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# High Speed Flight Vehicle Structures: An Overview

Christopher L. Clay*
*Air Force Research Laboratory, Wright–Patterson Air Force Base, Ohio 45433*

**High-speed vehicle structures have been of interest since the earliest days of aviation. Every since that fateful day at Kittyhawk, North Carolina, the quest for speed has been inseparable from the world of aviation. Some of the advances in structures technology are explored specifically for hypersonic flight vehicle structures during the first century of aviation. In particular, those advances pointing toward the next century hypersonic vehicles are examined.**

## I. Introduction

WITH the renewed interest in high-speed flight, the research and development community is once again focusing on structures technology to survive in these environments. Structures technology for high speed can encompass those structures technologies where the load environment includes extremes of temperatures (both hot and cold) or extreme temperature combined with significant acoustic loading, depending on the application. These structures technologies are not new. The best examples of these types of structures are usually found within the engines of high performance aircraft such as the F-15, rocket launch vehicles for access to space such as the space shuttle, reentry vehicles/capsules such as employed in the Apollo program, or earlier experimental vehicles such as the X-15. More recently, programs such as the X-30 and X-33 (although never flown) have considerably advanced reusable hypersonic structures. The future of high-speed (hypersonic) reusable systems is inextricably linked to our ability to design and build advanced, low-cost, and thermally resistant/durable structures. Thermal protection systems (TPS), hot structures, actively cooled structures, and heat pipe structures are some of the key structural components required in high-speed flight. The focus of this paper will be on fully reusable high-temperature structures for high-speed vehicle applications.

Many attempts have been made to develop high-speed systems with limited success. The most recent example is the NASA X-33 program, but examples may be found well back into the 1960s. Figure 1 shows some of these prior attempts.

The high-speed flight environment leads to much larger areas of airframe structure subjected to combined mechanical, thermal, and acoustic loads than have been typically faced by designers. In most rocket-based systems such as the X-33, the peak temperatures occur out-of-cycle with the peak acoustic loads.

The peak acoustic loads typically occur during lift off and can exceed 168 dB on the airframe. The structures, including the TPS, are usually at or below the ambient temperature. The peak temperatures, however, will occur during reentry. Peak temperatures may range from 2400 up to 2800°F for most concepts under study today.

Airbreathing concepts typically operate with temperatures up to 3000°F with acoustic loads approaching 174 dB on the airframe with much higher acoustic loading possible on the engine structure.

The challenges associated with structures for these environments as just described are many. Competing requirements for cost, durability, and weight are foremost in the mind of the designer. Increased durability can reduce the operations and support cost burden found with these structures, but the added weight could make a hypersonic system nonviable. In the following sections of this paper, we will discuss TPS, hot structures, heat pipe structures, actively cooled structures, and cryogenic tank structures associated with high-speed flight.

## II. TPS[1]

There are many differing types of TPS for reusable vehicle applications. Probably the one most familiar to us is that in use on the U.S. space shuttle (Fig. 2).

The space shuttle TPS employ both fibrous blankets and ceramic tiles bonded to an aluminum primary structure. The tiles make up the bulk of the lower surfaces that experience the highest temperatures on reentry. The lower temperature tiles and the blankets comprise the bulk of the remaining TPS. Although these systems are proven in operation, the cost to operate and maintain these systems is considered to great for many of the planned systems in the future. For the tiles alone, approximately 18,000 work hours are needed to ready them for the next flight.

There are over 20,000 tiles on the Space Shuttle Orbiter. In addition, there is over 4000 ft$^2$ of blanket TPS on the orbiter. Removal and replacement of these TPS are due to numerous factors. Approximately one-third of the remove and replace actions are due to flight-induced damage such as those caused by impact. Another third is due to modifications that have to be implemented on the orbiter. The remainder of the removal and replacement actions is due to access issues or TPS found to be out of tolerance. Future improvements and alternate approaches will be discussed next.

### A. Blanket TPS

One approach to address blanket TPS shortcoming is the development of mechanically attached blanket TPS. The use of mechanically attached TPS concepts instead of bonded TPS concepts will reduce vehicle operations and support cost while reducing the vehicle turnaround time. The ease of removal of the TPS will allow rapid replacement of damaged areas and provide quick access to doors and ease inspection of the structure underneath. In Fig. 3 the titanium frame and corner plates can be seen in the x-ray image. The attachment studs to which the fasteners will attach are also shown. The attachment studs can be adhesively attached to the structure or welded as appropriate.

Vibroacoustic and arcjet testing have been performed to verify both structural integrity of the mechanical attachment concept and blanket, as well as the thermal performance of the system. The vibroacoustic test conditions were a sound pressure level up to 170 dB with a frequency spectrum of 50–500 Hz random. Other improvements to the blanket system include advanced fibers such as Nextel 440 and improved emissivity coatings. A side benefit of the coatings is an improved durability of the surface of the blankets.

Other areas to be addressed include seals between blankets and other TPS concepts (tiles and panels), waterproofing protection, surface toughening, and refinement of the quick-release fastener concept.

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*Deputy Chief, Aeronautical Systems Sector. Senior Member AIAA.

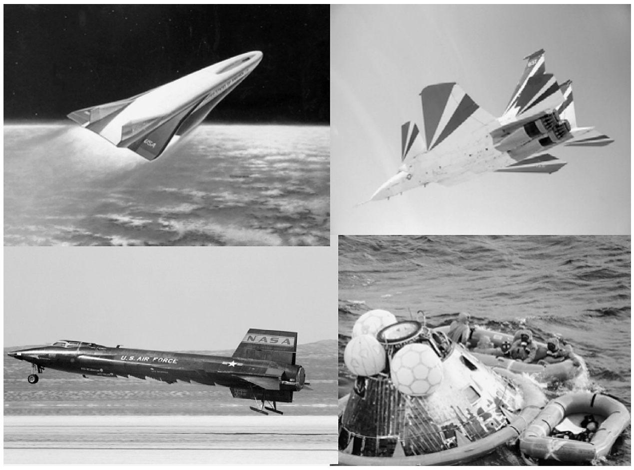


Fig. 1 High-speed vehicles.

