

ADVANCES IN HOT-STRUCTURE DEVELOPMENT

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ABSTRACT

The National Aeronautics and Space Administration has actively participated in the development of hot structures technology for application to hypersonic flight systems. Hot structures have been developed for vehicles including the X-43A, X-37, and the Space Shuttle. These trans-atmospheric and atmospheric entry flight systems that incorporate hot-structures technology are lighter weight and require less maintenance than those that incorporate parasitic, thermal-protection materials that attach to warm or cool substructure. The development of hot structures requires a thorough understanding of material performance in an extreme environment, boundary conditions and load interactions, structural joint performance, and thermal and mechanical performance of integrated structural systems that operate at temperatures ranging from 1500°C to 3000°C, depending on the application. This paper will present recent advances in the development of hot structures, including development of environmentally durable, high temperature leading edges and control surfaces, integrated thermal protection systems, and repair technologies.

The X-43A Mach-10 vehicle utilized carbon/carbon (C/C) leading edges on the nose, horizontal control surface, and vertical tail. The nose and vertical and horizontal tail leading edges were fabricated out of a 3:1 biased, high thermal conductivity C/C. The leading edges were coated with a three-layer coating comprised of a SiC conversion of the C/C, followed by a CVD layer of SiC, followed by a thin CVD layer of HfC.

Work has also been performed on the development of an integrated structure and was focused on both hot and warm (insulated) structures and integrated fuselage/tank/TPS systems. The objective was to develop integrated multifunctional airframe structures that eliminate fragile external thermal-protection systems and incorporate the insulating function within the structure. The approach taken to achieve this goal was to develop candidate hypersonic airframe concepts, including structural arrangement, load paths, thermal-structural wall design, thermal accommodation features, and integration of major components, optimize thermal-structural configurations, and validate concepts through a building block test program and generate data to improve and validate analytical and design tools.

Another effort has been focused on improving the fabrication and cycle mission life of ceramic matrix composite (CMC) control surfaces. The objectives of the work were twofold: (1) to develop and demonstrate technologies associated with the joining of separate CMC control-surface segments, and (2) to design, fabricate, and perform flight qualification testing of a CMC body flap control surface.

Recent work has been focused on developing on-orbit repair technologies for the Space Shuttle Wing Leading Edge (WLE) system that an astronaut can install during an extravehicular activity (EVA). The plug repairs incorporate C/SiC cover plates that are attached through the damage in the wing using a refractory metal attachment mechanism. These plug repairs were fabricated, flew on the Shuttle Return to Flight mission, STS 114, and are currently stowed on the ISS.

1. INTRODUCTION

The National Aeronautics and Space Administration (NASA) has developed hot structure technology for several hypersonic vehicles. Significant reductions in vehicle weight can be achieved with the application of hot structures which do not require parasitic thermal protection systems (TPS). Hot structures are also more durable than current tile and blanket TPS, are easier to inspect, and require less maintenance and repair.

The most significant technical issue that must be addressed in hot structure design is the development of cost effective, environmentally durable and manufacturable material systems capable of operating at temperatures from 1500°C to 3000°C, depending on the application. The development of these durable and affordable material systems is critical to technology advances and to enabling future economical hypersonic vehicles.

In this paper, recent advances in hot-structure technology will be discussed including; the X-43A leading edges, the X-37 body-flap control surface, and repair concepts for the Shuttle WLE. In addition, hot-structure and TPS technology advances will also be presented.

2. HYPER-X (X-43A)

The Hyper-X hypersonic research program aimed to demonstrate scramjet air-breathing engine technologies that promise to increase payload capacity—or reduce vehicle size for the same payload—for future hypersonic aircraft and reusable space launch vehicles. (A scramjet is a supersonic combustion ramjet, which operates by burning fuel in a stream of supersonic air compressed by the forward speed of the aircraft with the rapid expansion of hot air out the exhaust nozzle producing thrust. Unlike conventional aircraft engines, scramjets have no rotating parts.)

The Hyper-X flight demonstrator program consisted of three flights. Two were to be flown at Mach 7 and the third was to be flown at Mach 10. A mishap during the boost phase of the first Mach 7 flight resulted in the flight being prematurely terminated prior to the free flight of the research vehicle. However, the subsequent second Mach 7 flight was successful, leading to the decision to complete the flight demonstrator program with the Mach 10 flight.

