channel filled with room-temperature-vulcanized (RTV) silicone rubber. The rubber lining prevents gas from leaking into the cavity on this side of the seal. The other side of the metallic barrier presses against the surface on the opposite side of the gap. Tabs are spot-welded to adjacent telescoping seal sections, forming a continuous barrier along the entire gap.

The new barrier is particularly effective in sealing small gaps [these were 0.25 to 0.32 in. (6.3 to 8.1 mm) on the Space Shuttle]. It is also effective at high temperatures at which nonmetallic seals cannot be used. In addition to its possible uses in homes, the thermal barrier might also be utilized by metal-processing industries to insulate part of a metal section during intermediate processing steps.

This work was done by John Bellavia and John O. Kane of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 76 on the TSP Request Card.*

MSC-18135



This **Metallic Barrier** fits into **gaps between** panels to prevent hot gas from escaping. The barrier compri**ses** U-**shaped** strips secured to one end of the gap by screws. The opposite end is spring-compressed for a flush fit. Tabs, spot-welded to adjacent barrier elements, form a continuous seal.

Coating for Hot Sliding Seals

Heat-resistant paint reduces friction, improves reliability.

Lyndon B. Johnson Space Center, Houston, Texas

A paint has been found to be an effective surface coating for a sliding seal that must operate at elevated temperatures. Unlike other coatings, the paint is easy and economical to apply and offers minimal friction. Ceramic coatings, for example, must be applied by plasma arc spraying; and unless such coatings are ground or lapped to a fine finish, they are abrasive in a sliding seal. Moreover, ceramic coatings can interfere with subsequent manufacturing processes such as brazing.

High-emittance surfaces have been prepared on iron/cobalt/nickel heat- and corrosion-resistant alloy using Pyromark black refractory paint. (An equivalent paint could be used.) The paint is applied in successive 1-mil (0.025-mm) layers and is cured and vitrified at about 200° F (111° C) above the operating temperature. It can be used at temperatures as high as 2,000° F (1,093° C).

In rubbing tests against impregnated graphite at 5 psi (34.5×10^3 N/m^2) loading, the coefficients of friction were 0.25 (static) and 0.1 (dynamic) at room temperature and at 900° F (482° C). Wear on paint and graphite surfaces was negligible.

This work was done by Julius Stock of Fairchild Industries Inc. for **Johnson Space Center.** *No further documentation is available.*
MSC-16529

High-Temperature Insulation

Lightweight insulating material works over a very broad temperature range.

NASA Tech Briefs, Fall 1979

Marshall Space Flight Center, Alabama

A new insulating material developed for NASA's Space Shuttle main-engine nozzle protects the nozzle skin from engine plume heat. The material is unaffected by moisture or hydraulic oil and is usable at temperatures ranging from 2,200° F (1,200° C) to cryogenic levels. It could be readily applied to a number of high-temperature and cryogenic processes.

The material is a partially sintered batting using 2-percent-dense [10 lb/ft^3 (189 kg/m^3)] 8-μm wire blanket available in 1/8 in. (3.2 mm) thickness. Two commercially-available wire materials that may be used include CRES (corrosion-resistant steel) for operation up to 1,400° F (760° C) or 80Ni-20Cr Nichrome for operation up to 2,200° F (1,200° C) temperature.

In both cases, the material was successfully tested for 10 cycles of dipping in liquid nitrogen (LN_2) and thawing at 300° F (150° C). It also withstood two cycles of hydraulic liquid soaking, LN_2 dipping, and 1,600° F (870° C) oven thawing.

Other tests included prolonged thermal exposure to 2,200° F (1,200° C), emissivity tests prior to and after the thermal exposure, thermal-conductivity tests to 2,000° F (1,100° C), vibration to simulate anticipated nozzle-operation levels, and salt-spray and corrosion tests. No other lightweight insulation was found to withstand all of these tests.

This work was done by R. E. Mowers and A. C. Peterson of Rockwell International Corp. for **Marshall Space Flight Center.** *No further documentation is available.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Marshall Space Flight Center [see page A5]. Refer to MFS-19498.

Transport and Installation of Fibrous Insulation

Two methods preserve the fiber orientation before and during installation.

Lyndon B. Johnson Space Center, Houston, Texas

Two new techniques simplify the transport and installation of oriented-fiber thermal insulation. Other applications involving oriented fibers or loose fillings might also be able to utilize the methods.

It is necessary to fill spaces around the Space Shuttle reaction-control thrustors with insulation fibers in specific orientations. Each thrustor has to be wrapped with fibers able to withstand 2,500° F (1,370° C) and having the proper density so that the temperatures of thrustor components can be kept within safe limits.

In one installation method, layers of uncured phenolic-impregnated batting in various shapes are wound around a mold and then compressed to the required density by a mandrel. The mold and insulation are then heated to 425° F (218° C) to cure and evaporate part of the binder. The insulation has the minimum practical binder content — 4-1/2 percent, just enough to prevent the parts from breaking up in transit. The rest of the binder is volatilized after installation, when the thrustor is heated to 600° F (316° C) for 48 hours.

The second method uses no resin. Instead, the insulation is compressed in a mold and saturated with water. The wet insulation is then frozen in the shipping container. After the frozen insulation is installed around the thrustor, it is heated until the water is driven off.

This work was done by Sieg Borck, Frederick L. Falconer, and Raymond V. Loustau of The Marquardt Co. for **Johnson Space Center.** *No further documentation is available.*
MSC-20074

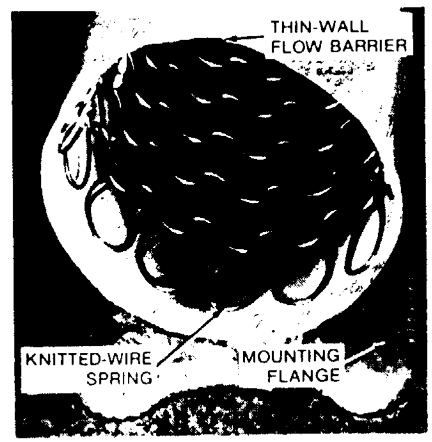
Thermal Seal for High and Low Temperatures

A composite seal remains flexible between -423° and +500° F.

Lyndon B. Johnson Space Center, Houston, Texas

A composite bulb seal can sustain temperatures ranging from −423° to +500° F (−253° to +260° C). The seal (see figure) incorporates a knitted-wire tubular spring core surrounded by a thin-wall flow barrier of polytetrafluoroethylene or polyolefin. Because of its wide temperature capability, the seal outperforms conventional elastomeric seals used in industrial freezers, environmental chambers, refrigerated trucks and railcars, and aircraft doors.

The knitted-wire tubular spring is resilient and can be deflected to an extreme without permanently deforming. The 304 stainless-steel wire has a constant modulus of elasticity over a very wide temperature range, and the spring force can be easily adjusted for specific applications by varying the wire diameter and knit density.



The **Bulb Seal**, designed for a wide temperature range, is made from a tubular knitted-wire spring core surrounded by a thin-wall polytetrafluoroethylene or polyolefin flow barrier.