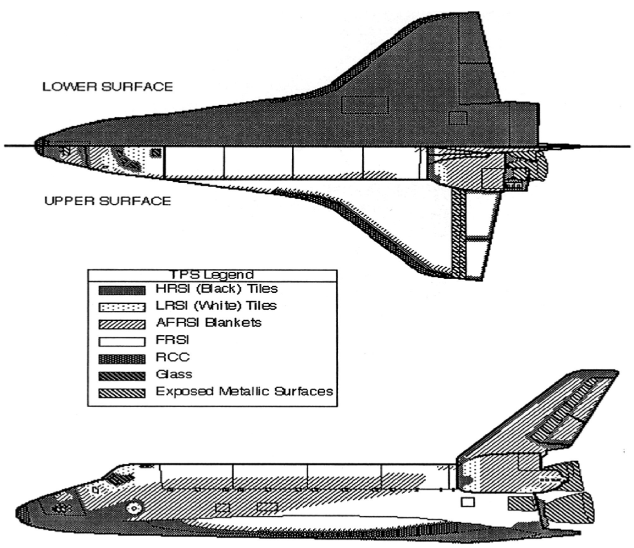


Fig. 2 Space shuttle TPS distribution.

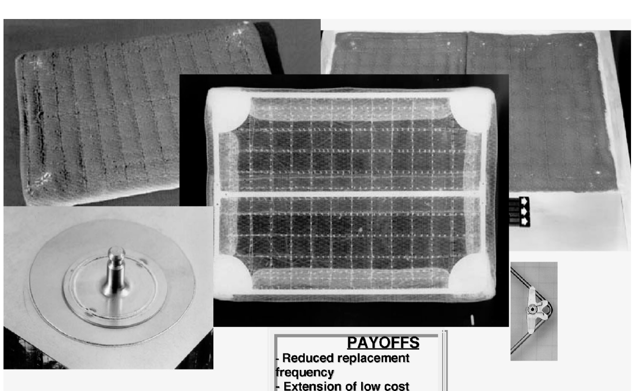


Fig. 3 Mechanically attached blanket TPS.

## B. Tile TPS

The development of tile TPS has largely been along the same lines as that experienced in the drive to develop blanket TPS. Many of the technical issues are the same. Tiles experience excessive damage, require waterproofing, and slow vehicle turnaround times. Many approaches to reduce tile damage include toughening of the exterior surfaces to improve durability. Potential concepts to toughen tiles include ceramic coatings and encasement of the tile in a thin ceramic matrix composite outer layer. The use of a ceramic matrix composite outer layer provides an avenue to the development of a mechanically attached tile. In Fig. 4, a tile with a ceramic matrix composite external surface after combined thermal/vibroacoustic testing is shown. The concept successfully survived its design environment.

Fig. 4 Ceramic matrix composite encased thermal tile.

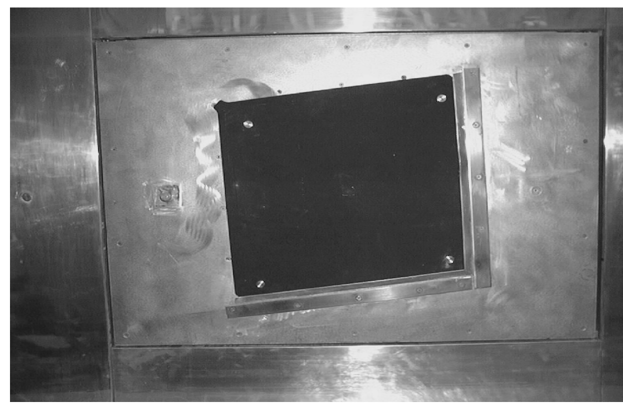

Fig. 5 Metallic TPS concept.

Other benefits of the ceramic matrix composite wrapped tile include the elimination of rewaterproofing requirements. These systems will begin to make their way into advanced systems in the next century.

## C. Standoff TPS

For many advanced systems under consideration such as the military Space Operations Vehicle, even greater durability requirements with much lower operations cost will drive TPS systems. For many of these systems, standoff TPS with either metallic or ceramic matrix composite designs may be needed. The metallic standoff TPS design (Fig. 5) is much more advanced than the ceramic matrix composite designs. These metallic designs have been under development since the early 1980s.

For lower temperature regimes (below than 1500°F) titanium, because of its lightweight, is preferred. As the temperature increases, superalloys and other high-temperature metallics may be considered. Typical construction consists of a high-temperature honeycomb core with high-temperature brazed on facesheets. Multiwall construction has also been investigated primarily with titanium materials.

Although not as mature as tile and blanket TPS concepts, standoff concepts in both metallic and ceramic composite are receiving a growing investment in their development. Several key issues remain before the use of standoff TPS in high-speed systems is practical. The metallics are limited in size due to their thermal growth during heating. This growth is usually managed through keeping the designs small (18–20 in. typical). This increased part count in panels and fasteners may lead to indirect cost and maintenance burdens than are higher than desirable. The coatings needed to ensure high emissivity, noncatalyticity, and oxidation protection could also affect turnaround times. In their favor, however, is the lightweight that can be attained.

The ceramic matrix composite TPS is usually significantly heavier than its metallic, blanket, and tile counterparts. Its advantage is that much larger part sizes (two to three times) than tiles or metallics can be utilized. Also, for the highest temperatures on the vehicle, it may be the only way to ensure adequate life without resorting to active cooling. Issues with the ceramic composites include the immaturity of the basic materials, especially in the fiber–matrix interface, the inability to assess life adequately, and the durability of oxidation protection coatings.