The leading-edge flight hardware for the X-43A Mach 10 flight vehicle consisted of eleven pieces; a nose, two forward chines, two aft chines, two horizontal tailpieces, two upper vertical tailpieces and two lower vertical tail pieces, as shown in Figure 1. For the Mach 7 flight vehicles, only seven of the leading-edge pieces were fabricated out of C/C since thermal analysis indicated that the four vertical tailpieces would not be subjected to high enough temperatures to require C/C, and thus could be fabricated from a Haynes alloy. For each of the two Mach 7 flights, the seven leading edge C/C flight hardware pieces were fabricated by Goodrich Corporation, Santa Fe Springs, California, USA.

Various views of the X-43A flight vehicle are shown in Figure 2. The front and side views show the sharp leading edges. The desired nose tip radius on the Hyper-X flight vehicles was 0.030 in. Aerothermal heating on

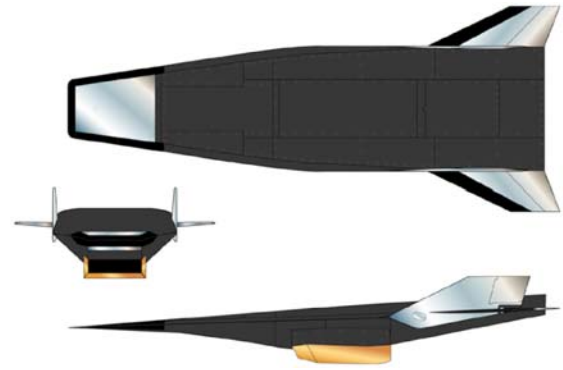


Figure 2. Three views of X-43A flight vehicle.

sharp leading edges such as this produce high temperatures and high thermal gradients.

In order to reduce the nose tip temperature and reduce thermal gradients, it was decided to construct the nose leading edge using high thermal conductivity carbon fibers woven in an unbalanced weave to give more fibers perpendicular to the leading edge. A K321 fiber woven in a 4:1 unbalanced weave was baselined for the nose leading edge of the Mach 7 vehicle.

Thermal analysis of this baselined construction indicated that the nose maximum temperature would only get to 3000°F, so a silicon carbide (SiC) oxidation coating system was deemed viable. Even though a 4:1 unbalanced weave was baselined, the Mach 7 nose pieces were fabricated from a 2-D billet of K321, 5:1 fabric. The difference in substrate weave architecture resulted from the 5:1 fabric being more readily available than the 4:1 fabric, and that it would conduct more heat away from the nose tip. The two horizontal tail control surface pieces for each Mach 7 vehicle were fabricated from quasi-isotropic K321, and coated with SiC while the four side chines were fabricated from 3-D needled C/C PAN-based fiber and coated with SiC. Figure 3 shows the assembled Mach 7 nose and forward chine flight hardware.

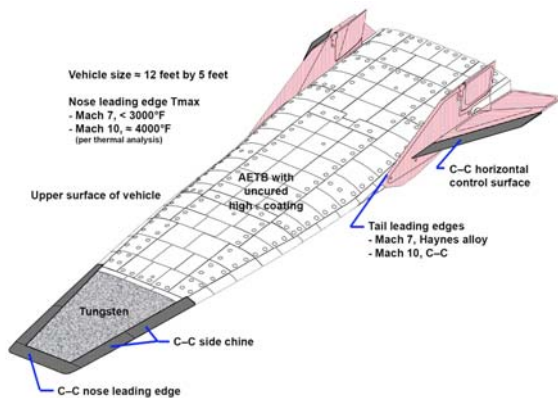


Figure 1. Thermal protection for the X-43A Mach 7 vehicle.

2.1. Development of Mach 10 Leading Edges

2.1.1. Coating Evaluation

Thermal analysis for the Mach 10 vehicle, with a 0.030 in. nose radius, predicted temperatures that would approach 4000°F at the nose tip. The 4000°F temperature greatly exceeds the use temperatures of SiC-based coating systems even for a short duration, single flight. To identify a suitable leading edge for the Mach 10 vehicle, arc-jet testing was performed on leading-edge segments fabricated using thirteen different material systems in the H2 arc-jet facility at the Arnold Engineering Development Center (AEDC),

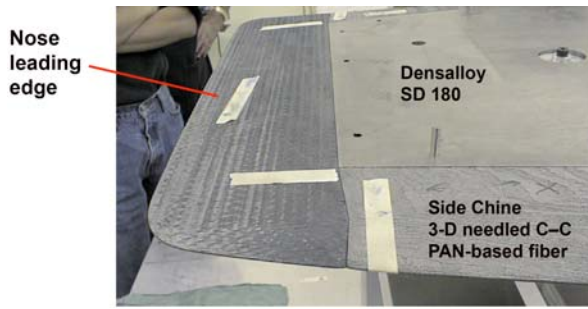


Figure 3. Goodrich X-43A Mach 7 C/C nose leading-edge flight hardware assembly.

