Research in Structures and Materials for Future Space Transportation Systems—An Overview

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This paper provides a review of some of the advances of the past decade in structures and materials that have application to future space transportation systems. The paper concentrates on metallic thermal protection systems and structures that could provide the structural efficiency, reliability, and durability dictated by future space utilization requirements. The need for completely reusable, onboard cryogenic fuel tanks is cited as the most challenging technical opportunity, and potential thermostructural concepts for tanks are identified. Other critical areas needing technology advances are discussed. Finally, the unique research opportunities offered by the Shuttle Orbiter experiments program for testing structures and thermal protection systems are explored.

Introduction

THE success of space exploitation rests to a large extent on the ability of structural designers to meet the challenge of providing lightweight structures for large Earth-to-orbit space transport vehicles that are repeatedly subjected to severe aerodynamic heating, and yet must have structural efficiencies, reliability, and durability approaching that of commercial aircraft. The present Space Shuttle represents a significant advancement in system reusability; however, future requirements will be ever more demanding and may require the increased durability afforded by metallic exterior surfaces. Although the resources devoted to metallic thermostructural research in the seventies have been minimal, there have been modest advances in both materials and structures that will impact future space transportation vehicles.

Recent studies[1,2] cite the structure as the largest single element of vehicle inert weight (approximately 60%) and emphasize the need for additional advances in the field of materials and structures to meet the nation's future requirements for space transportation systems. Another study[3] has identified the lack of developed and verified concepts as the major thermostructural technology deficiency for hypersonic vehicles. This finding is not too surprising when it is realized that a "proven" concept is the culmination of a highly interactive process involving different disciplines and extensive analytical and experimental verification. Many of the concepts just now reaching the required maturity for flight readiness were first conceived in the early sixties.[4-8] Past experience indicates that five to ten years are minimum to develop and verify a concept with an additional five to seven years for first vehicle production. Thus research is needed now to provide advanced technology for vehicles that are to be operational in the year 2000.

Configuration Impacts

Overall vehicle configuration is intimately involved in the selection of a thermostructural concept and vice versa. Systems studies,[1,2,9] which have examined the nation's future space needs, have identified three general configurations as potential candidates for future space transportation systems: horizontal takeoff/horizontal landing, vertical takeoff/horizontal landing, and vertical takeoff/vertical landing.

The horizontal takeoff/horizontal landing configuration relies on aerodynamic lift for the initial boost through the atmosphere and, therefore, must have relatively large wings to support the heavily loaded, fully fueled vehicle. This results in a low-heating-intensity, long-duration entry trajectory that permits use of metallic surface structures. Metallic surfaces accommodate relatively high total heat loads by reradiating heat to space, but cannot tolerate extremely high temperatures.

The vertical takeoff/horizontal landing configuration relies on thrust for the initial boost through the atmosphere. Mass, not lift, becomes critical, which tends to drive the design to a high wing-loading re-entry configuration that is constrained to a relatively high-heating-intensity, short-duration entry trajectory. This, in turn, dictates the use of high temperature surface materials, such as ceramics, to protect lower temperature structural materials. (This approach, of course, will be recognized as the one being used on the current Shuttle.) Although vertical takeoff inherently favors high wing loadings, it currently appears that future vertical takeoff vehicles will have wing loadings that are 20-40% lower than the present Space Shuttle Orbiter.[10] Thus metallic surface structures may also be viable options for these configurations.

The vertical takeoff/vertical landing configuration is constrained to a ballistic or semiballistic trajectory and will therefore experience the highest intensity, shortest duration heating pulse during entry. The extreme surface temperatures will exceed existing material capabilities and require the use of some type of cooling; the presence of fuel for final retrofire provides a potential heat sink for such cooling.

The continuing drive for low life cycle cost for any of these vehicles will place increased emphasis on long life and complete reusability, which in turn impacts the thermostructural concept and the overall vehicle configuration. In particular, the requirement for complete reusability dictates the use of large onboard cryogenic fuel tanks which pose challenging thermostructural problems.

Materials and Manufacturing Technology

Materials and manufacturing technology provide the basic building blocks for structures, and any successful thermostructural concept must comply with materials and manufacturing limits. Initial selection of materials which may be considered for application on future space transportation

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systems requires that the boundaries of service life, the range of operating temperatures, and the accumulated time at maximum operating temperatures be known. A reasonable estimate of these conditions is:

Maximum surface temperatures	580-1900 K
Service life	500 missions
Accumulated time at elevated temperature	250-1000 h

With these service requirements, the general materials applicable to future space transportation systems are shown in Fig. 1. This figure, developed from Refs. 3 and 11, also represents an estimate of the current status of the materials classes with respect to their operating temperature range. The following discussion highlights some recent developments with emphasis on improvements that show promise of increasing the maximum service temperatures of the various material classes.

