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METALLIC THERMAL PROTECTION SYSTEM TECHNOLOGY DEVELOPMENT: CONCEPTS, REQUIREMENTS AND ASSESSMENT OVERVIEW

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ABSTRACT

A technology development program was conducted to evolve an earlier metallic thermal protection system (TPS) panel design, with the goals of: improving operations features, increasing adaptability (ease of attaching to a variety of tank shapes and structural concepts), and reducing weight. The resulting Adaptable Robust Metallic Operable Reusable (ARMOR) TPS system incorporates a high degree of design flexibility (allowing weight and operability to be traded and balanced) and can also be easily integrated with a large variety of tank shapes, airframe structural arrangements and airframe structure/material concepts. An initial attempt has been made to establish a set of performance based TPS design requirements. A set of general (FAR-type) requirements have been proposed, focusing on defining categories that must be included for a comprehensive design. Load cases required for TPS design must reflect the full flight envelope, including a comprehensive set of limit loads. However, including additional loads, such as ascent abort trajectories, as ultimate load cases, and on-orbit debris/micro-meteoroid hypervelocity impact, as one of the discrete-source-damage load cases, will have a significant impact on system design and resulting performance, reliability and operability. Although these load cases have not been established, they are of paramount importance for reusable vehicles, and until properly included, all sizing results and assessments of reliability and operability must be considered optimistic at a minimum.

INTRODUCTION

One of the major goals of NASA has been to develop enabling technology for future launch vehicles. The emphasis has been on a vehicle that would be lightweight, fully reusable and easily maintained under the assumption that such a vehicle would achieve low-cost access to space. The proposed Lockheed-Martin VentureStar™, shown in Figure 1, is one concept for a single-stage-to-orbit (SSTO) reusable launch vehicle (RLV) and has a goal of reducing the cost of placing payloads into orbit by an order of magnitude.[1] The economic viability of this, and other RLVs depends to a large extent on two critical factors: 1) achieving target gross and empty weights for the vehicle, and 2) meeting a series of Commercial-Transport-Like operations goals. It is anticipated there will be many parallels between aviation industry operations and maintenance (O&M) and future reusable launch vehicle industry practices.[2] However, the area of RLV O&M is in the early stages of evolving, and it is anticipated that the Federal Aviation Administration (FAA) or some other regulatory agency will ultimately develop regulations to ensure that RLVs are safe to re-fly. The resultant costs to the industry to comply with these necessary regulations may have a significant effect on the economic viability of the industry.[2] Compliance with FAA regulations for an RLV will require a vehicle design, operations and maintenance philosophy that is consistent with economic viability yet meets safety requirements. This philosophy must then be decomposed into a comprehensive set of specific requirements, criteria and compliance methods that can be used for vehicle design.

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One way to accomplish the first critical factor, to minimize weight, is to develop innovative and integrated concepts for the major airframe structural components, including the Thermal Protection and its Support System (TPS and TPSS) and the cryogenic propellant tanks. Taken together, these airframe components account for a significant portion of the RLV empty weight, and minimizing the TPS and TPSS weights are important because of the large surface area of a SSTO RLV. The TPS is attached to the tanks and intertank structures through the TPSS. Depending on the tank structural concept and stiffening arrangement, the TPSS may be attached to external stiffeners, such as ring frames and longerons, or to the outer skin of a sandwich tank structure. An early LaRC Metallic TPS Concept, with two different TPS Support Structure concepts has been compared to a baseline Metallic TPS developed for the X-33 flight vehicle.[3] Although the weights reported were preliminary, the results demonstrated that there are design alternatives for a metallic TPS system that are weight competitive with the X-33 design. Achieving the second critical factor, meeting a series of Commercial-Transport-Like operations goals, requires a concept that has flexible design features, so that performance (while mitigating critical failure modes) can be tailored to enable acceptable levels of operations and maintenance activity.

Subsequent to developing the early LaRC Metallic TPS concept[3], a technology development program was conducted to evolve the TPS panel design, including: improving operational features, increasing adaptability (ease of attaching to a variety of tank shapes and structural concepts), and reducing its weight. One objective of this paper is to introduce and review a suite of integrated airframe TPS/TPSS/Tank concepts that were developed for different airframe geometries and tank materials, and illustrate the adaptability of the TPS concepts for different vehicle packaging configurations (Figure 2). The resulting Adaptable Robust Metallic Operable Reusable (ARMOR) TPS system is designed with the goal of meeting the high flight rates and quick turn-around times required for an economically viable RLV. A key feature of the ARMOR TPS concept is that it allows weight and operability to be traded and balanced. As a commercial reusable launch vehicle with Commercial-Transport-Like turnaround, the RLV may be required to meet the structural design requirements of Federal Aviation Regulations (FAR) Part 25[4] or a similar set of regulations developed specifically for RLVs. Since the requirements in FAR Part 25 are not sufficient by themselves, additional requirements and design objectives must usually be developed. A second objective of this paper is to illustrate the development of these design requirements and objectives for TPS and decompose them into criteria that can be used to design the ARMOR TPS. A third objective is to define a comprehensive RLV loads envelope (including ascent and entry trajectories) from which time- and location-consistent loads can be developed. The loads envelope will be the basis for defining a subset of critical load cases and associated design loads for TPS sizing.

The results of this paper were used in additional, more detailed, studies of specific ARMOR TPS configuration, design and sizing issues. ARMOR TPS analysis, sizing and sensitivity trade studies were performed[5] and ARMOR TPS panels were designed, analyzed, fabricated and tested for a specific vehicle application.[6] More detailed analyses addressing specific design issues associated with ARMOR TPS panel flutter[7] and response to hypervelocity impact[8] were also performed. A more fundamental investigation of generic critical ARMOR TPS design issues and sensitivity of the panel performance to changes in design parameters has also been completed.[9]

MISSION AND VEHICLE DESCRIPTION

Mission and Service Objectives

The vehicle chosen for developing the ARMOR TPS technology is a Single Stage to Orbit (SSTO) Reusable Launch Vehicle (RLV) meeting VentureStar™ class performance (payload, achievable orbits, etc.) and operational (system flight rate, vehicle lifetime, reusability, etc.) requirements. In order to meet the goal of reducing the cost of placing a pound of payload in Low Earth Orbit (LEO) by an order of magnitude over current costs (reduction to ~ $1000/lb), Commercial-Transport-Like operations are needed. For the VentureStar™, only vehicle safing, propellant replenishment, and payload integration were deemed acceptable as routine operations to be performed after every flight.[10] In the same document, operations/maintenance manpower levels were defined as a total of 200 base personnel, with 50 of them being "hands-on" maintainers for both the two and seven day turnaround scenarios. These operational requirements will have a significant impact on which structural and material concepts are ultimately chosen for the RLV airframe. Some of the key mission and operational requirements defined for the VentureStar™ include:

- The system shall be designed for functional life of 20 years minimum.
- The flight vehicle airframe shall have a minimum design life of 100 reference missions.
- Post-flight maintenance and preflight operations are performed in 7 days or less.
- Under selected rapid turnaround conditions, vehicle turn around can be preformed in 48 hrs.
- At least 20 flights must be flown before any scheduled maintenance must be performed.
- The time to perform scheduled maintenance shall not exceed 14 days.
- The number of scheduled maintenance periods should not exceed 3 per year.

Configuration Geometry and Packaging

The lifting body configuration used to generate integrated airframe (with accompanying TPS) concepts, is a 2.62 million-pound gross liftoff weight (GLOW) class RLV vehicle with linear aerospike engines (Figure 1). The engines burn a mixture of liquid hydrogen (LH2) and liquid oxygen (LOX) with the tank volumes based on a 6.6/1.0 (averaged over entire ascent trajectory) LOX-Weight/LH2-Weight engine mixture ratio. The payload bay is located external to the tanks (as opposed to nestled between two LH2 tanks for the X-33 configuration) to take advantage of more efficient structural load paths and reduced tank weight.[11] The vehicle has a single LOX tank and a single LH2 tank, with the LOX tank packaged forward, and an intertank (not shown) connecting the two (Figure 2). Because of the potential for accommodating more efficient integrated TPS/TPSS/Tank concepts[3], the semi-conformal tank geometry is being considered along with the lobed-tank geometry (Figure 2) for generating TPS concepts. The tank planform is generated by a fifteen-degree included angle for this particular lifting body shape and the tank depth (through the vehicle thickness) is 157 inches.[11]

Flight Envelope and Trajectory Definition

A comprehensive analysis of the RLV airframe would require calculating or determining the loads acting on the structure for ascent and entry flight maneuvers (including flight in turbulence), landing, and ground handling conditions. A complete flight envelope for an RLV would encompass all possible ascent/entry trajectories that could be flown by the vehicle at all possible vehicle conditions. Enough points on and within the boundaries of the flight envelope must be investigated to ensure that the critical load condition sizing each part of the airframe is obtained. The resulting applied aerodynamic loads and vehicle accelerations (load factors) are a direct consequence of the set of ascent and entry trajectories flown.