Arnold Air Force Base, TN in early 2000. The objective was to evaluate potential coatings for single use on a C/C substrate at Mach 10 heating conditions for 130 seconds. The flight conditions simulated were those of the Mach 10 flight.

The K321 fiber, 5:1 C/C substrate used in the Mach 7 fabrication was used by many of the vendors. Some vendors selected other substrates. Most of the coating systems provided for evaluation were Hf, Zr, Si, and Ir based materials. The range of materials and processes evaluated are shown below:

Substrates

- C/C (5:1, K321 fiber, P-30X)
- Functionally graded material (5:1, K321 fiber)
- W-1% La, TZM, ZrB₂/20%SiC

Coating components

- HfC, HfO₂, HfB₂, ZrC, ZrB₂, SiC, Si₃N₄, MoSi₂
- Ir, Re, ZrC/W-Re

Coating processes

- Chemical Vapor Infiltration (CVI), Chemical Vapor Deposition (CVD), Chemical Vapor Reaction (CVR), Reaction sintered, Molten salt bath, Plasma spray, Paint on, Hot pressing

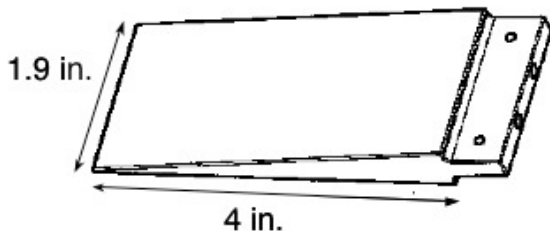


Figure 4. Specimen test size for material evaluations in the 2000 AEDC arc-jet test program.

Figure 4 shows the dimensions of the test specimens; 1.9 in. wide, 4 in. long with a tip radius of 0.030 in.

As a result of the testing, a three-layer coating comprised of a SiC conversion of the top surface of the C/C substrate, followed by a CVD layer of SiC, followed by a thin CVD layer of HfC was selected for the flight vehicle.

2.1.2. Weave, Layup, and Heat Treatment Temperature Selection

A numerical model of the Hyper-X nose components was developed at NASA Langley to calculate the aerothermal heating, thermal response, and structural response. A typical finite element model of the nose is shown in Figure 5.

As illustrated in Figure 6, the high thermal gradient at the tip of the nose leading edge leads to high thermal stresses in the spanwise direction, parallel to the tip of the leading edge. The stress of primary concern is the weak axis direction compressive stress along the leading edge. Available existing data on high thermal conductivity pitch fiber C/C composites implied that the weak axis compressive stress along the leading edge could cause failure for an unbalanced 4:1 C/C.

The use of heat-treated pitch fiber P-30X, unbalanced 4:1 weave was motivated by the desire to achieve the highest possible chordwise direction thermal conductivity by having the greatest possible fiber volume percentage oriented in the chordwise direction. Due to the concerns regarding potential spanwise direction compressive stress failure at the tip of the leading edge, a 3:1 reinforcement was used for the Mach 10 nose instead of the 4:1 reinforcement. This

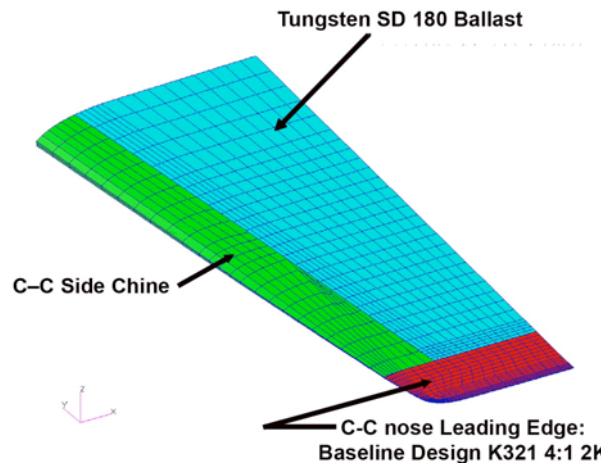


Figure 5. Finite element model of Hyper-X nose component.