The flow barrier remains flexible down to −423° F, which exceeds the performance of silicone-rubber seals. The latter are limited to about −150° F (−101°C), below which the material undergoes a glass transition [at approximately −170° F (−112° C)].

The seal can be used as is with the flow barrier and the spring, or it can have a mounting flange or facet bonded to the flow barrier.

This work was done by James F. Collipriest, Jr., and Donald M. Fell of Rockwell International Corp. for **Johnson Space Center**. *For further information, Circle 76 on the TSP Request Card.*
MSC-16151

Flexible Sliding Seal

Composite seal withstands temperatures up to 1,600° F.

NASA Tech Briefs, Winter 1979

Lyndon B. Johnson Space Center, Houston, Texas

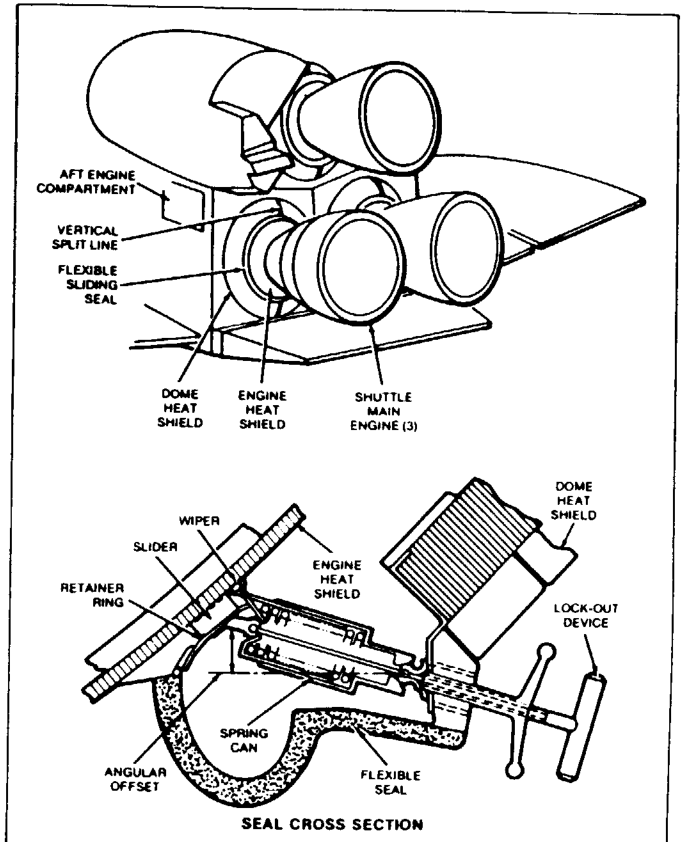
A circular seal both slides and flexes to accommodate relative motion between two sealed members. Originally developed for the Space Shuttle orbiter (see figure), it contains a sliding seal to accommodate engine gimbaling and a flexible seal that absorbs the forward motion at high thrust of the engine heat shield relative to the airframe. Other possible applications are in the support structures of heavy machinery and vehicle engines.

The seal can withstand temperatures up to 1,600° F (871° C) while sealing against moderate positive (crush) and negative (burst) pressures. It vents the interior to the exterior at a crush pressure difference of 1.32 psi (9,100 N/m^2) and a burst pressure difference of 2.65 psi (18,200 N/m^2).

The sliding seal consists of a silicone-rubber wiper coated with PTFE and a series of electrodeposited-nickel-on-graphite slider blocks. Forty-eight spring cans in a retainer ring exert pressure on the graphite block so that they slide and seal against the spherical surface of the engine heat shield. The retainer ring is held in position by four articulated links anchored to the airframe heat shield. The links allow the sliding seal to follow the forward movement of the engine heat shield as the engine thrust builds up.

The flexible seal consists of an outside layer of silica-fiber fabric, a layer of thermal insulation, a layer of glass-fiber fabric for pressure sealing and structural support, a polyimide film for a pressure-seal backup, and iron/nickel-alloy mesh for lightning protection. The layers are sewn together with quartz thread. Since the seal flexes only slightly as the pressure differential changes from positive to negative, the layers are not subjected to wrinkling or shape reversals and therefore last longer.

The seal can be removed with relative ease so that the engine can be inspected or replaced. A special "lockout" tool is inserted in each spring and turned to lock the can to the airframe heat shield. The retainer ring is then disconnected at the seal meridian splices. (Like the heat shields, the seal is in two half-circle sections.) The airframe heat shield (plus seal) is removed in two major assemblies, and the engine heat shield is removed in two halves.



The **Flexible Sliding Seal** is a ring about 7 feet (2.1 meters) in diameter at the interface between each Space Shuttle main engine and the engine compartment. The design can be resized for other applications. The cross-sectional view shows the structure of the sliding and flexible portions. The curvature of the flexible seal supports both positive and negative pressures without wrinkling.

This work was done by Edwin L. Wallenhorst of Rockwell International Corp. for **Johnson Space Center.** *To obtain a copy of a report describing the development of the Space Shuttle aft-heat-shield seal, Circle 65 on the TSP Request Card.*
MSC-18467

Flexible Heat-and-Pressure Seal

A thermal/pressure seal accommodates transverse and lateral motion between the sealed surfaces.

Lyndon B. Johnson Space Center, Houston, Texas

A new seal withstands both heat and pressure and accommodates relative motion between the sealed surfaces. It consists of a flexible tube filled with a thermally insulating material and coated with a pressure-resistant material.

The new seal withstands temperatures of 1,950° F (1,066° C) on one side while maintaining the temperature on its "cool" side at less than 350° F (177° C). It seals gases at pressures up to 5.0 psi (34×10^3 N/m^2). One such seal, fabricated for a 1.5-inch (3.8-cm) gap, accommodates about 2.5 cm of side-to-side movement of the surfaces and transverse motion of about 3.5 cm without disrupting the heat and pressure barrier.

As shown in Figure 1, the core of the tube is filled with alumina/silica batting for heat insulation. The batting is held in a sheath of knit iron/nickel-alloy wire, which gives the necessary strength, flexibility, and resilience. A ceramic-fiber sleeve is braided snugly around the wire sheath for further insulation, and the sleeve is enclosed in a glass-fiber cover coated on the outside with a layer of silicone as a pressure sealant.

The tube is held in place by a ceramic retaining strip on both of the sealed members and by a tensioning rod. The seal nests between the ceramic strips with the end flaps of the outer covering held in place by the rod.

If the sealed members move, the seal deforms (Figure 2), changing shape to accommodate the relative motion. A synthetic felt strain-isolator pad is bonded between each member and its retaining strip.

This work was done by John Bellavia, Jr., and John O. Kane of Rockwell International Corp. for **Johnson Space Center.** *For further information, Circle 77 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Johnson Space Center [see page A5]. Refer to MSC-18134.