A single nominal ascent and entry trajectory, which represents a limit load case, was used for the current conceptual-level TPS development effort. The trajectory was generated by the Vehicle Analysis Branch at LaRC for a lifting body configuration designated 003c.[12] More severe loads, due to launch abort scenarios for example, could also be generated to represent ultimate load cases, and hypervelocity impact of on-orbit debris or micrometeoroids could be used to generate Discrete Source Damage load cases. Although both of these additional sets of load cases would have a significant impact on vehicle operations and TPS design, neither is included as a design load condition in the current study. This omission is due to the lack of a comprehensive set of FAR-type requirements, which will be addressed in a subsequent section of this paper.

The ascent trajectory was designed to maximize the payload weight inserted into the target Space Station Freedom orbit of 50 x 248 nmi at a final inclination of 51.6 degrees.[13] The trajectory was optimized by adjusting the pitch attitude, engine power level and engine mixture ratio flight profiles, subject to constraints on angle-of-attack (-4 degrees $\leq \alpha \leq$ 12 degrees) and engine operating limits (0.2 $\leq$ engine power level $\leq$1.0, and 5.5 $\leq$ engine mixture ratio $\leq$ 7.0). An additional constraint was imposed on the mixture ratio profile; it had to be varied such that total fuel and oxidizer weights were consistent with the overall value of 6.0 used to size the tank volumes. Additional flight constraints imposed were: vehicle axial acceleration $\leq$ 3.0 g's; dynamic pressure, q, limited $\leq$ 600 psf; the quantity $|q\text{-}\alpha| \leq 1500$ psf; and the liftoff thrust-to-weight ratio = 1.35.

The entry trajectory is designed to limit the laminar heating to levels within the capability of the proposed metallic TPS, and to delay the onset of transition such that turbulent heating levels would not exceed those experienced during the preceding laminar phase of the entry. A reference heating rate limit of 45 Btu/ft^2-sec was used in the trajectory optimization process to constrain the radiation equilibrium temperatures to roughly 1800 degrees F on the windward side metallic TPS and 2000 degrees F on the nosecap and chine regions. The entry trajectory was also designed to meet a 750 nmi. cross-range constraint. Transition onset, angle-of-attack, bank angle, and control surface deflection limits were also included in the trajectory optimization process.[13]

INTEGRATED AIRFRAME CONCEPTS

Integrated Concept: Motivation and Definition

The subcomponents (defined previously) making up the airframe include the TPS, the TPSS, and the cryogenic tanks. Successfully defining integrated concepts requires understanding of: the parameters that most significantly impact each subcomponent design, performance and cost; interactions (and associated sensitivities) between each of the subcomponents; and, interactions between the integrated concept and other vehicle components and/or system parameters. Especially important is to identify and take advantage of synergistic effects and interactions between subcomponents and components in the complete system. As an example of a vehicle/integrated airframe concept interaction, the total depth of the integrated airframe concept (measured inward from the vehicle Outer Mold Line) interacts with the vehicle volumetric efficiency, and the vehicle weight is significantly impacted by this effect.[3] Thus, an integrated airframe concept that is thinner, but heavier than a thicker concept, may be better from a system perspective because the net effect of the additional airframe weight, when coupled with a decrease in vehicle weight (due to the increase in volumetric efficiency), could result in a lighter weight vehicle. Strong interactions may also occur between an integrated concept and safety or cost. As another example, the ARMOR TPS has the ability to protect the tank structure from on-orbit debris and micro-meteoroid impact.[8] Improving the protective capability of the ARMOR TPS might incur an increased weight for the TPS, but could lead to an increase in vehicle safety and perhaps a decrease in operations costs if the need for tank inspection/repair was reduced. The philosophy of concept integration asserts that: if done at the outset, defining and optimizing a complete airframe concept, in the context of the entire system, can lead to a better product when compared to suboptimizing each individual subcomponent (where little or no consideration was given to subcomponent impacts and interactions with each other). This paper will focus on integrated airframe concepts and their features, but recognizes that additional benefits at the vehicle/system level (in terms of performance, safety and cost) are likely to also be incurred based on the examples given.

Concepts Menu And Concept Features

An integrated airframe concept can generically be defined as a layered series of subcomponents that must provide an integrated set of functions. These functions (see Table 1) include, proceeding from the outer surface inward; react aerodynamic pressure, withstand aerodynamically-induced heating, protect/isolate inner components from high temperatures, support and transition between adjoining layers, react vehicle flight loads, accommodate thermal gradients within and between subcomponents, store thermal energy, transmit loads between subcomponents, condition propellant (fuel and oxidizer) to minimize losses, and contain propellant under pressure. The layer that performs a particular function will depend on details of the Integrated Airframe Concept definition. The options available to perform each of the functions are numerous and vary in their materials, performance and design features, as illustrated in Table 1. Other features will also have an impact on an integrated airframe concept definition. For example, any difference between the vehicle OML and the tank surface geometry must be accommodated by an intermediate layer within the system. The desired size and geometry of the TPS will interact with lower supporting layers, as will the orientation of any seams or seals to accommodate or mitigate flow effects.

A menu of integrated airframe concepts was developed with the goal of maximizing the combination of options for each layer. The focus of this technology development effort was exclusively on airframe concepts that included the ARMOR TPS configuration. At the highest level, a distinction was made between integrated concepts for lobed and semi-conformal tank geometries (Figure 2). The lobed tank geometry, at a minimum, would require bridging structure to span the valleys between the lobes, while the semi-conformal tank geometry allows for the possibility of completely eliminating bridging structure.

Two key geometry parameters that affect the definition of an integrated airframe concept are the shape and size of the TPS panel. Both parameters impact, and can be traded against performance and weight, operations and cost, and safety and reliability, as summarized in Table 2. In general, for all shapes, increasing panel size would reduce total panel part count (and also, number of spares) and reduce the number of attachment points and amount of associated hardware (impacting operations and inspection).

A primary feature of the ARMOR TPS panel concept is that the pressure seals are on the lower, cooler surface. Thus, the subcomponent supporting the panel must provide a surface for the seals to react against. Selection of the TPS panel size and geometry must be done with consideration for providing these sealing surfaces, and consideration of the tank geometry, tank stiffening requirements, and general arrangement of the tank structure. Three general classes of integrated concepts have been developed using the ARMOR TPS concept. For semi-conformal tanks, if sandwich tank walls develop sufficient stiffness and strength such that no additional stiffening is needed, TPS panels can be attached directly to the tank wall, thus precluding any need for TPSS, as shown in Figure 3a. Also, for semi-conformal tanks, if an integrally stiffened concept is used and the stiffeners are external, the TPS can be directly attached to the stiffeners. In this case, the size and shape of the TPS panels must be compatible with the geometry and spacing of the stiffeners (Figure 3b). For general stiffened-skin tank walls, whether semi-conformal or non-conformal, the size and shape of the TPS panels must be compatible with any required TPSS and bridging structure. Square and hexagonal TPS panels can also be integrated with the frames of a stiffened-skin tank wall (Figures 3c and 3d respectively). Additional features, pros and cons associated with each of the three concept classes are summarized in Table 3.

ARMOR TPS Concept

The ARMOR TPS concept was evolved from the early LaRC Metallic TPS concept[3,9] in order to improve operations features, increase adaptability (including ease of attaching to a variety of tank shapes and structural concepts), and reduce its weight. The early LaRC Metallic TPS concept was a superalloy honeycomb sandwich TPS consisting of lightweight fibrous insulation encapsulated between two honeycomb sandwich panels, as shown in Figure 4. The panels were designed to be mechanically attached directly to a smooth, continuous substructure. Each panel was vented to local pressure so that the substructure, rather than the outer honeycomb sandwich of the TPS, would carry aerodynamic pressure loads. The outer surface consisted of a foil-gage Inconel 617 honeycomb sandwich and the inner surface was a titanium honeycomb sandwich with part of one facesheet and core removed to reduce weight. Beaded, foil-gage, Inconel 617 sheets formed the sides of the panel to complete the encapsulation of the insulation. The perimeter of the panel rested on a RTV (room temperature vulcanizing) adhesive coated Nomex felt pad that prevented hot gas from flowing beneath the panels, provided preload to the mechanical fasteners, and helped damp panel vibrations.

