NASA-TM-87626 19860003881

NASA Technical Memorandum 87626

# STRUCTURES AND MATERIALS TECHNOLOGY ISSUES FOR REUSABLE LAUNCH VEHICLES

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OCTOBER 1985

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## Abstract

Projected space missions for both civil and defense needs require significant improvements in structures and materials technology for reusable launch vehicles: reductions in structural weight compared to the Space Shuttle Orbiter of up to 25 percent or more, a possible factor of 5 or more increase in mission life, increases in maximum use temperature of the external surface, reusable containment of cryogenic hydrogen and oxygen, significant reductions in operational costs, and possibly less lead time between technology readiness and initial operational capability. In addition, there is increasing interest in hypersonic airbreathing propulsion for launch and transatmospheric vehicles, and such systems require regeneratively cooled structure. These technology issues pose quite a challenge to the structures and materials community, especially since most industry and DOD organizations have not been working these technology issues for 15 years, and the NASA base R&T program has been supported at a relatively low level of effort. This paper addresses the technology issues, giving brief assessments of the state-of-the-art and required activities to meet the technology requirements in a timely manner.

## Introduction

The Air Force Wright Aeronautical Laboratories (AFWAL) and the National Aeronautics and Space Administration (NASA), and their predecessor organizations, have worked the problems of high speed flight since the late 1940's. The US had extensive programs in the 1950's and 1960's aimed at various missions requiring flight of reusable vehicles at hypersonic speeds [1-8]. Due to accelerating costs and formidable technical challenges most programs were terminated, and except for the NASA Space Shuttle Program, R&T for reusable launch vehicles has been a low priority effort. In spite of this limited activity, mainly conducted by NASA at the Langley Research Center, significant progress has been made in key technologies. For example, developments in structures include composite and high temperature materials, airframe structures/TPS, hypersonic airbreathing propulsion structures, structural heat transfer analysis, aerothermal loads, and high temperature test techniques for TPS and airbreathing hypersonic propulsion structure [9-16].

In addition, advances have been made in more generic, but essential, areas such as aeroservoelasticity [17] and interdisciplinary analytical design methodology [18] as well as in other key technology areas such as rocket and hypersonic airbreathing propulsion systems and aerothermodynamics. These technology advancements, coupled with the development of and experiences with the space shuttle [19, 20], now make feasible (with significant continued R&T efforts) reusable launch vehicles that meet the anticipated demanding performance requirements and significantly reduced launch costs of various potential DOD and NASA missions currently being defined as a part of the National Space Strategy.

In response to the National Space Security Directive, planning exercises are underway to develop the National Space Strategy. One of these planning exercises is to develop the technology issues and define plans and resources to resolve these issues. The purpose of this paper is to present some historical perspectives of work in structures and materials on hypersonic vehicles in the 1950's and 1960's, to review technology developments made during the 1970's and early 1980's, to assess this technology to identify technical issues in structures and materials, identify high pay off technologies, and, in a broad sense, define what is required to bring these technologies to a state of readiness that will resolve the technical issues.

## Structures and Materials Technology Needs

A number of studies [21-31] have revealed the significant challenges facing the materials and structures communities for advanced reusable launch and orbital transfer vehicles. Briefly stated, systems studies reveal a need for up to a 25 percent reduction in structural mass compared to the current Space Shuttle Orbiter, a factor of five increase in mission life, reduced turn around time and time to launch, reduced costs per launch (by up to a factor of 10) and per pound of payload to orbit, and flexible launch/landing site requirements.

In addition, for some missions the performance requirements are such that higher maximum use temperatures and overall heat loads must be accommodated; generally maximum use temperature is the limiting factor on desired performance for such missions. Also, for reduced operational cost and mission flexibility, reusable containment of cryogenic LOX and LH2 and durable, long life surface structures are required. There is increasing interest in hypersonic airbreathing propulsion for launch systems, thus structural concepts and thermal management systems for such engines must be developed. These requirements dictate that loads be known to an accuracy never before required to permit knowledgeable reductions in margins of safety. Also the structural concepts designer must wring out every ounce of nonoptimum structure. Such requirements dictate a large data base on aerothermal loads and accurate and efficient structural and heat transfer analysis codes. In addition, test facilities, techniques and instrumentation must be provided to verify analysis codes, advanced lightweight concepts, and overall structural performance.



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The current structures and materials technology needs list thus includes:

- o Advanced Materials
- o Advanced Structures/TPS (Airframe)
- o Flightweight, Reusable Cryogenic Tanks
- o Hypersonic Airbreathing Propulsion Structures
- o Loads/Criteria
- o Interdisciplinary Analysis
- o Test Requirements/Facilities

Structures and materials efforts are also required for advanced rocket propulsion and recovery systems for large boosters, but these technologies are covered elsewhere in the National Space Strategy technology plans and will not be discussed in this paper.

## Advanced Materials

The leading candidate materials for hot structures and thermal protection systems include polymer matrix composites, advanced aluminum and titanium alloys, aluminum and titanium metal-matrix composites, superalloys and carbon-carbon composites. The best material for a particular application depends on a number of factors including maximum service temperature, oxidation conditions, maximum loads expected, design life-time of the structure, and the ability to fabricate the required shapes and configurations. Generally a substantial data base is required on the candidate materials to consider them for hardware applications. However, the level of support available for high temperature materials research during the post shuttle technology era has been very small and consequently there have been very few technology advancements in high temperature materials for airframe applications during the past decade.

Significant technology issues exist for all of the material systems of interest for space transportation systems. A discussion of these issues and possible approaches for their resolution are presented in the following sections. Selected examples are included to illustrate the general level of maturity for composites, high temperature metallics, and carbon-carbon composites.

### Composites

High temperature polymer matrix composites are candidates for structural applications on space transportation systems in locations where the temperature does not exceed approximately 600°F. Trade studies [32] have shown that substantial structural weight reduction and performance gain could be realized for the Space Shuttle by taking advantage of the large strength-to-weight and stiffness-to-weight ratios of advanced composites. Savings of 25 to 30 percent of the total structure/ TPS weight could be realized if graphite/polyimide (Gr/PI) could be used to 600°F to replace the baseline aluminum structure (350°F structural allowable). Because of the higher allowable temperature of the composite structure less TPS is required and the TPS tiles could be bonded directly to the Gr/PI substrate because of the thermal compatability and the higher specific strength and stiffness of the composites. A total weight savings of approximately 15,000 lbs. was projected.

Projections for these types of benefits from composites assumes that a mature technology base is available to allow for confident design with composites. Such a data base does not exist for high temperature composites. The CASTS program (Composites for Advanced Space Transportation Systems) [9] is an example of the type of material development programs that are required to foster the development of a new family of materials. This program was conducted by NASA during the mid seventies to develop Gr/PI for high temperature (300-600°F) structural applications. Fabrication procedures and specifications for NR150B2, PMR-15, and LaRC-160 polyimide matrix composites were developed. Prepreg specifications, cure and post-cure cycles, manufacturing methods applicable for building full scale structure components and nondestructive inspection procedures were included in this effort. Items shown in fig. 1 were built to demonstrate the fabrication technology. Flat laminates up to 0.125 in. thick and 2x4 ft. in area, chopped fiber moldings, honeycomb core sandwich panels, skin-stringer panels and a built-up component, a shuttle aft body flap segment, were fabricated using PMR-15 and LARC-160. The initial results of the fabrication technology contracts awarded for each of the three Gr/PI composites are reported in [9]. The capability to successfully fabricate structural components was demonstrated with the PMR-15 and LARC-160 Gr/PI composites.

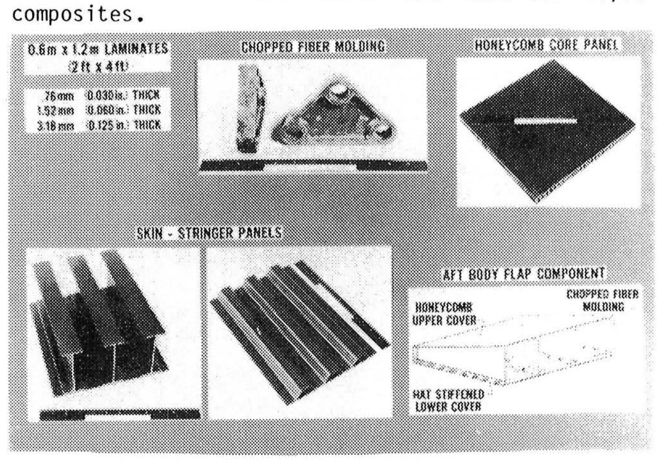



**Fig. 1.- Casts fabrication development components.**

NASA Lewis has pioneered the development of the PMR-15 system emphasizing improvements in processability and applications [33]. Modifications have been made to improve prepreg tack, to lower the curing temperature to improve the thermo-oxidative stability and resin flow, and improve shear strain. PMR-15 composites have been used to fabricate a variety of structure components ranging from small compression-molded bearings to large autoclave-molded aircraft engine cowls and ducts.

Although structural components have been successfully fabricated using polyimides they are substantially more difficult to process than epoxies. Also, the higher the thermal stability of the polymer the less processable they tend to be.

Thermal aging studies have been conducted at NASA Langley to establish the long-term upper-use limits of the graphite/polyimide composites [34-35]. Celion 6000/PMR-15 and Celion 6000/LARC-160 composites have been exposed in air circulating ovens at temperatures of 220°F, 500°F, and 550°F for times up to 25,000 hours. Weight loss measurements, short beam shear and flexural tests were conducted to characterize material changes. Unidirectional laminates, of both LARC-160 and PMR-15, cracked and degraded preferentially at the specimen edge perpendicular to the fibers [35] as shown in fig. 2. Because of this type of degradation the results obtained for small specimens were much more severe than results for large panels which are more representative of structural parts.

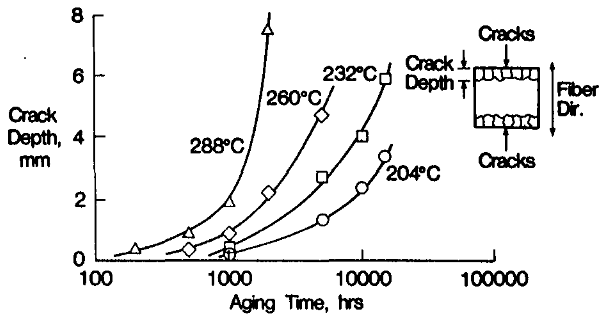



**Fig. 2.- Edge crack growth of aged Celion 6000/LARC-160 laminates.**

Another durability concern with Gr/PI composites is their tendency to microcrack during thermal cycling. Reference 36 reports on an experimental program where Celion 6000/PMR-15 laminates were thermal cycled between -250 and 600°F. Thermal cycling induced transverse microcracks and delaminations in the laminates. Compression, shear and flexure strength were significantly reduced by thermal cycling.

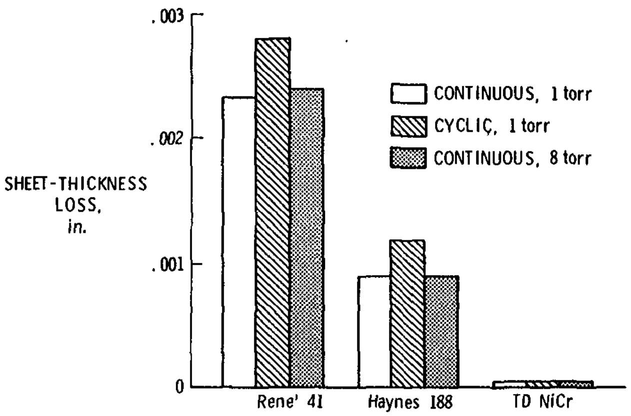
The basic trend observed in these studies is that the maximum allowable temperature for long-term service is generally significantly less than expected from early short-term exposure tests. The lifetime of composites is dictated by the combined time, temperature and stress conditions to which the composite is subjected. Additional testing of stressed specimens is required to establish the upper temperature use limits of composites.

During the past several years NASA Langley has developed a number of novel new approaches for high-temperature polymers with improved processability and thermo-oxidative stability, [37-41]. Several high temperature thermoplastics exhibit excellent adhesive and composite properties at elevated temperatures after long term aging at high temperature in air. For example, a polyimide displayed high adhesive properties with no loss in performance at 450°F after 37,000 hours at 450°F in air. The high temperature thermoplastics offer an attractive combination of properties such as high toughness (no microcracking as in the addition polyimides such as PMR-15) and no volatile evolution during fabrication (essentially void-free bondlines and composites). However, the present form of these high temperature thermoplastics are somewhat difficult to process (relative to epoxies). Further research is required to improve the processability of thermoplastics, to develop innovative methods to prepare adhesive tapes and composite prepregs, to optimize the fabrication of bonded components and composites, and to fabricate and test a demonstration component. To improve the processability, work will concentrate on molecular weight control, termination with inert stable end groups, polymer blends and plasticizers. The adhesive tape and composite prepreg effort will evaluate new methods such as powder impregnation. Fabrication work will concern the optimization of process conditions to produce high quality parts. A large scale component should be fabricated and tested to demonstrate the successful development of a viable structural resin system for high temperature applications.

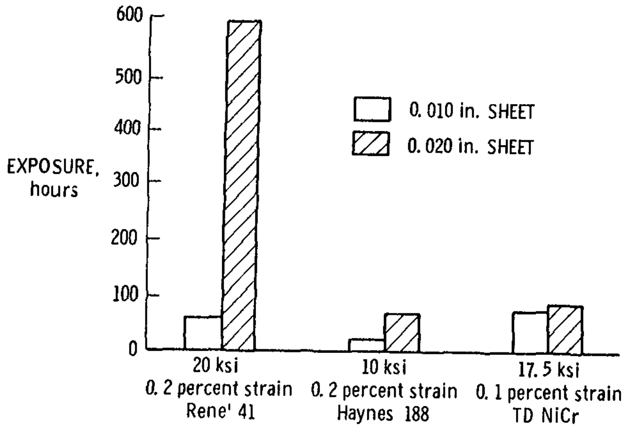
## Metallics

During the late sixties and early seventies NASA sponsored research on high temperature superalloys for supersonic and hypersonic transport structure, shuttle metallic thermal protection systems, and aircraft turbine applications. Work on the superalloy sheet materials focused on oxidation and creep, elevated temperature properties, high temperature fatigue of titanium alloys, and fabrication of structural panels for various thermal protection system concepts. The decisions not to build a supersonic transport and to baseline the ceramic tile system for the Shuttle TPS resulted in termination of most of the research on high temperature materials for airframe applications. Thus the technology of high temperature alloys for airframe applications has not dramatically changed during the past decade (with a few exceptions like superplastic forming (SPF) [42]). Also some of the promising high temperature alloys such as TD-NiCr are no longer commercially available. Because of the lack of new research programs many high temperature research facilities were dismantled, or placed in storage. Thus a new thrust in high temperature airframe materials will require a significant investment in new and updated controls, furnaces, and automation to restore the capability lost by inactivity during the past decade.

Two key technology areas requiring additional work are development of advanced processing to fabricate efficient light-weight metallic structures for high temperature applications and residual mechanical property data on candidate alloys after exposure to the environmental conditions imposed by the flight requirements to establish the upper temperature use limits of the materials. An example of the type of residual property data [43] required to establish the upper temperature use limits of thin-sheet superalloy for metallic thermal protection systems (TPS) is shown in fig. 3. Results of 100 hours. cumulative exposure at 1800°F on .020 in. thick oxidation specimens subjected to continuous and half-hour cyclic exposures are shown in fig. 3a. The effect of sheet thickness on creep resistance of selected superalloys at 1400°F is shown in fig. 3b. For both René 41 and Haynes 188 alloys at 1400°F the 0.010 in. sheet had a markedly lower creep resistance than the 0.020 in. sheet. For the René 41 alloy the difference was an order of magnitude. These results are particularly important because they illustrate that data generated on sheet material may not be very useful for predicting the lifetimes of foil gage alloys which are being considered for some of the new metallic TPS concepts, such as multiwall (discussed in a later section).