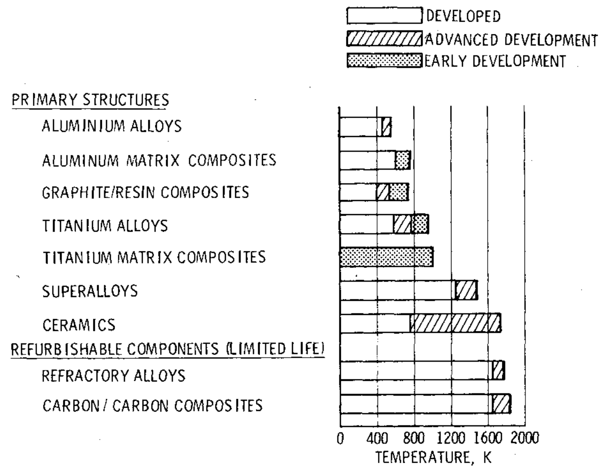

Fig. 1 Material and manufacturing technology status.

Aluminum Alloys

Recent work[12] on boron fiber-reinforced aluminum matrix composites suggests that they may be applicable for temperatures up to 560 K if they are not subjected to high-temperature fatigue loadings in ambient pressure air. Borsic/aluminum honeycomb core panels have been successfully exposed for 100 h and then tested at 533 K,[13] suggesting that many of the early fabrication problems with borsic/aluminum have been solved. Specimens of a polycrystalline alumina fiber/aluminum matrix composite have been exposed to temperatures up to 590 K for 2500 h.[14] Fiber properties were not degraded, but a need for an improved matrix alloy was noted. This materials system has the potential of service temperatures up to 750 K.

Graphite Resin Composites

Elevated temperature matrix degradation by oxidation restricts graphite/epoxy composite materials to maximum service temperatures of approximately 425 K. However, the graphite/polyimide systems under investigation show promise for use at temperatures up to 560 K for periods up to 1000 h.[12] The problems associated with the fabrication of complex structural components with polyimide resin systems are being actively investigated and progress is being made.[13] Newly formulated polyimide resins may substantially improve the processability of polyimide composite systems without significantly reducing their thermal stability.

Titanium Alloys

Titanium alloys are well developed for service up to 700 K. Two new areas of investigation for titanium alloys show promise of significantly increasing the maximum use temperature and the ease of fabricating complex structural components. The high temperature creep strength and oxidation resistance of titanium alloys can be significantly increased by ion implantation/plating of noble metals such as platinum.[11] This technique holds the promise of raising the maximum use temperatures of titanium alloys for long-term applications up to 975 K. The ease of fabricating titanium structural components and the complexity of the geometrics which can be produced may be significantly increased as a result of current investigations (for example, Contract NAS1-15527) to develop superplastic forming and codiffusion bonding of titanium alloys. Fiber-reinforced/titanium alloy matrix composites offer the potential for high specific strength and stiffness in high-temperature applications. Various titanium alloys and fibers have been evaluated. The potential of a service temperature up to approximately 1000 K has been established for a borsic/titanium composite.[15] However, interfacial chemical reactions between the fiber and the matrix continue to pose a problem. If the fiber/matrix interface interactions can be minimized by the use of barrier coatings, such as tungsten, on the fibers and/or by selection of other fiber materials, such as silicon carbide, titanium composites have a potential application temperature of 1140 K. This assumes that adequate oxidation resistance will be provided by coatings, such as ion-plated noble metals.

Superalloys

Nickel and cobalt base superalloys can currently be considered adequate for service to 1250 K. Dispersion-strengthened materials, such as TD-NiCr, have been successfully fabricated and tested as thermal protection systems (TPS) for temperatures up to 1370 K.[16] Although not increasing maximum service temperature, the development of a new class of superalloys known as long-range-order (LRO) superalloys, which have high strength and significantly greater creep strength than conventional superalloys,[11] may reduce system weights.

Ceramics

Ceramic materials have long had the potential for structural application up to 1650 K. However, until the recent effort to utilize ceramics in turbine engines, only a few successful structural applications of ceramics have been reported. Recent successes in the turbine program have significantly improved the outlook for use of ceramic materials in structural applications.[11]

Refractory Metals and Carbon-Carbon

In the area of limited-life components, refractory metal alloy TPS have been demonstrated at temperatures up to 1600 K for up to 50 simulated Shuttle missions.[17] With additional development of the coated columbium alloy systems, it is reasonable to project a service life of at least 100 missions at a maximum service temperature of 1750 K for future space transportation systems. Coated carbon-carbon composites will be used for the refurbishable leading edges and nose cap of the current Shuttle at maximum service temperatures in excess of 1600 K.[18] Continued development of coated reinforced carbon-carbon (RCC) materials is underway to improve strength and oxidation resistance. The goal of this work is a 500-mission life at temperatures up to 1800 K.