## III. Hot Structures[1]

Generally, for a given design application, a hot structure must be evaluated along with the competing design approaches of thermally protected structure and actively cooled structure. The major deficiency of actively an cooled structure is the weight, design complexity, and, hence, cost, associated with the coolant arrangements, as well as the ensuing reliability issues associated with proper operation of the system. The thermally protected structure is a more mature technology and is flying today. For a thermally protected structure, however, the thermal barrier is usually parasitic, resulting in additional weight beyond that required by the primary load-carrying structure alone. Also, a thermally protected structure can have extensive inspection and maintenance actions required to verify integrity and continued safe operation of the thermal shield as discussed earlier. The major attribute of a hot structure is that generally it is the minimum weight design possible for an intended application. If the hot structure can be designed to be reliable and robust without sacrificing its weight advantage, then a superior solution will have been attained. Potential hot structure solutions will be discussed next.

### A. Carbon–Carbon Hot Structures

Because of its high specific properties at elevated temperature, carbon–carbon (C–C) is an attractive material for high-speed application. One particular area where C–C is usually a candidate is in the control structures area. Here, a hot structure such as C–C is particularly appealing for aerodynamic surfaces because it can often more easily meet the stringent volume restrictions imposed by thin surfaces.

One such C–C design is shown in Fig. 6. This design provides a proper balance between minimize weight, low risk, and ease of fabrication.

To evaluate the design, a test article was fabricated from two-dimensional advanced C–C (ACC-4). The major detail parts were two rib-stiffened, upper and lower panels, a C–C torque tube, 10 C–C attachment rings, a C–C closeout channel, and a Rene' 41 actuator lug fitting. The rib stiffened panel details were laid up and processed together, eliminating the need for mechanical joining at the rib/skin interfaces. The two panel halves were then joined by secondary bonding with the remaining C–C processing and coating cycles performed on the assembly as a whole. Rene' 41 fasteners and cleats were used to attach the actuator lug fittings to the C–C torque tube. Overall dimensions of the test article were 38.7 in. span, 56 in. chord, and a depth tapering from 14 in. at the forward edge to 6.2 in. at the aft edge.

As fabricated, the ACC-4 test article represented a segment of a full-scale elevon control surface and was designed to sustain a maximum surface temperature of approximately 3000°F.

### B. Three-Dimensional C–C Wing Box

Traditional two-dimensional C–C suffers from weak interlaminar material properties. One way of overcoming this weakness is three-dimensional fiber architecture. In Fig. 7, an example of C–C hot structure, a three-dimensional angle interlock weave was employed in the design, fabrication, and testing of a 3-spar, wing torque box structure. In the three-dimensional angle interlock process, most of the fibers remain in the plane of the material. A smaller percentage of the fibers are woven through the thickness to increase the interlaminar tension and shear properties.

A side benefit of this fiber architecture is the ability to weave integrally certain design features of the structure as a unitized whole. In this particular example, the spar caps and spar webs were integrally woven. Additionally, the skins, in the area of the

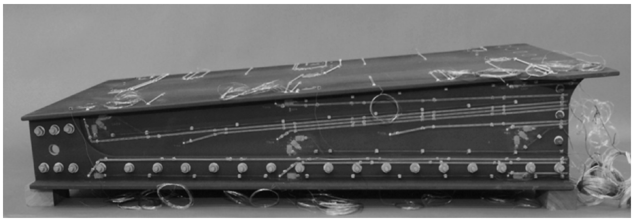

Fig. 7 Three-dimensional C–C structure (pretest).

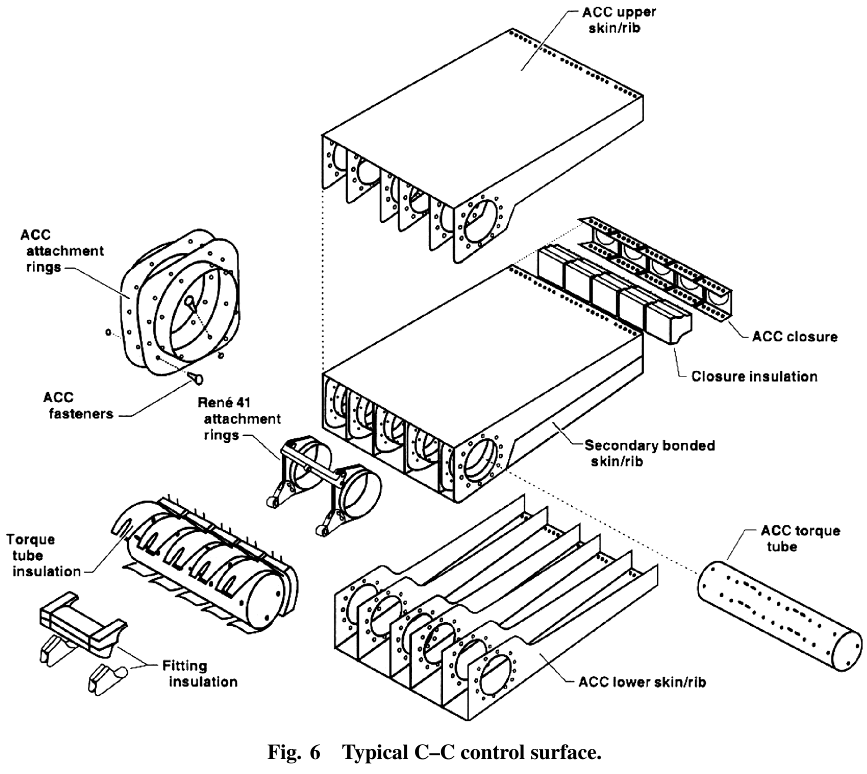


Fig. 6 Typical C–C control surface.

spar caps, were woven as a bilayer preform. This allowed one layer to be cut and the spar cap inserted into the intervening space, resulting in a coprocessed, fastenerless skin-to-spar joint. For the closure ribs, traditional two-dimensional laminated C–C composite was used. To prevent oxidation, a multilayer, silicon-carbide-based coating was applied to all surfaces. Elimination of mechanical fasteners through the skins allowed the outer surfaces to present an uninterrupted, coating interface to the erosive and oxidative flow.