**(a) Results of 100-hour cumulative exposure at 1800° F on 0.020-in. thick oxidation specimens subjected to continuous and half-hour exposures.**




**(b) Effect of sheet thickness on creep resistance of selected superalloys at 1400°F.**

**Fig. 3.- Residual mechanical property data of sheet material after thermal exposures.**

An example of the types of issues that can be encountered in the fabrication of honeycomb panels using foil-gage materials is given in [44]. Liquid interface diffusion (LID)-bonded honeycomb sandwich panels were fabricated using 0.028-in. thick Ti-6242S outer skins and 0.002 in. thick Ti-6Al-4V core. Characterization tests on these panels and other coupon specimens showed several adverse effects of the LID bonding process including: reduced static strength and elongation at -50°F and RT, lower RT fatigue strength, higher fatigue crack growth rates especially in the thinner gages, and welds through the LID-treated material developed delayed weld cracking without loads being applied. This work illustrates that joining of foil gage materials will require additional work to establish procedures and processes that are optimized for the alloys, section thicknesses and use temperatures.

Development of advanced processing techniques and procedures for forming, joining, and thermal mechanical treatment for existing and new alloy systems is a key technology for successful development of efficient and cost effective hot metallic structures. The ability to economically fabricate complex structures such as the fuel strut for a hypersonic scramjet engine is a high priority technology development activity. Although technology advancements have demonstrated that complex structures can be fabricated [11], the durability of such structures under realistic temperature and loading conditions has not been fully demonstrated. Thus the durability of hot structures and the residual mechanical properties of selected alloys needs to be further investigated to establish the upper use limits of alloys in the as-fabricated condition. Because processing can substantially alter the microstructure of alloys it is essential that materials durability testing be performed on alloys after they have been subjected to realistic processing conditions. The desire to reduce weight of hot structures and push for the maximum structural efficiency will naturally lead to reduced thicknesses of metal cross-sections and higher stresses in the materials. The demand for higher performance will require a better definition of the upper temperature use limits of alloys in their as-fabricated condition.

For a given reentry environment the surface temperature is governed primarily by the emittance and catalytic activity of the surface. Good progress in lowering the catalytic activity of superalloys for heat shield applications has recently been reported [45]. Data for Inconel 617 and a dispersion strengthened iron base superalloy MA-956 is shown in fig. 4. The borosilicate coatings applied to the surface in a thickness of a few hundred angstroms resulted in a dramatic reduction in the catalytic activity of the MA-956 surface resulting in a 600°F decrease in the equilibrium surface temperature for the particular exposure conditions selected for this test. Research of these and similar coatings is continuing and arc-jet tests for realistic size panels are planned to verify results obtained on small 1 in. diameter disk specimens.

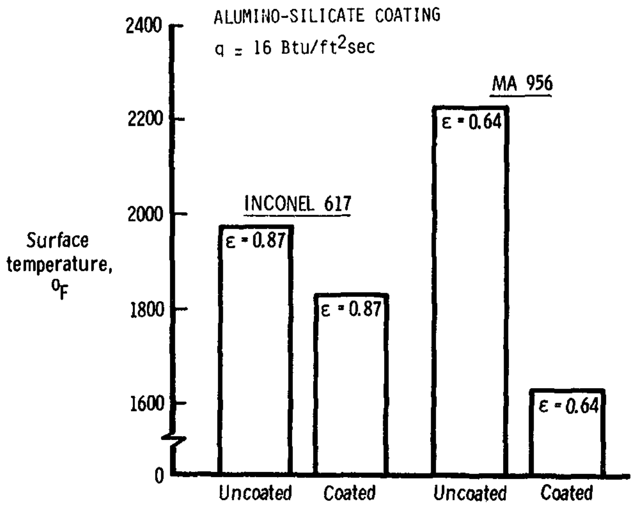



**Fig. 4.- Effects of coatings on catalysis of superalloys.**

The upper temperature use limits of titanium alloys will generally be determined by either creep or oxidation. Oxygen contamination of titanium can result in large reductions in ductility. Recent studies of the oxidation of Ti have been reported [46, 47]. A mathematical model of the oxidation process was developed and verified experimentally for commercially pure Ti, Ti-6Al-4V and Ti-6242 [46]. For given exposure conditions the oxygen behavior of the alloy can be calculated including the thickness of the surface oxide, the amount of oxygen in solid solution in the alloy, and the concentration profile of the oxygen in the alloy substrate. This allows the effective contamination depth to be calculated from which the embrittlement of the substrate can be estimated. A new nondestructive technique was also developed to measure oxygen concentration profiles in Ti substrates [48]. Also a new class of coatings has been tested [47] which substantially reduces the rate of oxygen diffusion into the alloy substrate. The beneficial effects of a few hundred angstroms of Al and Ti in reducing oxygen pick up by the substrate is shown by the results presented in fig. 5. The figure shows the residual room temperature tensile elongation of Ti-6242 foil after exposure to simulated Space Shuttle reentry conditions. Reentry simulation was by static oxidation at the indicated temperatures for times determined by the analysis of [46] to equal the cyclic conditions of actual reentry. Data are shown for uncoated alloy after 100, 200, and 400 reentry missions at temperatures of 1000, 1100, 1150, and 1200°F. The sensitivity of titanium embrittlement by oxygen is shown graphically. A single data point for coated alloy is shown for 200 missions exposure at 1150°F. That data shows the potential for reducing the embrittling effects of oxygen on titanium by adding as little as half a micron of aluminum coating to the alloy using these coatings. Ti alloys can be used to higher temperatures provided they are not limited by some other consideration such as creep.

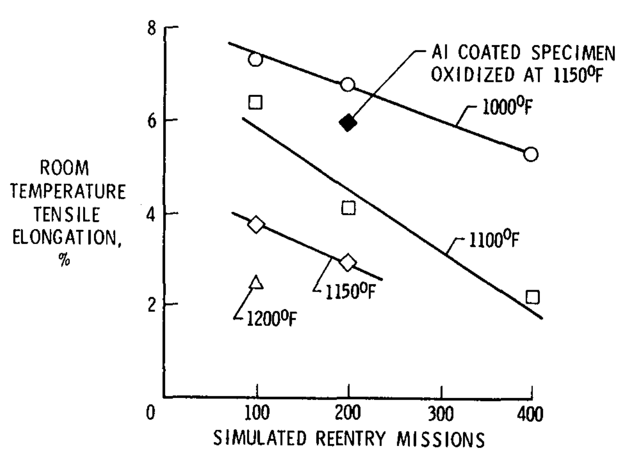



**Fig. 5.- Effect of aluminum coating on oxygen embrittlement of titanium alloy.**

New material systems which have high potential for improving the performance of hot structures include: rapid solidification rate (RSR) technology [49, 50], development of ductile intermetallic compounds [51], formulation of new dispersion strengthened alloys [52], and development of metal matrix composites. RSR technology will likely result in a new family of alloys that will have dramatically improved elevated temperature properties and retain these properties for much longer periods of exposure time because of better stabilization of the microstructure. Typical of the level of property improvements possible by RSR technology is the data for aluminum alloys shown in fig. 6. Similar improvements in titanium alloys are possible and research is underway to develop RSR Ti alloys.

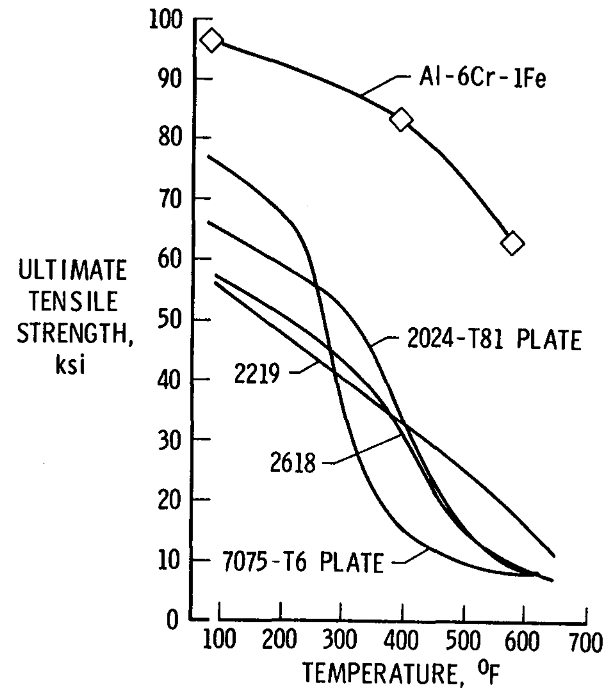



**Fig. 6.- Ultimate tensile strength as a function of temperature for developmental aluminum alloys.**

Metal-matrix composites (MMC) are a relatively new class of materials which can be produced with specific strength and stiffness properties superior to those of monlithic metals, and frequently to those of polymer matrix composites, especially at elevated temperatures. The ability to tailor both mechanical and physical properties is a unique and important feature of these materials. The principal reinforcement for metals include continuous fibers, discontinuous fibers, whiskers, and particulates. Key continuous fibers include graphite, silicon carbide, alumina, and boron. Because these fibers have good high temperature strength, and stiffness MMC made with these fibers can have outstanding creep resistance and elevated temperature strength and stiffness. These features make MMC an important new class of materials for hot structures. However, additional resources are required to address a significant number of technology issues to permit successful utilization of these materials in aerospace vehicles.

Carbon-Carbon

Carbon-carbon composites are attractive candidate materials for hot structures and thermal protection systems for future launch vehicles because of their strength retention at high temperatures. The effect of temperature on the specific strength of several classes of high-temperature materials [53] is shown in fig. 7. Three levels of carbon-carbon strength efficiency are shown. Reinforced carbon-carbon (RCC) was developed for the Space Shuttle Orbiter's nose cap and wing leading edges (baselined in 1973) [54].

Even though this material is made from low-strength rayon-based carbon fibers, its strength efficiency is superior to both superalloys and ceramics at temperatures higher than 1800°F. Development of Advanced Carbon-Carbon (ACC) under the sponsorship of NASA-Langley [55] resulted in a 100 percent increase in in-plane strength over that of RCC. The ACC material which is currently being evaluated by a number of different laboratories is made using woven carbon cloth made with PAN-based graphite fibers. If unidirectional carbon fiber tapes are interplied with woven cloth to create a hybrid ACC, the strength in at least one direction can be increased to 50,000 psi or more. However, the use of unidirectional tapes could degrade interlaminar properties.

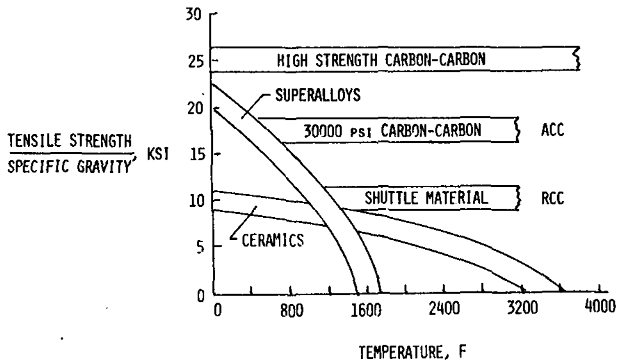
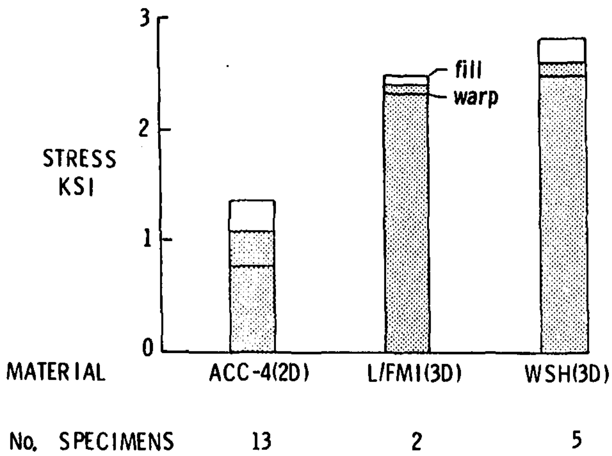



**Fig. 7.- Comparison of carbon-carbon specific strength with other high temperature materials.**

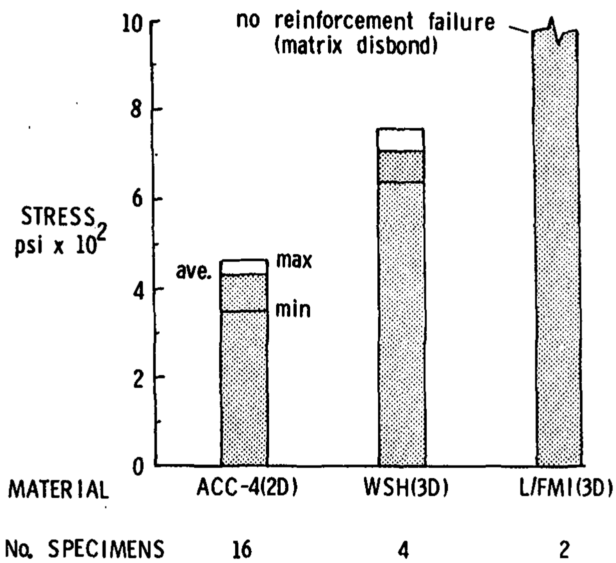
Attainment of higher strength was one of the primary objectives behind the research leading to the development of advanced carbon-carbon (ACC). Although significant improvements in in-plane strength were achieved by utilizing stronger fibers and a more efficient weave architechture, the matrix-dominated out-of-plane properties are somewhat lower than for RCC. Weak interlaminar properties could become a major limitation in the overall design of a structural part and prevent full utilization of the available in-plane strength. Approaches for improving interlaminar properties include alternate fiber and matrix materials, enhancement of fiber-matrix interface bonding, variations in composition and processing to impart higher through-the-thickness strength, and use of three-directional (3-D) reinforcement.

The latter two approaches are currently under investigation at NASA Langley to improve out-of-plane tension and interlaminar shear strengths of carbon-carbon composites. To date, two types of three-directional carbon-carbon composites have been fabricated and tested along with two-directional advanced carbon-carbon (ACC-4(2-D)) [56]. The first 3-D material was fabricated at Langley using a 3-D preform of polyacrylonitrile yarns from Fiber Materials, Inc.; hence the designation L/FMI (3-D). The architecture is a true 3-D orthogonal weave with fibers in the warp, fill and thickness (Z) directions. The second 3-D material was fabricated by Woven Structures-Hitco (WSH(3-D)). The architecture is described as an angle-interlock weave and has no fibers oriented parallel to the through thickness (Z) direction. The warp yarns pass through the thickness at about 45° angles and contain fewer fibers than in-plane yarns.

Fig. 8 shows out-of-plane tensile strength and interlaminar shear strength comparisons of three materials. For both properties, the 3-D materials had much higher strengths. For out-of-plane tensile strength, WSH(3-D) was 65 percent stronger than ACC-4(2-D). The true strength of the L/FMI(3-D) material could not be determined because of failure of the adhesive bond between the specimen and the test fixture. These failures occurred at stresses up to 975 psi, which is 130 percent stronger than ACC-4(2-D). For interlaminar shear strength, the 3-D materials were twice as strong. Thus, inclusion of fibers in the out-of-plane direction significantly improves out-of-plane mechanical properties.




(a) Interlaminar shear strength




(b) Out-of plane tension strength

**Fig. 8.- Strength comparisons of two-and three-directional carbon-carbon reinforcements.**

For application in oxidizing environments, such as on the Shuttle, carbon-carbon parts must be coated and sealed to protect them. For the Shuttle application the outer surfaces of the parts are converted to silicon carbide (SIC) in a high-temperature diffusion coating process. Because of differences in thermal expansion between the silicon carbide and the carbon-carbon part, the coating develops microcracks when the part is cooled from the coating temperature. To reduce the entry of oxygen through these cracks the SIC surface is coated with tetraethylorthosilicate (TEOS) which forms a viscous glass sealer.