In summary, the improvements in materials most beneficial for future space transportation systems are: 1) increase oxidation resistance, thermomechanical stability, and creep strength for all the metallic systems discussed; and 2) decrease complexity and cost of fabricating composite materials and titanium alloys.

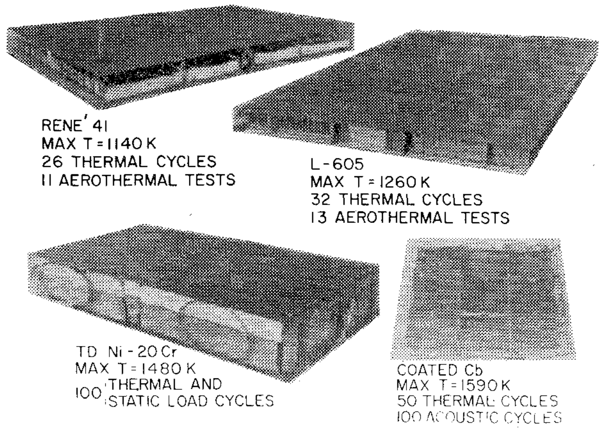

Fig. 2 Metallic standoff TPS concepts.

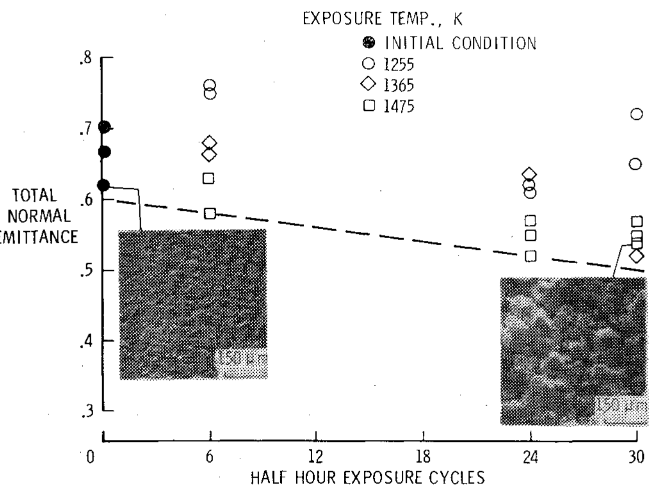

Fig. 3 Emittance/morphology of TD-NiCr after simulated re-entry.

Thermal Protection Systems

Reusable Surface Insulation

Although this paper concentrates on metallic structure and thermal protection systems, it would be incomplete to proceed without some comment on the status of a reusable surface insulation (RSI) type of thermal protection currently being used on the Space Shuttle. These insulations have undergone extensive development as part of the Shuttle program in an attempt to overcome the inherent fragility, water absorptivity, and thermal strain incompatibility with metals characteristic of porous ceramic materials. Most recently, there have been significant advances at the NASA Ames Research Center in the development of a materials system designated fibrous refractory composite insulation (FRCI) which will have twice the strength and approximately one-half the shrinkage rate of the LI series RSI currently being installed on the Shuttle.[19] It remains to be seen, however, if any ceramic-type insulations will have the durability and reliability that will be demanded by future space transportation systems.

Standoff Metallic TPS

Over the past decade there has been a broad-based technology program to advance the state-of-the-art of radiative metallic thermal protection systems to the point where designers can make rational decisions on thermal protection materials and concepts for future space transportation systems.[20] This effort has, for the most part, concentrated on standoff-type TPS (Fig. 2), which consists of a metallic heat shield typically corrugated and mounted on flexible supports to accommodate thermal expansion, and an

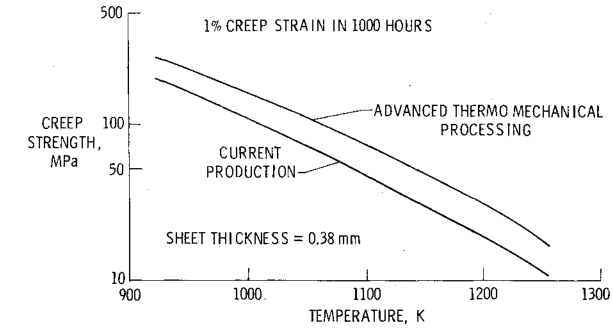

Fig. 4 Creep strength of Haynes HS 188 sheet.

insulation package to reduce heat flow to the primary structure. As indicated by the figure, standoff TPS of various materials for service at temperatures up to 1600 K have been subjected to durability and performance tests,[21-24] including aerothermal tests in the realistic flight environment of the Langley Research Center 8-Foot High Temperature Structures Tunnel. Based on the confidence gained with these TPS, a second generation of TPS have been designed, fabricated, and tested. Results of aerothermal tests of the higher temperature Haynes 188 TPS are presented in Ref. 25; results of the Rene' 41 TPS tests have not yet been reported. The second-generation TPS are as much as 34% lighter than the earlier designs. The mass savings are the result of detail design of the heat shield, supports, and insulation package, which results in more efficient use of materials.