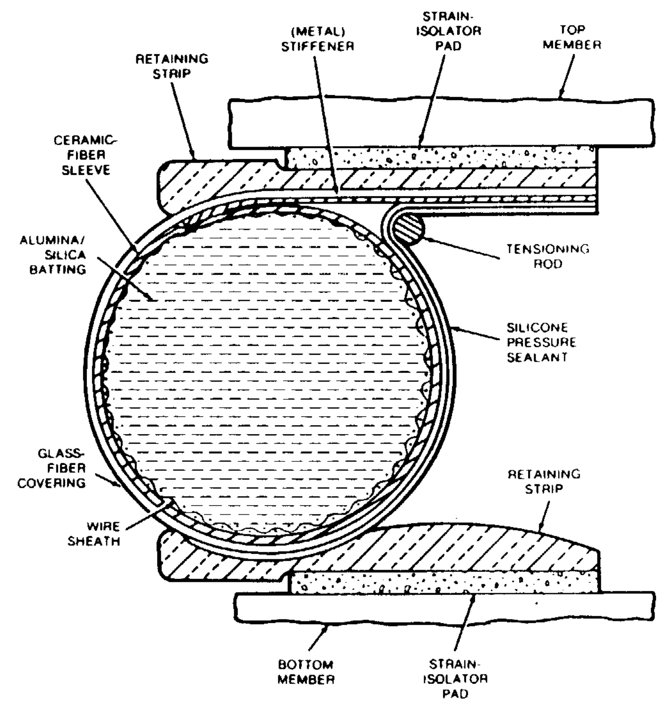


Figure 1. The **Heat/Pressure Seal,** shown here in cross section, uses an outer silicone layer to keep out gases. The pressure limit, 5 psi in the original design, can be raised by increasing the thickness of the silicone layer. The ceramic fibers and inner batting block heat flow. A wire sheath is the structural framework for the seal.

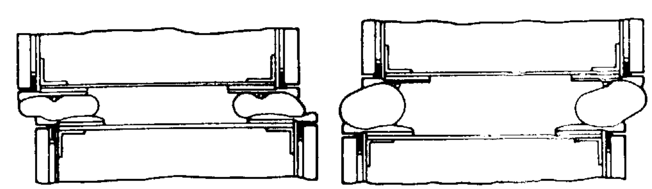


Figure 2. The **Seal Deforms To Accommodate Relative Movement** of the sealed members — left: lateral movement with minimum spacing between the members, right: lateral movement with maximum spacing between the members.

Heat/Pressure Seal for Moving Parts

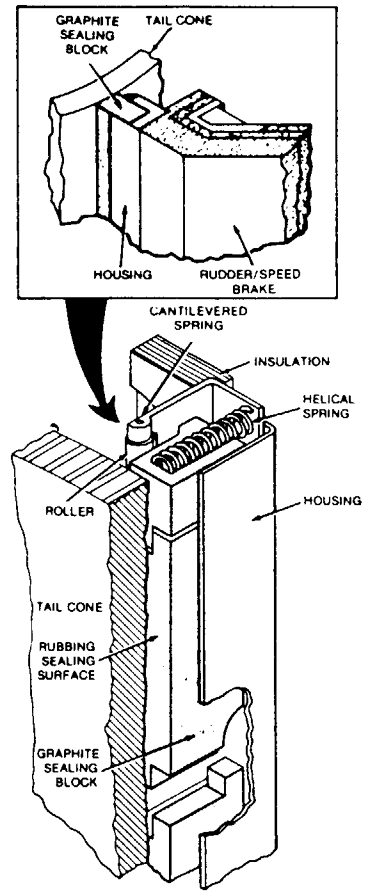
Articulated graphite segments accommodate dimensional variations.

Lyndon B. Johnson Space Center, Houston, Texas

A prototype seal keeps hot gases from leaking between large, adjacent parts in relative motion. The seal withstands temperatures greater than 1,000° C (1,800° F) and accommodates heat- and pressure-caused distortion of the parts. It is nonabrasive, creates little resistance to the movement of parts, and causes minimal wear and damage to surface coatings.

The seal was developed for the tail of the Space Shuttle orbiter, where it protects rudder/speed-brake components from high temperatures during reentry. It could also find application in manufacturing processes requiring thermal barriers between large moving parts.

The seal (see figure) consists of a housing, for attachment to one of the two adjacent moving parts, and an articulated graphite seal block, located in a channel within the housing. There are two sealing surfaces on the seal block: a rubbing surface that is held against one of the moving parts by helical springs, and a sliding surface that is held against the seal housing by a cantilever spring and a roller. The helical-spring force varies only slightly with deflection so that essentially constant pressure is maintained on the rubbing surface. In addition, the spring force is low enough to minimize rubbing friction and high enough to ensure an adequate seal under atmospheric and aerodynamic pressures.



Graphite Sealing-Block Segments move on the Space Shuttle tail cones as the rudder/speed brake moves. Mounted on the rudder/speed brake, the housing contains springs that bias the block segments against the tail cone and housing sealing surface. The block segments accommodate dimensional changes caused by variations in temperature and pressure.

The Shuttle seal is 20 ft (6.1 m) long. The overlapping joints between adjacent seal block segments allow the blocks to conform to the tail cone surface. The opening created by the angle between blocks is so small that only an insignificant amount of gas leaks through. On the Space Shuttle, the seal permits 2-ft (0.61-m) relative rotational motion of the rudder/speed-brake leading edge around the tail cone.

Because of the high-temperature difference between the graphite blocks and the metal housing, each block is positioned by a guide slot and locating pin and is held against the housing by a cantilever spring. The locating pin is attached to the side of the housing, and the guide slot is between projections of the graphite blocks. This allows relative expansion and contraction of the housing without interrupting the seal.

This work was done by Martin L. Stevens of Fairchild Republic Co. for **Johnson Space Center.** *For further information, Circle 74 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Johnson Space Center [see page A5]. Refer to MSC-18422.

Improved High-Temperature Seal

A flexible seal has four
barriers against leakage.

NASA Tech Briefs, Summer 1981

Lyndon B. Johnson Space Center, Houston, Texas

The high-temperature seals on the Space Shuttle orbiter elevons will be improved by a new design that increases the number of leak barriers, allows for thermal expansion, and cuts the weight by more than two-thirds. The improved seal should be useful in other applications where it is necessary to seal gaps between moving surfaces.

The main components of the seal, seen in Figure 1, are constructed from Inconel (or equivalent alloy) metal foil. An outer foil wrapper maintains contact between the surfaces to be sealed, which are the cylindrical tube and the baseplate in this mockup. The foil wrapper is formed into telescoping sections, allowing for length adjustment to compensate for temperature changes. There is a double thickness of foil wrapper over most of the length.

A secondary seal, which is enclosed within the outer seal, consists of a ceramic cloth sleeve wrapped around a foil tube containing an alumina mat filler. As shown in Figure 2, the seal presents four barriers to airflow. The previous design, described in "Improved Wrapped-Curtain Seal" on page 435 of *NASA Tech Briefs*, Vol. 4, No. 3, had only two flow barriers. A weight reduction of the Space Shuttle orbiter elevon seals from 138 to 32 lb (63 to 20 kg) is expected.