This sealer is effective at the higher temperatures (1600-2900°F), but does not perform as well in the intermediate temperature range (1000-1600°F) [57-58]. In this intermediate temperature range, newly developed sealants show promise of dramatically reducing the mass loss due to oxidation. The Vought Corporation, under contract to Langley, has developed two techniques that have greatly improved oxidation resistance [55].

Fig. 9 shows mass loss after 10 hours of exposure at 1000°F for these two new materials compared to the Shuttle baseline material. The RCC/SiC/TEOS Shuttle baseline represents the state of the art in oxidation resistance. The ACC/SiC/TEOS/MAP is basically the Shuttle baseline with a substrate fabricated with PAN base graphite fibers instead of rayon base fibers and an additional overcoat of monoaluminum phosphate (MAP) applied to the surface. The addition of the MAP sealer reduced the mass loss rate to approximately 30 percent of the baseline. A further modification of the material was made which increased the oxidation resistance even more. The SiC coating was doped with boron. The doped and sealed coating (ACC/DSiC/TEOS/MAP) had 25 times the oxidation resistance of the baseline material in this temperature range. The sealed and doped material has potential uses in many other applications in addition to future Shuttle thermal protection systems.

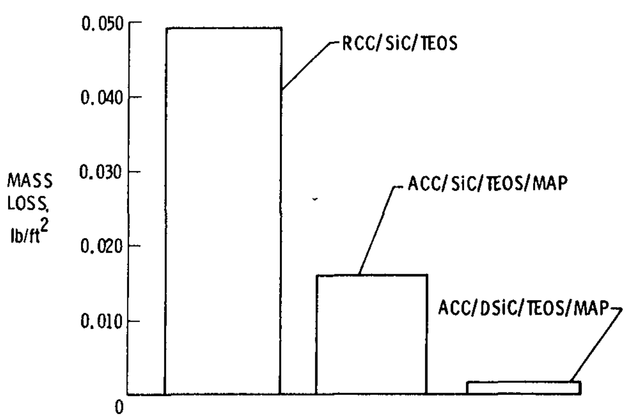



**Fig. 9.- Comparison of mass loss for three carbon-carbon material/coating systems after 10 hours exposure at 1000°F.**

The ACC substrate and coating research activities have lead to significant improvements in the oxidation resistance of coatings for ACC. Matrix strength retention after exposure to the expected use environment will continue to be an important criterion for coating improvements. Continued research on doped silicon carbide coatings and advanced coating sealants is expected to yield additional improvements in matrix oxidation protection and increase maximum use temperatures. Coatings with 3500°F use temperatures will be required for future applications.

Although progress has been made in developing improved coatings for carbon-carbon and approaches for obtaining higher interlaminar properties, essentially no progress has been made in simplifying the processing of carbon-carbon. For example, fabrication of the Space Shuttle Orbiter nose cap and wing leading edge components is a multi-step process [54] requiring hundreds of hours to complete. First the woven graphite fabric, which is preimpregnated with phenolic resin, is laid up as a phenolic-graphite laminate in a mold and is autoclave cured. Once cured, the part is pyrolyzed to form a carbon matrix surrounding the graphite fibers. The part is then densified by multiple furfural alcohol reimpregnations and pyrolyzations. The process is very time consuming. For instance, a single pyrolysis step may take more than 70 hours in an inert-atmosphere furnace. Research is currently underway to reduce the processing time and consequent cost of carbon-carbon parts.

If carbon-carbon is to be an economically viable structural material, new approaches for simplifying and shortening the processing time must be developed. Thus processing science is a key technology need for advancement of carbon-carbon technology. Development activities have been highly empirical in nature and have not resulted in a fundamental understanding of structure/property relationships. Fundamental research to acquire this type of data is needed if the full potential of carbon-carbon composites is to be realized. Such research is underway but additional resources are required to conduct these programs in a timely manner to make carbon-carbon technology available for the next generation of reusable launch vehicles.

## Advanced Structures/TPS

Material developments over the past 10 years provide the designer with several promising systems for saving weight. Also, advanced fabrication techniques such as superplastic forming and diffusion bonding [59] permit fabrication of low-cost, high-geometric-efficiency structures that were not possible with older materials and fabrication techniques. This area of structures technology has received more attention in the past 15 years than any of the other technology issues. Studies of hot structure [30, 60], insulated structure [61, 62], and, for hypersonic cruise applications, actively cooled structures [12] have included design, fabrication and testing efforts. The "best" concept depends on many variables and the criteria used to determine what is best. Future vehicles are likely to be a mix of various concepts if minimum mass is a significant driver and the thermal environment is extremely severe. Typical efforts for hot structures and durable thermal protection systems will be briefly reviewed in the following sections.

Concepts Development Chronology

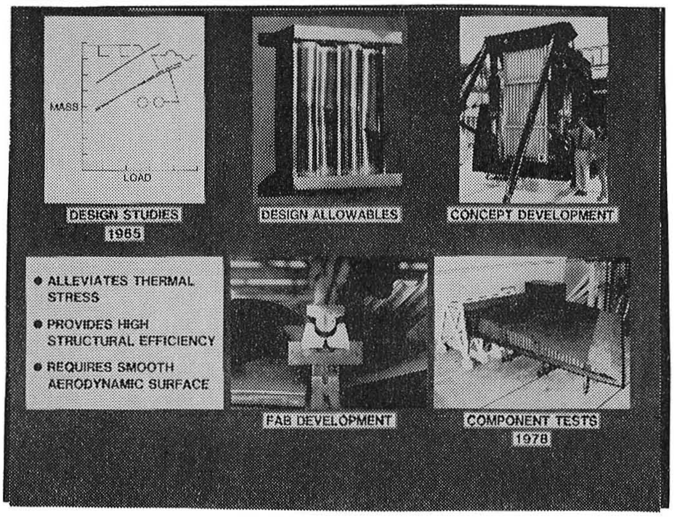



**Fig. 10.- Primary structure concepts development chronology for tubular panels.**

Considerable effort has gone into the development of geometrically efficient structural concepts for high-temperature applications. One concept that uses curved elements to provide axial stiffness while alleviating transverse thermal stresses was conceived in 1965 as part of a hypersonic aircraft study [61]. The concept has undergone subsequent development, shown in fig. 10, including design optimization, fabrication development, structural allowables testing, and large component testing under combined mechanical and thermal loads. The latter were completed late in 1978 in the Flight Loads Research Facility at the Dryden Flight Research Facility. During these tests the wing was subjected to combined thermal and structural loads, including structural temperatures up to 1340°F and loads up to the maximum design conditions (fig. 11). Analysis of the data [62] indicates that the structure performed as expected and it may be concluded that curved-element hot structures represent a mature technology ready for flight demonstration. This same process is required to evaluate other concepts and will be an expensive undertaking.

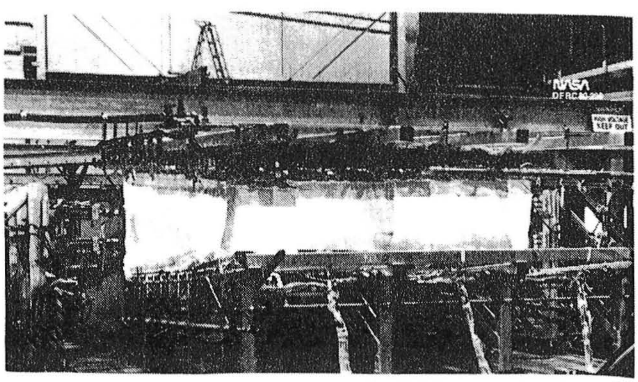


**Fig. 11.- Hypersonic wing test structure heating test at Dryden Flight Research Facility.**

The complete process from conceptual design to large component testing took 13 years. This time period is excessive as the work was done while hypersonics technology was on the decline, and such efforts were given low priority. Although the time can be suppressed significantly if the activity is given sufficient priority and support, the overall process is still time consuming. Taking "shortcuts", on the premise you don't have to know everything about the current activity before proceeding to the next phase of work, often leads to unanticipated problems and longer performance periods than if the work were done in a systematic step-by-step manner. Program planners must avoid falling into the trap of thinking "nine women can make a baby in a month."

Efficient Panel Geometries

For a structural panel to have maximum geometric efficiency, the principal load-bearing area (caps) should be symmetrical about the neutral axis, have a high local buckling coefficient (curved caps and clamped edges), have a low density web between the caps (low core density), and have its core material supporting load. The three panel geometries shown in fig. 12 have improved geometries from left to right. The tubular panel satisfies three of these factors, but not the low core density. The beaded web corrugation does not have a load-bearing core whereas the truss-core web corrugation satisfies all four geometry efficiency factors. Experimental results were in good agreement with structural analysis for the first two panel types. The truss-core web configuration is the subject of current design and fabrication research. More detailed results on the tubular and beaded web configurations are given in [62-64].

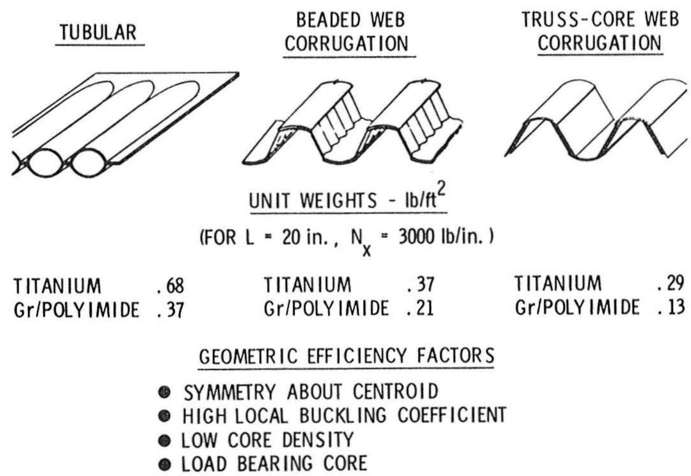



**Fig. 12.- Geometrically efficient structural panels.**

Carbon-Carbon Control Surface Structure

"Carbon-carbon" material has attractive features for applications to future reusable launch vehicles, particularly for control surfaces that are subjected to relatively low mechanical and high thermal loads. These materials retain their strength at high temperatures and have a lower density than aluminum. Carbon-carbon is currently used as a load-bearing heat shield on the nose cap and wing leading edges of the Space Shuttle orbiter.

The Shuttle Orbiter aft body flap was selected for a conceptual design study to determine the feasibility of using carbon-carbon hot structure for lightly loaded control surfaces. The design uses only carbon-carbon components to reduce weight and to eliminate the inherent thermal expansion mismatch between metallic and carbon-carbon structures. The flap is approximately 21 feet wide, 7 feet long, and 20 inches deep, and is connected to the Orbiter at four actuator attachment points. A section of the baseline body flap and that of the carbon-carbon flap are shown in the lower left of fig. 13. The baseline design consists of upper and lower honeycomb core panels which are supported by aluminum ribs every 20 inches and connected to a full depth honeycomb core sandwich trailing edge. The aluminum structure is protected from entry heating by thick reusable surface insulation (RSI) tiles on both the lower and upper surfaces.

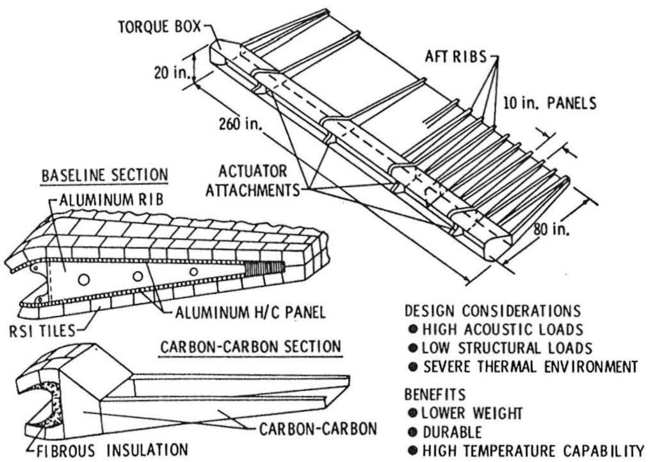



**Fig. 13.- Conceptual design of carbon-carbon body flap.**

The carbon-carbon body flap design concept consists of a torque box and tapered, flanged ribs which support the continuous lower skin. Each rib extends into the torque box to form a bulkhead. Because the leading edge of the body flap is sealed, no significant air flow passes over the upper surface of the flap, and therefore the upper skin was removed to save weight, and to allow more heat to be radiated from the lower skin, thereby reducing peak temperatures.

There are several key considerations for the body flap design. The body flap is subjected to high acoustic loads from the main engines during liftoff. The structure is lightly loaded by aerodynamic pressure, but is subject to high entry heating. The concept has been sized for static loads and the predicted weight is 610 pound, 850 pounds less than the current insulated aluminum body flap weight of 1460 pounds. Analysis shows the carbon-carbon body flap to be stiffer than the baseline body flap, and will have a peak temperature of 2370°F, 330°F less than the baseline body flap. More detailed information on the body flap design is given in [65].

An initial concern was the joining of carbon-carbon with metal fasteners. However, a fastener can be shaped to eliminate the thermal stress which would otherwise result from differential thermal expansion between dissimilar fastener and sheet materials for many combinations of isotropic and orthotropic materials. A theoretical basis has been developed for the design of such fasteners [66]. In general, such a fastener has curved sides; however, if both materials have isotropic coefficients of thermal expansion, a conical fastener is free of thermal stress. Experimental results for a conical fastener are given in [67].

Durable TPS Concepts

Although the Reuseable Surface Insulation (RSI) currently used on the Space Shuttle is an excellent insulation, it may not be durable enough for future applications. However, extensive work is underway at NASA Ames Research Center to develop more durable systems, both rigid and flexible, as well as higher temperature use capability [68, 69]. Fig. 14 summarizes a program to develop a more durable Thermal Protection System (TPS) using metallic concepts for temperatures from 700°F to 2000°F, and using Advanced Carbon-Carbon (ACC) above 2000°F [70]. The goals of the program are to develop TPS that have durable surfaces, are mechanically attached, have covered/blocked gaps between panels to reduce gap heating, and are mass competitive with current Shuttle TPS. The graph in the figure shows that the durable TPS concepts indicated by the symbols are mass competitive with RSI indicated by the cross-hatched area.

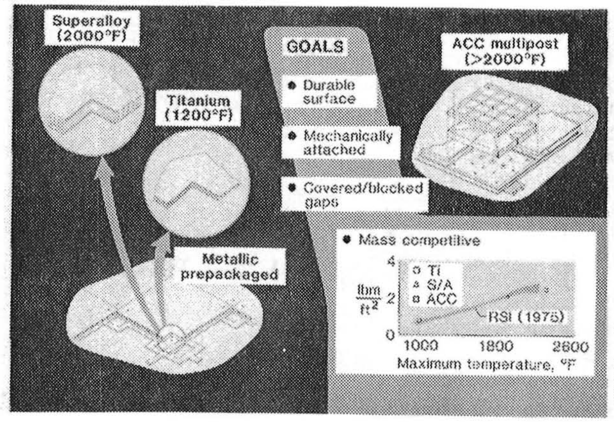



**Fig. 14.- Metallic and carbon-carbon TPS concepts.**

Metallic TPS Concepts: The two metallic prepackaged concepts are discrete panels that have a strip of RTV-covered Nomex felt beneath the perimeter of each panel to prevent hot gas flow beneath the panels. The titanium multiwall concept (maximum surface temperature < 1200°F) consists of layers of dimpled titanium foil Liquid Interface Diffusion (LID) bonded together at the dimples with a flat foil sheet sandwiched between each dimpled sheet. The superalloy honeycomb concept (maximum surface temperature 2000°F+) consists of an Inconel 617 honeycomb outer surface panel, layered fibrous insulation, and a titanium honeycomb inner surface panel. The edges of the two metallic concepts are covered with beaded closures to form discrete panels nominally 12 inches square.