Although a significant technology base has been established for metallic standoff TPS, not all potential problems have been fully resolved and further improvements are possible. For example, recent studies[26,27] have indicated that under service conditions the emissivity of heat shield materials may not be as high as the values used in design calculations. As shown in Fig. 3, tests of TD-NiCr sheets after exposure to simulated Shuttle service conditions (½-h cycles in an arc-heated wind-tunnel) yielded average total normal emissivities (ϵ_{TN}) that were initially less than 0.6 (whereas 0.8 is often assumed) and tended to decrease as the number of simulated missions increased. As illustrated by the two photomicrographs, emittance appears to be related to surface morphology—the coarser the texture of the oxide on the surface, the lower the emittance. Since the heat flux which a candidate TPS material can tolerate is directly related to emittance, it is important to determine if this behavior in a flowing environment is typical of other nickel and cobalt base alloys. If so, then modifications of surface chemistry and/or coatings will have to be developed to produce a stable and high ($\epsilon_{TN} \geq 0.8$) surface emittance for these materials.

Another subject warranting further study is the mass increase resulting from water retention by the fibrous insulation. Rain leakage through joints in the shields and condensed water vapor in tank areas are potential sources of water to be excluded by dry gas purge or sealing techniques.

On the positive side, another study[28] indicates creep properties of high-temperature structural alloys can be improved through advanced thermomechanical processing procedures. Early in the Shuttle technology program creep strength was identified as a possible design constraint for metallic TPS and studies were initiated to improve the creep strength of HS 188 (a cobalt base TPS candidate material). These studies have led to the improvements in the low-strain creep strength of HS 188 sheet shown in Fig. 4. The figure shows that with advanced thermomechanical processing the creep strength (1% strain in 1000 h) of the material is increased approximately 50% over the complete temperature range of interest (920-1250 K), or, alternatively, design life has been increased by an order of magnitude for a constant stress level. These improvements were obtained using an

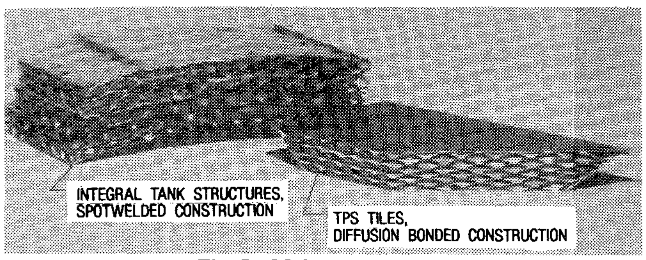

Fig. 5 Multiwall samples.

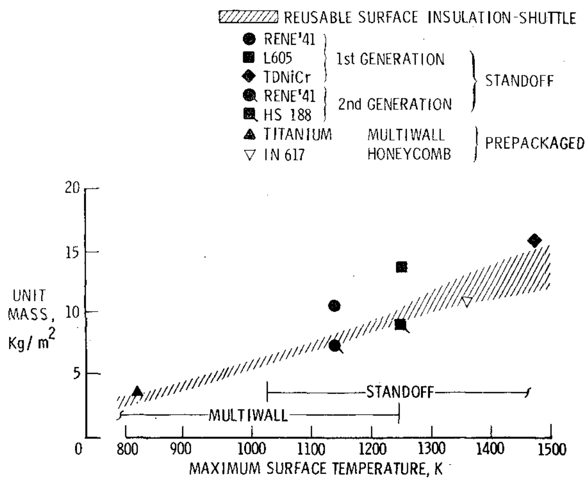

Fig. 6 Unit masses of TPS sized for a Shuttle trajectory. (Solid symbol denotes tested TPS; open symbol denotes fabricated TPS.)

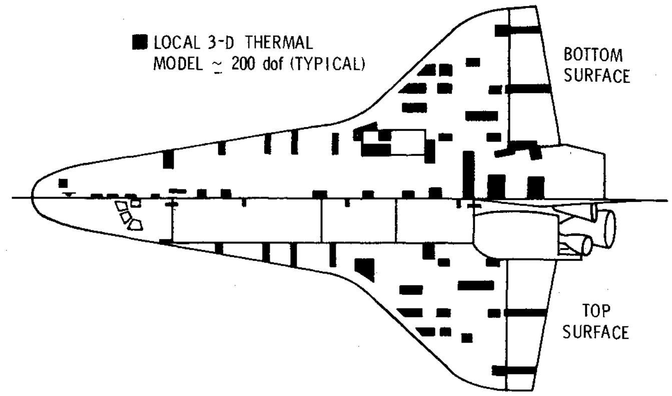

Fig. 7 Overall thermal model of Shuttle Orbiter.