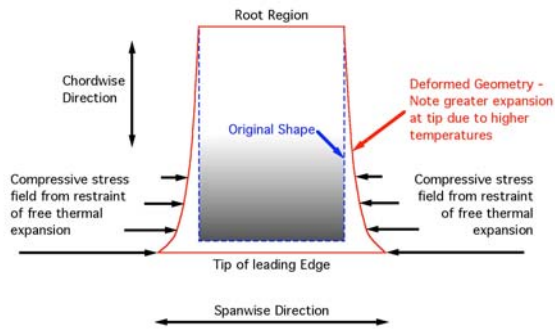


Figure 6. Schematic diagram of planform view indicating compressive stress field parallel to leading edge.

change allowed more fibers to be oriented in the spanwise direction, thus increasing the fiber-dominated compressive strength. This recommendation was accepted by the M10LEC.

Table 1. Leading-Edge Design and Requirements

Part	Nominal Size, in.	Max. Temp., °F	Major Concerns
Nose	18 x 5 x 0.6	3800	High thermal gradient
Chine	18 x 4 x 3	1300	Thick
Horizontal Tail	33 x 5 x 0.6	3200	High temperature at root, clearance
Vertical Tail	14 x 5 x 0.7, 8 x 5 x 0.6	2800	Fixed, clearance

Table 1 shows the leading-edge design requirements and indicates the major concern of each part type. Major concerns for the nose were tip temperature and high thermal gradients leading to high compressive stresses. The chine maximum temperature was only 1300°F, however, there was a major concern about the fabrication of such a thick part. The horizontal tailpieces were very long, 33 in. leading to concerns of coating uniformity. High temperature at the root clearance was another possible concern for these pieces. The upper vertical tail pieces were 14 in. long, 5 in. wide and 0.7 in. thick while the lower vertical was 8 in. long, 5 in. wide and 0.6 in. thick. A fixed clearance was the major concern with these parts.

The chine material was a conventional 2D C/C for two primary reasons. First, the heating rate was low

enough that high thermal conductivity was not required to reduce the temperature. Second, since the thickness was large, a conventional 2D composite had the best chance of surviving processing without delamination. Hence, the use of a conventional, balanced 1:1 fabric and a quasi-isotropic lay-up was utilized. Bending loads were also relatively low at the tang so quasi-isotropic strength was sufficient and a warp-aligned composite was not required.

2.2. Slotting Evaluation

The C/C utilized for the nose leading edge of the Hyper-X Mach 10 vehicle was anticipated to have large compressive thermal stresses in the spanwise direction. The large thermal stresses were due to the extremely large chordwise thermal gradients at the nose. Numerical analysis indicated that there might be a problem with material failure due to stresses and/or strains above the strength of the material. These concerns over the possibility of leading-edge failure at the nose tip led to a consideration of possibly slotting the nose leading edge to relieve the stresses.

It was then decided to test the C/C material utilizing a 4-point bend test. The flexure specimen configuration is illustrated in Figure 7. This test was developed by MR&D and SRI, and was designed to induce a compressive stress state in the spanwise direction equal to the spanwise direction compressive stress at the tip calculated from the temperature gradient of the flight condition. SRI conducted the elevated temperature 4-pt bending tests on uncoated material at 3000°F and 3800°F.

In both the 3000°F and 3800°F tests, the maximum compressive strain was above the predicted strain during flight. (Strain was used to evaluate the material instead of stress since it was felt to be a better gage of material capability for the leading edge conditions.) There was no indication of compressive failure in the tip

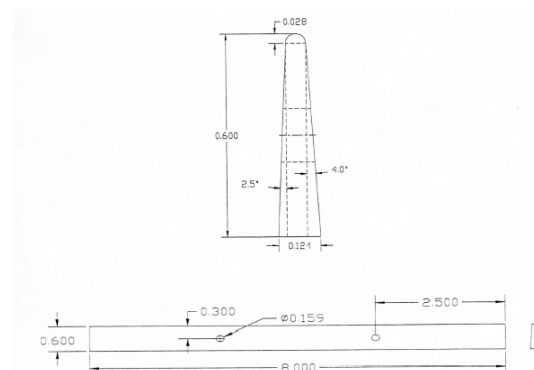


Figure 7. Four-point bend flexure specimen design details for compressive strength testing.

region. The test specimen failed at the loading holes, preventing increasing the load until tip failure.