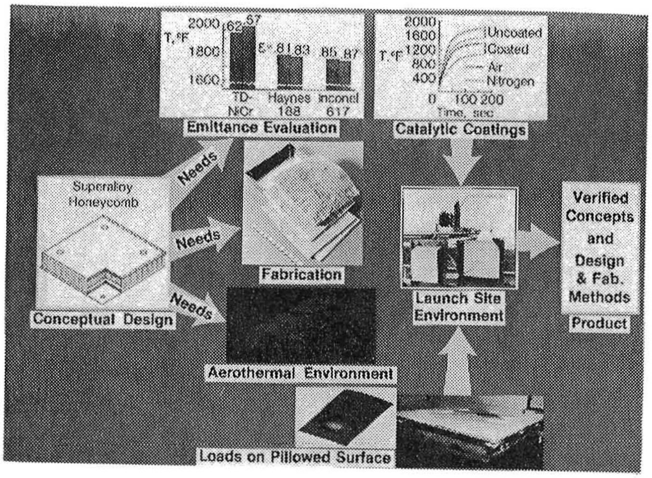



**Fig. 15.- Interaction of disciplines - example for metallic TPS development.**

Fig. 15 shows the highly interdisciplinary nature of the design process for TPS, which really are systems requiring a systems approach for successful design. This lesson was painfully learned during the Shuttle RSI tile development [71]. Starting with a concept, illustrated here by a prepackaged superalloy honeycomb concept, the designer needs to consider material performance such as surface emittance and catalysis, fabrication developments, and various environmental tests such as aerothermal and launch site contamination, durability to resist impact, and lightning strike, etc. Since a metallic TPS will have gaps and usually have a wavy surface, or as in the example, deform into a pillowed surface, such surface roughness effects on loads and heating must be determined. The product of such research should be verified concepts, and verified analytical design and fabrication methods.

One of the objectives of aerothermal tests is to determine if temperatures in the gaps between panels will be increased by exposure to the flow. Such an increase would indicate that the panel edge overlap which covers the gap is not adequate by itself to prevent gap heating when the flow is parallel to the gap.

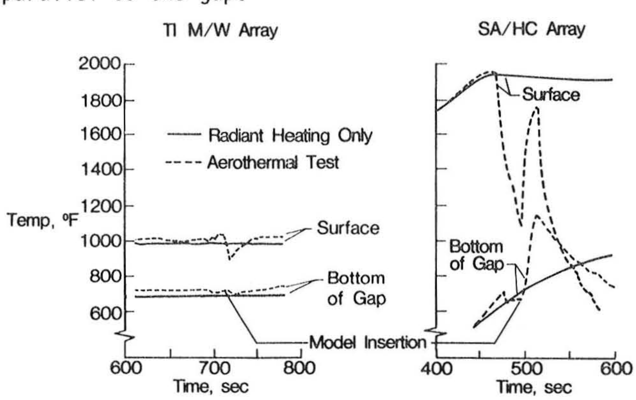



**Fig. 16.- Effect of aerothermal exposure on TPS gap temperatures.**

Surface temperatures and temperatures at the bottom of a gap are shown in fig. 16 for 20-panel arrays of titanium and superalloy TPS panels tested in the Langley 8-Ft. High Temperature Tunnel (HTT). The dashed curves show temperatures during a 200 second portion of an aerothermal test when the array was inserted into the tunnel stream. The solid curves show temperatures recorded at the same locations and time intervals during a static radiant heating test. At the time interval shown, the surface and gap temperatures of the titanium multiwall model were at equilibrium. When the model was inserted into the flow, negligible temperature perturbation occurred at the bottom of the gap thus indicating no additional gap heating occurred. However, immediately after the superalloy model was inserted into the flow, the temperature at the bottom of the gap increased to a level considerably greater than it was before the tunnel started. This high, quick temperature rise indicates that hot gases flow in the gaps between panels. Thus, when the edges of the superalloy panels were parallel to the flow, the overlapping edges do not provide an adequate seal. Superalloy honeycomb panels may be more susceptible to gap heating because the gap is much larger than the gap between titanium multiwall panels. Consequently, when thermal expansion closes the top of the gap, the bottom of the gap remains partly open because it is much cooler.

Even though much of the surface of Shuttle-type reusable launch vehicles is flat or nearly flat, some locations, such as the chine areas, are necessarily curved. The fabrication of curved TPS panels often presents complexities not encountered in fabricating flat panels, and the design of curved panels must include large surface pressure gradients and factors contributing to thermal stress which are normally not important in the design of flat TPS.

A curved titanium multiwall panel has been fabricated to demonstrate that the multiwall concept will lend itself to curved panels, and an array of curved superalloy panels was fabricated for aerothermal tests to evaluate their performance in a high-surface-pressure gradient environment. The curved 20-panel array shown in fig. 17 was installed into the cavity of the Curved Surface Test Apparatus (CSTA), so that the surface of the array was flush with the surface of the CSTA. The array was tested in the 8-Ft. HTT to determine if heating occurs in the gaps between panels. Based on the results for flat panels, metal tabs, one of which is identified on the single panel in the figure, are located at the corner intersections of the panels to block flow in the gaps.

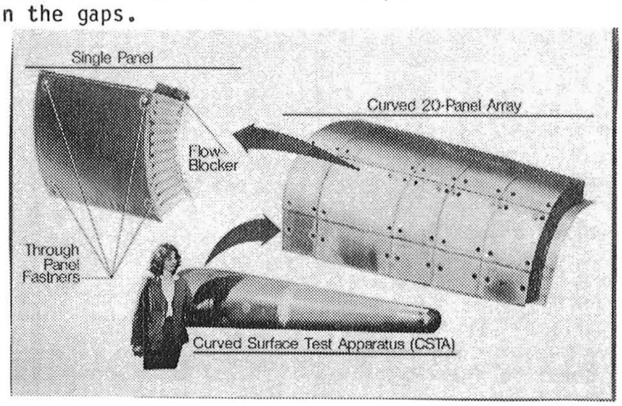



**Fig. 17.- Curved superalloy honeycomb TPS panels.**

Results from the tests (which were completed July 1985) indicate that high heating occurred in the gaps between panels. However, Nomex felt strips beneath the panel perimeter were not properly coated with RTV to eliminate porosity, and the aft seal of the cavity into which the array was installed failed. Preliminary evaluation of the data indicates that the cause of the high gap heating cannot be distinguished between the imposed surface pressure gradient, leakage through the cavity seal, and/or leakage beneath the panels due to the porous felt. Before making additional aerothermal tests, model and cavity seal repairs are necessary. Modifications of the concept to further inhibit gap heating are also under consideration. Once the gap heating problem has been resolved for flat and curved surfaces, the more difficult problem of intersecting surfaces needs to be addressed.

Carbon-Carbon TPS Concept: The Advanced Carbon-Carbon (ACC) multipost concept (maximum surface temperature > 2000°F) consists of a rib-stiffened ACC sheet attached to the vehicle primary structure by posts with fibrous insulation packaged in a ceramic cloth between the ACC panel and the vehicle structure. (Venting and waterproofing of the insulation package is a problem, but currently is not being studied.) The surface of the single ACC panel is nominally 36 inches square. A 1 ft. by 2 ft. ACC test model (fig. 18) represents the intersection of four panels, and has a joint design intended to preclude gap heating. The test model was subjected to thermal vacuum tests and aerothermal tests in the NASA Langley 20MW Aerothermal Arc Tunnel. Conditions were selected which gave a 2300°F surface temperature on the front of the model. The hot gas flow was 45° to the edges of the panel.

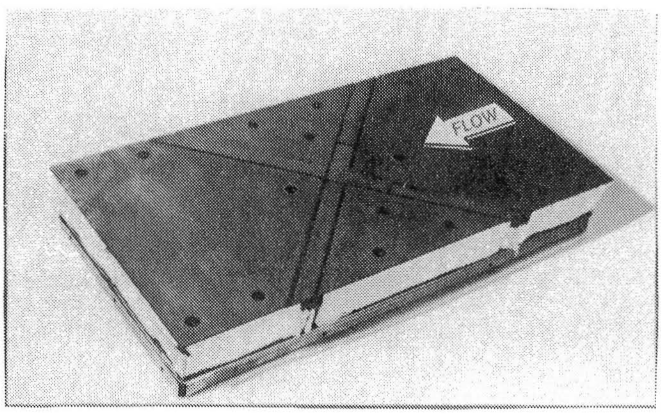



**Fig. 18.- ACC multipost TPS test article.**

Temperatures obtained during the arc-tunnel tests and those obtained during a thermal/vacuum test were compared; the comparison indicated that slight heating due to flow occurred in the gap region where one panel overlaps another [70]. The ACC model was not damaged by either the thermal/vacuum or arc-tunnel tests.

Future Efforts

Considerable work, requiring considerable time and resources, is required to develop efficient structural/TPS concepts for the various missions required by the National Space Strategy. This effort includes design optimization, fabrication development, structural allowables testing and large component testing under combined thermal and mechanical loading conditions and possibly under hypersonic flow conditions. This type of effort will be iterative in that a discovery in one phase of work may require modifications to an earlier phase of work. An example of this iterative nature is given in [72] where optimized conceptual designs based on the relatively new difussion bonded fabrication process had unanticipated low fatigue life discovered in the test phase. The poor fatigue life was due to the stress raisers caused by the very small fillets that result when members are joined by diffusion bonding which required reoptimizing a very detailed part of the design.

Carbon-Carbon is a promising material for TPS applications and for lightly loaded structures such as control surfaces. However, considerable work is required including the difficult hurdle of scaling up from research size to full size components.

Cryogenic Tanks

The cryogenic fuel used by the only existing reusable Space Transportation System (STS), the Space Shuttle, is carried in the expendable external tank, but future systems designed for full reusability will undoubtedly carry their own cryogenic fuels internally. Consequently, structural design of new fully reusable systems must necessarily address problems associated with containment of cryogenic fuel and LOX as well as the conventional considerations of thermal protection and support of vehicle mechanical and thermal structural loads.

Since reusable launch systems will become virtually flying cryogenic tanks, the design of such structure will dominate the airframe structures effort for such vehicles. The primary problem, which distinguishes hydrogen tanks from other cryogenic tanks, is the proclivity of liquid hydrogen to condense other gases because of its extremely low temperature (-423°F). Air, or any purge gas other than helium, condenses on the tank surface and produces a partial vacuum which pumps additional gas to the surface where it is condensed. This cryopumping (as it is called) transmits large quantities of heat to the fuel causing hydrogen boiloff and, if the gas is air, produces a potential safety hazard because of the selective liquefaction of oxygen from the air. No large, flightweight, reusable cryogenic tank has ever been flown; in fact only one has ever been built [73]. The tank was a double-bubble (lobed) non-integral half-scale (6000 gal) tank. It was subjected to limited testing and no combined loads (thermal and structural) tests were conducted. A variety of hydrogen tank concepts have been proposed in conceptual studies in the past [28-31] including many which have not been documented. However, the technology for such concepts has received little attention, and none of the concepts have been proven completely acceptable for multiple reuse applications. Thus cryogenic tanks are one of the key and least developed technologies for reusable launch vehicles.

Both insulated and hot-structure design approaches have been studied. A hot-structure concept [24] for a single-stage-to-orbit vehicle is shown in fig. 19. This concept has followed the design philosophy of accepting the recently developed Space Shuttle main engines as the propulsion system and striving for improvements in airframe structural mass fraction. The integral tank/fuselage structure combines the functions of fuel containment, thermal protection, and support of vehicle thrust and aerodynamic loads. The vehicle is designed for a low planform loading, which results in a higher altitude entry trajectory than that flown by the Shuttle Orbiter. This high-altitude, gliding entry results in maximum surface temperatures of about 1400°F which are within the operating range for the nickel-base superalloy Rene' 41.

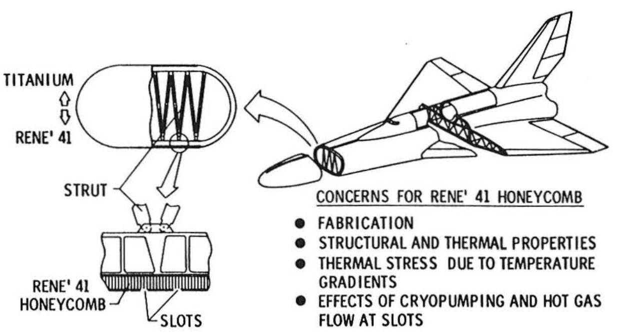



**Fig. 19.- Integral tank/fuselage hot structure concept.**

Integral Tank/Fuselage Hot Structure Concept

The structure (fig. 19) consists of a vacuum-sealed-cell honeycomb sandwich with the inner skin of the fuselage at a temperature of -423°F due to exposure to the cryogenic fuel and the outer skin at a temperature of 400°F due to exposure to the boost aerothermal environment. These temperature gradients produce large thermal stresses which must be accommodated in the design. The thermal stresses have been partially relieved by slotting the outer face sheet on the windward surface of the vehicle fuselage. These slots eliminate biaxial compression stress which would otherwise occur from thermal and pressure loads. The slots are sized to be nearly closed when the outer surface of the panel is heated to 1400°F. The effect of these slots in the cryogenic environment during ground hold and boost and in the hypersonic environment during entry was a concern. Pressure loads in the noncircular section are carried by tension struts at each frame location. Although Rene' 41 (or another superalloy) is required on the hotter, windward surface, a material with a better strength-to-weight ratio may be preferred on the cooler leeward surface to save weight. (The study of [24] considers using titanium honeycomb on this surface.)

Combined Loads Test: A Rene' 41 honeycomb panel, 1 ft. by 6 ft., was tested under combined thermal and bending loads at the NASA Dryden Flight Research Facility. The purpose of these tests was to evaluate the life of a panel when exposed to cyclic combined thermal and mechanical stresses representative of high elastic stresses seen at a fuselage frame attachment. The test apparatus and the test panel are shown in fig. 20.Quartz lamp heaters were used to produce temperature histories representative of both boost and entry cycles. For safety reasons, the cryogenic temperature required on the inner face sheet during the ascent cycle was represented by using $LN_2$ at -320°F. The use of $LN_2$ in place of $LH_2$ has only a small effect on the thermal strains which occur during the test.

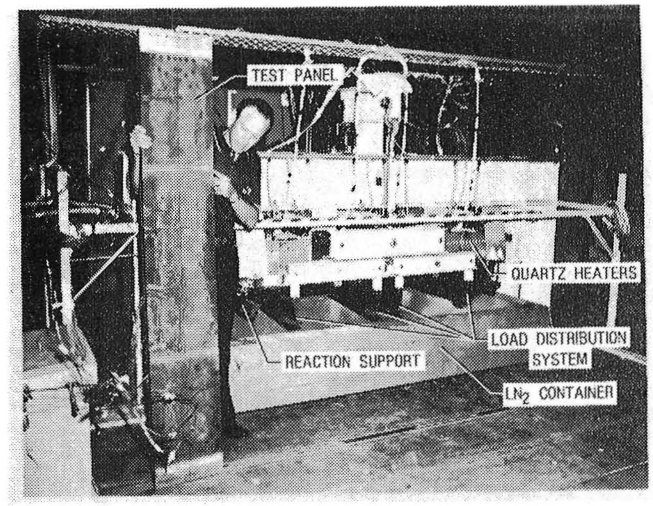



**Fig. 20.- Combined loads test fixture and panel.**

The panel was exposed to 500 boost cycles and 500 entry cycles. For each cycle, the mechanical load was held constant while the thermal load was applied. The maximum compressive strains were up to 80 percent of the proportional limit. After the first entry cycle, a small bow in the panel in the longitudinal direction was observed. This residual bowing gradually increased with additional exposure to entry cycles, but it was unaffected by additional boost cycles. No damage occurred as a result of the 500 boost and entry cycles, but the panel was left with a permanent center displacement of 0.58 in.over the length of 6 ft. with the concave face on the hot side. Additional details are given in Reference [30].