This work was done by Kenneth E. Wood, Paul P. Zebus, and Alan R. Olson of Rockwell International Corp. for **Johnson Space Center.** *No further documentation is available.*
MSC-18790

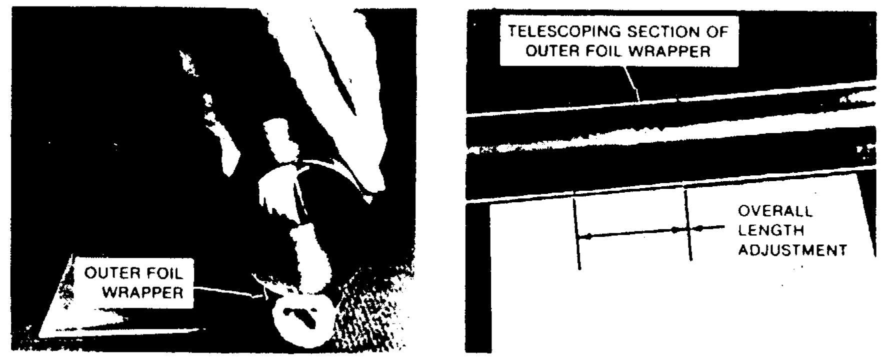


Figure 1. **Components of the Telescoping Elevon Seal** are shown in a mockup. Here, the cylindrical tube and the baseplate represent the surfaces to be sealed.

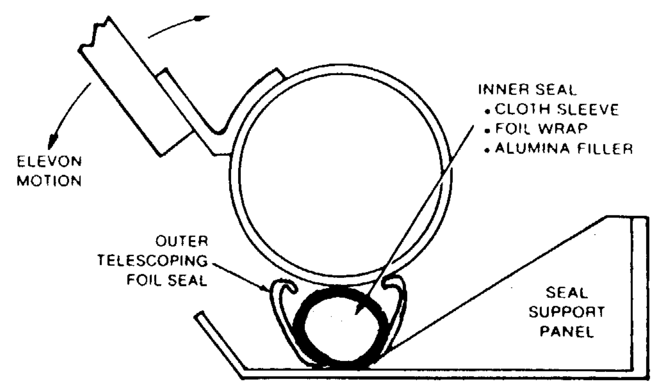


Figure 2. The **Improved Seal** presents four barriers to airflow, instead of only two as in the previous design.

Ceramic-Cord Gas Seal

High-temperature gasket material seals at temperatures above 1,100° C.

NASA Tech Briefs, Fall 1982

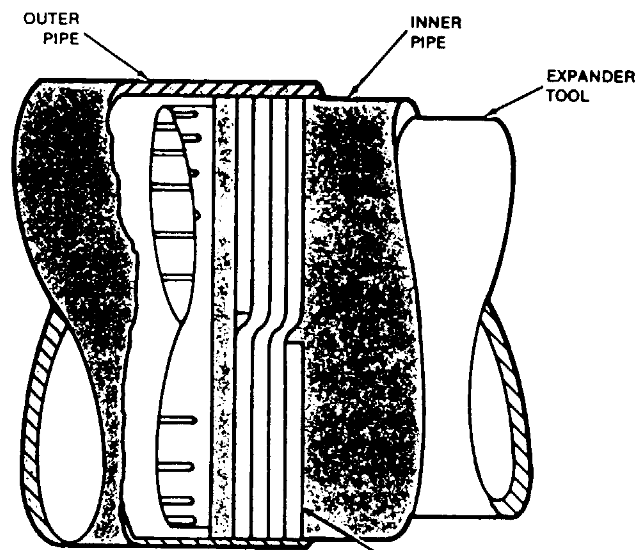
Lyndon B. Johnson Space Center, Houston, Texas

Braided cord made of ceramic fibers can form a low-pressure gas seal for temperatures up to 1,160° F (630° C). The alumina/boria/silica cord has been used as an exhaust-duct seal on the Space Shuttle orbiter. Those designing high-temperature seals may find the braided ceramic superior to asbestos in some applications.

The ceramic-cord seal between concentric pipes is shown in the figure. First, the cord is wrapped around the inner pipe. The wrapped inner pipe is then slid into the outer pipe. Following insertion, an expander tool is placed inside the inner pipe and used to expand the inner pipe, thus compressing the cord between the two pipes to form a seal.

The ceramic-fiber cord is adaptable for emergency repairs, quick replacement, or permanent installation. It requires less-stringent machining tolerances than are demanded by metal or elastomeric O-rings. Typical applications might include engine exhaust ducts or hot pipes passing through firewalls.

This work was done by Charles W. Etzel of Rockwell International Corp. for **Johnson Space Center.** *For further information, including a specification for preparing the braided ceramic cord, Circle 46 on the TSP Request Card.*
MSC-20200



Concentric Exhaust Pipes are typical of applications in which ceramic-cord seals might be used. The cord is crushed to form a seal between the inner and outer pipes when the inner pipe is expanded into place.

Metallic Panels Would Insulate at 2,700° F

Insulation under development for the Space Shuttle could have other applications.

NASA Tech Briefs, Spring 1981

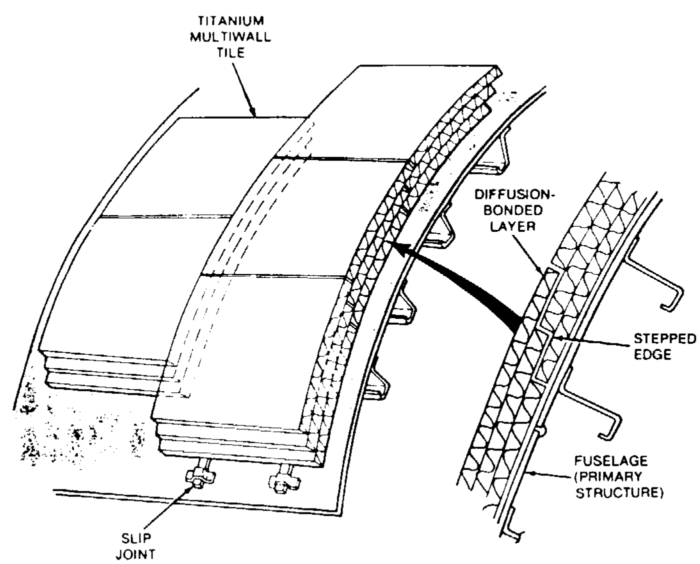
Langley Research Center, Hampton, Virginia

Multiwall metallic panels now under development as replacements for the ceramic surface-insulation tiles of the Space Shuttle could eventually be used in other aircraft and possibly even as thermal protection in ground-based applications. Various configurations of the basic multilayer sandwich are expected to protect against temperatures ranging from 700° to 2,700° F (370° to 1,480° C). With assistance from heat-pipe cooling, the panels should withstand temperatures to 3,500° F (1,930° C); however, the heat pipes would not exceed 1,600° F (870° C).