Following the above tests on uncoated material, test results indicated that the compressive properties of the HfC/conversion coated C/C material actually exceeded those of the uncoated C/C material. The coated C/C composite compressive strengths resulting from the 4-point bend testing provided a positive stress margin of safety and gave confidence that the leading edges would not fail due to the high thermal gradient at the tip of the leading edge.

2.3. Tip Erosion

The initial AEDC arc-jet test in 2000 was conducted on a specimen only 1.9 in. wide. The final design of the nose leading edge had several differences from the original specimen tested in 2000 including a different material lay-up, a different heat treat temperature, and a different coating process. The coating process was changed because the full scale hardware could not fit into the original apparatus used to coat the original, smaller test specimens. Because of these differences, it was felt prudent to repeat the 2000 test with the new hardware configuration. It would be ideal to test the full-scale nose so that thermal stresses would match flight conditions; however, the largest model span length that could be accommodated in the AEDC, H2 facility at the required test conditions was approximately 6 in. Therefore, the two test specimens were fabricated with a span length of 5.89 in., and the test were performed at identical conditions as the initial test. A post test picture of one of the test models is shown in Figure 8. As can be seen in the figure, the nose tip eroded. Buckling or shear failure was not observed due to the span length limitation, but erosion along the full length of the leading edge was observed on both models.

After the unexpected erosion in the AEDC arc-jet test, various options were considered and a modified approach was selected for supplying the nose leading edge flight hardware. It was decided to machine a new nose leading edge out of an existing C/C billet. This nose leading edge would be redesigned to have a 0.050-in. radius, and the new nose leading edge would be heat treated at a temperature to be specified by the M10LEC.

2.4. Delaminated Chine

During the fabrication cycle of the parts, periodic fit checks were required to ensure the parts fit on the vehicle. Fabricating the parts to the exact required tolerances was not a trivial task. Thermal expansion of the parts, substrate contraction and expansion during the fabrication and coating process, plus coating thickness all had to be considered. During installation, gaps had to be maintained to allow for thermal expansion so as not to create undue stresses. Accurate thermal expansion

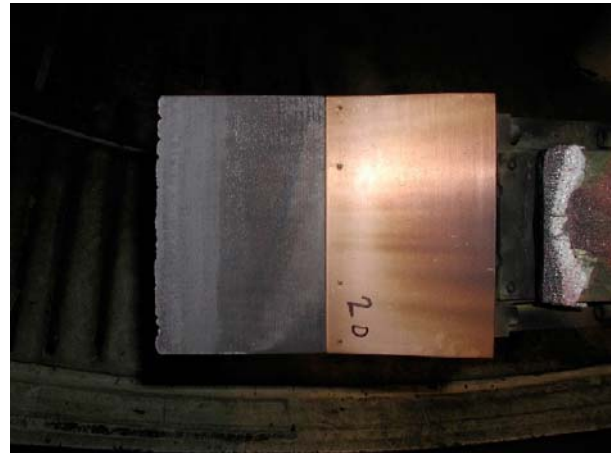


Figure 8. Arc-jet model tested at AEDC in February 2004.

data in all directions was required to calculate the expected maximum thermal expansion that would occur in each part and in each direction, such that the gap sizes needed to account for part expansion, could be calculated.

During one of the fit checks, a delamination was found in a forward side chine. The problem was solved by fabricating a replacement part. However, the concern generated by discovering this flawed part late in the process raised questions as to whether other parts might also be flawed.

The damaged (delaminated) forward side leading edge chine and one of the nose samples tested in the arc heater were sent to NASA Langley for evaluation. The NDE personnel at Langley had considerable experience evaluating shuttle RCC; however, there was uncertainty if they could image a thick component such as the side chine. They were able to image the chine, and the NDE test showed density variations in the side chine and no density variations in the nose. Because it was not possible to quantify the severity of the density variations from the NDE images, the M10LEC decided to perform a load proof test on the side chine. Reference 3 gives full details of the load proof test. The chine was supported as it would be on the vehicle and was uniformly loaded to over two times the expected flight load. The chine passed the test and no damage was observed, indicating that the density variations that were observed in the NDE tests were not detrimental to part integrity.