$LH_2$/Boost Heating Environment: A 21x25 in. panel was fabricated under NASA sponsorship with several slots in the outer skin and tested to evaluate the effects of the slots in the boost environment. The panel formed the bottom of a container which was partially filled with $LH_2$. A frame was brazed to the inner skin on the panel centerline. Quartz lamps were used to expose the panel to the boost heat cycle. Hold times between cycles ranged from 10 min to 1 h during which cryodeposits accumulated on the panel surface and in the slots. The panel was inspected after cycle 10 and cycle 30, and no damage to the panel was observed. After 36 cycles, a fire occurred in the test fixture but caused no damage to the panel. The panel was examined visually and by X-ray and C-scan. Sections cut from the panel were examined by metallographic inspection. No structural damage was found.

During the hold times water frost was observed depositing on the -200°F panel surface, and temperatures less than -300°F measured during the tests indicated that liquid air formed in the regions of the core open to the atmosphere. These results indicate that proper attention must be given to sealing honeycomb core splices to prevent passage of air into the core from the slots. Without such sealing, considerable liquid oxygen may flow within the honeycomb structure. The results also show that a honeycomb core sandwich with a slotted outer skin integrally fixed to an inner frame can withstand the localized thermal environment imposed by the boost trajectory proposed in [24].

The USAF is sponsoring further work in this area. The program is a three-phase program. Phase

I was a preliminary design analysis and trade study phase. The results include a review of the state of the art materials, manufacturing technology and structural design concepts. At the end of Phase I a panel design was selected for detail design and fabrication in Phase II. The design selected for fabrication is a 30x80 in. panel of brazed Rene 41 honeycomb. The large panel is made by electron beam welding together four smaller panels each approximately 15x40 in. During Phase II extensive work was carried out to optimize the braze process. This work was oriented at developing a process for brazing large panels in a production environment. The panel being fabricated is to be a flight weight panel. A detailed weight tracking program is underway to assess the final panel weight for comparison with previously developed estimated weights. The panel will be tested in a simulated ascent and reentry environment. The panel will be subjected first to 100 simulated reentry cycles followed by 100 simulated ascent cycles. After completion of this testing by the contractor the panel and associated test fixtures will be delivered to the Flight Dynamics Laboratory where the panel will be subjected to additional testing of 400 ascent and 400 reentry cycles in order to demonstrate the 500 cycle design life of the panel. The contractor testing will be carried out during the last quarter of CY-85. Air Force testing will be carried out during CY-86.

Mach 7 Aerothermal Tests: A slotted panel was tested to evaluate the effect of localized heating in the region of the slots during entry. The panel was designed to be exposed to a Mach 7 stream in the Langley 8-Ft. HTT. Two slot cover concepts were evaluated in addition to the open slot. Preliminary results from these tests (completed in January 1985) indicate that no significant heating occurs in the slots since the thermocouples located directly beneath the slots showed no unusual temperature rise during exposure to the hypersonic flow. However, television cameras recorded greater brightness at the surface of the slots which suggests that the slots augmented heating near the panel surface.

## Integral and Non-Integral Concepts

Sections through two typical proposed fuselage and tank walls of a reusable launch vehicle are shown in fig. 21. The packaged fibrous insulation for reduced heating to the tank and structure is common to both concepts. However, the amount of fibrous insulation required varies depending on the temperature limit of the exterior tank surface. In both concepts, the tank walls are welded to provide leak-free containment of the cryogenic propellants.

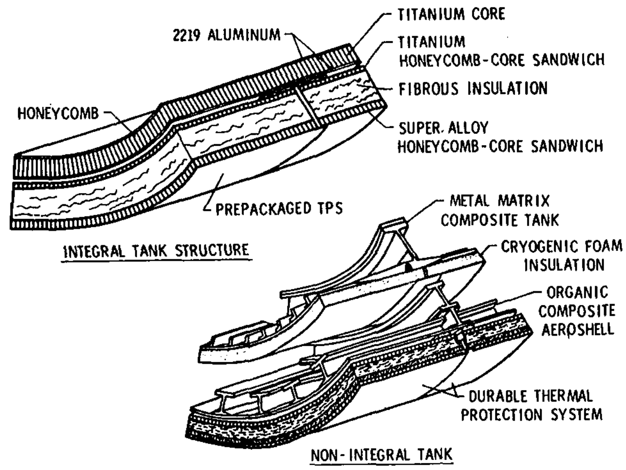



**Fig. 21.- Integral and non-integral tank/structure/TPS concepts.**

The integral tank concept uses a near-art evacuated honeycomb core sandwich structure protected by a durable prepackaged TPS [70]. The honeycomb carries both tank and fuselage loads and provides the insulation for the cryogenic liquid. The honeycomb sandwich structure has aluminum face sheets supported by ring frames but the core material is titanium to reduce conduction between the honeycomb faces. The core thickness is varied to provide the required cryo insulation function. The aluminum faces are compatible with the propellants or LOX and the honeycomb structure is more efficient than a more conventional Z-stiffened structure.

The non-integral structure is a more advanced concept using a welded metal-matrix composite tank with a 400°F closed-cell organic foam providing the cryogenic insulation and an organic composite fuselage structure. The foam thickness is varied to meet the insulation function. The foam may be on the interior of the tank wall, for $LH_2$. However, the organic foams burn in the presence of LOX with very little ignition energy. Therefore, for LOX tanks the foam is required to be on the exterior tank wall.

## Future Efforts

Most work involving hardware in the past 10 years has been on a hot integral tank concept. Tests to date on flat panels with stress relieving gaps on the outer surface have revealed no structural problems or augmented heating in the gaps, but ice and LOX could form during ground hold which could be an operational problem. There has been no effort to date on curved panels. This concept is limited to maximum temperatures of about 1400°F and hence is not applicable to some reusable launch vehicle missions currently under study. A number of paper studies have been done for TPS/structure/tank concepts which could be required for the higher temperature missions. A wide variety of concepts have been studied, only two of which were reviewed herein. If the concept needs a cryogenic insulation, development of a reusable, high temperature (> 175°F) material is a high priority technology need. Flaw-size crack-growth criteria for aircraft applications could impose big weight penalties on some cryogenic tank designs and a reassessment of the criteria is needed.

Additional research is required 1) to ascertain and improve the compatability of the tank materials with the fuel and oxidizer, and 2) to develop the facilities and techniques for fabricating and verifying the thermostructural performance of large tank components. Compatibility with $LH_2$ and LOX (especially the latter) is a fundamental problem that may impact a material system selection. Some preliminary examinations indicate that aluminum matrix composites may be suitable, and beneficial from a mass standpoint. However, data for aluminum matrix composites for such applications is limited, and additional materials development and characterization would be required. Considerable resources will be needed to satisfy the second requirement, both in developing the techniques for forming and joining panels into a complete tank structure and for verifying the thermostructural performance of load-carrying cryogenic tanks. Performance verification will require test facilities that can handle cryogenic hydrogen and simultaneously load and heat large tank structures. Such capability will be required regardless of the thermostructural concept that is ultimately selected for future space transportation vehicles.

## Hypersonic Airbreathing Propulsion Structure

Work on hydrogen-cooled engine structures at the Langley Research Center began with the Hypersonic Research engine (HRE) Program of the 1960's which culminated, from a thermal/structural standpoint, in tests of a complete flight-weight hydrogen-cooled Structural Assembly Model (SAM) in the Langley 8-Ft. HTT. These tests [74] confirmed the suitability of the basic approach for research purposes. However, two major thermal/structural problems were uncovered that must be solved before a hydrogen-cooled scramjet can become a practical reality: (1) the coolant requirements must be reduced (the HRE required almost three times as much hydrogen for coolant as for fuel) and (2) the thermal fatigue life must be increased (HRE had an anticipated fatigue life of only 135 operational cycles). Both of these problems stemmed, at least in part, from the annular design and high compression ratio of the engine which resulted in large areas being exposed to an intense heating environment. A fundamental goal of the continuing research program was to develop an engine concept which required only a fraction of the total fuel heat sink for engine cooling.

In parallel with the HRE project, a major engine manufacturer conducted a comprehensive study of regeneratively cooled panels under NASA sponsorship [75, 76]. These studies included analysis, fabrication and test and were intended to define the problems associated with the design and fabrication of structurally efficient regeneratively cooled panels. In this program the coolant was hydrogen and the panel loading conditions were representative of the internal and exteneral surfaces of hypersonic aircraft. Material selection was found to be an important consideration in the design of the cooled panels. In addition to the usual requirement for oxidation resistance and high-temperature strength, the selected material must be compatible with the coolant (hydrogen) and with the forming and joining methods employed. The heat exchanger material selection is strongly influenced by the elevated-temperature ductility which is a primary factor in determining thermal fatigue life.

Studies in the 1970's of airframe-integrated scramjets with high potential performance led to the sweptback, fixed-geometry, hydrogen-fueled, rectangular scramjet concept shown in fig. 22. The scramjet modules are integrated with the airframe and a number of aerodynamic/propulsion advantages are obtained with this concept [77]. Structural advantages include the fixed geometry and reduced wetted surface area and heating rates. By 1971 propulsion technology for the airframe integrated scramjet had advanced sufficiently to warrant development of the required advanced thermal/structural technology. A preliminary thermal/structural design analysis study [78] based on HRE technology indicated viability from both an engine structural mass and coolant requirement standpoint. This study revealed a number of critical areas (e.g., panel-to-panel seals, fuel injection struts) and reemphasized the need for advances in fabrication and materials technology to obtain reasonable structural life. A more detailed study of this scramjet concept was undertaken by a major engine manufacturer while the effort at Langley concentrated on the fuel-injection strut.

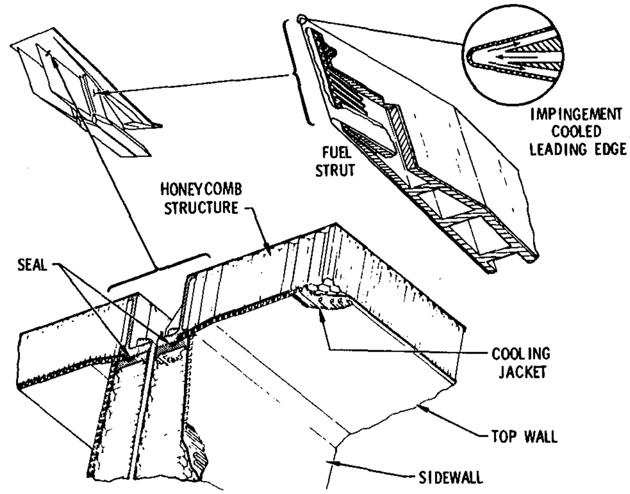



**Fig. 22.- Cooled scramjet structure concept.**

## Cooled Scramjet Structure

All engine surfaces wetted by the airstream are regeneratively cooled by circulating the hydrogen fuel through a cooling jacket before injecting the fuel into the combustor. The cooling jacket, which is brazed to the primary structure, consists of the aerodynamic skin and multiple straight-fin or pin-fin coolant passages; straight-fin passages are shown as part of fig. 22. Three basic engine shell concepts were investigated: two frame-stiffened honeycomb-core sandwich panels and a deep-core honeycomb sandwich panel. A deep-core honeycomb concept was selected as shown in fig. 22 as the baseline design primarily because it exhibits the least deflection in the sidewall and nozzle areas and is the least complex structure. Analytical results indicate relative displacements between adjoining components are generally small which permits the panel corners to be rigidly joined allowing the use of a simple static seal or even a welded corner.

The results indicate that the basic shell concepts have a significant temperature gradient through the thickness during thermal transients (maneuvers, combustion shutdown) which may significantly impact the final design of both the seals and basic shell structure.

The fuel-injection struts presented the most formidable cooling and structural problems. The struts must simultaneously support a large side load, contain high-pressure hydrogen at two temperature extremes, and withstand the high thermal stresses resulting from complex aerodynamic heating as well as convective heating from the hot hydrogen in the internal manifolds. To compound these problems the cross sectional area and contour cannot be altered without significantly changing the engine propulsion performance.

## Thermal Fatigue Life

The fabrication and material technology required to obtain reasonable thermal fatigue life for the cooling jacket was developed and experimentally validated [79]. The goal for the airframe-integrated scramjet is 1000 hours and 10,000 cycles of hot operation which represents an improvement of two orders of magnitude over the HRE. Analytical predictions of the fatigue life as a function of the temperature difference between the hot aerodynamic skin and the back surface are presented in fig. 23. The life goal appears attainable through a number of factors such as engine design, fabrication, and material selection. The improvements attributable to these factors are graphically illustrated in the figure. The bottom curve indicates the anticipated life of the Hastelloy X coolant jacket for the HRE. The solid symbol at the right denotes the HRE design point and the open symbols indicate experimental data. A fundamental change in engine design to decrease the heat flux intensity and thus the temperature difference, as indicated by the horizontal arrow, is the first factor to increase the life of the airframe-integrated scramjet. An additional increase, as indicated by the vertical arrow, is obtained through an advanced fabrication technique. In this technique, the fin coolant passages are photo-chemically etched into the aerodynamic skin which eliminates the strain concentration caused by local thickening of the skin by the fin and eliminates the hot skin to fin braze joint present in the HRE configuration. (The braze joint to the cooler primary structure remains, however.) The two candidate configurations fabricated by this process are shown in the figure. Finally, another increment in life is attained through the selection of a material with high thermal conductivity which decreases the temperature difference, and with high ductility which increases the fatigue life directly. To date Nickel 201 and Inconel 617 or 718 appear to be the most attractive materials.

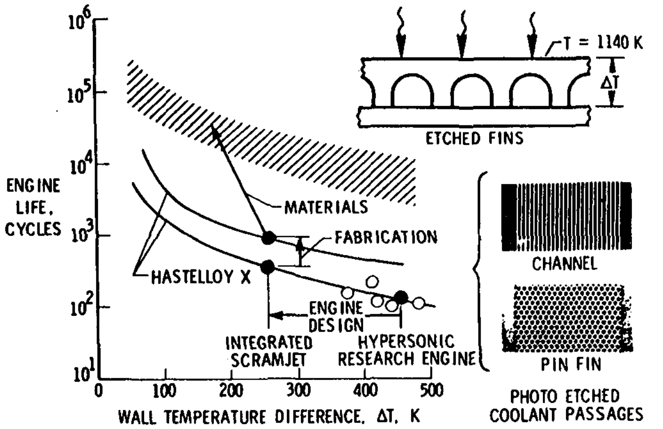



**Fig. 23.- Scramjet structure thermal fatigue life.**

## Future Efforts

A major engine manufacturer is currently building a fuel injection strut for testing in the NASA Langley Combustion and Mixing Research Apparatus. These tests are expected to be completed in late 1986 or early 1987. Additional studies are underway to re-examine the trades between rectangular duct and circular duct configurations, as systems studies of hypersonic airbreathing launch systems indicate propulsion structure weight is large, and thus a significant mission driver. Advanced propulsion structure is a high payoff technology deserving considerable R&T effort in the future. Continued development of materials, analysis and fabrication capability, detailed design studies and life cycle verification tests of various components in realistic environments, while expensive, is essential.

## Loads/Criteria

If the more demanding performance goals of future missions are to be met, the designer must have better loads definition than he has been given in the past to avoid both undesirable (possibly catastrophic) damage and weight penalties due to unduly conservative safety (or ignorance) factors. Also, design criteria established for aircraft or missiles may not be applicable to reusable launch vehicles, and the criteria developed for hypersonic vehicles in the 1960's must be examined to determine their current applicability.

## Detailed Aerothermal Loads

The Langley Research Center initiated a program in detailed aerothermal loads in the late 1970's. The rationale for this program is that there are detailed, generic surface irregularities such as gaps and protuberances that will exist on most high speed vehicles, yet there has been little systematic exploration of the flow disturbances and augmented heating caused by such irregularities. This lack of data led to excessive cost and delays in component development for the Shuttle project. Urgent requests for tests and analysis resulted in "reaction to crises" type studies rather than normal, methodical research and development efforts.