A major objective for the new panels is to survive the expected 100-mission lifetime of the Space Shuttle. They should be more durable than the fibrous or ceramic tiles, which have to be carefully monitored for surface fraying, erosion, and cracking. The new metal panels should withstand thermal and mechanical stresses better, without adding to the weight of the vehicle.

Each panel attaches to the primary structure with a bayonet mounting at its corners (see figure). This simple slip joint allows for expansion to relieve thermal stress. Furthermore, each panel is isolated from the strains of the primary structure, with adequate space for thermal expansion and mechanical tolerances. To prevent vibration, felt strips are compressed at the edges of the panels. Each panel is vented to the local static pressure through a hole in the felt strip at the trailing edge on the cooler surface.

The insulating panels have different structures, depending on the local temperature to be encountered. For the temperature range from 700° to 900° F (370° to 480° C), multiwall metal (usually titanium) panels consist of alternating flat and dimpled sheets, joined at dimple crests as shown in the figure. Heat conduction through these panels is minimal because the layers touch only at the dimple crests, resulting in a contact area less than 0.2 percent of the surface area. The long conduction path and low heat conductivity of titanium contribute to the thermal barrier. Radiative heat transfer is inhibited by the multiple layers, each of which is a radiation barrier. Heat transfer by convection is avoided by keeping the cells small. Gaseous conduction can be eliminated by evacuating the cells and sealing them; however, when vented they are weight-competitive with the ceramic tile system.



Insulating Panels are attached to a surface to protect the interior from high temperatures. In this illustration, the panels are multiwall in which layers of dimpled foil alternate with smooth ones, forming thousands of small cells within the panel.

For the temperature range from 900° to 1,600° F (480° to 870° C), a metallic enclosure supports a fibrous insulating filling. The pressure loads are supported by either an outer foil-gage superalloy dimple-core sandwich or a superalloy honeycomb sandwich and an inner titanium sandwich, joined by beaded edge seals.

For the range from 1,600° to 1,900° F (870° to 1,040° C), the outer sandwich is a superalloy honeycomb sandwich with a thicker facesheet to resist oxidation.

In the range of 1,900° to 2,200° F (1,040° to 1,200° C), the outer layer is a flanged waffle of oxidation-dispersion-strengthened alloy, which has excellent oxidation resistance.

At 2,200° to 2,700° F (1,200° to 1,480° C), the outer layer is a coated refractory-metal-flanged waffle or a rib-stiffened advanced-carbon/carbon panel. While the carbon/carbon is lighter in weight, the refractory metal panels may last longer due to better resistance to damage.

Nickel-alloy heat pipes can be used where localized heating may produce temperatures as high as 3,500° F. The heat pipe would transfer the heat to a large, cooler area, from which it can be radiated; and the heat pipes operate at about 1,600° F for long life.

The basic technology required to produce the panels has been or is being developed, and models have been built. They await large-scale testing of assemblies on representative structures.

This thermal protection system was conceived by L. Robert Jackson of **Langley Research Center.** *For further information, Circle 80 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Langley Research Center [see page A5]. Refer to LAR-12620.

Ablative-Filled Honeycomb Composites

Two techniques reduce fabrication cost and complexity.

NASA Tech Briefs, Summer 1976

Langley Research Center, Hampton, Virginia

Two new techniques have been developed to simplify and reduce the expense of fabricating ablative-filled honeycomb composite structures. One technique, called net surface molding, permits ablative insulators with sculptured tapers to be produced over a substrate having unpredictable irregularities. Thickness may easily be controlled with a precision of ± 0.010 in. (± 0.025 cm). Previous methods, which were very costly and not very reproducible, required a final machining of the entire outer surface after curing the ablative to an oversize thickness. This required that the depth of the molded material, to the substrate, be measured, followed by a rough machining and a final hand finishing to produce a tapered surface of precise thickness.

The second technique, called subsurface molding, results in an ablative surface below the honeycomb face. Then the composite is machined using the honeycomb face as a tooling reference surface. The method may also be used to fabricate composites having thin strata of different densities throughout their core thicknesses. This makes it possible to tailor the heat-char and ablation rates to satisfy varying requirements and to produce localized islands of high or low density without costly premolding or multiple-cure processing.

The steps of the processing sequences for both techniques are similar.

1. The core is machined in detail to final thickness using simple plane-surface machining methods, and the core thickness is verified. The core may be tapered as desired at that time.
2. The core details are hot formed and are bonded together. Alternatively, this may be done prior to machining.
3. The substrate and all details are cleaned.
4. The core is bonded to the substrate. Simple depth measurement techniques may be applied then to verify the core thickness.
5. The core is primed, and a base coat is applied.
6. The preweighed quantities of the desired filler are loaded to given surface areas using loading frames, a slipsheet, and a leveler bar.

At this stage the two techniques differ. If the first process is being used, proceed directly to step 10.; if the second process or a variable density core filler is desired, then:

7. Uncured silicon gum-rubber sheets are positioned over the surface to be molded and serve to drive the filler into the core, producing controlled depressions.
8. The filler is cured under a vacuum bag in a convection oven.

This completes the subsurface molding process. For a variable density filler that completely fills the honeycomb core, the steps are:

9. The gum rubber is then stripped away, and step 6. is repeated, using a filler of a different density.
10. The last of the filler is pressed into the cells which are then covered with a flexible pressure membrane. A controlled pressure is applied over the membrane, assuring uniform cell filling and the control of the filler densities.
11. The flexible pressure membrane is removed, and the excess filler extending beyond the core surface is removed using flexible strips of metal shim stock as scrapers.
12. Rigid caul sheets are positioned over the surface to be molded and serve to bridge over the core cell width. This prevents the core pressure membrane from pulling into the cells, thus precluding the possibility of an uneven filler thickness occurring across the open cell faces.
13. The filler is finally cured under a vacuum bag in a convection oven.

The resultant ablative net thickness has a molded finish which requires no additional machining.

This work was done by H. L. Linebarier of Martin Marietta Corp. for **Langley Research Center.** *No further documentation is available.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Langley Research Center [see page A8]Refer to LAR-11180.

Oxidation-Resistant Slurry Coating for Carbon-Based Materials

New process uses a paint sprayer and vacuum furnace
to produce a silicon carbide outer layer.