3. HOT AND INTEGRATED STRUCTURES

In the area of hot and integrated structures, the Next Generation Launch Technology (NGLT) program developed technology with a focus on ceramic matrix composite (CMC) materials development for application to structures, and the development of wall

structural concepts. Efficient and reliable hot wing structures with low maintenance and fabrication costs were part of the long-range goals of this element. This element consisted of five tasks. Integrated Airframe Structures was focused on the long term development and validation of structural systems that show the best potential for a “wall that does it all.” The CMC Control Surfaces task was focused on reproducible CMC materials with improved mechanical reliability and cyclic durability for control surfaces.

The Integrated Airframe Structures task was focused on both hot and warm (insulated) structures and integrated fuselage/tank/TPS systems. The objective was to develop integrated multifunctional airframe structures that eliminate fragile external thermal protection systems and incorporate the insulating function within the structure. The approach taken to achieve this goal was to develop candidate hypersonic airframe concepts including structural arrangement, load paths, thermal-structural wall design, thermal accommodation features, and integration of major components; optimize thermal-structural configurations; and validate concepts through a building-block test program and generate data to improve and validate analytical and design tools.

The structural arrangements considered include both integral, where the tank carries internal and external loads, aerothermodynamic loads, and nonintegral, where the tank carries only internal pressure loads and the tank can expand and contract. An integrated wall construction is an approach, or design philosophy, where the entire structure (the tank, insulation, TPS, etc.) is designed together to account for thermal and mechanical loads. This task considered all options for an integrated structure, including TPS, cold structure, hot structure, tanks, insulation, and all types of material systems. An illustration of a truss core sandwich concept is shown in Figure 9.

The CMC Control Surfaces task was focused on improving the fabrication and cycle mission life of ceramic matrix composite (CMC) control surfaces. A high payoff application presently under study is a CMC control surface. In June 2001, Materials Research & Design, Inc. (MR&D) was awarded the NASA Next Generation Launch Technologies (NGLT) contract entitled, “Design, Fabrication and Test of Ceramic Matrix Composite (CMC) Control Surface Structure and Joining Technology.” The objectives of the contract were twofold: 1) to develop and demonstrate

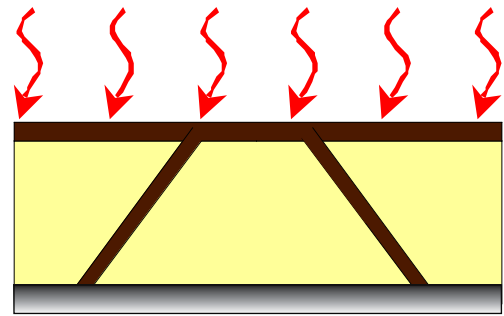


Figure 9. Schematic of truss core sandwich concept.

technologies associated with the joining of separate CMC control surface segments, and 2) to design, fabricate, and perform flight qualification testing of a CMC body flap control surface. The first objective is required when a given hot structure control surface is too large to be fabricated within single CMC processing facility. Relative to the second objective, the NASA/Boeing X-37 long duration orbiting vehicle (LDOV) is a potential flight demonstration vehicle.

The contract was performed by a joint industry and government team lead by MR&D, the prime contractor. For the subelement test articles, the industry participants included two separate fabrication teams. For one team, General Electric Company Power Systems Composites (GE PSC) of Newark, DE, was the partner responsible for the CMC fabrication, while Textile Engineering And Manufacturing (T.E.A.M.) of Slatersville, RI provided the T-300 carbon fiber 2D fabric and 3D woven textile weaving and preforming for the reinforcement of the silicon carbide matrix composites fabricated by GE PSC. For the second team, Refractory Composites, Inc. (RCI) of Glen Burnie, MD fabricated the C/SiC subelements using T-300 carbon fiber fabrics and 3D woven preforms woven and preformed by Albany International Techniweave (AIT) of Rochester, NH. Southern Research Institute (SRI) of Birmingham, AL performed non-destructive examination of all of the C/SiC composite subelements manufactured by both GE PSC and RCI. Non-destructive examination (NDE) was performed on the C/SiC subelements before and after mechanical testing. Government participants in this study have included NASA Langley Research Center (LaRC) for the testing of the C/SiC subelements, NASA Dryden Flight Research Center (DFRC) for the combined thermal and mechanical load testing of the C/SiC subcomponent, and NASA Johnson Space Center (JSC) for guidance on the re-entry environmental conditions.