The Shuttle experience was not an exception, but is the rule all too familiar to project managers of previous and on-going high speed vehicle projects. Since such surface irregularities can be anticipated and defined in a generic sense, a combined experimental and analytical program is in progress to provide a data base and prediction/extrapolation techniques for use in design of future high speed vehicles. The type of efforts required will be illustrated by some of the work to date on understanding the flow in a wing elevon cove. This example is chosen because of the difficulty it poses for the analyst, and because higher than expected cove heating was encountered on Space Shuttle flights [80]. Post-flight inspection of STS-1 revealed thermally damaged insulation in the cove that required replacement by a material capable of withstanding higher temperatures.

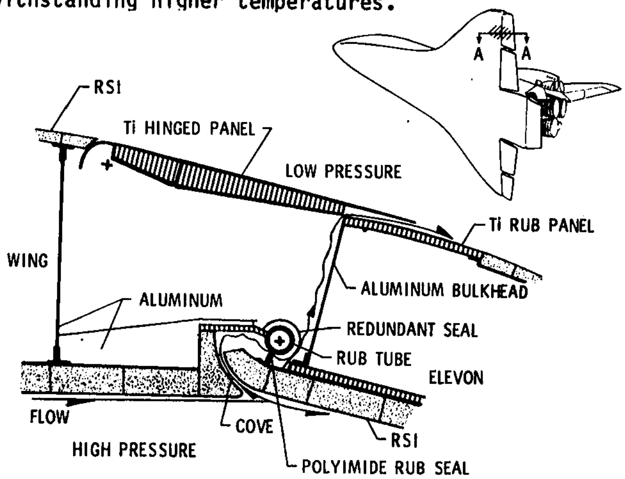



**Fig. 24.- Shuttle wing elevon cove.**

Wing Elevon Cove: The ingestion of hot gas and the possible disastrous consequence of the flow of these gases in control surface gaps in hypersonic flow has been a continuing concern in the design of the current Space Shuttle Orbiter. One specific concern was the flow in the spanwise gap or cove between the wing and the elevon. The potential problem for the wing elevon cove is depicted in fig. 24 which shows a sketch of the current Space Shuttle Orbiter and a simplified cross-sectional view of its structure at the juncture between the wing and elevon. Without the spring-loaded polyimide rub seal, differential pressure between the windward and leeward surfaces of the wing would drive a portion of the boundary-layer into the cove where it would contact the unprotected aluminum load-bearing structure. The environment inside the cove must be known in order to properly size insulation up to the rub seal, and to design the rub seal to avoid thermal distortions which could cause leaks. Experimental and analytical work on this problem at Langley is briefly reviewed below.

The model, illustrated by the sketch at the top of fig. 25, consisted of a fixed wing-cove housing, a rotatable elevon, and aerodynamic fences at the sidewalls to channel the upstream surface flow across the cove entrance. The cove channel gap height and radii duplicated that of the Shuttle Orbiter. Seal Leakage was simulated by rectangular slots in a rub seal located at the end of the channel.

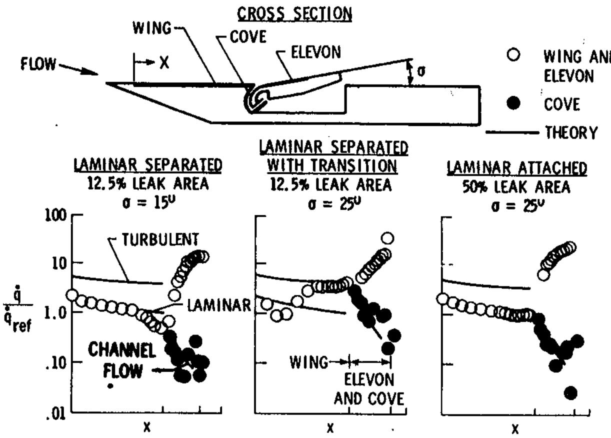



**Fig. 25.- Heating distributions for wing-elevon cove with leakage.**

Three basic results emerged from attached flow tests: (1) the cove environment is sufficiently hostile to require thermal protection of the cove surfaces, and positive sealing to exclude hot gases from the unprotected interior aluminum structure is essential; (2) the cove aerothermal environment is dependent on the approaching wing boundary layer and leak area and not on elevon deflection as long as the wing flow is attached; and (3) the convective heat transfer in the cove increases with time. The latter occurs because, in contrast to the external flow where the energy source is essentially infinite, the cove flow consists of a relatively small portion of the wing boundary layer and hence has a finite energy content. As the upstream cove walls approach thermal equilibrium (steady state), increasing amounts of energy are retained by the ingested mass. Consequently the potential for increased heat transfer further in the cove interior exists.

An investigation was initiated to define cove response to flow separation, even though this phenomenon is not anticipated for Shuttle flight conditions [81]. Similar to the attached flow results, the level of heating within the cove is highly dependent upon flow conditions on the wing at the cove entrance, as shown in fig. 25. With laminar-flow separation near the cove entrance (left graph) cove heating rates (filled symbols) diminish along the cove length by an order of magnitude. Increased elevon deflection angle moves the flow separation point upstream and, as shown by the rising wing heating rates (center graph), the separated laminar boundary layer transitions to turbulent flow ahead of the cove entrance. Consequently, cove heating rates are an order of magnitude greater than for purely laminar flow separation at the same cove seal leak area. As shown in the right graph, for the same elevon deflection, if the leak area is sufficiently large boundary layer suction can force the separated boundary layer to reattach, thereby reducing cove heating rates.

A laminar, inviscid 2-D flow analysis over the cove gap was conducted using finite element methods [82]. The finite element solution is shown in figs. 26 and 27. The pressure contours (fig. 26) are relatively smooth and clearly define the oblique shock which was captured over 8 elements. Fig. 27 compares the finite element solution for pressure along the inner cove wall and elevon with experimental data. The experimental data shows that the pressure is nearly constant in the cove and rises gradually along the elevon. The finite element solution predicts the constant cove pressure with a steep pressure rise along the elevon. Downstream of the computational domain, the experimental data appears to be asymptotically approaching the oblique shock theory pressure. The finite element solution slightly overestimates the oblique shock theory pressure. The actual flow phenomena is complicated at the cove entrance by viscous effects not included in the present analysis. Viscous effects may account for the disagreement in the pressure predicted by the inviscid finite element analysis and the experimental data along the elevon (s>7). Work on viscous and 3D codes is in progress.

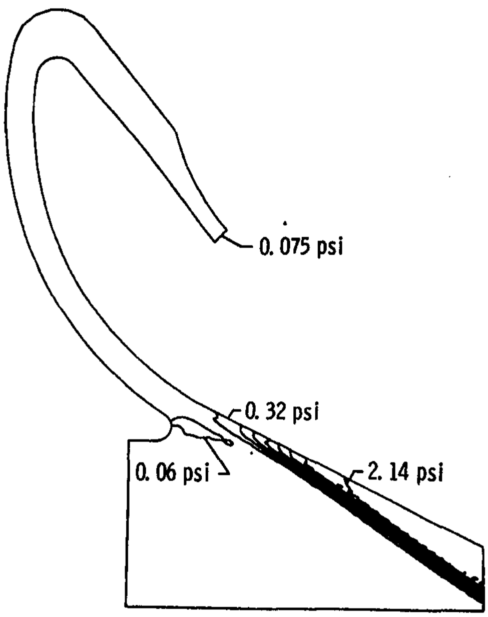



**Fig. 26.- Pressure contours for flow in wing-elevon cove.**

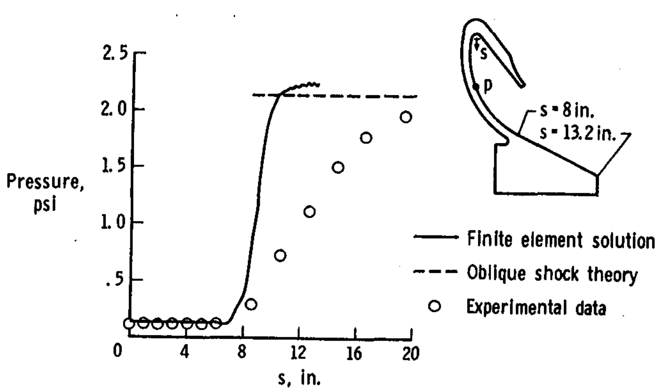



**Fig. 27.- Comparison of finite element predicted pressures and experimental data for wing-elevon cove.**

A considerable insight into the environment in a leaking wing elevon cove has been gained by the research conducted to date. But more remains to be done. Specifically, viscous flow effects need to be considered in the analysis, and no 3D results, analytical or experimental, have been obtained. But with continued progress an acceptable understanding of the environment will be obtained permitting the designer to intelligently design future systems in a timely manner.

## Acoustic Loads

The aeroacoustic environment and the interaction of the structure with the environment must be considered early in the design and development phases of future reusable launch vehicles. Previous Shuttle acoustics technology provides only a starting point for the new technology base because of new roles, operating scenarios, and overall size of an advanced system. The design driver for greatly increased performance will require increased use of light-weight/high-strength materials for the structure, propulsion systems and thermal protection systems. The ability to predict and control intense acoustics loads and structural response to those loads is critical to the success of the systems because they can lead to severe structural fatigue and payload damage.

Determination of launch acoustics loads and potential response problems are critical and may require different technologies rather than an extension of existing Shuttle experience. The aeroacoustic ascent loads may be significantly different because of shape, size, and operating profiles, such as horizontal takeoff. The probable utilization of advanced metallic TPS will require the consideration of the combined thermal and acoustic reentry environment due to the potential for fatigue or less than useful lifetime of the TPS. In addition, it may be necessary to develop new acoustic simulation facilities to meet combined environment requirements. The feasibility of developing new simulation facilities or conversion of existing facilities must be considered early in the technology development phase.

## Criteria

Formulation of design criteria (the ground rules, design procedures, design philosophy, and basic mission configuration data necessary for vehicle design) is by nature part art and part science. As the many interrelated aspects of design criteria grow, there is an increased need for a more systematic approach, one that relies less on engineering art. This is particularly true for reusable launch vehicles for which there is a limited data base of experience, and for which thermal loads, and containment of cryogenic fluids (unfamiliar to the general community of airframe designers) are significant design factors. There are a number of documents in the literature on design criteria such as the NASA series of special publications on space vehicle design criteria (structures) developed in the late 1960's and early 1970's. Several documents address specifically design for thermal loads [83-85]. A recent USAF sponsored study [86-87] addressed the various heat transfer methods used to calculate missile structural temperatures, and developed a set of design guidelines and recommended practices for air launched missiles.

In addition to the usual criteria for aircraft (such as load factors, mission profiles, weight, structural life) other parameters (such as heating rate, heat input, time of exposure) must be considered for high speed and reusable launch vehicles particularly with the need for increased structural efficiency. Ref. [85] discusses the individual parameters and their criticality to design criteria. Although [86-87] specifically addresses missile airframes, many of the parameters affect reusable launch vehicle criteria development. Factors deemed crucial for reusable launch vehicles include: combined loads and thermal effects; factors on loads and temperatures to reduce conservatism to a minimum; determining load factors and temperatures over the entire missile profile - perhaps using ground simulation as an aid for nominal and off-nominal conditions; conducting parametric studies to determine the sensitivity of the criteria to uncertainties; and defining the "hostile" environment for military vehicles so that vehicle survivability may be addressed.

There is currently no well accepted method for sizing flight-weight reusable cryogenic tanks because of a lack of design criteria for fracture. The method of tank sizing used in [27-29, 31] requires a nondestructive inspection to find and repair all detectable flaws in the material before installation of the tank into the vehicle. The analysis assumes that the largest possible flaw, which cannot be found by inspection, exists in the material. For 2219 aluminum tanks, this procedure leads to assumed flaws no larger than .050 in. deep. This minimum detectable crack size is an extension of a military specification found in [88], which applies to closely inspected regions near holes and cutouts. Inspection of large surface areas for .050 in. cracks will require large area, small flaw size inspection procedures which may have to be developed or improved. In addition, there is a requirement for adverse weather capability, but a definition of adverse weather and a definitive durability criteria for external surfaces exposed to such weather does not seem to exist. Continued efforts in developing and updating design criteria are essential to the design of safe, minimum weight structures for future reusable launch vehicles.

## Interdisciplinary Analysis

The interdisciplinary and iterative nature of aircraft design, which is steadily becoming more complex, has resulted in the evolution of interdisciplinary computing systems and extensive research on interdisciplinary analytical design. One of the first major systems developed by a major airframe company [89], evolved after the frustrating and time consuming experiences with aeroelastic analyses for the national SST program. This type of experience lead NASA Langley and AFWAL Flight Dynamics Laboratory around the 1970 time frame to begin R&T efforts into optimization and interdisciplinary analytical design methods as indicated by the typical results given in [90-92]. Significant progress has been made as indicated by the papers contained in [18]. Although work has been done on a wide variety of vehicles, relatively little has been done on the type of vehicles and some of the problems expected for future reusable launch vehicles. In particular the software and in-house expertise is aimed more at conventional subsonic aircraft and fighter aircraft with limited supersonic capability. Steady progress has been made by a relatively low level-of-effort activity on integrated thermal-structural analysis, and more recently, integrated flow-thermal-structural analysis, as reviewed briefly in the next two sections.

## Integrated Thermal-Structural Analysis

Generally finite elements are used for structural analysis and finite difference techniques are used for heat transfer analysis [13]. To perform such analyses on a complex geometrical model, a single numerical method is desirable to eliminate transferring data between different analytical models. With the wide acceptance of the finite element method in structures and its rapid growth in thermal analysis, it has been found particularly well-suited for such analyses. Yet often an incompatibility between the thermal and structural analyses exists because the finite element thermal model and finite element structural model require dissimilar discretizations.

Work is underway to address this problem. Specifically, the hierarchical finite element approach has shown promise of resolving the problem by allowing for a common discretization and seeking improvement in the accuracy of the analyses by: (1) improving the accuracy of the thermal analysis by using hierarchical temperature interpolation functions to converge the thermal solution, (2) using the converged temperature distribution to compute consistent equivalent thermal forces, and (3) using hierarchical displacement functions to converge the structural solution. A hierarchical element is one where the interpolation function (P) increases from linear (P=1) to quadratic (P=2), and so on until the required convergence is achieved. Details of the approach are given in [93].

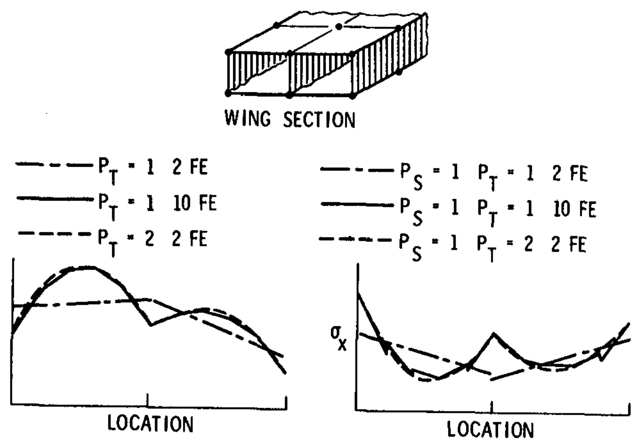



**Fig. 28.- Hierarchical integrated thermal-structural analysis.**

Results for a simple wing box are shown in fig. 28. A common discretization to suit the geometry was first developed. For this simple problem that is a simple task, but the thermal-structural analyst quickly runs into problems where the power of finite element modeling becomes essentially mandatory to work the problem. The thermal solution is obtained with a sequence of analyses (if needed). The accuracy is improved via thermal hierarchical elements (PT=1, 2, ...). Results for the temperature distribution are shown in the lower left of fig. 28. For PT=1, a linear interpolation function, and for a model consisting of two finite elements along the chord, a bi-linear distribution is achieved. Increasing the number of finite elements to 10 results in a much different distribution. Essentially this same distribution is achieved with the original two element model by increasing the hierarchical interpolation parameter to quadratic (PT=2). With the correct thermal distribution now determined, consistent thermal forces are calculated; this capability is generally not available in existing analysis computer programs.