There were 33 simulated mission cycles of heating and mechanical loads successfully applied to the box. Also a final room-temperature test to failure was performed, with the box successfully exceeding its 150% design limit load objective.

### C. Titanium Matrix Composite Hot Structure

For lower temperature applications where C–C is not required, titanium matrix composites may be considered. Titanium matrix composites (TMC) were chosen as the principal material system for the X-30 fuselage and empennage structures. Maximum use temperature of the material was to be 1500°F. Use of TMC for the X-30 was shown to offer a weight savings of 50% vs state-of-the-art superalloys and high-temperature steels, a weight advantage that was of critical importance to the vehicle's intended single-stage-to-orbit operation.

The specific TMC material system selected for the National Aerospace Plane (NASP) program used coated silicon carbide fibers in a Beta 21S titanium matrix.[2] The matrix alloy, Ti–15Mo–2.8Nb–3Al–0.2Si, was developed by Timet specifically to meet the oxidation resistance requirements of NASP. The fibers selected included Textron Specialty Material's SCS-6 silicon carbide fiber, chosen as the baseline fiber, and SCS-9. The SCS-9 fiber is essentially the same as SCS-6 except for a smaller diameter. The SCS-9 fiber has lower mechanical properties but its density is lower also. Both fibers were able to meet the acceptance criteria of 50 h exposure at 1500°F without unacceptable degradation of the fiber protective coating. This exposure time was greater than what would be encountered for the expected X-30 lifetime.

Titanium was feasible as a matrix material because of its limited exposure at the maximum temperatures and also because the maximum temperatures were to be reached at a high altitude where the availability of oxygen and hydrogen would be reduced. Thermomechanical fatigue (TMF) testing was performed which demonstrated the ability of the SCS-6/Beta 21S baseline material system to survive more than two NASP lifetimes. Testing also demonstrated the Beta 21S matrix material was sufficiently oxidatively resistant to survive the environment without a protective coating. Exposure to casual hydrogen in TMF testing at the low anticipated hydrogen pressures additionally demonstrated the material would not be unacceptably degraded due to hydrogen embrittlement.

The processing and manufacturing operations necessary to produce consistently high-quality material and built-up parts using the TMC material was also developed. Resistance spot welding, brazing, and weld brazing joining techniques were successfully developed to enable the manufacture of minimum weight X-30 fuselage hat-stiffened skin designs. Mechanical fastening techniques were also developed to enable the joining of structural parts into an assembly. The fastener technology utilized the high-temperature materials Haynes 230 and Ti-1100. Typical TMC parts are shown in Fig. 8.

During the fabrication development, various structural details were produced, which demonstrated the ability to fabricate stiffened skin panels. On a larger scale, several shear/compression panels were built and tested to validate the stiffened skin design approach. Fabrication techniques were also developed for a TMC-reinforced sandwich panel taking advantage of the ability to diffusion bond a superplastically formed titanium core to a TMC facesheet. TMC components of multifeet dimensions were also successfully fabricated and tested demonstrating viability of TMC technology for larger scale structures. These components were a splice subcomponent, a stabilator torque box, a full-scale panel assembly, and an integrated fuselage/tank article.

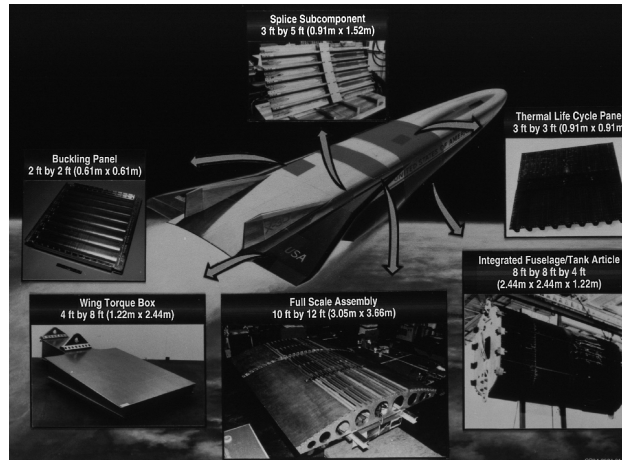


**Fig. 8 X-30 TMC fabricated parts.**

Additional, larger, and more complex structural components need to be fabricated and tested, but since the end of the X-30, work has been minimal. Additional data on fasteners and joints are required, as well as durability and damage tolerance investigations. In addition, manufacturing cost reductions would make these concepts more viable.

### D. Heat-Pipe-Cooled Structures[2]

Leading edges of high-speed vehicles experience extreme thermal environments during flight. The materials and cooling techniques for vehicle leading edges must be chosen carefully to minimize cost and maximize performance. These high-heat flux regions can be cooled passively (radiation), actively, or with heat pipes, depending on the magnitude of the heat flux. Passively cooled structures have by far the lowest cost, complexity, and weight, but also have the lowest capability to withstand extreme local heating. Once the passive heat-flux limit is surpassed, hypersonic vehicles must utilize either heat-pipe or active cooling. Active cooling is well developed and has the highest heat-flux capability, and actively cooled structures generally operate at the lowest temperatures for a given heat flux. However, active cooling is also costly, complex, heavy, and require numerous auxiliary systems.

A viable solution to thermal–structural problems associated with stagnation regions of high-speed vehicles may be obtained with heat pipes. Heat-pipe-cooled leading structures combine features of both passive and active cooling and generally have a heat-flux capability, cost, complexity, and weight between passive and active cooling. Like a passive leading edge, the maintenance required for a heat-pipe-cooled leading edge is negligible. Because of the high localized heating on a leading edge, and the ability of a heat pipe to redistribute the heat passively, leading edges are an ideal application for heat pipes. The use of heat pipes results in a lighter weight, lower cost, and lower maintenance leading edge than an actively cooled leading edge.