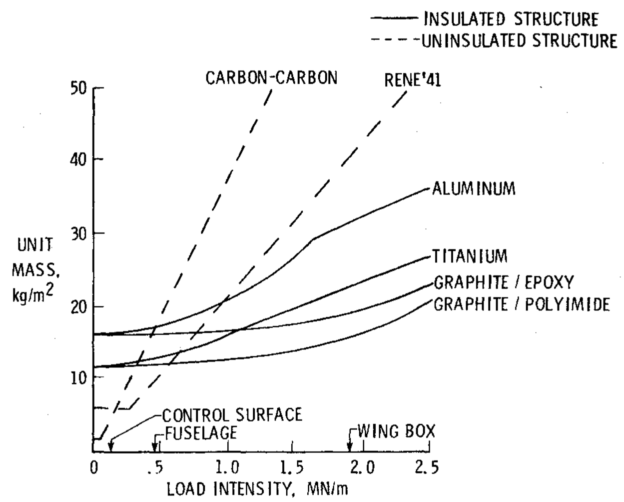

Fig. 8 Unit masses of thermostructural systems optimized for transient heating (T_{max} = 1090 K).

advanced thermomechanical processing procedure that severely cold-works the sheet material prior to a high-temperature (1420 K) solution anneal. This processing does not significantly degrade other mechanical or physical properties. It remains to be seen if the creep properties of precipitation-hardening nickel base alloys, such as Rene' 41, can also be improved by advanced thermomechanical processing.

Multiwall Tiles

Advances in manufacturing technology permit an examination or re-examination of a previously proposed concept for metallic TPS. This concept, called multiwall, shown in Fig. 5, was originally proposed as a combined TPS and structure for cryogenic tankage.[29] It was abandoned, however, when a vacuum-tight structure could not be obtained.[30] Repeated attempts to seal cracks, which emanated from the spotwelds used to join the dimpled sheets of the specimen on the left with repair welds, proved fruitless. Recently crack-free multiwall specimens of titanium have been fabricated (specimen on the right in Fig. 5) using the newly developed liquid interface diffusion bonding (LID) and super plastic forming diffusion bonding (SPF/DB) processes. Calculations[31] indicate that titanium multiwall can provide an efficient thermal barrier at moderate temperatures even when vented. The multiple layers are effective radiation shields, the dimpled foil construction provides low through-metal conduction, and the small cell size virtually eliminates gas convection leaving only gas conduction. The latter can, of course, be eliminated entirely if a vacuum-tight configuration can be maintained. As presently envisioned, multiwall would be used as individual insulating tiles.[31] A comprehensive review of the development of the multiwall concept is presented in Ref. 32.

As shown in Fig. 6, masses of the multiwall and standoff TPS are comparable to the RSI used on the current Shuttle. At lower temperatures, multiwall appears to be the better choice of metallic TPS, whereas at higher temperatures standoff TPS appears superior. The limits of the region of applicability of the two concepts have not been precisely defined and would probably vary for different applications. The use of multiwall to the upper limit shown would be dependent upon the development of fabrication techniques for bimetal construction (superalloy outer layers; titanium inner layers), which preliminary studies indicate is an attainable advancement (Contract NAS1-15646). The importance of design refinement is clearly indicated by the advanced standoff TPS designs. Although not attempted because of costs and availability of materials, application of the same refinements to the TD-NiCr configuration should result in comparable mass savings for the 1480 K design. Thus it appears that up to at least 1480 K metallic TPS are mass and thermally competitive with ceramic-type insulations, while providing the durability and reusability inherent with metallic surfaces. Additional research is required, however, to investigate the effects of curved and intersecting surfaces, shock impingement, local hot spots, etc., on TPS performance.[25]

Thermostructural Analysis

Shuttle experience has shown that the design of large structures for the hostile re-entry environment taxes the capabilities of existing thermal and structural analysis methods. The overall thermal model of the Shuttle Orbiter (Fig. 7) consists of 118 three-dimensional, lumped-parameter local models each having about 200 nodes. Temperatures are computed in each of the local models indicated by the shaded areas and interpolated to obtain temperatures in the unmodeled regions. To determine the times of occurrence of the critical combinations of thermal and mechanical internal loads, it is necessary to inspect temperature-time histories from the thermal analyses. This procedure, while somewhat standard for analysis of such complicated structures, constitutes a tedious, laborious, and expensive task. For future vehicles with potentially higher structural temperatures, longer heat pulses, and more closely integrated thermal and structural functions than the current Shuttle, the analysis problems will be magnified.