The improved ARMOR TPS panel that has been fabricated[6] is shown in Figures 5 and 6. The outer surface consists of an Inconel 617 honeycomb sandwich. The outer facesheet of the honeycomb sandwich extends past the sides along two edges to cover the gap between adjacent panels. Mechanical fasteners are located at the four corners of the TPS panel. The ARMOR TPS panel fabricated for the technology development program[6] was chosen to be the size of a nominal X-33 panel: 18-in. square. Other materials could be used for the outer sandwich to decrease the TPS weight. In the maximum temperature range of 1400ºF to 1500ºF for example, gamma titanium aluminide, which has less than half the density of Inconel 617, could be used. However, a number of fabrication and joining issues need to be solved to make gamma titanium aluminide an affordable, efficient sandwich structure. Some of the oxide-dispersion-strengthened (ODS) alloys may also be used to extend the operating temperature range of the TPS if they can be made into an efficient structure for the outer sandwich surface.[9]

One of the main improvements of the ARMOR design is the use of bowed, compliant sides and support brackets instead of the stiff beaded sides of the earlier superalloy sandwich concept. The compliance of the sides decouples the thermal expansion mismatch of the upper and lower surfaces. In addition, the nonstructural compliant sides can be made thinner than the beaded sides, thus saving weight. Radiation heat transfer in the gaps between the panels can be greatly reduced by bowing or bulging the sides to fill the panel-to-panel gap. The support brackets (Figure 6) are built into the TPS panel and are sized to transmit pressure loads from the outer surface to the attachments while providing minimal restraint to thermal growth of the upper honeycomb sandwich. This thinner support bracket also minimizes the direct conduction heat short from the upper surface into the support structure.

The improved ARMOR TPS panel has several other features. A lightweight titanium frame on the lower surface provides support for the compliant sides and for a titanium foil sheet that closes out the bottom of the panel. The frame also provides a stiff edge around the bottom of the panel to seal against the felt strips under the panel perimeter (similar to the arrangement in Figure 4). The size and geometry of the fastener holes in the titanium frame are designed to decouple the frame from any mechanical strains in the vehicle structure, as well as the thermal expansion mismatch between the frame and the underlying structure. This decoupling is achieved by slotting the holes so that the panel can expand freely from one of its corners. Bellows-type fastener access tubes (Figure 6) provide access from the outer surface to the mechanical fasteners, yet avoid significant structural coupling between the outer honeycomb sandwich and the attachments. Snap-in fastener access covers close out the top of the fastener access tubes to provide a smooth outer surface. The interior of the panel can be filled with fibrous insulation, advanced multi-layer insulation, or combinations of non-load-bearing insulation materials.

METALLIC TPS DESIGN REQUIREMENTS

Designing and manufacturing economically viable vehicles that successfully balance performance, safety and cost is accomplished on a regular basis by commercial (aircraft) transport manufacturers (Figure 7). However, safety, in the form of demonstrating compliance to applicable Federal Aviation Regulations (FARs), is not negotiable for transports, and successfully achieving economic viability requires an ability to trade cost and performance to meet safety requirements as the design matures. The performance and costs of a particular airframe system are intimately linked with each other through details of the design implementation, usually in an opposing manner. Traditional launch vehicle design has emphasized developing high performance systems that operate at or close to design limits, with little or no emphasis on operations and economics. In order to achieve economic viability, full life-cycle system costs must now be considered, and to achieve Commercial-Transport-Like operations requires establishing FARs (or something equivalent) for commercial launch vehicle development.

The economics of a launch vehicle will depend on the total launch cost, which for a RLV, is the sum of the following six individual cost components: 1) amortization of nonrecurring development cost, 2) amortization of vehicle production cost, 3) total cost of flight operations (per flight), 4) recurring cost of recovery, 5) refurbishment cost, and 6) cost of launch insurance.[14] An attempt has been made to address cost in the current RLV program by defining operational goals in the form of a set of mission and service objectives (listed in previous section of this paper). The magnitude of each individual cost component will then depend on the specific design implementation of the system which meets the mission and service objectives. The critical link between the two, which establishes the traceability between operational goals and the design implementation, is a set of (FAR-type) airframe design requirements and compliance methods. These must be established as performance-based requirements rather than design-specific requirements. Items such as acceptable design philosophies (damage tolerance and safe life for example) and associated implications, flight envelope definition (speed, load factor, and mass conditions), design load conditions (limit, ultimate and discrete-source-damage) and associated factors of safety, damage scenarios and requirements (lightning strike, hail strike, engine rotor burst, etc.), inspection intervals and levels, etc. would be included.

Unfortunately, due to a lack of knowledge for this new class of vehicle (RLV), this critical link is absent from the current RLV design process. Thus current airframe concepts are being defined, designed, assessed and compared to competing concepts without verifiable and traceable assurance that safety requirements and cost goals are being met. An initial attempt at establishing this critical link for TPS is made in this paper.

General Requirements

Because the TPS forms the external (aeroshell) surface of an RLV, it will be exposed to (and must be designed for) environments in addition to those experienced during entry. The type and nature of these additional environments will derive from the operations that are required to meet the final set of mission and service objectives to make the vehicle economically viable. A proposed set of general requirements that apply to an external TPS system are described in this section. These requirements use wordage similar to statements made in FAR Part 25.[4] Although the specific content of the proposed requirements is open to debate, the final list should be comprehensive if the safety of the vehicle and ability to obtain certification is to be assured. This list can serve as an initial point of departure, but should by no means be considered as all inclusive at this early stage of RLV development.

Durability and Damage Tolerance Requirements

(a) General. An evaluation of the strength, detail design, and fabrication must show that failure due to fatigue, corrosion, manufacturing defects, or accidental damage, will be avoided throughout the operational life of the airplane[4]. This evaluation must be conducted in accordance with the provisions of paragraphs (b) and (e) of FAR Part 25, Section 25.571, except as specified in paragraph (c) of that section, for each part of the structure that could contribute to a catastrophic failure (such as wing, empennage, control surfaces and their systems, the fuselage, engine mounting, landing gear, and their related primary attachments).

Inspection thresholds for the following types of structure must be established based on crack growth analyses and/or tests, assuming the structure contains an initial flaw of the maximum probable size that could exist as a result of manufacturing or service-induced damage:
(i) Single load path structure, and
(ii) Multiple load path "fail-safe" structure and crack arrest "fail-safe" structure, where it cannot be demonstrated that load path failure, partial failure, or crack arrest will be detected and repaired during normal maintenance, inspection, or operation of an airplane prior to failure of the remaining structure.

(b) Damage-tolerance evaluation. The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. Repeated load and static analyses supported by test evidence and (if available) service experience must also be incorporated in the evaluation. Special consideration for widespread fatigue damage must be included where the design is such that this type of damage could occur. It must be demonstrated with sufficient full-scale fatigue test evidence that widespread fatigue damage will not occur within the design service goal of the launch vehicle.

Metals Durability An S-N based durability analysis shall be performed to demonstrate that (fatigue) cracking within the design service objective will be highly unlikely. The fatigue capability of the structure shall be demonstrated by full scale, or sub-component testing, for four lifetimes of spectrum fatigue loading.

Primary bonded metallic structure shall be capable of sustaining ultimate load, after one lifetime of fatigue cycling with “small” levels of damage in the structure. This shall be demonstrated through element and component testing. “Small” shall be defined as the threshold inspection resolution for the material/structural concept, or a 0.5-inch diameter delamination, whichever is larger.

In addition to the ultimate load requirement in the paragraph above, adhesively or metallurgically bonded structure shall be capable of sustaining limit load with a disbond between the facesheet and core or between skin and stiffeners. The disbond size may be limited by design concepts which are shown to contain growth of the damage (e.g., splices, fastener rows), if it can also be shown that the resulting loads do not cause failure in adjacent panels. It must be shown that damage will not propagate to critical levels within four inspection opportunities.

Damage Tolerance Requirements Vehicle structure shall meet the following damage tolerance requirements (defined in FAR 25-571). As an objective, primary structure, including the fuselage, main wing box, intertank, thrust structure, and tanks, shall be capable of sustaining limit load with obvious damage. In general, obvious damage will be considered as one of the following:
- For tension structure, either a two-bay skin crack with central failed support member (e.g. a stringer), or a 12-inch skin crack centered on a failed support member (if applicable), whichever is smaller.
- For compression structure, a six inch wide discrete damage penetration, except as noted below.
- For sandwich concepts (e.g. honeycomb), the damage shall be through the full depth of the sandwich. This design objective applies to all material concepts in all areas of the vehicle, and shall be validated though analysis and test.
- For primary structure not meeting the obvious damage objective above, it shall be demonstrated that damage will be detected prior to the structure degrading to the extent that regulatory loads can no longer be sustained.
Note: In areas susceptible to discrete source damage (in the vicinity of engines, engine pumps, and other rotating machinery), the damage sizes shall be dictated by the threat, when larger than the criteria above.