Heat-pipe-cooled structures have been developing since the early 1970s. Efforts have evolved from blunt, all metallic heat-pipe-cooled leading edges for the Space Shuttle Orbiter to sharp leading-edge heat pipes embedded in refractory-composite materials for future high-speed systems. Heat-pipe-cooled leading edges were estimated to provide a 50% weight savings over active cooling for the X-30 program. The use of heat-pipe-cooled leading edges and nose cap on the Space Shuttle Orbiter could extend the range of the orbiter up to 400–500 miles in the event of an aborted mission by allowing the vehicle to fly a lower angle of attack. The current C–C leading-edge maximum temperature limits the angle of attack that the orbiter can fly.

An additional constraint on future vehicle designs is the requirement to service vehicles rapidly for additional mission activities. Turnaround times of less than 8 h are desired to minimize fleet size and maintain a high operational tempo during wartime. This service requirement essentially eliminates ablative approaches to achieving

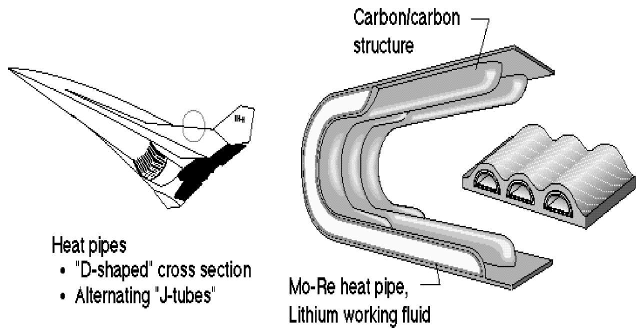


**Fig. 9 Schematic refractory composite/heat-pipe leading edge.**

sharp leading edges due to the refurbishment time. In addition, ablative concepts may well result in some blunting of the leading edges and a reduction of $L/D$.

Many different design concepts have been explored. A half-scale heat-pipe-cooled leading-edge test article was fabricated from Hastelloy-X for a shuttle-type leading-edge requirement. Both aerothermal and radiant testing was conducted. Haynes 188 leading-edge-shaped heat pipes were fabricated and tested to heat fluxes of 220 Btu/ft$^2 \cdot$ s (250 W/cm$^2$). A niobium vapor chamber heat pipe was fabricated for evaluation as a nose cap to X-30 conditions.

Each of these events, as well as other developments in this area, has paved the way for heat pipes as an important concept for future high-speed systems. Embedding individual heat pipes in a refractory-composite structure such as C–C or C–SiC are of particular interest. The high-temperature capability of these materials may provide some additional measure of fail safety in case a heat pipe fails to operate properly. The maximum operating temperature capability of coated refractory-composite materials for the primary structure of the leading edge is high (~3000°F) relative to refractory metals, which are typically limited to approximately 2400°F. The higher operating temperature increases the radiation heat-rejection efficiency of the heat-pipe-cooled leading edge to permit reductions in the mass of the leading edge for a given leading-edge radius. In addition, the higher operating temperature increases the total heat load that can be accommodated passively by the heat pipe that is, no forced convective cooling required. For many space access applications, the high operating temperatures help eliminate the need for active cooling and the associated mass and operability penalties.

Refractory-metal heat pipes embedded in a refractory-composite structure offer an integrated thermal/structural design solution. As shown in Fig. 9, the heat pipes can be oriented normal to the leading edge and have a D-shaped cross section, with the flat part of the D forming the wing-leading-edge outer surface. The flat external surface optimizes heat transfer from the external surface to the heat pipe and also allows for other heat-pipe orientations if desired.

The leading edge comprises J-tube heat pipes, with a J-tube heat pipe being a heat pipe with a long leg on one side of the nose region and a short leg on the other side of the nose region, as shown in Fig. 11. An alternating J-tube configuration can be utilized to minimize heat-pipe spacing in the nose region where heating is the highest and to provide greater heat-pipe spacing on the upper and lower surfaces where heating is lower, which also serves to reduce mass. The refractory composite structure sustains most of the mechanical structural loads and also offers ablative protection in the event of a heat-pipe failure.

The technology over the last 30 years has progressed from individual metallic heat pipes to metallic heat pipes embedded in a composite structure. The state-of-the-art refractory-metal/refractory-composite heat-pipe-cooled leading edge builds on prior work to offer an integrated thermal–structural design solution for future high-speed vehicles.

### E. Actively Cooled Structures[2]

In the past 15 years or so, a great deal of interest in low-cost access to space and rapid-strike/global-reach vehicles has developed. This interest has manifested itself in a number of programs ranging from component technology to propulsion and vehicle programs. Among these programs, several have included major activities focused on the development of structures for applications in the most extreme thermal environments of the vehicle. Foremost among the structural concepts developed for these high-heat-flux environments have been actively cooled structures. Designs in numerous material systems have been developed and tested to address the thermal challenges associated with internal ramjet and scramjet engine surfaces, engine external inlet and nozzle surfaces, and wing and engine leading edges. Some of the most recent programs to include development of actively cooled structures have been the NASP program, the NASA/Lockheed Martin Venture Star program, and the U.S. Air Force hypersonic scramjet engine technology program. Within these programs, advancements in materials, coatings, and designs have been made that have significantly improved the state of the art in reusable cooled structures for high-speed vehicle applications.

**Table 1 X-30 design conditions**

| Condition | Value |
|---|---|
| Heat flux (face) | >2500 Btu/ft$^2 \cdot$ s |
| Heat flux (gap) | ~60% of face |
| Acoustic pressure | >176 dB |
| Coolant temperature | Cryogenic to 1100°R |
| Coolant pressure | 4000 psi |

Design requirements and conditions vary widely depending on the application. The X-30 propulsion system encompassed probably the most demanding requirements and conditions of any past or present system design. The engine heat-flux levels reached, and in local areas exceeded, values as high as 2500 Btu/ft$^2 \cdot$ s, whereas other environmental conditions also pushed the extremes of materials and design capabilities (Table 1). In comparison, heat flux levels for the VentureStar engine reach levels of only 1200 Btu/ft$^2 \cdot$ s. In addition to the thermal and mechanical environment to which these designs are exposed, design life and fuel compatibility are major considerations in an individual design. For those applications involving hydrogen coolant, compatibility of the materials with the coolant must be considered. Many common materials that would be thought of as initial candidates for a design quickly prove to be too negatively effected by hydrogen exposure. For these materials, hydrogen exposure results in loss of ductility, strength, or one or more of a whole range of detrimental effects. As a result, the list of materials that may be used in a reusable application is severely limited.