Consequently, three NASA centers (Dryden, Johnson, and Langley) have undertaken a coordinated effort to develop improved thermostructural analysis techniques. These improvements will include: 1) methods to automate the more tedious aspects of generating lumped-parameter and finite-element thermal models; 2) faster solution techniques for large-order matrix equations governing nonlinear, transient heat transfer; 3) methods to transfer temperatures from a thermal finite-element model to a dissimilar structural finite-element model; 4) methods of automating the determination of the times at which the critical combinations of thermal and mechanical internal loads occur; and 5) improved modeling procedures to reduce model size. This effort will also include the verification of procedures through comparison with experimental results obtained from large representative structural components. A variety of papers providing a status report on efforts to improve thermostructural analysis techniques were presented at a recent symposium.[33]

Although in the past the thermal and structural functions of thermostructural concepts have been analyzed separately and interactive effects have been handled approximately, advances in optimization processes[34] portend the development of effective integrated thermostructural analysis and design capability. With such capability, the selection of optimum structural systems for a future space transportation system can be enhanced at an early stage in the design process. An example of such optimization is illustrated in Fig. 8. The figure shows the unit mass, including both insulation and structural mass, for insulated structures of various materials optimized for a transient heating pulse that produces a peak surface temperature of 1090 K (the carbon-carbon and Rene' 41 structures are uninsulated since they can operate at the 1090 K temperature). The carbon-carbon material, shown for comparison with the more ductile materials, may be applicable to primary structures when satisfactory oxidation and fail-safe features are developed. The results are interesting in that for low loading intensities, typical of control surfaces and fuselage structures, the high-temperature, uninsulated materials are the lightest; whereas for higher loading intensities the insulated concept, even with relatively low-temperature materials, is the lightest. The beneficial effects of increasing the maximum use temperature of the primary structure is also evident as indicated by the superior performance of a graphite/polyimide composite over the graphite/epoxy.

Primary Structures

Composites

Over the past few years there has been an extensive effort devoted to the development of a firm technology base for a graphite/polyimide primary structure. This work has been carried out under the Composites for Advanced Space Transportation Systems (CASTS) program and was the subject of a recent conference.[35] The program includes material characterization, fabrication development, and structural design. As a focal point, the CASTS program has centered on replacement components for the Shuttle elevons and body flap. As indicated by Table 1, the replacement of the baseline Shuttle body flap with a graphite/polyimide body flap would result in a mass savings of over 25%.

The major portion of this mass savings is the result of the higher temperature capability of polyimide, which permits a significant reduction in TPS mass. Exploitation of the higher temperature capability has required the development of a higher bondline temperature adhesive for the RSI. Initial efforts in this area[36] have successfully demonstrated a new class of silicon-based adhesives that has an operational range of 90-640 K for bonding RSI to graphite/polyimide panels.

Hot Metallic Structures

Considerable effort has also gone into the development of structural concepts for high-temperature applications that employ curved elements to provide axial stiffness while alleviating transverse thermal stresses. These concepts, which are highly efficient, were conceived in 1965 as part of a hypersonic aircraft study.[37] The concepts have undergone subsequent development, including design optimization[38-40] and concept verification testing at both the elemental and component levels.[40-42] The latter, which were completed late in 1978, includes tests of both beaded and tubular panels as part of a hot wing structure that was tested in the Flight Loads Research Facility at Dryden Flight Research Center. During these tests the wing was subjected to combined thermal and structural loads, including temperatures up to 1000 K and loads up to the maximum design conditions. Although analysis of the data is incomplete, preliminary results indicate that the structures performed as expected and it may be concluded that curved-element hot structures represent a mature technology ready for flight demonstration.

Tankage Structures

As indicated previously, the containment of cryogenic fuel and oxidizer poses the most challenging thermostructural problem to the designers of future space transportation systems. There was, of course, considerable attention devoted to this problem in the preliminary design phase of the Shuttle; however, with the decision to use expendable tanks, the problems of reusable cryogenic tanks were circumvented. With the increased emphasis on reusability indicated by recent system studies,[1,2] there has been a renewal of interest in cryogenic tankage. The cryogenic tankage problem actually

Table 1 Mass reduction—shuttle aft body flap

Material	Structural mass, kg	TPS mass, kg	Total mass, kg
Aluminum	204	396	600
Gr/PI	161	279	440
Δ Mass	43	117	160

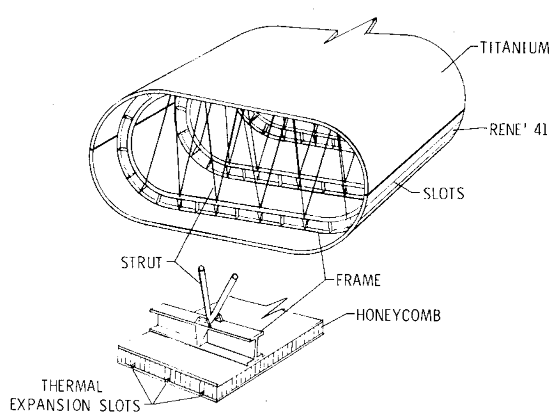

Fig. 9 Metallic honeycomb cryogenic tankage concept.