Ground Hail Zone and Requirements

Exposed/external structure (such as TPS) shall meet ultimate load requirements, and not require immediate repair following impacts spaced 12 inches apart, as specified in Table 4. Impact energies for surfaces between vertical and horizontal may be calculated as:

$$E = E_v + (E_h - E_v) \sin\theta \qquad (1)$$

where θ is the angle between the surface and the vertical plane.

Runway Debris Zones and Requirements

Primary external structure exposed to runway debris shall have strength and reparability characteristics equivalent to structure on current commercial transport airplanes.

Lightning Strike Zones and Requirements The external structure shall be protected against the catastrophic effects of lightning (FAR 25.581).

Provisions such as the addition of conductive materials shall be made to ensure the survivability of all structure and systems to the aforementioned lightning strike conditions. Weight statements for each structural or material concept shall include lightning protection requirements.

Rain and Rain Erosion The vehicle TPS and structure shall be capable of launch, entry and landing in rain at the rate of TBD in/hr with no damage due to rain erosion. If susceptible to moisture ingression, the TPS and or structure shall be either protected against detrimental effects, or, it shall be demonstrated that damage will be detected prior to the structure degrading to the extent that regulatory loads can no longer be sustained.

In-Flight Discrete Source Damage Requirements

The vehicle shall be capable of successfully completing the flight during which likely structural damage occurs as a result of any of the following:

- Impact with a 4-lb bird, at likely operational speed and altitudes up to 8000 feet (FAR 25.571) during ascent and entry.
- Impact with an 8-lb bird on the empennage, when the velocity of the vehicle equals a likely operational speed at sea level (FAR 25.631) during ascent, entry, and/or landing.
- Uncontained engine failure and engine disc component or rotating machinery penetration of any structural element in its possible path. It will be assumed that a failed rotor disc or other fragment will have infinite energy and will pass through anything in its path.
- A loose, or thrown tire tread from a tire rotating at landing speed. Assume a loose tread size of 10 inches long by 15 inches wide. Assume a thrown tread size of 7 inches square.
- Tire burst in the wheel well at any flight altitude.

Hyper Velocity Impact The vehicle shall be capable of successfully completing the flight during which likely structural damage occurs due to Hyper Velocity impact from on-orbit debris and/or micrometeoroid particles up to a size of To-Be-Determined (TBD). (For particles larger than TBD, it must be demonstrated that the vehicle has the means of detecting and avoiding collision with the particle.) Analyses shall be performed to define particle size and velocity distributions that take into account vehicle surface area, orbital altitude and inclination, duration on orbit, etc. Analyses shall be performed to determine the size of resulting damage zones in the TPS[8] and underlying primary airframe structure, for use in residual strength analysis. The potential for tank loss-of-pressure on orbit and the associated consequences (negative, i.e. crush tank pressure) on entry must be accounted for.

Material Compatibility with Environments

Materials selected for TPS and tank systems must be compatible with the fluids and other exposure environments.[15] Possible effects to be evaluated include (but are not limited to):

- Material degradation due to exposure (metal corrosion for example)
- Catalytic decomposition of propellants
- Hydrogen embrittlement
- Material contamination
- Stress corrosion
- Galvanic corrosion
- Ignition of materials (in presence of oxidizers).[15]

Metallic TPS: Specific Requirements

Minimum Gage Minimum gage for the Metallic TPS sandwich panel facesheet and core material, based on manufacturing considerations, is given in Table 5. It may be necessary to increase minimum gage values to satisfy hail, lightning strike, bird strike, etc. requirements.

Deflection Deflection limits are established for the following conditions:

Transition. During entry flight above Mach 5.0, the external surface deflection for an individual panel, and gaps between adjacent panels of the TPS shall be no greater than the values listed in Table 6 (unpublished data obtained from BF Goodrich) to prevent transition of the boundary layer from laminar to turbulent. Below Mach 5.0, no panel deflection requirement is imposed. The deflection limits may be waived if a turbulent boundary layer is assumed. However, the assumption must be consistent with that being used to calculate the aerothermal loads.

Local Heating. If deflections are no greater than those listed in Table 6, additional local heating due to panel deflection is assumed to be negligible.

Insulation. In order to prevent permanent compaction of the fibrous insulation in the metallic TPS (which is located between the external surface sandwich panel and the lower surface picture frame box), the deflection of the upper sandwich panel shall be less than 10 percent of the total TPS panel thickness at all times.

Oxidation The effects of oxidation shall be addressed either by coating the TPS surfaces exposed to the oxidation environment during entry, or accounting for loss of panel strength, stiffness, material thickness, etc. in the panel design. For coatings, end-of-life performance and integrity must be validated through appropriate durability testing. For uncoated panels, end-of-life material properties resulting from oxidation exposure, shall be used for all margin-of-safety calculations.

Creep The effects of creep must be addressed during design of the metallic TPS panels and permanent creep deflection, stress and/or strain included in appropriate assessments. The creep deflection, stress and/or strain at end-of-life will be added to the elastic panel deflection, stress and/or strain when determining compliance to design criteria.

Low Cycle Fatigue Durability analyses will be performed, using a typical mission profile, to demonstrate that the initiation of cracks within the design service objective is highly unlikely in all TPS and TPSS components. The fatigue capability shall be demonstrated by full scale, or subcomponent testing, for two lifetimes of spectrum fatigue loading.

High Cycle Fatigue Durability analyses will be performed to demonstrate that the initiation of cracks within the design service objective is highly unlikely in all TPS and TPSS components due to the acoustic environment induced by engine noise. The total exposure duration per flight shall be consistent with assumed engine operations during liftoff and ascent. The fatigue capability shall be demonstrated by full scale, or subcomponent testing, for two lifetimes of spectrum fatigue loading.

LOADS FOR TPS DESIGN

The loads described in this section are derived from a single nominal ascent and entry trajectory (as described previously in the mission description) and represent limit loads. This must suffice for the conceptual-level design and analyses currently being performed, until agreement to establish and use a consistent and more comprehensive set of loads that include ultimate (due to abort for example) and Discrete Source Damage are established. In general, separate loads need to be derived for the windward and leeward sides of the vehicle because of the vastly different conditions experienced by the two surfaces. A full set of TPS applied limit loads would include: 1) aerodynamic pressure, both normal and parallel (drag) to the surface, 2) inertial loads, 3) heating rate, 4) acoustic pressure, 5) vibration, and 6) shocks (especially on seals). Derived loads needed to size the TPS[5] at specific locations are the heating rate (which will determine the temperature of the various TPS components as a function of time), and the applied pressure (which will size the upper TPS sandwich panel and the support brackets). The applied pressure is the total of the aerodynamic-induced and the engine-acoustic-induced on ascent, and the aerodynamic-induced and aerodynamic-acoustic-induced on entry. Material properties are modified according to an element's temperature, and thermal stresses and deflections can be introduced due to differences in temperatures (thermal gradients) between elements. The maximum temperature reached by the TPS material, together with the time at temperature is used to select the TPS material at each location on the vehicle.

Vehicle Surface Aerothermal Loading

Aerothermal loads were derived based on the ascent and entry trajectory designs described previously. The loads, consisting of heating rates (as a function of time) and normal static surface pressures, were generated for both ascent and entry on the windward and leeward vehicle centerline. On ascent the angle of attack is always low, so the leeward and windward results are very similar. The windward-side radiation equilibrium temperature distribution induced on the vehicle by the entry (generated using LATCH, an inviscid boundary layer method[13]) is shown in figure 8 for the peak laminar heating condition, which occurs at Mach 20, and 237 kft altitude. An emissivity of 0.86, representative of Inconel, is used to generate these temperatures. The entry environment is based on a low heating rate trajectory that is optimized for a metallic (versus a ceramic) TPS. A conservative value (225) of the transition parameter, $R_{e\theta}/M_e$, was selected for use in the trajectory optimization process. Unlike the length-based Reynolds numbers typically used for a vehicle at a constant cruise condition, this parameter, the momentum thickness Reynolds number divided by the local Mach number accounts for angle-of-attack effects, which are critical for determining transition. The trajectory was designed to delay the transition to turbulent flow, such that turbulent heating levels would not exceed the peak laminar heating levels and thus not require a change in materials to accommodate higher temperatures. The ascent and entry aerothermal environments were generated using the engineering analysis code, MINIVER.[16] Radiation equilibrium temperatures were assumed to approximate the surface temperatures for determination of the applied heating rates. Because the local pressures generated within MINIVER are based on compressibility effects, these results were not used at transonic and subsonic conditions. Thus, a value of 1.5 was chosen as a cutoff Mach number value, below which MINIVER pressure results were not used.