In other designs, where alternate coolants such as hydrocarbon fuels are used, other issues become important. For instance, hydrocarbon-cooled designs for a number of reasons require that the fuel be cracked within the actively cooled panels. These high-temperatures may lead to coking of the coolant passages, a condition that can be made worse depending on ones material selection.

To date, the main list of materials considered for hydrogen- or JP-cooled structures includes those listed in Table 2.

Previous programs explored many of these advanced materials in support of lightweight, durable heat exchangers for high-speed application. In broad categories, these material systems included metallics, composites, microcomposites, and refractory composites. Titanium alloys and copper microcomposites were dropped due to the adverse affect of hydrogen on material properties. Beryllium was dropped after the catastrophic failure of a 20 × 20 in. test panel emphasized the need for ductile heat exchanger materials. In contrast, materials such as C–C and C–SiC, Mo–Re, Lockalloy, and copper graphite composite demonstrate significant promise for application in actively cooled structures and are discussed further next.

### F. Refractory/Ceramic Composites

Several activities have investigated refractory composite materials for use in actively cooled panels. These included C–C, C–SiC, and SiC–C. Of these, only C–C and C–SiC were extensively studied. The temperature limit of SiC–C at 2500°F was felt to be too limiting when compared to the nearly 3000°F limits of the other

Table 2 Material capabilities

| Material | Material temperature range | Coolant temperature range | Heat flux | Hydrogen compatibility |
|---|---|---|---|---|
| Aluminum | Low | Low | Low | Yes |
| Beryllium | Low | Low | Low | Yes |
| Lockalloy | Low | Low | Low | Yes |
| Titanium | Low | Low | Low | No |
| Cu | Low | Low | High | Yes |
| Cu Microcomposites | Low | Low | High | No |
| Cu/GR | Low | Low | High | Yes |
| H188 | Medium | Medium | Medium | Yes |
| IN625 | Medium | Medium | Medium | ? |
| IN718 | Medium | Medium | Medium | No |
| IN909 | Medium | Medium | Medium | Yes |
| MoRe | High | High | High | Yes |
| C–C | High | High | Medium | Yes |
| C–SiC | High | High | Medium | Yes |

materials. Work focused on extensive development of the C–SiC material and heat exchanger design. Two- and three-dimensional composite panels were procured from several vendors and subjected to mechanical tests and inspections. Of these designs, a composite using a T300 fiber/three-dimensional angle interlock material with a chemical vapor infiltration of a SiC matrix was found to have excellent mechanical properties. Warp tensile and compressive ultimate strengths were 38 and 54 ksi, respectively.

In parallel with the work accomplished for the composite, the development of embedded coolant tubes was also required. Materials including tungsten, tungsten rhenium, molybdenum, and molybdenum rhenium alloys were studied. Work on tungsten tubes was short lived because of the brittleness of the material. Tungsten rhenium tubes were also dropped from the program when the layered approach (alternating layers of deposited tungsten and rhenium that were then annealed to consolidate the structures) was found to be too costly to continue and the deposition process resulted in large nonuniform grains. Molybdenum tubing was dropped after tests of material exposed to 2000°F temperatures exhibited a 20% reduction in axial strength. Molybdenum rhenium was the only material system to complete program evaluations. Tests of several alloys from several suppliers were conducted to identify the best alloy system with the Mo–47.5Re material being selected.

The final step in these concept developments was the validation of thermal performance of these designs. During various validation tests a proof of concept panel fabricated with C–SiC and Mo–Re tubing was exposed to heat fluxes up to 100 Btu/ft$^2$ · s using cold water as a coolant. Results from the tests indicated good heat transfer between the composite and the coolant.

Another approach using Haynes-188 tubing slip fit into a refractory or ceramic composite has been developed as well. The purpose of this conceptual approach was to attempt to decouple the thermal strains developed between the dissimilar materials of the tubes and composite on heating. A total of 11 of these articles were then tested under a wide range of coolant and heat flux conditions. A total of 58 tests in 2 test series were run on the 11 specimens. In these tests coolant pressure mass flow rate and temperature was varied to study the effects of these parameters on heat-exchanger performance. Inlet pressure was adjusted from a low of 1900 to a high of 3000 psi while mass flow rates were from 0.01 to 0.03 lbm/s and coolant temperatures were from −200 to 700°F. Incident radiation heat flux in these tests ranged from a low of 52 to a high of 894 Btu/ft$^2$ · s, 2.8 times the design requirement.

### G. Copper (NARloy-Z) Development

NARloy-Z is probably the best-characterized and best-tested material to be used as a flat actively cooled panel. The material has a long history of use by Rocketdyne in rocket engine applications. With the material so well characterized for previous applications, most of the recent efforts focused on designs and manufacturing processes for flat non-integral panels and on thermomechanical validation testing. Its high conductivity, nearly that of pure copper, made it one of the best possible materials for the high-heat-flux environment of the airbreathing high-speed vehicle combustors where heat fluxes may exceed 2000 Btu/ft$^2$ · s.