NASA Tech Briefs, Winter 1983

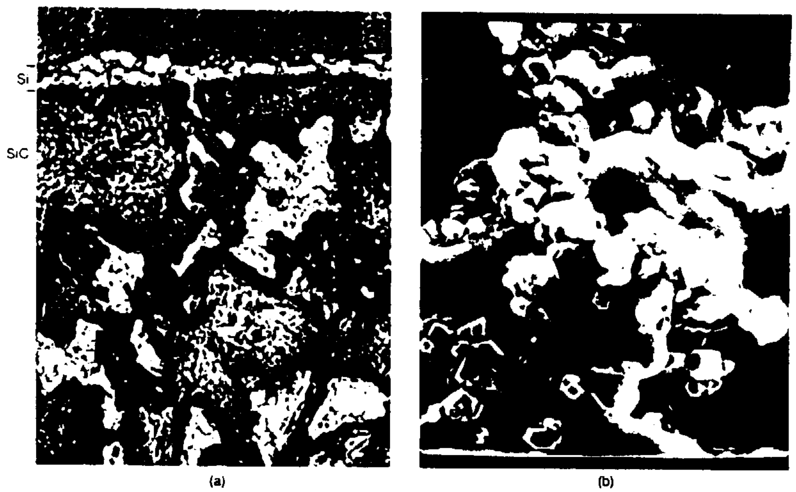
Lewis Research Center, Cleveland, Ohio

Carbon-based materials, such as graphite or carbon/carbon composites, are very lightweight, possess excellent mechanical properties at high temperatures [2,200° to 3,300° F (1,200° to 1,800° C)], and are resistant to thermal shock. However, above about 900° F (480° C), these materials oxidize rapidly unless they are coated with oxidation-resistant materials. Most commercially available coatings are based on chemical-vapor deposition (CVD) processes, which require specialized equipment to produce a silicon carbide (SiC) outer layer.

The process described here also produces an oxidation-resistant SiC outer layer; but it may be produced more easily, using only a paint sprayer and a conventional vacuum furnace. Fine nickel and silicon powders are blended in proper proportions, mixed with nitrocellulose lacquer, and sprayed on the graphite parts to be coated. The coatings are air-dried and vacuum-fused at temperatures near 2,400° F (1,300° C). The resultant coatings wet the graphite, penetrate any open porosity, and produce a SiC reaction zone. A top surface of silicon (see figure), which has a metallic luster may also exist. Upon exposure to air at high temperature, the Si + SiC coating protects the graphite part from catastrophic oxidation by forming an impervious slow-growing silicon dioxide (SiO_2) surface scale.

Preliminary 2,200° F cyclic oxidation tests showed that the coatings themselves are very oxidation resistant and protected the graphite for about 5 to 10 hours. Uncoated graphite by comparison was totally consumed by oxidation after only 5 hours at 2,200° F. However, localized coating infiltration of the excessive porosity in the graphite resulted in premature oxidation of the substrate. Use of less-porous carbon/carbon composite substrates will produce more protective coatings.

It is expected that these materials and coatings could find application in areas where short exposures to high-temperature oxidative gases are experienced.



The **Microstructure of a Fused Nickel and Silicon Slurry Coating** on a graphite substrate is shown here. In the cross section (magnified 500 ×) of a silicon and silicon carbide reaction zone (a), a top layer of silicon adheres to the silicon carbide layer. In (b), crystals are prominent on the melted top surface (magnified 3,000 ×) of a slurry coating.

The fused-slurry process offers some flexibility over conventional CVD processing in terms of coating composition and localized application. The process may be especially useful in coating repair.

This work was done by James L. Smialek of **Lewis Research Center.** *For further information, Circle 32 on the TSP Request Card.*

Inquiries concerning rights for the commercial use of this invention should be addressed to the Patent Counsel, Lewis Research Center [see page A5]. Refer to LEW-13951.

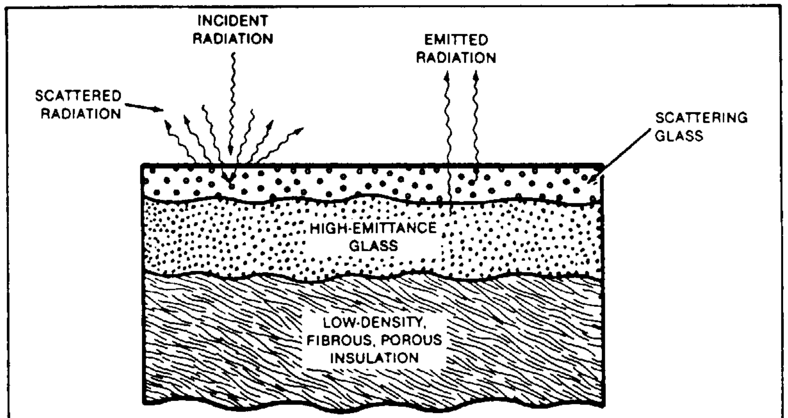
Two-Layer Glass Thermal-Control Coating

Optical properties endure at high temperatures.

NASA Tech Briefs, Winter 1983

Ames Research Center, Moffett Field, California

New all-glass thermal-control coatings withstand repeated exposure to temperatures as high as 2,000° F (1,093° C) with only minimal degradation of optical properties. In contrast, older formulations contain organic binders that degrade above 500° to 600° F (260° to 316° C). The coating helps to prevent excessive solar heating of spacecraft or other objects at moderate operating temperatures, typically room temperature to 150° F (66° C). It also radiates heat away during convective heating at typical spacecraft reentry temperatures around 1,200° F (649° C). The new coating may have industrial uses in solar-energy equipment, some high-temperature chemical processing systems, laboratory equipment, and high-temperature instrumentation.



The Glass Thermal-Control Coating has an outer scattering layer and an inner high-emissivity layer. The absorptivity/emissivity ratio is less than 0.4. The coating can withstand repeated exposure to temperatures in excess of 2,000° F (1,093° C).

As shown in the figure, the coating is applied in two layers to a tile of low-density, fibrous, porous insulation. The inner layer, 0.3 to 0.45 mm thick, has high emissivity for cooling the surface by radiation. It is a frit made from a reactive borosilicate glass, an emittance agent such as silicon tetraboride, and a flux glass.

The outer layer is 0.013 to 0.13 mm thick, depending on optical requirements. It must scatter incoming solar radiation but be as transparent as possible to the predominantly-longer-wavelength thermal radiation emitted by the inner layer. It is a sintered high-silica glass containing particles of 1- to 4-mm size. The particles scatter radiation at wavelengths of 0.35 to 2.3 μm while permitting longer wavelengths to pass. The high silica content also gives this layer a low thermal-expansion coefficient that helps to minimize the generation of residual thermal stresses during fabrication.

The coating layers are prepared as water/alcohol slurries and sprayed in turn onto the fibrous insulating tile. After drying for a few hours at or slightly above room temperature, the coated tile is glazed in a furnace for 1 to 2 hours at 1,092° to 1,231° C. Because of the particular combination of thermal-expansion coefficients of the layers, the residual thermal stress is minimized by inserting the tile when the furnace is at the glazing temperature and rapidly removing the tile at the end of the glazing period.

Coatings of the new type have a ratio of absorptivity to emissivity of $\leq$0.4 at room temperature and a total hemispheric emissivity of 0.8 or greater at 1,200° F (649° C). Even after 100 cycles of heating to 2,300° F (1,260° C), one such coating still had an emissivity of 0.88, only slightly less than the original value of 0.92.

This work was done by David A. Stewart and Howard E. Goldstein of **Ames Research Center** *and Daniel B. Leiser of Stanford University. For further information, Circle 33 on the TSP Request Card.*

This invention has been patented by NASA (U.S. Patent No. 4,381,333). Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Ames Research Center [see page A5]. Refer to ARC-11164.