The heating rates during ascent for the windward surface of the vehicle are shown, as a function of time, in Figure 9a. The rates are shown for a number of vehicle stations (x locations), starting near the nose (x=0.0 inches). The peak heating rate along the vehicle centerline location for ascent is shown in Figure 9b. Other than near the vehicle nose (a stagnation point), the heating rate is very low (less than 0.5 BTU/ft^2-sec) on the vehicle surface and will not significantly influence the TPS material choice, thermal design and sizing. Heating rates and peak heating for the vehicle windward side for entry are shown in Figures 10a and 10b, and for the vehicle leeward side in Figures 11a and 11b. The entry simulation is initiated at an altitude of 400,000 feet and Mach = 28 (relative orbital velocity of 25,000 feet/second) and terminated approximately 2,100 seconds later at an altitude of 86, 000 feet and Mach = 2.55 (2,500 feet/second). The shapes of the heating rate curves for both the windward and leeward sides of the vehicle are very similar for all locations. The rate rapidly rises and reaches a peak value at between 400 to 500 seconds after entry, plateaus at the peak rate for approximately 1000 seconds, and then begins decreasing through the end of flight (Figures 10a and 11a). On the windward side, the peak heating rate drops rapidly with distance from the vehicle nose, decreasing by 50 percent (from 18 to 9 BTU/ft^2-sec) in the first 120 inches and below 25 percent at 500 inches (Figure10b). An even quicker reduction takes place on the leeward side, with the heating rate asymptoting to approximately 0.5 BTU/ft^2-sec beyond 500 inches (a level very similar to that experience during ascent).

The transient nature of the heating response suggests that thermal gradients will occur in the metallic TPS which should be evaluated during the entire entry, and thus considered when compiling load cases for sizing (especially with respect to any requirements or limitations on deflections). The large gradients in peak heating values along the vehicle centerline, for both the windward and leeward sides, as well as the large difference in magnitudes between the two sides, indicate that different TPS metallic materials, concepts and insulation thicknesses all need to be considered for different locations on the vehicle.

Local Static Pressure Loads

Ascent and entry aerodynamic surface pressures (static normal) for approximately 15 locations along both the windward and leeward centerlines on the vehicle were also calculated as part of the environments. In order to size the TPS, the applied pressure and element temperatures are needed at the time of occurrence of critical ascent and entry flight conditions. For the integrated TPS/Tank concept being considered (Figure 3c) a cavity exists between the base of the metallic TPS panels and the outer mold line of the tank wall. This cavity is assumed to be vented to the atmosphere to allow pressure equalization on ascent (pressure goes from 1 atmosphere to vacuum) and entry (pressure goes from vacuum to 1 atmosphere). Then, the pressure in the entire vehicle cavity would be equal to the local static atmospheric pressure at altitude. However, in a real vehicle, the flow paths, cavity location with respect to vents, and finite flow rates associated with vent areas would most likely lead to a time lag in pressure stabilization in some cavities. Without completing a detailed vehicle design, it would be difficult to estimate the pressure lags and differentials that might occur due to this effect. As a result, it has currently been assumed that there is no pressure lag. Thus the local static aerodynamic pressure differential at any location on the vehicle, is assumed to be

$$\Delta p_{aerodynamic} = p_{local\ static} - p_{atmospheric} \qquad (2)$$

The values of equation (2) are given for ascent and entry on the windward and leeward centerlines in Figures 12a – 12c. Due to the limitations of the analysis methods used by MINIVER to calculate the local static pressures cited previously, the pressures for Mach numbers below 1.5 are not accurate, leading to very low Δp values at the beginning of ascent and the end of entry, as shown in the figures. A method is needed to generate Δp values for Mach numbers below 1.5, since the maximum values occur at those speeds during both ascent and entry, as shown in Figures 13a – 13c. The aerodynamic component of the pressure load data, for three flight conditions, and used for Venture Star™ TPS initial sizing was obtained (unpublished data from BF Goodrich-see Table 7). Since the times of the flight conditions and the location or locations for the pressures given in Table 7 were not available, the following assumptions were made:

1) The conditions all occur at the same location on the vehicle with that location being on the forward windward centerline.

2) The maximum Δp on ascent, for all locations on the windward and leeward sides of the 003C vehicle occurs between approximately 60 and 80 seconds, when the vehicle is at Mach 1.4 (Figure 13a). The ascent condition listed in Table 7 is also assumed to occur at M<1.5 during ascent.

3) For the 003C vehicle entry, peak heating begins at approximately 450 seconds and ends at approximately 1390 seconds with the rate remaining essentially constant between the two times. During that time, the corresponding value of Δp reaches its maximum at the end of the peak heating period. It is assumed that the Peak Heating condition in Table 7 refers to the beginning of the peak heating period when the pressure is lowest.

4) From the 003C entry aerothermal data, and for vehicle locations between STA 0 and STA 290, the peak Δp occurs at approximately 2250 seconds at Mach 1.3, with a smaller peak at Mach 6.8. At locations greater than STA 290, the peak Δp occurs at the Mach 6.8 condition, with the peak at Mach 1.3 becoming smaller. However, on the leeward side, the maximum Δp for all locations occurs at Mach 1.3. The pressure listed for Entry in Table 7 is assumed to also occur at similar representative mach numbers.

Δp Modification For Aerodynamic Pressure

For any ascent/entry condition at M<1.5, the value of Δp given in an aerothermal data set (such as that for the 003C) was replaced with a modified value, based on pressure ratios derived from Table 7, as follows.

1) If the entry $\Delta p_{max,entry}$ occurs at M>1.5 and is considered accurate, then the maximum Δp for **ascent** is modified to be:

$$\Delta p_{max,ascent} = 2.3 * \Delta p_{max,entry} \quad (3)$$

(2.3 = 2.3/1.0 - values from Table 7)

2) If the entry $\Delta p_{max,entry}$ occurs at M<1.5, modified Δp values are calculated for both ascent and entry using the Δp for peak heating as follows:

$$\Delta p_{max,ascent} = 15.3 * \Delta p_{peak\ heating} \quad (4)$$

(where 15.3 = 2.3/0.15)

$$\Delta p_{max,entry} = 6.7 * \Delta p_{peak\ heating} \quad (5)$$

(where 6.7 = 1.0/0.15)

3) To calculate the pressure for any ascent condition other than $\Delta p_{max,ascent}$, and that occurs at a Mach number M<1.5, the pressure at the condition can be divided by the pressure at maximum Δp to obtain a ratio. That ratio is multiplied by the modified $\Delta p_{max,ascent}$ from step 1 to obtain the corrected pressure.

Note: if the data allows process 1 to be used to calculate $\Delta p_{max,ascent}$, a similar value should be obtained if process 2 is used, and the two values can be compared as a verification.

TPS Design Pressure

Equivalent static pressures induced by engine acoustics at liftoff and aerodynamic acoustics during ascent and entry were calculated. The resulting pressures at three locations on the vehicle are summarized in Table 8. For a given flight condition, the total applied pressure used for TPS design and sizing is a function of the aerodynamic pressure and the acoustic pressure. The aerodynamic pressure is always assumed to be acting inward, while the acoustic pressure is oscillating and can be acting inward or outward. Thus, the following two total pressure values are calculated:

$$\Delta p_{ultimate,TPS}+ = [\ \Delta p_{aerodynamic} + 3\ \Delta p_{rms,\ acoustic}\ ]\ 1.4 \quad (6)$$

$$\Delta p_{ultimate,TPS}- = [\ \Delta p_{aerodynamic} - 3\ \Delta p_{rms,\ acoustic}\ ]\ 1.4 \quad (7)$$

where,

1.4 is the factor of safety,
$\Delta p_{aerodynamic}$ is from equation (2), modified if necessary using the procedure from the last section,
$\Delta p_{rms,\ acoustic}$ is the equivalent rms pressure due to acoustics. Two conditions are defined:

Liftoff - induced pressure is due to engine acoustics, aerodynamic pressure is zero.