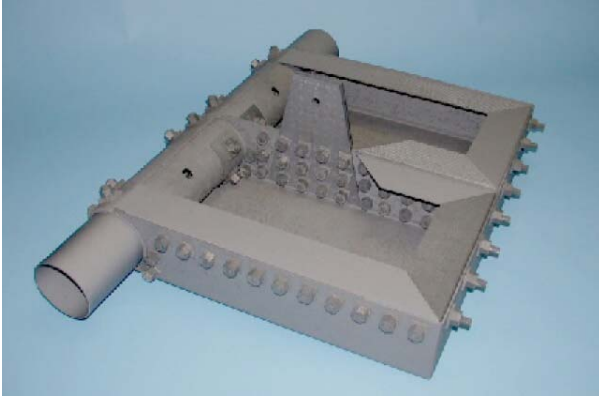


Figure 10. C/SiC body flap subcomponent assembly thermal-mechanically tested at NASA DFRC.

For the C/SiC subcomponent, a half-scale non-tapered hot structure body flap, the fabrication was performed entirely by GE PSC with reinforcement woven and preformed by T.E.A.M. Figure 10 shows the C/SiC body flap subcomponent designed by MR&D and fabricated by GE PSC. SRI performed NDE on the C/SiC subcomponent prior to testing at NASA DFRC. Post-test NDE was performed by GE PSC using infrared thermography. In addition to coordinating the activities of all of the industry and government participants, MR&D also performed the material and thermo-structural design and analyses of the C/SiC components, including each of the C/SiC subelements and the C/SiC subcomponent.

In November 2003, the C/SiC body flap subcomponent was subjected to combined thermal and mechanical testing, see Figure 11, by means of simultaneous 2060 F heating and 100% design limit (mechanical) loading (DLL). The simultaneous combined thermal and mechanical testing performed by NASA DFRC was the first combined load testing

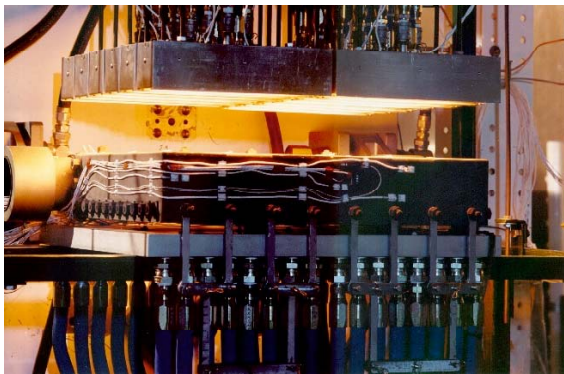


Figure 11. Photograph of C/SiC subcomponent during testing at NASA Dryden.

conducted on a CMC control surface. Figure 11 is a photograph of the body flap subcomponent test article under combined loading at NASA DFRC.

Two additional sub-tasks focused on the applications of C/C's for control surfaces. The first of these focused on the development of integrated hybrid hot structures, comprised of a ceramic matrix composite (CMC) face sheet/insulating foam core/polymer matrix composite (PMC) substructure, which would be load-bearing as well as eliminate the need for an external, parasitic TPS. The second task had as its objective the development of ceramic matrix composites with improved durability under cyclic conditions in oxidizing environments. Both efforts shared a goal of enabling a wider choice of vehicle flight profiles and increasing operational margin by providing enhanced thermal load capability, and increased safety and reliability, while decreasing vehicle weight.

4. SHUTTLE WLE REPAIR

The Columbia Accident Investigation Board's (CAIB) final report identified that damage to one of the wing leading edge (WLE) panels of the Space Shuttle Columbia's left wing resulting from an impact by foam shed from the External Tank during ascent allowed the inflow of hot plasma gasses into the wing during reentry and precipitated the tragic loss of Columbia and her crew. The CAIB recommended that an on-orbit WLE repair capability be developed prior to the return of Shuttle to flight. Several technologies were pursued in an extensive effort to develop an On-orbit WLE repair resulting in two capabilities that flew on STS-114, the return to flight mission. Those two capabilities included a material that can be applied to fill small cracks in the coating and substrate of the refractory carbon-carbon (RCC) and the plug repair kit that provides the capability of repairing holes in the RCC as large 10.16-cm in diameter. Only the plug repair kit will be discussed in this paper.