Thermal-Stress-Free Fasteners for Orthotropic Materials

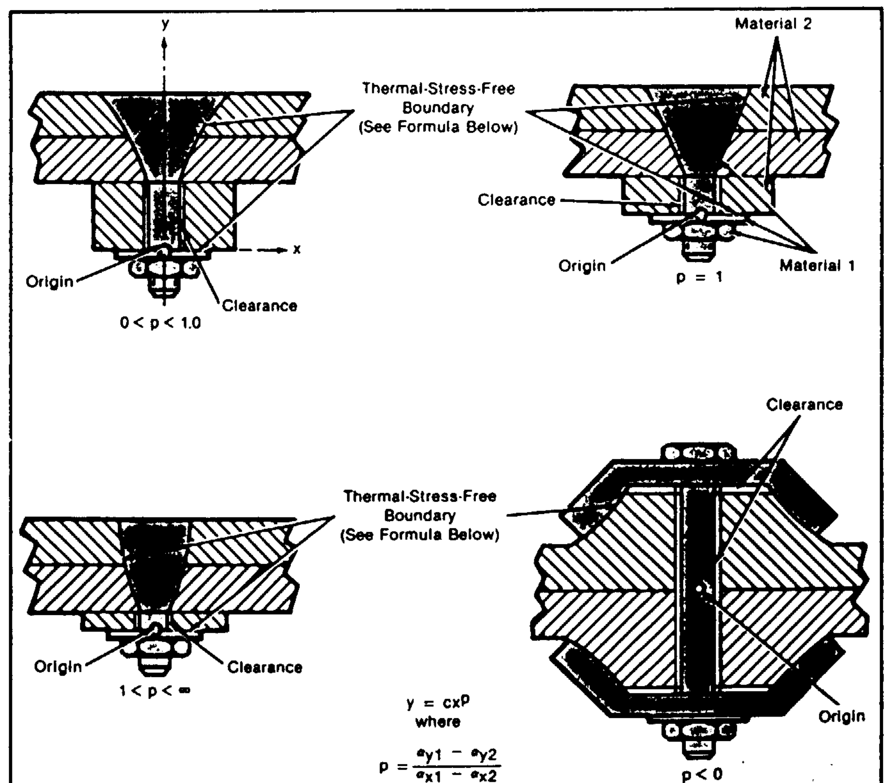
Two-dimensional analysis defines the shapes of interfaces between materials.

NASA Tech Briefs, July/August 1986

Langley Research Center, Hampton, Virginia

A common method of attaching structural components in aerospace systems is by riveting or bolting. If the rivet or bolt is made from a material with a coefficient of thermal expansion substantially different from those of the components being attached, then it is possible to encounter fairly high thermal stresses (or, conversely, a loose joint) if the temperature is increased or decreased significantly. In the case of metal components, this is generally not a problem, because the rivet or bolt is made from the same or a similar alloy. However, with structures made from composite materials, it is difficult to fabricate fasteners from the same material. Also, structural metals are generally isotropic; that is, the thermal-expansion coefficient is the same in all directions. Composite materials, on the other hand, are often anisotropic. Hence, when riveted composite materials heat up or cool down, the rivets can come loose or be subjected to high compressive stresses. Even the state-of-the-art DAZE conical fastener, which by its unique geometry can successfully join isotropic materials with dissimilar rivets, has found only limited application with composites, because most composites are not isotropic.

A theoretical basis for the design of thermal-stress-free fasteners has been de-



Thermal-Stress-Free Joints can be made in a variety of configurations, depending upon the thermal-expansion characteristics of the materials.

veloped. The analysis yields the equation of a two-dimensional thermal-stress-free interface between two materials with orthotropic coefficients of thermal expansion. If both materials have coefficients of thermal expansion that are isotropic in one plane, the two-dimensional analysis can be used to design thermal-stress-free fasteners, made from one material, which are used to join pieces of the other material. The two materials remain in contact as the temperature increases, forming a tight joint without interference and providing effective shear transfer. The simplest general shape is an axisymmetric, curved-side fastener. If the exact shape of the thermal-stress-free fastener is nearly conical, it can be approximated by a conical fastener with the vertex slightly offset.

Several types of thermal-stress-free joints are shown in the figure. The material being joined (material 2) has unequal thermal coefficients of expansion (COE) in the inplane and thickness directions, denoted in the figure as α_{x2} and α_{y2}, respectively, which also differ greatly from the inplane and thickness COE's of the fastener material (material 1), α_{x1} and α_{y1}, respectively. These conditions will cause adverse thermal stresses or looseness in the joint if conventionally shaped fasteners are used. This design technique determines fastener shapes that will maintain a tight thermal-stress-free joint while the joint undergoes a uniform temperature change.

The equation defining the initial interface shape is shown in the figure as a power curve in which the exponent is dependent on the COE's of the fastener (material 1) and of the structural material (material 2). Suitable minimum and maximum fastener diameters are obtained by varying the washer thickness of material 2 and the coefficient of the power term. Varying the COE values results in shapes having either negative curvature ($0 < p < 1$), conical shapes (the shape of the DAZE fastener with $p = 1$), positive curvature ($p > 1$), or a negative slope with negative curvature ($p < 0$). A steel fastener designed to join pieces of graphite survived four thermal cycles to 1,600 °F (870 °C), whereas a conventional fastener failed after only one cycle, demonstrating the validity of this concept.

This work was done by Max L. Blosser and Robert R. McWithey of **Langley Research Center** *and Thomas F. Kearns of the Institute for Defense Analyses. Further information may be found in NASA TP-2226 [N84-13614/NSP], "Theoretical Basis for Design of Thermal-Stress-Free Fasteners."*

Copies may be purchased [prepayment required] from the National Technical Information Service, Springfield, Virginia 22161, Telephone No. (703) 487-4650. Rush orders may be placed for an extra fee by calling (800) 336-4700.

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Langley Research Center *[see page 29]**. Refer to LAR-13325*

Insulation Debond Detection

Load-cell and acoustic responses indicate the bonding condition nondestructively.

NASA Tech Briefs, Summer 1984

Marshall Space Flight Center, Alabama

A nondestructive test for determining whether insulating material is bonded to its substrate or debonded involves measuring sound and load-cell responses to an impact. The technique was used to confirm the bonding of thermal-protection insulation to the Space Shuttle orbiter external tank.

In the new test, the sound and load-cell responses to a controlled impact on a small area of the insulation surface are analyzed for their frequency content. The load-cell measurement is the primary determinant of proper bonding, with the sound measurement used for confirmation. If the frequency response is above a certain threshold, the insulation is properly bonded; below this threshold, the insulation is either debonded or only partially bonded.

Figure 1 shows a version of the bond detector. The calibrated load cell, which measures the impact force, is mounted on the tip of a spring-driven plunger. The plunger is released when the operator presses the trigger. A microphone measures the sound of the impact.

Both signals are processed by a fast-Fourier-transform analyzer. It takes the amplitude-versus-time responses from the load cell and microphone and converts them in about 2 seconds to their respective amplitude-versus-frequency plots.