Ascent/Entry - induced pressure is due to aerodynamic noise. Nominal entry values were calculated[17] and are given in Table 8 for three locations on the windward and leeward surfaces of the vehicle.

For the preliminary TPS sizing currently being performed, any additional pressure induced by oscillating acoustic shocks are ignored since none of the ascent or entry load cases occur at transonic conditions. However, it is recognized that detailed panel design for a flight vehicle would need to include oscillating shocks, especially for sizing the overlapping seals between panels.[7]

Load Conditions For TPS Sizing

Knowledge of the ascent and entry flight envelope, aerothermal environments, acoustic environments and details of the particular integrated airframe design are used to define a set of critical load conditions for TPS sizing. Detailed sizing would require a comprehensive set of loads that interrogate the entire flight envelope, including limit, ultimate and discrete-source-damage cases. However, for the preliminary sizing of the ARMOR TPS described in reference 5, some general observations, based on the information from this study, were used to guide compilation of a small subset of loads that captured critical aspects of the pressure and heating profiles. First, large local static pressures are induced by engine acoustics at liftoff for all locations on the vehicle. As a result, these pressures must be calculated and will be critical for sizing many of the TPS panel elements. Since heating on ascent is much more benign than entry, no other ascent load cases are likely required. Second, at the initial onset of the peak heating plateau, the metallic TPS external surface will quickly rise to it's peak temperature. Since it will take time for the heating to penetrate into the rest of the system, the maximum thermal gradients are likely to occur at or near this time. Although the normal pressure is very low and stresses are probably also low, the maximum thermal gradient condition should be included because deflections may be critical at this point (especially for boundary layer transition). Third, at the end of the peak-heating plateau, most components in the TPS will be at their maximum temperature. Although normal pressure is still low (relative to ascent), degradation of material properties at elevated temperature and creep are both of concern and this condition should also be included. Fourth, and finally, the maximum static pressure differential during entry will occur while the TPS materials are at elevated temperature. Significant static pressure, coupled with elevated material temperatures (and associated degradation of properties) require that this condition also be included.

Initial Sizing Locations

The aerothermal and pressure environments vary dramatically over the vehicle surface during ascent and entry (Figures 8 – 13), and for final TPS sizing on a flight vehicle, environments would likely be defined for each panel being built. However, to support preliminary sizing in the ARMOR TPS trade study[5], it was desirable to understand the impact of various components of the environment on TPS sizing in an effort to learn which have the largest impact. Thus, a small number of locations on the vehicle were defined that emphasized different load components. Other considerations also influenced the choice of locations. For example, since integrated TPS/TPSS/Tank airframe concepts were being developed , the locations had to be points that were on the tank barrels (locations over tank domes for example would need additional TPS support structure). Since the vehicle definition includes a carbon-carbon nose cap that extends to station 120 (x = 120 inches) on the vehicle windward side, a location must be beyond station 120 to include metallic TPS. Thus, based on the preceding requirements, the desired location of three points were chosen on the windward centerline, as shown in Figure 14.

Station (STA) 264 is the most forward location on the vehicle that also includes a tank barrel and will experience the highest heating environments and pressures during entry. STA 827 is approximately mid-way along the vehicle and should have average environments (Figure 8). STA 1238 should have the most benign aerothermal environment during entry, but will experience the most extreme acoustic environment, and thus the largest normal pressure, during ascent. Aerodynamic pressures were calculated at 15 locations as described previously, but these locations did not exactly match those on figure 15. Thus, the next closest point forward on the vehicle was chosen, to ensure that the environment was no less severe than that which exists at the desired location. As a result, sizing was performed at STA 240, STA 802 and STA 1199[5].

CONCLUDING REMARKS

One of NASA's major goals has been to develop enabling technology for future reusable launch vehicles (RLVs) with the goal of reducing the cost of access to space. A technology development program was conducted to evolve an earlier metallic thermal protection system (TPS) panel design with the goals of: improving operations features, increasing adaptability (ease of attaching to a variety of tank shapes and structural concepts), and reducing weight. The resulting Adaptable Robust Metallic Operable Reusable (ARMOR) TPS system is designed to meet the high flight rates and quick turn-around times required for an economically viable RLV.

The ARMOR TPS concept allows a high degree of design flexibility in; the materials that can be used to construct the panel, the sizes and shapes of panels that can be produced, and the gages of materials and thickness of insulation packages that can be accommodated. The ARMOR TPS can also be easily integrated with a large variety of tank shapes, airframe structural arrangements and airframe structure/material concepts. A key feature of the ARMOR TPS concept is that its design flexibility allows weight and operability to be traded and balanced.

The ARMOR TPS has incorporated features that successfully improve on previous metallic TPS designs in the areas of; primary and secondary sealing against hot gas flow, blocking radiation in the gaps between panels, and reducing conduction through heat shorts. Mismatches in thermal expansion due to thermal gradients through the panel thickness have been mitigated and aerothermal-induced loads in the TPS panel are decoupled from flight-induced airframe loading. The panel outer surface support, and transfer of pressure loading into the supporting airframe structure have also been improved.

Achieving the ultimate goal of an "economically viable" RLV will eventually require developing Federal Aviation Regulation (FAR)–type performance-based requirements and certification by the FAA, as is currently done for commercial transports. Because these requirements do not exist, there is currently no verifiable and traceable link between TPS design implementation and resulting performance, safety and cost. An initial attempt has been made in this paper to outline a set of performance-based TPS design requirements. A set of general (FAR-type) requirements have been proposed, focusing on defining categories that must be included. However, many details are lacking and there is no consensus among the RLV community (including the FAA) on what requirements to include and the specific content of those requirements. Where applicable, wording from FAR Part 25 (for commercial transport aircraft) has been included as a point of departure.

Current TPS and airframe design does not address critical requirements that will have a profound effect on the economic viability of an RLV in terms of level of inspection, inspection intervals, and meeting mission flight rates and response times. Examples include requirements for ground hail strike, lightning strike, bird strike and rain/rain erosion. Perhaps most critical, the airframe (including TPS) design cannot be considered valid until requirements for on-orbit debri/micro-meteoroid hypervelocity impact have been established and design compliance validated for the vehicle. As competing technologies are proposed and assessed for application to an RLV, it is imperative that they all be designed to a single set of design requirements and validated against a comprehensive set of compliance criteria.

Currently, metallic TPS sizing has been performed using a single nominal trajectory as a limit-load case. Including ascent abort trajectories as ultimate load cases and on-orbit debri/micro-meteoroid hypervelocity impact as one of the discrete-source-damage load cases will have a significant impact on system design and resulting performance, reliability and operability. These load cases are of paramount importance for reusable vehicles, and until properly included, all sizing results and assessments of reliability and operability must be considered optimistic at a minimum.

When designing vehicle trajectories and evaluating resulting environments, the focus has been on entry and the associated aerothermal heating. However, for TPS design and sizing, peak aerodynamic pressures can occur on ascent, and for both ascent and entry, they generally occur at low supersonic or subsonic speeds. Also, at many locations on the vehicle, the critical static design pressure is induced by engine acoustics at liftoff. Thus, at a minimum, these conditions must be included in the set of TPS load cases if the TPS sizing is to be valid.

In general, on entry, the leeward side of the RLV experiences a much more benign aerothermal environment, compared to the windward side. Although difficult to predict because of turbulence and separation, accurately defining these environments and compiling the associated loads is critical to selecting TPS materials and structural concepts and ensuring the leeward TPS sizing is optimized, especially with the focus on minimizing vehicle weight.

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Table 1. Potential options for developing integrated airframe concepts.

Layer	Function	Examples
Thermal Protection System	React aerodynamic pressure React induced heating Reradiate stored thermal energy Protect/isolate inner sub-components from high temperature Accommodate thermal gradients (in-plane) Store thermal energy	Metallic Panels Ceramic Tiles Ceramic Blankets Carbon-Carbon Structure Carbon-Silicon Carbide Structure
Bridging Structure	Isolate inner sub-components from high temperature Accommodate temperature and strain differences between layers Transmit loads between subcomponents Support another layer	Standoffs Lattice Work Carrier Panel Frames/Stiffeners
Cryogenic Insulation	Isolate inner sub-components from high temperature Accommodate thermal gradients Reduce ground-hold heat flux Prevent air liquifaction	Foam Honeycomb Core Foam-filled Honeycomb Core Gaseous Purge
Tank Structure	React vehicle flight loads Accommodate thermal gradients Store thermal energy Contain propellant (under pressure)	Stiffened Skin Integrally Stiffened (Isogrid, Orthogrid) Sandwich

Table 2. Panel size and geometry summary.