The plug repair kit consists of several 17.78-cm-diameter carbon-silicon carbide cover plates of various curvatures that can be attached to the refractory carbon-carbon WLE panels using a TZM refractory metal attach mechanism (see Figure 12). The attach mechanism is inserted through the damage in the WLE panel and as it is tightened, the cover plate flexes to conform to the curvature of the WLE panel within approximately 0.050 mm (see figure 13). An astronaut installs the repair during an extravehicular activity (EVA). After installing the plug repair, edge gaps are checked and the perimeter of the repair is sealed using a proprietary material developed under a separate effort to fill cracks and small holes in the WLE.



Figure 12. Plug repair components and kit.

In developing the plug repair concept several issues had to be addressed including material, design, performance and operability. An Oxyacetylene torch, shown in the photograph in Figure 14, was calibrated to produce the heat required to heat a specimen to WLE entry temperatures and was used to screen candidate repair materials. Promising materials were then tested in the NASA Johnson Space Center arc-jet test facility to determine their resistance to oxidation in a hypersonic environment. C/SiC was selected as the cover plate material due to its superior strength and resistance to oxidation. In order to raise its operational temperature limit, a proprietary oxidation barrier coating was developed. TZM was selected as the attach mechanism material due to its manufacturability and structural performance as well as its ability to withstand the plasma environment when coated with a proprietary oxidation barrier coating.



Figure 13. Plug flexes to conform to WLE panel.



Figure 14. Typical torch and arc-jet tests.

Careful attention was paid to the design of the plug repair to maximize flexibility (minimizing the number of cover plates required) and minimize the protuberance of the repair to prevent excessive aerothermodynamic heating. Non-linear finite element analyses, including contacting surfaces, were used to model the plug during installation and operation. CFD and thermal analysis were used to predict plug temperatures during entry. Typical results of these analyses are shown in Figure 14.

The ability of the astronaut to safely handle and install the repair during an EVA was also a significant consideration during the design process. Attention to handling, tools and the ability to check the correctness of a repair after installation were considered in developing the repair and the tools required to affect a repair during an EVA operation.

Plug repairs that were prepared in the Human Thermal Vacuum test facility at NASA JSC were tested to verify performance in the hypersonic environment. The tests were performed in the arcjet test facilities at both NASA Johnson Space Center and NASA Ames Research Center. Pre-test and post-test photographs of one of these test samples is shown in Figure 15.

The plug repair kit was flown aboard STS 114, the return-to-flight mission, and is now stowed on the ISS. During the mission, astronauts successfully practiced installation of a plug repair inside the orbiter to

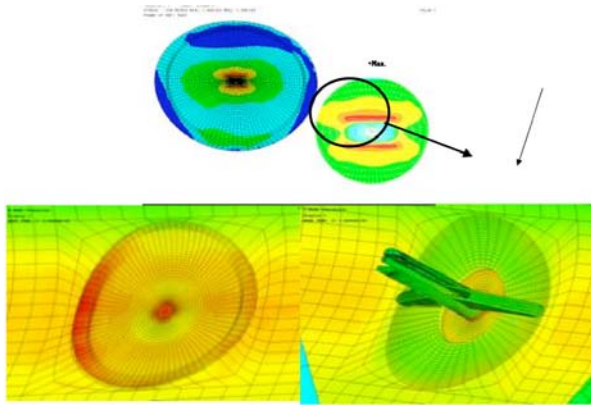


Figure 15. Typical structural and thermal analysis results.

demonstrate operation of the attachment mechanism in the microgravity environment.

5. SUMMARY REMARKS

NASA has successfully developed hot-structure components for vehicles including the X-43A, the X-37 and the Space Shuttle Orbiter. NASA has also investigated advanced integral hot structures and CMCs for application to hot structure. Recently, NASA has even used hot-structures technology to develop repairs for the Space Shuttle Orbiter WLE that can be installed on-orbit.



Figure 15. Arc jet tested plug repair.

Hot structures are lighter and require less maintenance than insulated cool structures and will enable future hypersonic flight, space access and entry vehicles. Environmentally durable high-temperature materials that can be manufactured and are inexpensive are an enabling technology for hot structure and should be the focus of near-term research. It should also be noted that a state-of-art material is not a state-of-art structure, especially when discussing high temperature materials and structures. Once a material is matured, a significant amount of work remains to be performed to obtain an operable structure.

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