With the consistent thermal forces known, the structural solution for displacements and stresses is obtained. These results are shown on the lower right of the figure. The procedure is the same as for the thermal solution: sequence of analyses; accuracy improved via structural hierarchical elements (PS = 1, 2, ...). For the two element model and PS = PT = 1, a bi-linear solution is obtained. If the finite element model is increased to 10 elements, a much different stress distribution is obtained. Note, however, that essentially the same distribution is obtained with the original two element model for PT = 2, PS = 1. The potential suggested by this simple example is significant, and applications are planned for more complex problems to fully explore this potential.

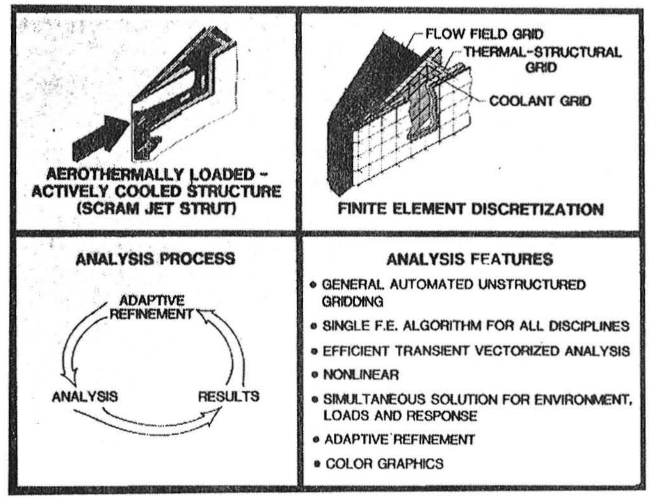



**Fig. 29.- Integated flow-thermal-structural finite element analysis.**

Integrated Flow-Thermal-Structural Analysis

Recently finite element flow analysis work has been extended to exploit the potential of integrated flow-thermal-structural analysis. A nonlinear finite element algorithm developed for the fluid dynamics model was recognized as a viable alternate algorithm for the thermal-structural model as well.

This methodology is illustrated in fig. 29. The upper part of the figure shows the physical model of the Langley airframe integrated scramjet fuel injection strut (left side), and the finite element discretization of the external fluid flow and the structural model (including the hydrogen coolant) for thermal and structural analyses (right side). The process is shown on the lower left of the figure. Starting with an original math model, analyses are done using the same algorithm for defining the external environment including the aerodynamic heating to the structure, the temperature distributions within the structure, and the response in terms of displacements and stresses due to the thermal loads. These results can be examined to see if there are high gradients or unacceptable errors suggesting the need for solution refinement, particularly in the external flow model where shocks may exist whose location is not known a priori, and adaptive refinement techniques exist to automatically alter the discretization [94] or the hierarchical interpolation parameters and the process repeated until satisfactory results are achieved.

This capability has not been proven by doing an all-in-one analysis, but efforts are underway now to exploit what appears to be feasible, based on completed separate analyses using a common methodology. This capability and other developments in computer hardware and software should lead to a very powerful tool for the designer that should drastically reduce the time required for a complete single loads cycle, which takes from 6 to 12 months for the Shuttle. However, this capability will probably remain a research tool for some time while development efforts are continued, and efforts are initiated to demonstrate to project managers that the improved capability is worth the effort required for their staff to learn new tools and procedures.

Aeroservoelasticity

Avionics and controls were enabling technologies for the Space Shuttle [20]. There were some control-structure interaction problems which are a subset of the larger technical area of aeroservoelasticity, which combines the disciplines of aerodynamics, servo control systems, and structural response or elasticity. Currently the Shuttle uses its ailerons during ascent for wing load aleviation.

Because of their tendency to have very far aft center of gravities, winged vehicles, especially single stage to orbit vehicles, will require stability augmentation systems [95]. Although some work has been done to determine the aeroelastic effects on the performance of hypersonic single-use re-entry vehicles [96], aeroservoelasticity in its most general form has not received much attention for reusable launch vehicles. Some of the challenges are the wide range of operational environments and parameters, thermal deformations as well as elastic deformations due to airloads, and actuators that can operate at high rates and at higher temperatures than current vehicles require.

Test Requirements/Facilities

Early Days at Langley

Reference 1 describes some of the early research on structural problems produced by aerodynamic heating conducted at the Langley Aeronautical

Laboratory of the National Advisory Committee for Aeronautics (NACA) from 1948 until 1958 when NACA became NASA. Development of test equipment and facilities began along with the initiation of research projects and accelerated along with their expansion. Combinations of furnaces and testing machines were the principal generators of data on materials and structural elements. Starting in 1951, efforts were directed towards searching for ways to simulate or duplicate aerodynamic heating in the laboratory. A variety of devices for radiative and convective heating of structures were evaluated. One of the goals was to achieve initial heating rates of 100 Btu per square foot per second. This was derived from calculations of the heat transfer rate to airplanes accelerating to M = 3 or M = 4 at 50,000 feet. Tungsten filament lamps met the requirements and were the heat source used in most future heating tests.

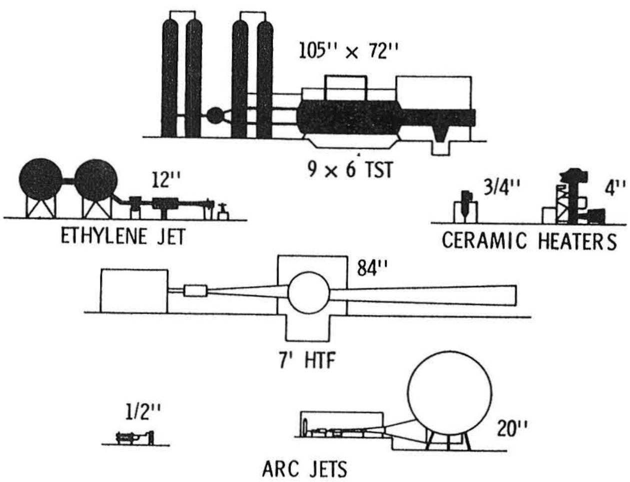



**Fig. 30.- Operational (black) and planned hot jets and tunnels at NASA Langley in April 1959.**

Convective heating to simulate or duplicate aerodynamic heating was investigated from the beginning of the facility development program and a variety of techniques were tried. Hot subsonic air flow and radiant heating panels were proposed to heat structures in a large test chamber. Further study and testing, however, revealed that a true-temperature, M = 3, blowdown wind tunnel was the best approach. This became the 9 x 6 Foot Thermal Structures Tunnel (fig. 30). Its basic characteristics were established in March 1952, the tunnel became operational in 1957, and research testing began in the summer of 1958. This facility was used to test a wide variety of structural models, many of which were evaluated for panel flutter. Ethylene jet and ceramic heaters were very high-temperature supersonic jets for testing materials and small models (fig. 30). The electric-arc powered jets subsequently carried this capability to extremely high temperatures. Their original development was motivated by the long-range ballistic missile program, but these Langley facilities made their major contribution later to the manned space flight programs, including the Space Shuttle. The facility labeled 7' HTF in Fig. 30 is the initial concept of the facility now known as the 8-Foot High Temperature Tunnel. It is a true-temperature, M = 7 blowdown wind tunnel. Construction began in 1960 and high-temperature testing began in 1968. Note that nearly 10 years elapsed between concept and research.

Fig. 31 shows one of the more spectacular tests conducted at Langley on a large lifting body type structure [97]. This test was the last of any significant size conducted at Langley as NASA, in the 1960's, began development of test capability at Dryden Flight Research Facility for heating and loading large structures including complete aircraft.

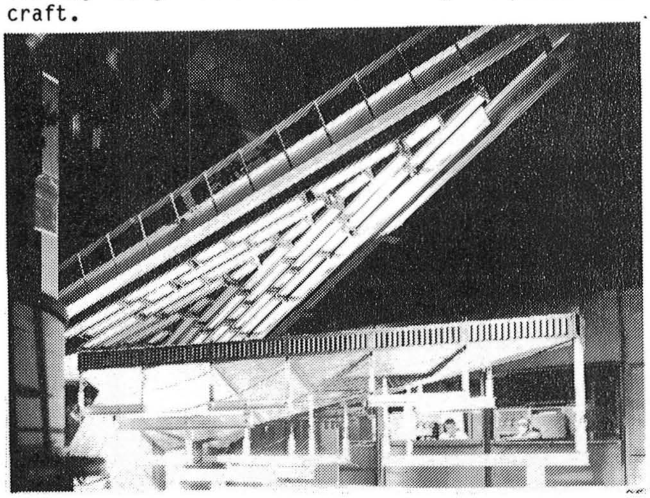


**Fig. 31.- Combined thermal mechanical loads test on lifting body hot structure at NASA Langley, Circa 1960.**

## Early Days at Dryden

Activities at NASA Ames-Dryden associated with aerodynamic heating began with the X-15 flight test program in the early 60's. Minor structural failures of the canopy and other areas, along with attempts to measure loads with strain gages in a thermal environment, drove the development of a structures laboratory test facility capable of combined loading and heating of flight test vehicles. The Flight Loads Research Facility (FLRF) was completed in 1966 and a series of loading and heating tests of X-15 tail and wing structure were conducted over the next several years, using conventional, analog controlled, quartz lamp heaters [98]. During this same period a sophisticated data acquisition and control system was obtained which employed a digital computer to perform adaptive digital control of 512 heating channels, using 20 megawatts of power, and acquisition of 1200 channels of data. This system became operational in 1970 and was used for the YF-12 Flight Loads Research Program which involved gathering strain and structural temperature data in flight, installing the flight vehicle in a laboratory heating set up in the FLRF, (fig. 32), and subjecting the vehicle to the flight measured thermal environment using 16,000 quartz lamps grouped in 512 control zones. The objectives of the tests were successfully achieved [99] and the vehicle was returned to flight operation.

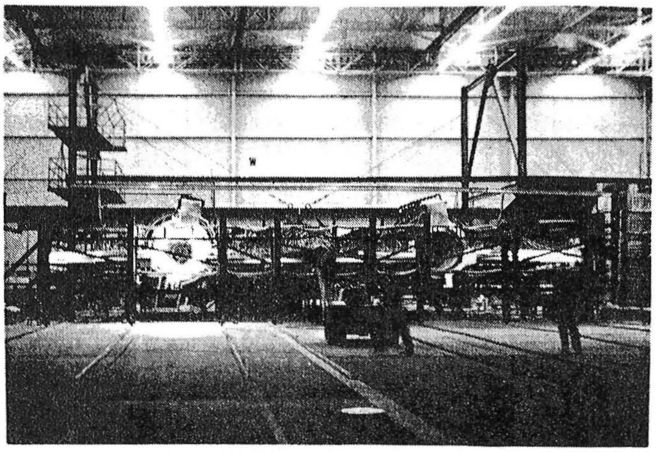


**Fig. 32.- Laboratory heating of complete YF-12 aircraft at NASA Dryden.**

In the later 1960's, NASA LaRC sponsored a study producing the preliminary design of a hypersonic cruise vehicle wing structure [61]. In 1973, NASA Ames-Dryden instituted the fabrication of the Hypersonic Wing Test Structure (HWTS) [60] based on this study. Combined loads and heating tests of this specimen (fig. 11), to 1800°F for the external surface which was metallic radiative TPS panels, were conducted providing data for validation of concepts and of computer codes for predictions of structural temperatures and thermal stresses [62].

Early Days at Ames

Arc jet utilization at Ames began in the late 1950's with early testing of a generic nature followed by exploratory tests of various candidate heat shield materials. Future requirements for combining radiative heating with convective heating was recognized, and carbon arc radiation sources were combined concentrically with the arc jets. Later, argon plasma radiation sources replaced the carbon arcs. In the early 1960's the Ames-developed constricted, segmented arc heater was perfected. Ames supported the Mercury, Gemini, and Apollo programs and a considerable amount of basic, generic research was done at both ends of the heating spectrum. Low density, glass fiber reinforced polymers were studied as afterbody heat shields and very definitive, basic research was done on the ablation of graphite at temperatures in excess of 7000°F. Near the end of this time period, a considerable amount of testing was carried out in support of Pioneer Venus probe heat shields.

Ames, like Langley, was also interested in hypersonic flight and in the mid-60's had the Linde Company design and build the Linde N-15,000 Arc Heater with a design capability of 15 megawatts to match the Ames existing d.c. power supply capability at that time. The Air Force later borrowed the Linde N-15,000 and learned quickly that it was rather simple to drive it up to in excess of 50 megawatts. That design then became known as the Flight Dynamics Lab 50 megawatt arc jet.

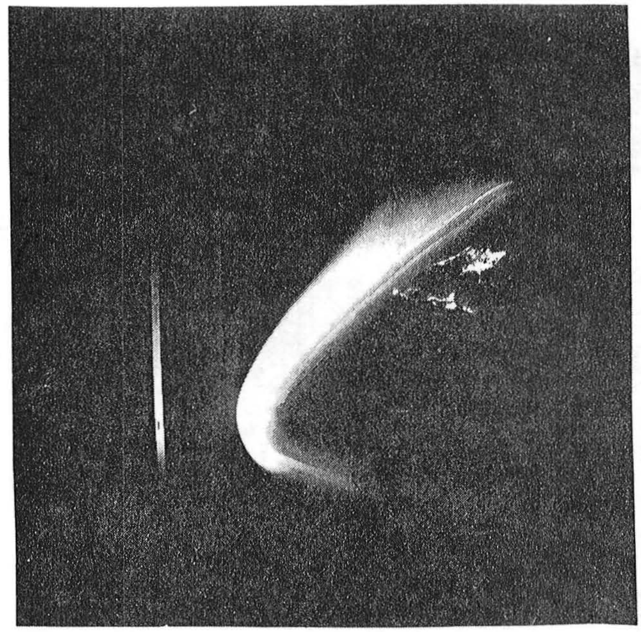


**Fig. 33.- Full size Shuttle leading edge segment in the Ames 60MW facility.**

In 1969 it became clear that the proposed Space Shuttle would require large test articles and therefore large arc jet facilities with high power. Ames designed and built the 2x9 inch supersonic, turbulent flow duct to match the Linde N-15,000 to provide the required flow over test specimens of 8x10 and 8x20 inches in planform. At the same time, work was begun on scaling up the high-enthalpy constricted arc jet design to 20 megawatts in anticipation of approval for a 60 MW arc heated Shuttle test facility. All of these facilities were successfully built and operated to support the Shuttle program. An example of this support is shown in fig. 33 which is a photograph of a full size shuttle leading edge segment (RCC material) in the Ames 60 MW facility with a 41 in. nozzle.

Early Days at FDL

For more than 25 years the Flight Dynamics Laboratory (FDL) and its predecessor organizations at Wright Patterson Air Force Base have been active in furthering the technology required for the design, development and testing of hypersonic vehicles. All of this technology is directly applicable to today's reusable launch vehicle efforts. Most of the USAF's structures test experience with lifting body, hypersonic vehicles has been obtained in the Structures Test Facility of the FDL.

This experience began in 1959 with the structural tests of the Boeing "Hot Structure" and the Bell "Double Wall Structure" (fig. 34). These were end items of a manufacturing methods demonstration program and they represented structural concepts for the proposed X-20 (Dyna-Soar) Glider. From 1959 to 1961 these test programs were successfully accomplished. For the first time, programmed test temperatures simulating flight isotherms to 2000°F were applied to large structures. Predicted flight loads were simultaneously applied on a common time base and temperature and deflection measurements were successfully made in this temperature regime.