Panel Feature	Pros	Cons
SIZE - LARGER	Lower total part count Fewer number of attachments Fewer interfaces Less supporting structure More durable (increased gage) Reduced gap length/area	Heavier panels Larger panel-to-panel gaps (more difficult to seal) Higher loads at attachments Increased honeycomb core thickness, may increase amount of thermal bowing
SHAPE - UNIFORMITY	Uniform thermal expansion Uniform thermal bowing Lower cost tooling/manufacturing	Reduced flexibility in choosing support spacing
SHAPE - TRIANGLE Equilateral: A/P = 0.144L	Flat facets can be mapped to any curved surface geometry	Maximum gap-length/area (minimum A/P) No attachment redundancy (3 points) Long, continuous rows of seals Highest attachment part count and complexity (6 panels at each corner)
SHAPE – SQUARE OR DIAMOND Square: A/P = 0.250L	Reduced gap length/area Attachment redundancy (1 point)	Long, continuous rows of seals Additional thermal stress for non-uniform shape (diamond)
SHAPE - HEXAGON Uniform: A/P = 0.433L	Minimum gap-length/area (max. A/P) Multiple attachment redundancy (up to 3 points) Minimum number of panels connected at corner (3) No continuous rows of seals Minimum areal weight (6 supports)	Slightly more complicated shape Interlocking panels with 6 edges may be more difficult

Table 3. Integrated concept classes and associated pros and cons.

Integrated Concept Class	Pros	Cons
- Sandwich Tank Wall (No Additional Stiffeners) - Direct-Attach TPS * General	TPS panel size not linked to structural arrangement Continuous sealing surface Continuous panel support surface (panel deflection and stress) May have smallest total thickness May be lightest weight No cavities for gases to accumulate Can incorporate cryogenic foam in core	No purge gap available Cannot interrogate core-to-facesheet bondline integrity Difficult to inspect, replace or repair cryogenic foam if incorporated into core Limited thermal mass of tank structure (single facesheet) Increased thermal gradient in tank wall due to foam
* Bonded TPS Specific	No discrete heat shorts (TPS into tank)	Cannot interrogate bondline integrity Difficult to remove TPS panels Difficult to inspect tank structure Thermal mis-match must be accommodated by bondline
* Mechanically Fastened TPS Specific	Panels easily removed for replacement/repair Fastener integrity inspectable Easy to inspect tank structure Fastener holes can be slotted to accommodate thermal mis-match	Discrete hard-points difficult to incorporate in sandwich structure, more so for PMC Discrete heat shorts due to TPS fasteners
- Integrally-Stiffened Tank Wall (External) - Direct Attach TPS * General	Continuous sealing surface Continuous panel support surface (panel deflection and stress) Tank structure has no core-to-facesheet bonds Increased thermal mass of tank (skins and stiffeners) No external space for gases to accumulate (if filled with cryogenic foam) Can incorporate purge gap if desired by partially filling spaces with cryogenic foam	Panel size and shape linked to tank structural arrangement (square TPS > orthogrid, hexagonal TPS > isogrid)
* Bonded TPS Specific	No discrete heat shorts due to TPS attachment	Cannot interrogate bondline integrity Difficult to remove TPS panels Difficult to inspect tank structure Thermal mis-match must be accommodated by bondline
* Mechanically Fastened TPS Specific	Panels easily removed for replacement/repair Fastener integrity inspectable Easy to inspect tank structure Cryogenic foam easily inspected, replaced, repaired (remove TPS panel) Fastener holes can be slotted to accommodate thermal mis-match	Discrete heat shorts due to TPS fasteners

Table 3. Integrated concept classes and associated pros and cons (concluded).

Integrated Concept Class	Pros	Cons
- Stiffened-Skin Tank Wall - TPS Mechanically Attached To TPSS Or Bridging Structure	Purge gap easily accomodated Required for non-conformal tank geometries Panels easily removed for replacement/repair Fastener integrity inspectable Easy to inspect tank structure Cryogenic foam easily inspected, replaced, repaired (remove TPS panel) Fastener holes can be slotted to accommodate thermal mis-match	Strong interrelationship between panel size/shape and tank structural arrangement Higher system weight Higher system part count Discrete panel supports (panel deflection and stress) Increased system thickness (number of layers, clearance for assembly) Discrete heat shorts due to TPS attachments

Table 4. Ground hail requirements (commercial transports).

Structure Type	Surface Position	Hail diameter, inches	Impact energy, inch-pounds*
Fixed Primary	Horizontal	2.5	500
	Vertical	2.5	300
Removable Primary	Horizontal	1.5	100
	Vertical	1.5	60
Fixed Secondary	Horizontal	2.0	350
	Vertical	2.0	200
Removable Secondary	Horizontal	1.2	40
	Vertical	1.2	25

***NOTE:** Based on the lesser of 500 inch-pounds for horizontal surfaces, 300 inch-pounds for vertical surfaces, or the terminal velocity of an iceball, at standard temperature and pressure, assuming a C_D of 0.4.

Table 5. Metallic TPS foil minimum gage.

Material	Foil - Sandwich Facesheets	Foil - Honeycomb Core
Ti 1100	0.006"	0.004" (currently)
Gamma-Ti-Aluminide	0.006"	0.003"
Inconel 617	0.006"	0.0015"
602 CA	0.006"	0.002"

Table 6. TPS deflection limits for boundary layer transition.

Location	Leading Edge	Windward Forebody	Windward Aft Body	Leeward Forebody	Leeward Aft Body
Deflection/L	0.01	0.01	0.015	0.015	0.025
Gap, in.	0.030	0.045	0.050	0.045	0.075

Table 7. BF Goodrich TPS design (Ultimate) pressures.

Flight Condition	Aerodynamic Pressure (psi)
Ascent	2.3
Entry: Peak Heating	0.15
Entry	1.00

Table 8. Engine and aerodynamic acoustic pressure summary.

	Nose: STA 264	Middle: STA 827	Bottom: STA 1238
SPL_{OA}(dB)	154.7	159.6	170.7
P_{rms}(psi) - Engine	0.1575	0.277	0.994
P_{rms}(psi) – Aero.	0.0348	0.009	0.009

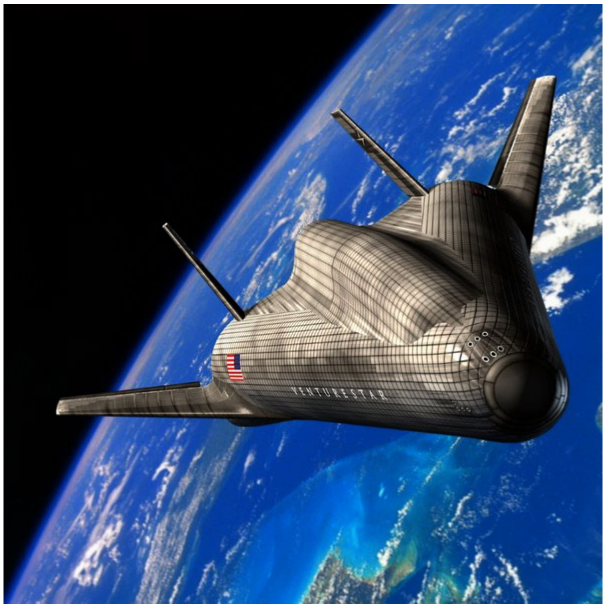


Figure 1. Example of single-stage-to-orbit reusable launch vehicle with metallic TPS.