Designs include several key features. To account for gap heating between individual panels, designs may incorporate a second channel along the edge. This channel performs several functions, the first and primary function is to control edge temperatures that otherwise would be hundreds of degrees higher than those found away from the edge of the panel. The second function was to reduce the heat load on the edges channels because without such help it was possible that severe flow maldistribution would develop between the edge and center channels. In such an event, edge temperatures would have risen further with respect to panel center channels possibly setting up a self-destructive cycle of increasing edge temperatures and flow maldistribution. A second feature of the designs is the use of an oxidation protection coating. To get the highest performing designs requires very high panel surface temperature. To avoid rapid oxidation and erosion of the panel surface at these temperatures, designs incorporating a Cu–Cr coating several mils thick were found to be useful. This coating functioned by producing a well-attached Cr-rich surface layer when exposed to high temperatures. This surface was shown in several subsequent test programs to be very durable with little loss in thickness over many cycles and hours of testing. The final unique feature of the design was the attachment method. Because of the high thermal expansion of NARloy-Z and the high backside temperatures of the panel, an integral structure using lightweight composites was not possible. Such a design would have suffered from high shear stresses, high composite temperatures, and as a result very short life. To alleviate these problems, nonintegral designs are necessary.

### H. Haynes 188 Development

Another material studied extensively for actively cooled engine panel applications was Haynes 188 (H188). This material was selected primarily for its high-temperature capability and its hydrogen compatibility. Designs incorporating this material focused on engine conditions with heat-flux levels of approximately 1000 Btu/ft$^2$ · s and coolant temperatures up to 1500°R. The primary areas of development activity for the H188 designs have been in fabrication, design analysis, and verification. Thermal behavior for or a typical H188 design is shown in Fig. 10.

Hydrogen coolants testing of H188-type designs have been encouraging. Over 160 thermal cycles without failure at heat fluxes exceeding 1000 Btu/ft$^2$ · s and coolant outlet temperatures averaging 750 to 800°F have been demonstrated. More work needs to be done in fabricating these thin-walled, narrow-channel, heat-exchanger structures, evaluating and controlling gap heating, and exploring their dynamic oxidation/erosion characteristics.

### I. Molybdenum Rhenium

Mo–47.5%Re has been found to be a potential material of choice for very high-temperature engine actively cooled panels within high-speed vehicles. Its placement at the end of an engine coolant

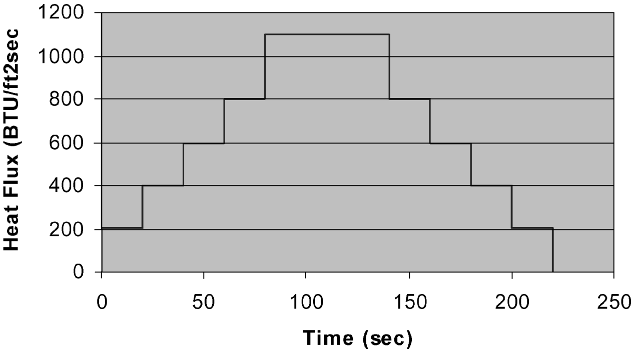


Fig. 10 H188 thermal profile (typical).

circuit provides the coolant/fuel temperature boost required to get peak combustion efficiency. However, to make this material practical in this application requires significant fabrication and coating technology.

The bulk of the fabrication effort for Mo–Re revolved around machining methods capable of economically producing coolant channels in large panels. Conventional milling processes were inadequate given the toughness of Mo–Re, and therefore, alternate methods such as high-speed, laser, and electrochemical machining processes have been explored. A typical machining/fabrication concept is shown in Fig. 11.

The electrochemical process was used to produce the platelet design. One drawback to this method was the cusping present in the bonded assembly. This cusping was the result of the more extensive loss of material at the surface of a platelet as compared to that experienced in the midplane of the platelet. This cusping is a