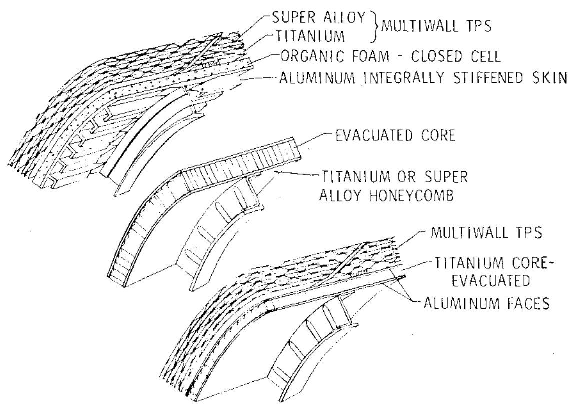

Fig. 10 Cryogenic tankage concepts sized for spacejet application.

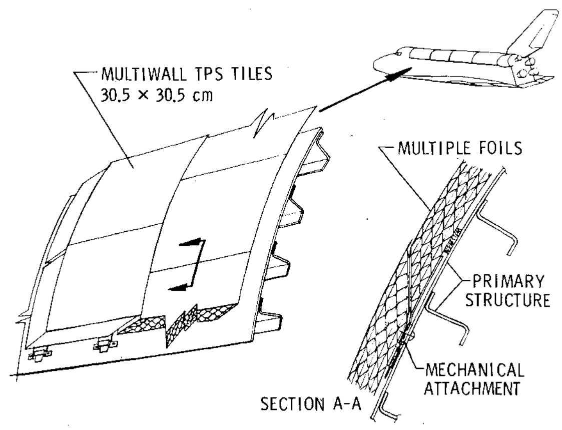

Fig. 11 Potential orbiter experiment—multiwall thermal protection.

resolves into two problems: 1) containing the relatively dense oxidizer (LOX); and 2) containing the much lighter, but thermally more demanding, fuel (LH_2). There appear to be compelling reasons[43] for containing the oxidizer in the wing of horizontal takeoff vehicles. However, the volumetric and thermal characteristics of liquid hydrogen necessitates its being stored in the fuselage and the demand for structural efficiency requires that the cryogenic tanks become an integral part of the primary structures.[1,2] The primary problem, which distinguishes hydrogen tankage from other cryogenic tankage, is the proclivity of liquid hydrogen to condense other gases because of its extremely low temperature (20 K). Air, or any potential 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. A variety of hydrogen tankage concepts have been proposed in the past,[5-7,44-46] including many which have not been documented. However, none of them has proven completely acceptable for multiple reuse applications because of mass, cost, durability, or operational complexity.

Most of the recent advanced tankage research has been directed toward an all-metallic honeycomb[2,47,48] structure. The choice of an all-metallic airframe system is based on the premise that only the inherent durability of metals can provide the high degree of reusability and reliability demanded by future space missions. The concept, shown in Fig. 9, uses evacuated honeycomb for the dual function of thermal protection and structure. Titanium honeycomb is used for the upper half of the fuselage tank which experiences only moderate heating. Slotted brazed Rene' 41 honeycomb is used for the lower half which experiences more severe heating. The wings, which comprise the oxidizer tanks, are of similar construction. Technology for the titanium honeycomb was developed as part of the supersonic transport effort; the brazing process for the Rene' 41 honeycomb was developed more recently by Boeing. A status report on subsequent Rene' 41 honeycomb structure development is presented in Ref. 48.

Additional research is required 1) to assess the thermal and structural performance of the longitudinal slots that are required to reduce thermal stresses, 2) to ascertain and improve the compatibility of the tankage materials with the fuel and oxidizer, and 3) to develop the facilities and techniques for fabricating and verifying the thermostructural performance of large tank components. Work to resolve the first of these requirements is currently underway. The compatibility of titanium with LH_2 and LOX (especially the latter[49]) is a more fundamental problem that may dictate a material system change. Some preliminary examinations indicate that the substitution of an aluminum matrix composite for titanium may be suitable and, in fact, even beneficial from a mass standpoint. However, as indicated earlier, the technology for aluminum matrix composites is not as advanced as that for titanium, and additional materials development and characterization would be required. Considerable resources will be needed to satisfy the third requirement, both in developing the techniques for forming and joining honeycomb panels into a complete tank structure and for verifying the thermostructural performane of load-carrying cryogenic tanks. Performance verification will require test facilities that can handle cryogenic hydrogen and simultaneously load and heat large tankage structures. Such capability will be required regardless of the thermostructural concept that is ultimately selected for future space transportation vehicles.