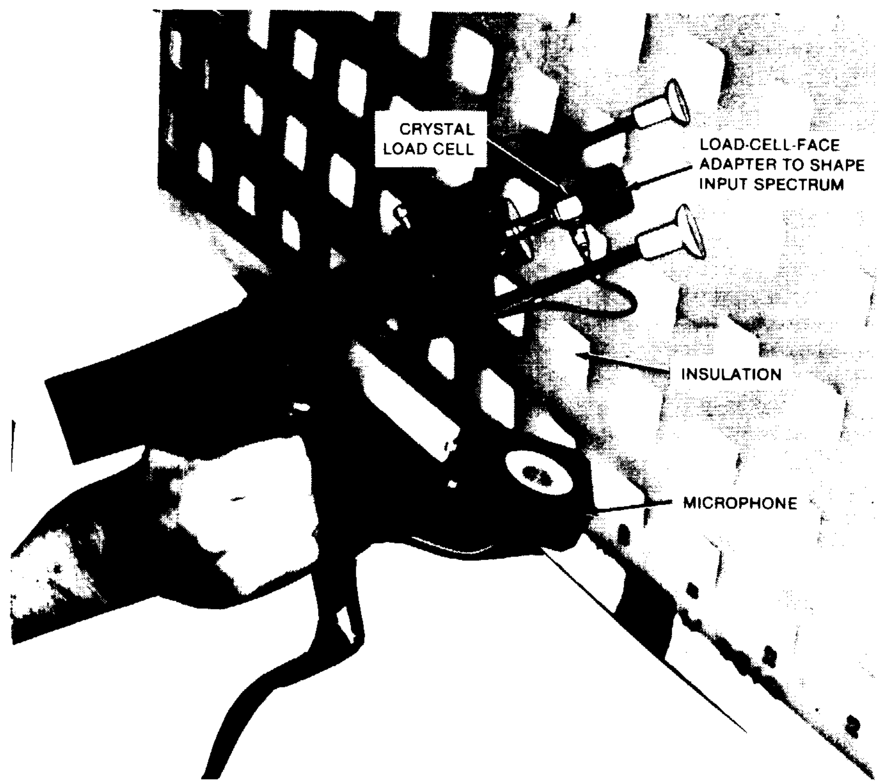


Figure 1. The **Load Cell and Microphone** measure the response of the insulation to a force pulse. The microphone measures sound created during the force period of vibration and during the subsequent period of free vibration. It is positioned at a 45° angle to the point of impact at a distance of 6 to 12 inches (15 to 30 cm) from the point of impact. The greater distance minimizes the structural modes of the substrate and allows the insulation response to be the dominant acoustic energy measured.

For each spectrum, the operator observes the point at which the force spectrum terminates (see Figure 2). In the case of the insulation used on the Space Shuttle, it is 425 Hz. This is the first indication of bonding. To confirm this indication, the operator glances at the sound spectrum in the region of the insulation frequency. For the Space Shuttle insulation this is in the range of 300 to 400 Hz for proper bonding.

This work was done by Garland D. Johnston, Archie D. Coleman, Joseph N. Portwood, Jerry M. Saunders, and Allan J. Porter of **Marshall Space Flight Center.** *For further information, Circle 91 on the TSP Request Card.*

This invention is owned by NASA, and a patent application has been filed. Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to the Patent Counsel, Marshall Space Flight Center [see page A5]. Refer to MFS-25862.

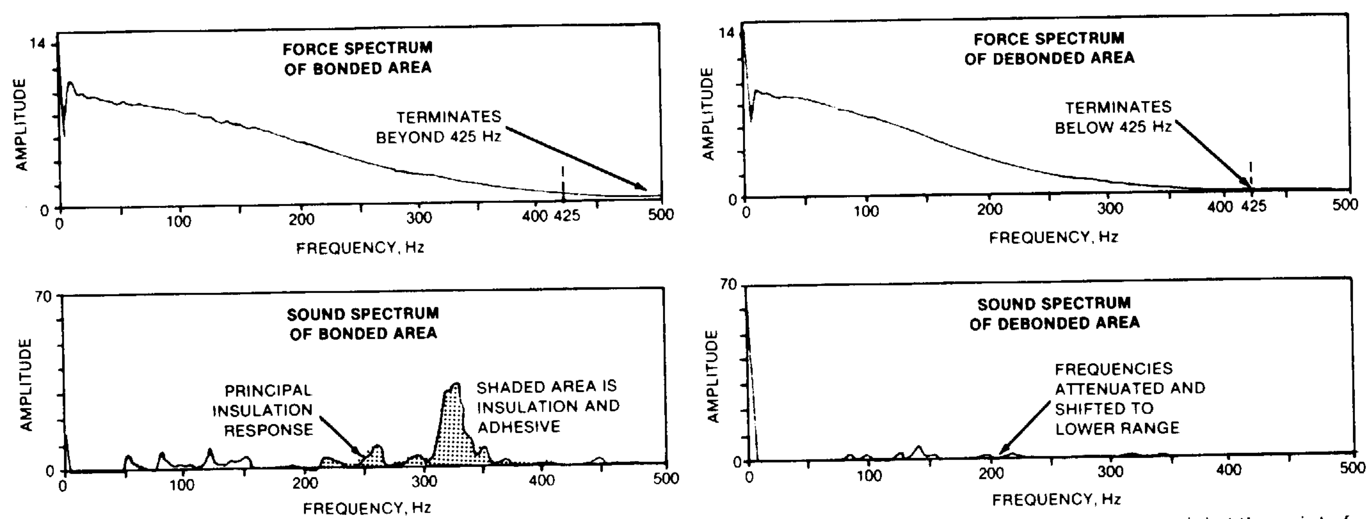


Figure 2. The **Signal Recorded by the Load Cell** is a direct and instantaneous measure of the local stiffness of the material at the point of impact. It is a separate and distinctly different measurement than that sensed by the microphone. The spectrum analysis of the pulse obtained from a debonded point will only show frequencies below 425 Hz because the insulation alone does not have the stiffness to support energy at higher frequencies.

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16. Abstract

The Thermal Protection System (TPS) of the Space Shuttle is one of the most important and successful accomplishments of the National Aeronautics and Space Administration. The TPS introduced and continues to incorporate many of the advances in materials over the past two decades. This review provides a comprehensive, single-volume summary of the TPS, including system design rationales, key design features, and borad descriptions of the subsystems of TPS (E.g., reusable surface insulation, leading edge structural, and penetration subsystems). Details of all elements of TPS development and application are covered--materials properties, manufacturing, modeling, testing, installation, and inspection. Disclosures and inventions are listed and potential commercial application of TPS-related technology is discussed.

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Thermal protection; insulation; materials; ceramics; composites; reentry; gap fillers; Space Shuttle; coatings; tiles; blanket; penetration; carbon; carbon composite; strain isolation; adhesive; radiation; plasma; heat transfer; waterproofing; seals.

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