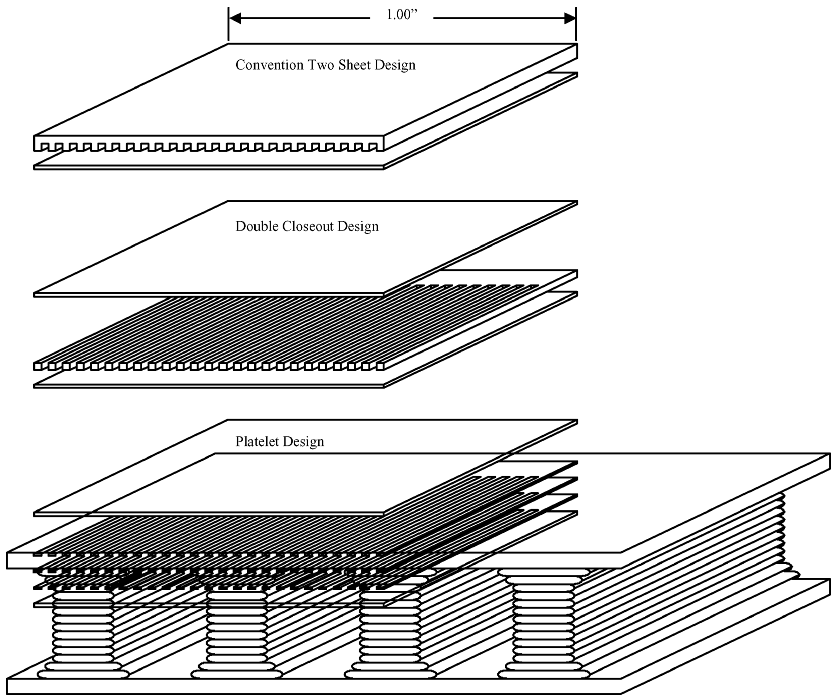


**Fig. 11 Mo–Re heat exchanger fabrication concept.**

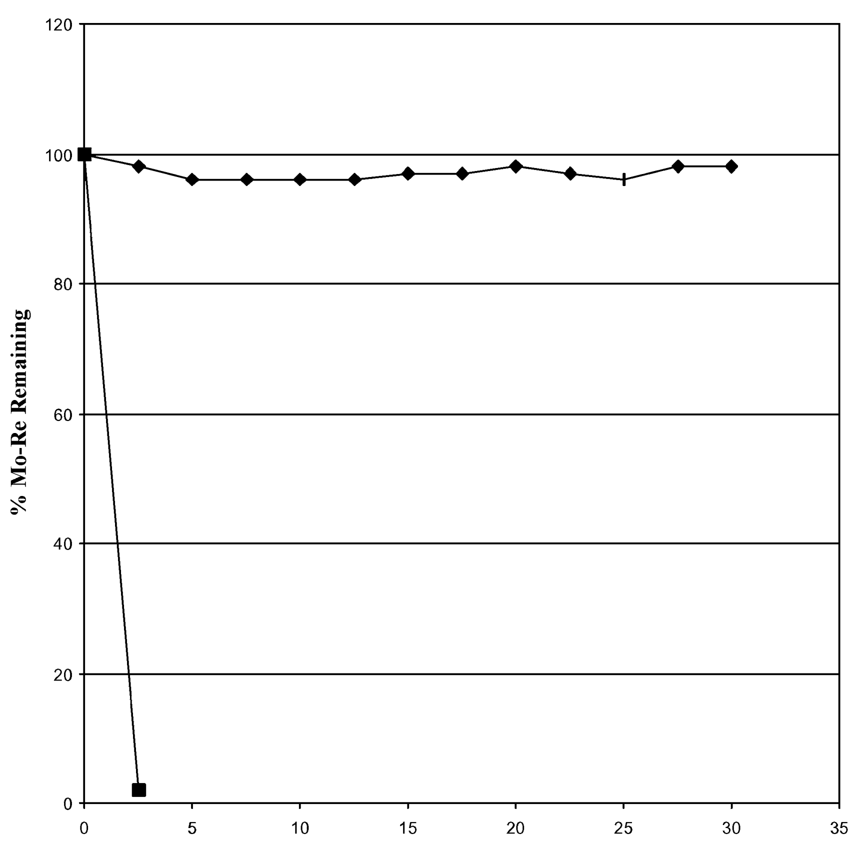


**Fig. 12 R512E oxidation evaluation at 2300°F: ◆, Hitemco R512E coating and ■, uncoated.**

Table 3 Metallic coating deposition methods

| Clad bonding | Aqueous plating |
|---|---|
| Aqueous brush plating | Magnetron ion deposition |
| Cathodic arc deposition | Ion assisted deposition |
| Ram bonding | Vacuum plasma spray |
| Low temperature arc vapor deposition | Molten salt bath |
| CVD fluidized bed | |

concern given that it may lead to premature cracking. As a result, laser machining was explored. The laser process was used to manufacture the double closeout design. In this process, the laser was used to mill the long narrow slots in the thick center sheet after which thin closeout sheets were diffusion bonded to either side to form the channels of the heat exchange.

Coating development for Mo–Re has probably been the most aggressively pursued technology effort for this concept. Unlike many materials that form a somewhat protective scale as they oxidize, Mo–Re alloys do not. Instead, at around 1250°F, Mo–Re oxides are gaseous and, therefore, provide no protective scale that might slow the rate of material loss. As a result, the requirement to use the alloy at temperatures approaching 2500°F requires the development of a dependable, defect-free coating that can be repeatably produced. To that end, a number of protective materials and a number of methods for applying them have been considered. The primary materials considered in the effort fell into two categories, glassy coatings and metallic coatings. The glassy coating used with success was R512E. One example of the effectiveness of this coating is seen in Fig. 12. In other tests conducted at a different facility, similar results were obtained providing additional confidence in the results.

The metallic coatings investigated included platinum, rhodium, and iridium. Experiments with these materials explored the benefits of these materials applied individually and in alternating layers. When applied in alternating layers, platinum was normally the primary (outer) protective material with undercoatings of rhodium and/or iridium. To apply these materials many deposition methods such as chemical vapor deposition (CVD) were tried as shown in Table 3. A layered metallic coating system using platinum as the primary protective material was found to be effective. An example of the effectiveness of platinum was demonstrated in early oxidation trials on hot isostatically pressed (HIP) clad Mo–Re, where it demonstrated reasonably good oxidation protection. The sample tested was a thin sheet of Mo–Re HIP clad inside a 0.005-in.-thick envelope of platinum. After 5 h in an oxyacetylene torch flame at 2350°F, the sample experienced a 0.4% weight change. In a complementary test, the same specimen type demonstrated a 0.4% weight change in 30 h of static conditions at 2350°F.

In parallel with fabrication and coating development, development of panel designs included a subelement design ($1 \times 4$ in.) and a full size ($20 \times 20$ in.) flightweight design. To support the out-of-plane pressure loads, the panel relied on pins and an egg crate style backup structure for attachment and to distribute loads to the primary structure.

A viable Mo–47.5Re actively cooled panel appears promising. Many of the technical challenges associated with manufacturing and oxidation coatings have been addressed at the specimen level. What remains is the validation of the design at the subelement and full-scale panel levels in test facilities that can replicate the harsh environments of flight.

## IV. Copper-Based Actively Cooled Structures

Copper graphite designs were explored as a means to achieve panels with the high conductivity of copper while avoiding the high weight associated with a monolithic copper alloy panel. Eventually, issues of compatibility between the graphite fibers and the copper matrix were overcome, and a lightweight panel design was developed and produced in $1 \times 4$ in. specimens for testing. However, these panels were never tested.

Copper microcomposites materials were also investigated. This work sought to develop copper materials strengthened through the addition of such high-temperature materials as niobium, tantalum, chromium, molybdenum, and rhenium. Processing of these materials yielded copper sheet materials with ribbons of one or more of these materials throughout the sheet intended to provide superior high-temperature strength. Of those material systems investigated, several including the Cu–Nb system showed some promise, although none were ever pursued far enough to result in panel tests.

This is an area where additional investigation for the future may have high payoff. Significant weight reductions and thermal performance gains may be realized.

## V. Summary

High-speed flight will only be practical when the challenges of materials and structures have been met and overcome. There are many different high-speed vehicle concepts that have been investigated or are under consideration today. Whatever the concept, structures technology figures prominently in the likelihood for success. There are many more concepts that this paper could have explored: only the surface has been touched. The purpose of this paper was to elucidate the issues and progress made in structures for high-speed flight in our first century of aviation.

## Acknowledgment

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