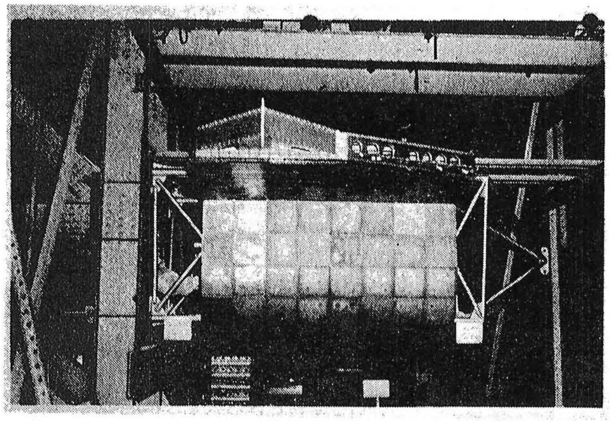


**Fig. 34.- Bell double wall end item installed in jig at FDL structural test facility, circa 1961.**

At the conclusion of these two test programs the FDL Structures Test Facility was committed to conduct all required structural tests for the X-20. At the time, it was anticipated that test temperatures as high as 3000°F might be required. The "Hot Structure" and "Double Wall Structure" test programs had indicated that extensive test techniques development would be required to push the state-of-the-art of heating from 2000°F to 3000°F. The same intensive development effort would be required for the loading and instrumentation systems and techniques. This development program was begun in 1962. The instrumentation efforts were primarily aimed at determining thermocouple types, sizes, and methods of attachment to coated refractory metal heat shields. High temperature strain gages for the anticipated test temperatures were not expected to be a likely development for use during these tests, but efforts to develop an 1800°F strain gage were pursued. A 3000°F deflection measuring capability was required. Thermal simulation required infrared heater development capable of 3000°F simulation. Loads were to be applied to the cooler upper surface and would involve development of loading methods in a 1500°F environment.

This development effort was conducted on a priority basis until the Dyna-Soar was cancelled in late 1963. Heater development efforts were continued in order to satisfy the 3200°F heating requirements for the Aeronca Thermantic Structure Test Program which began in December 1964.

Flame heating techniques were considered because it was questionable if infrared heating could be used for temperatures over 3000°F. Flame heating was discarded because of poor temperature control and an unbearable noise level when heating large areas. The thermantic structure was internally cooled and was to be loaded internally during the tests. Due to problems with the structure and some of the test equipment, all of the test goals were not met. However test temperatures up to 3195°F were achieved on a heated area of 70 square feet. This demonstrated the capability of accurately controlled heating of large structures to temperatures in excess of 3000°F and also revealed a great deal about the capabilities and proper handling of large scale heating test support equipment.

Hypersonic structures testing was continued in 1965 with the tests of an X-20 Elevon and side window. Both of these programs were part of fall-out efforts relating to Dyna-Soar hardware already fabricated prior to Dyna-Soar program cancellation. All ascent and reentry loading and heating environments were successfully simulated using the test techniques developed for the previously described programs.

In 1966 the Structures Test Facility began its most ambitious effort in hypersonic structures testing. A cryogenic fuel carrying test article, built by the Martin Company for the Advanced Structural Concepts Experimental Program (ASCEP), was subjected to real-time ascent and reentry condition simulation. This was the largest flightweight, airframe structure ever assembled of refractory metal (fig. 35). Additional facility capability for the liquid nitrogen fuel simulant was required and developed for this program. This USAF effort is documented in [3-8].

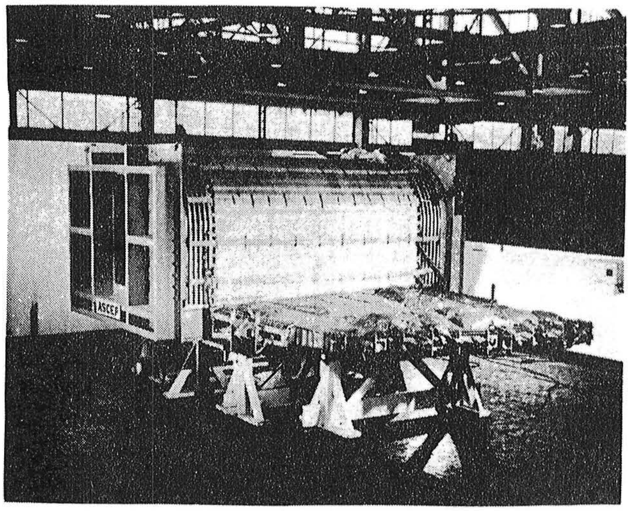


**Fig. 35.- Advanced structural concepts experimental program test article at FDL structural test facility, circa 1966.**

Current and Future Effort

Much of the laboratory test capability available in the 1960's and early 1970's has been unused and unimproved since then. A formidable task is to determine the realistic test requirements for the various vehicles that will be a part of the National Space Strategy, to assess the existing facilities and instrumentation to do such testing, define test capability and instrumentation that needs to be developed, design and build the facilities, and do it in a time frame consistent with the earliest IOC of the National Space Strategy, 1995. A comprehensive Hypersonic Research Facilities Study (HYFAC), was published in 1970 [100]. HYFAC identifies several facilities that are still needed, although possibly with modified capability, and the costs, in 1970 dollars, are staggering. Thus another formidable task will be the successful advocacy for funding of the required new facilities.

Airframe Test Capability: A number of steps are being initiated to prepare for the development of the next generation of hypervelocity flight vehicles. Data acquisition and control systems are being refurbished to incorporate the latest hardware and software. An industry-government survey is nearing completion that will compile data on existing facilities and test methods useful for hypervelocity structural testing. Particular emphasis is being put on ability to test with liquid hydrogen fuel in the test structure. Studies are being performed to investigate the feasibility of upgrading heating and cryogenics capabilities, adding vacuum capability, and the development of improved instrumentation for measurement of strain, temperature, pressure and deflection. Testing of structural concepts will continue, along with generic testing to provide data for assessment of analytical codes. Additional activities will focus on the definition of requirements for test qualification of flight components and vehicles under combined loads for static strength and fatigue, and probably for dynamic loads and frequency response data.

Since the NASA project Galileo heat shield testing would require heating rates one thousand time greater than Shuttle, Ames was given approval to build three new arc jets: the 110 megawatt Giant Planet Facility (operational); the 100 atmosphere 110 megawatt Transitional Flow Facility (under construction) and finally, the 165 megawatt High Enthalpy Entry Facility (under construction). In addition to the arc jets, Ames has a high power CO2 gasdynamics laser. The flexibility of the Ames facility is such as to provide a research capability for a range of entry technology problems. The Facility complex consists of seven test positions into which different arc heaters can be installed and with those arc heaters a myriad of nozzles of diverse shapes and sizes from two to 42 inches in exit size can be used. A combination can be found to provide a simulation for almost any NASA and many non-NASA problems. The primary use will continue to be support of TPS materials research and NASA and DOD TPS support programs.

Propulsion Structure Test Capability: Current facility capabilities for testing propulsion or missile systems at high Mach numbers are severely limited. The Marquardt Company, the Chemical Systems Division of United Technology Corporation, and Air Force facilities at Arnold Engineering Development Center (Aero Propulsion Test Unit and Aero Propulsion System Test facility) can accommodate full-scale ramjets and missiles up to about Mach 4. Other test facilities (for example, those at NASA Langley and General Applied Science Laboratories) can simulate flight Mach numbers up to 7, but they are small and can only accommodate incomplete subscale engines or engine components. No existing facility in the U.S. can provide both true-temperature, high-Mach-number flow and a large scale.

The NASA Langley 8-Ft. HTT has many of the attributes desirable for a propulsion test facility, in particular, size and true temperature simulation for Mach 7 flight; however, the high energy level required to simulate Mach 7 flight is obtained by burning high pressure methane and air and the resulting products of combustion are used as the test medium. Thus the test stream is oxygen depleted and will not support combustion. In addition it would be highly desirable to be able to test at lower Mach numbers nearer the range where transition from turbojet or rocket to ramjet or scramjet mode of operation is anticipated.

Reference 16 describes a planned modification of the 8 Ft. HTT to make it a unique national research facility for hypersonic air-breathing propulsion systems and discusses some of the ongoing supporting research for that modification. The modification involves: (1) the addition of an oxygen-enrichment system which will allow the methane-air combustion-heated test stream to simulate air for propulsion testing; and (2) supplemental nozzles to expand the test simulation capability from the current nominal Mach number of 7.0 to include Mach numbers 4.0 and 5.0. The modified facility will retain the present capability for aerothermostructural and aerothermal loads research for high speed vehicles [101].

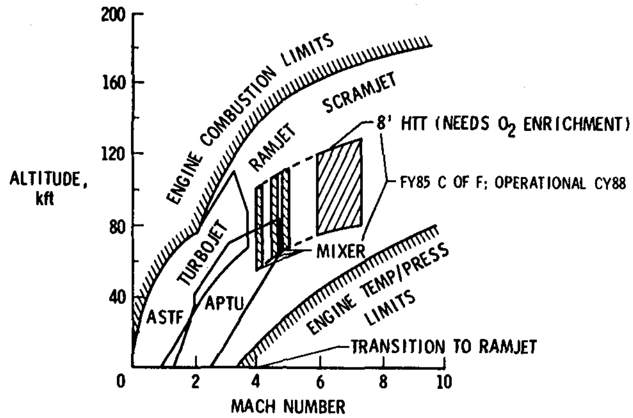



**Fig. 36.- Operational envelopes for propulsion facilities.**

The modified facility will complement the two large scale Air Force facilities at the Arnold Engineering Development Center as illustrated by fig. 36 in which the operating envelopes of the three facilities are superimposed on projected operating ranges of air-breathing propulsion systems. The Aero Propulsion Test Unit and Aero Propulsion System Test facility appear completely adequate for propulsion research for Mach numbers below approximately 4.5. The modified 8 Ft. HTT will extend that range to Mach 7 and provide coverage in the ramjet to scramjet transition region.

Another requirement under study at NASA Langley is the capability for life-cycle durability testing. The test facility must include the capability for 1) the extremely high heating rates encountered by engine leading edges, 2) cryogenic hydrogen at close to $LH_2$ temperatures, 3) sufficient run times for creep to occur, and 4) turn-around-time for testing to permit acquiring many thermal cycles (1000) in a timely manner.

Flight Test Capability: Under NASA contract a study was conducted to determine airframe structures technology needs and flight test requirements for hypersonic vehicles [102]. The study team was supplemented by two consultants: The late Professor E. E. Sechler of the California Institute of Technology and Professor Rene Miller of MIT. Prof. Sechler commented: "Discussions held with other members of the aerospace community and the advisors assigned to this study inevitably led to the same conclusion -- flight test can yield a large increase in confidence in thermal/structural design. Flight test and demonstration offers the only reasonably sure way to discover design "unknowns" before production vehicles become operational. In the case of thermal/structural design, confidence is not only important in verifying performance, but is necessary for flight safety. In summary, flight testing of hypersonic thermal/structural designs will be <u>an absolute necessity</u>."

Prof. Miller commented: "The cost effectiveness of such structural concepts is greatly dependent on solutions to the detailed design problems. In fact, it is likely that these detailed design problems as demonstrated in the X-15 program will prove to be the pacing item in the development of hypersonic aircraft. It is believed that many of these problems cannot be solved except through flight tests. For example, structural distortion in the presence of flight loads, uneven thermal expansion and high dynamic pressures occur in flight in a way which would be impossible to duplicate on the ground. Flight tests are important to provide this proof of concept information."

Air breathing propulsion testing requirements are even more demanding as no ground based facility can duplicate the flight environment above Mach 8. Serious proposals for flight testing have been made in the past including the National Hypersonic Flight Research Facility [103] and Shuttle launched entry research vehicles [104, 105]. Thus flight testing seems to be a firm requirement, and studies need to be initiated to determine what, how, and when.

## Concluding Remarks

Significant progress has been made in the area of structures and materials technology for reusable launch vehicles, considering the resources provided this area during the last 20 years. The ceramic and carbon-carbon thermal protection systems (TPS) on the Shuttle Orbiter are working and could be used for future launch vehicle applications provided they can be made to meet the durability requirements. Metallic TPS have received continuing attention and are beginning to reach maturity, at least from a R&T point of view. Progress has been made in hot structures for both wing and cryogenic tank/fuselage applications. This technology is at the point that fabrication of large test specimens is the next logical step for certain materials such as Rene' 41. For other materials such as GR/PI, metal matrix composites, RSR titanium and dispersion strengthened iron base alloys, work on materials and small test specimens is required. Some progress including fabrication and testing of hardware has been made for actively cooled airframe and regeneratively cooled engine structure, and major problems for future research have been documented. However, much remains to be done for this class of structure. A good start has been made in developing an aerothermal loads data base and integrated flow thermal structural analysis methods, but these disciplines will mature in a timely manner only if continued at an increased level of effort.

In spite of such progress, much remains to be accomplished. Research in high payoff areas of materials and structures has barely scratched the surface. One of the most exciting areas is the development and use of advanced carbon-carbon materials for acreage TPS and structural applications. Little has been done for cryogenic tanks (other than work on a Rene' 41 concept), especially composite tanks and cryogenic insulations. Also, the area of high temperature test facilities, techniques and instrumentation has been largely neglected by the aerospace community. Work in these areas, especially for testing of cryogenic tanks, and life-cycle testing of structures operating at high temperatures, must begin if such structures are to be considered for future space transportation systems.

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## Standard Bibliographic Page

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|---|---|---|
| 1. Report No. NASA TM-87626 | 2. Government Accession No. | 3. Recipient's Catalog No. |
| 4. Title and Subtitle<br>Structures and Materials Technology Issues for Reusable Launch Vehicles | | 5. Report Date<br>October 1985 |
| | | 6. Performing Organization Code<br>506-53-33-07 |
| 7. Author(s)<br>S. C. Dixon, Darrel R. Tenney, Donald R. Rummler, Allan R. Wieting and R. M. Bader | | 8. Performing Organization Report No. |
| 9. Performing Organization Name and Address<br>LANGLEY RESEARCH CENTER<br>Hampton, Virginia 23665-5225 | | 10. Work Unit No. |
| | | 11. Contract or Grant No. |
| 12. Sponsoring Agency Name and Address<br>NATIONAL AERONAUTICS AND SPACE ADMINISTRATION<br>Washington, DC 20546 | | 13. Type of Report and Period Covered<br>Technical Memorandum |
| | | 14. Sponsoring Agency Code |

15. Supplementary Notes
This paper presented at the EASCON 85 18th Annual Electronics and Aerospace Systems Conference, Washington, D.C., October 28-30, 1985.

16. Abstract
Projected space missions for both civil and defense needs require significant improvements in structures and materials technology for reusable launch vehicles: reductions in structural weight compared to the Space Shuttle Orbiter of up to 25% or more, a possible factor of 5 or more increase in mission life, increases in maximum use temperature of the external surface, reusable containment of cryogenic hydrogen and oxygen, significant reductions in operational costs, and possibly less lead time between technology readiness and initial operational capability. In addition, there is increasing interest in hypersonic airbreathing propulsion for launch and transmospheric vehicles, and such systems require regeneratively cooled structure. These technology issues pose quite a challenge to the structures and materials community, especially since most industry and DOD organizations have not been working these technology issues for 15 years, and the NASA base R&T program has been supported at a relatively low level of effort. This paper will address the technology issues, giving brief assessments of the state-of-the-art and proposed activities to meet the technology requirements in a timely manner.

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|---|---|---|---|
| 17. Key Words (Suggested by Authors(s))<br>Structures and Materials | | 18. Distribution Statement<br>Unclassified - Unlimited<br>Subject Category - 16 | |
| 19. Security Classif.(of this report)<br>Unclassified | 20. Security Classif.(of this page)<br>Unclassified | 21. No. of Pages<br>29 | 22. Price<br>A03 |

For sale by the National Technical Information Service, Springfield, Virginia 22161

NASA Langley Form 63 (June 1985)