The metallic honeycomb approach and two other concepts, shown in Fig. 10, have been the subject of a recent analytical study.[50] All of the concepts have been sized for a horizontal takeoff, turbojet-boosted spacejet application. The concept, which uses closed-cell organic form to perform the all-important function of eliminating cryopumping, is the heaviest of the three and represents the current state-of-the-art. The other two, which combine the cryogenic insulation function with the basic structure, require technology advances to ensure vacuum leak-free construction but offer potential mass savings. As pointed out in the reference study, the all-honeycomb concept, which combines the high-temperature thermal protection with the cryogenic and structural functions, has the advantage of simplicity, but may experience more severe operational constraints because of the higher thermal stresses for this concept. The recently proposed multiwall/honeycomb concept, which is in the embryonic stage, minimizes thermal stress and material compatibility problems and appears to be an attractive candidate for future research.

Future Research

Ground Tests

In general, research needs and ongoing research in various disciplines have been identified in the preceding text; however, there remains a basic question as to the extent of large-scale thermostructural testing, and the consequent buildup of facilities that will be required to validate structures for future space transportation systems. Virtually every commercial and military aircraft has had an extensive full-scale structural ground test program involving one or more complete aircraft and specialized test equipment. For the Shuttle Orbiter, however, a different approach has been taken.[51,52] The Shuttle approach makes maximum use of existing facilities and uses development and verification tests of carefully selected test articles to augment analytical capabilities that are relied upon for the ultimate certification of the vehicle.

The selected approach to certification is probably adequate for the Shuttle; however, it may not be satisfactory for future space transportation vehicles where, in all probability, the structural and thermal functions will be more highly integrated, more advanced concepts will be employed to increase structural efficiency, and the mission life will be increased by at least a factor of 5. For the Concorde, which experiences relatively moderate, supersonic flight heating but is designed for long life, a complex, single-purpose facility[52] has been constructed for simultaneously heating and loading a complete vehicle. In addition, experience with the YF12 hot structure[54] and the hot-wing test structure[42] indicates that thermostructural performance of these structures is not readily predicted. The problem apparently arises from difficulties in predicting internal structural temperatures in a complex structure with sufficient precision for the corresponding thermal stress analysis.

Work is being initiated to assess the validity of these early findings as part of the analytical capability development effort previously mentioned. This effort will also provide insight into large-scale ground test needs and thermostructural test techniques. Initial efforts will concentrate on an examination of existing analytical and experimental data supplemented by additional tests of the hot-wing structure. It would be desirable to expand the effort to include the instrumentation and thermostructural testing of the Orbiter 101 flight vehicle using heating blankets and the extensive thermal and structural control and loading capability of the Flight Loads Calibration Facility at DFRC.[55] The 101 vehicle tests could yield extensive information on large-scale testing and provide an invaluable diagnostic tool for correlating results of the Shuttle flight test program.

Flight Experiments

The availability of the present Shuttle and the proposed Shuttle-launched entry research vehicle (SLERV) as flying test beds offer unprecedented opportunities in the 1980's for experimental verification of thermostructural concepts in an actual re-entry environment. Plans are presently being formulated for a comprehensive thermostructural development program for future space transportation systems that will include flight tests as part of the Shuttle Orbiter experiment (OEX) program.[56]

One potential OEX experiment would consist of the selective replacement of RSI tiles with metallic TPS as indicated in Fig. 11. Highest priority would be given to titanium multiwall tiles, which are a candidate replacement for the low temperature reusable surface insulation. In subsequent experiments, the RSI in higher temperature areas would be replaced by bimetal multiwall or standoff TPS, whichever appears superior based on ground tests.

Concluding Remarks

The next generation of space transportation systems will require reusability structural efficiencies, and durability characteristics approaching those of today's commercial aircraft. These requirements lead to the following research opportunities: 1) onboard integral cryogenic tankage; 2) low-mass fraction primary structures; 3) structures with closely integrated thermal and structural functions; and 4) metallic exterior surfaces.

The development of integral cryogenic tankage presents many challenging thermostructural problems. There are a variety of research opportunities in this area. The need to closely integrate thermal and structural functions to produce low-mass thermostructural systems requires improvements in our ability to analyze and reliably predict the behavior of complex structures under transient thermal conditions. New materials systems, such as fiber-reinforced composites, can significantly improve primary structure mass-fraction, but there remain many challenges to improve our ability to fabricate reliable low-cost structures with these new materials. The challenge to provide durable exterior surfaces affords the opportunity to further refine metallic thermal protection systems and surface structures to take advantage of recent advances in both materials and fabrication technology.

Finally, the Shuttle Orbiter experiment (OEX) program offers unprecedented opportunities for experimental verification of thermostructural approaches by providing access to the actual re-entry environment the concepts must endure prior to commitment to a vehicle/project.

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