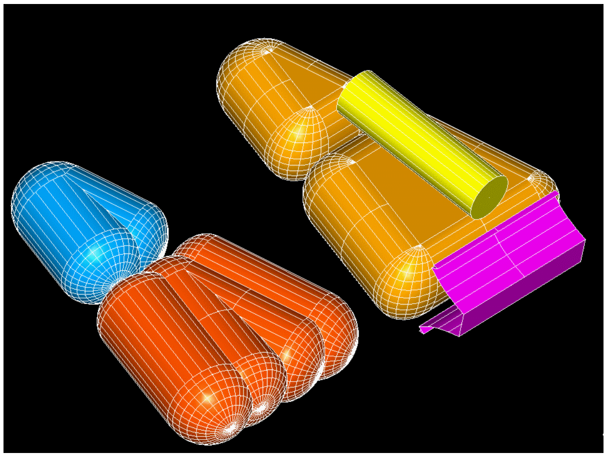
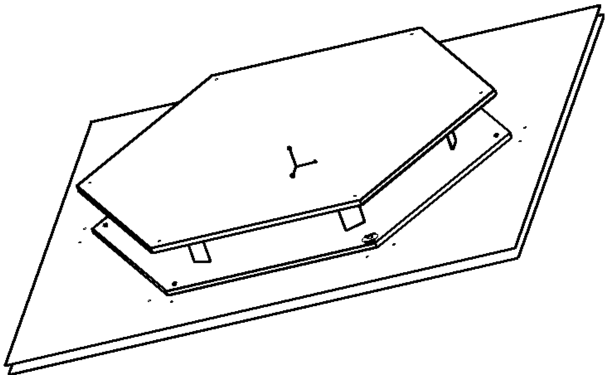
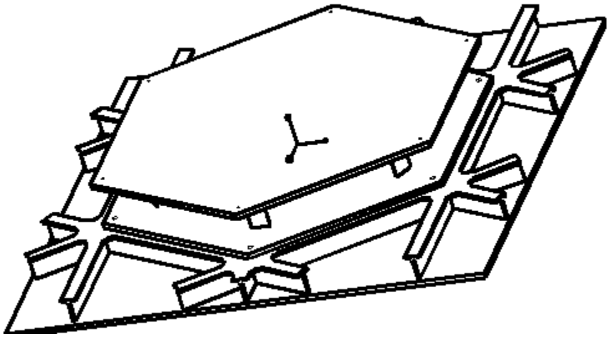


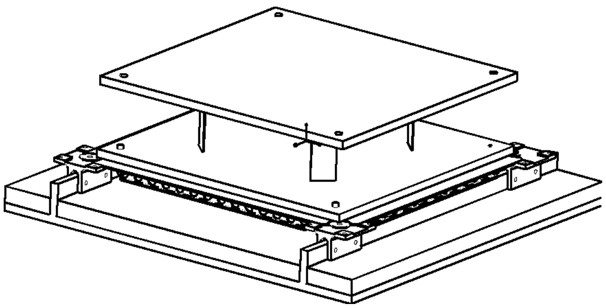
Figure 2. Vehicle packaging and tank geometries.



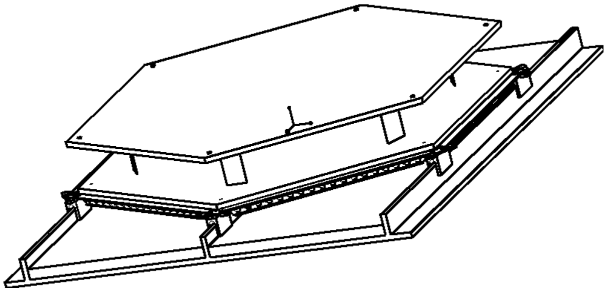
3a. Hexagonal TPS panel on sandwich tank.



3b. Hexagonal TPS panel on isogrid tank.



3c. Square TPS panel on stiffened-skin tank.



3d. Hexagonal TPS panel on stiffened-skin tank.

Figure 3. Integrated Tank/TPS Support/TPS concepts.

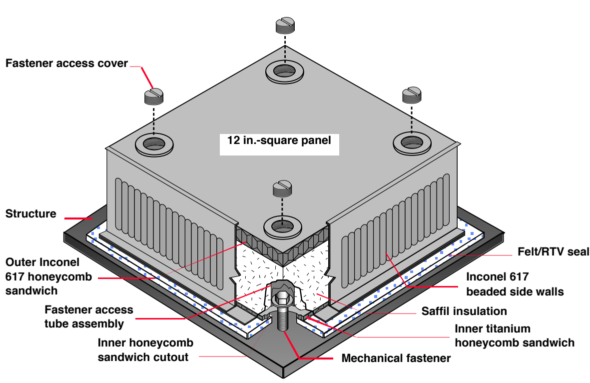


Figure 4. Early superalloy honeycomb-sandwich metallic TPS concept.

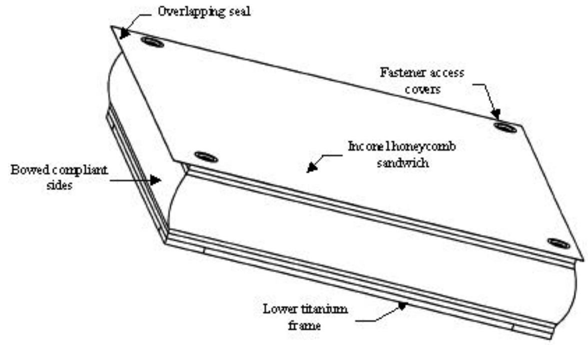


Figure 5. Outer surface of ARMOR TPS panel.

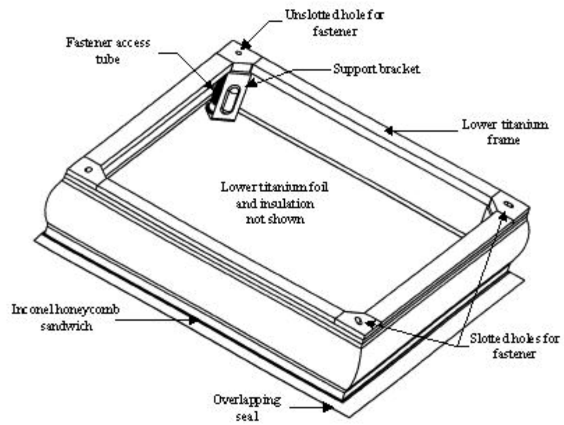


Figure 6. Inner surface of ARMOR TPS panel.

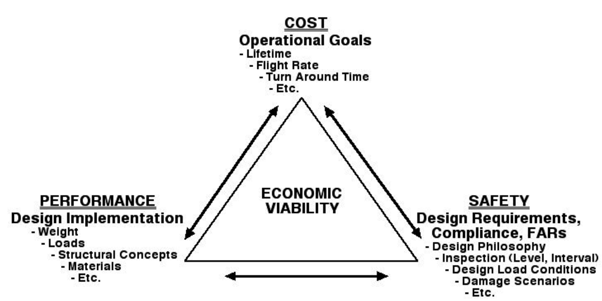


Figure 7. Model for achieving economic viability.

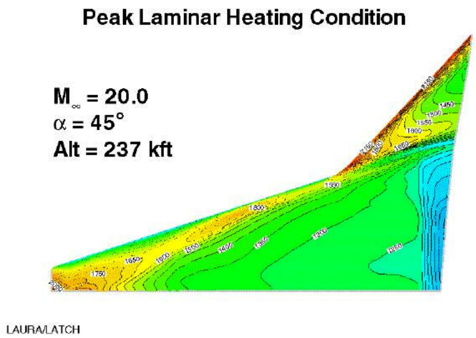
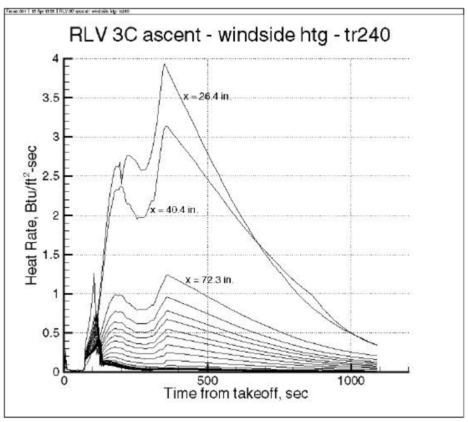
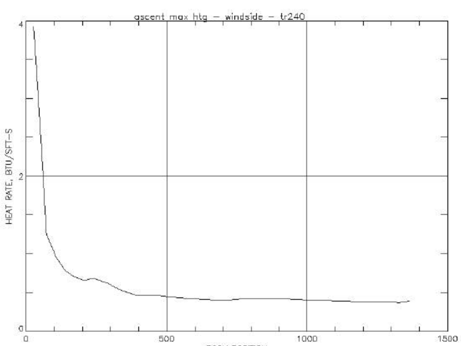


Figure 8. Windward temperature distribution for peak laminar heating condition.

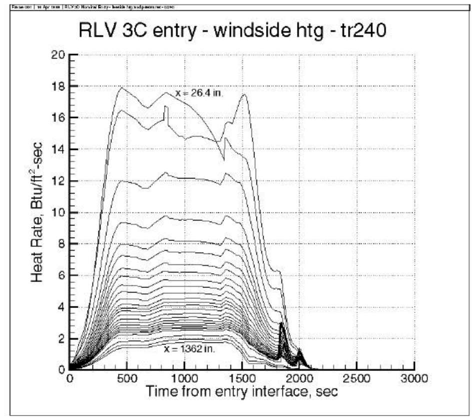


9a. Heating rate versus time.

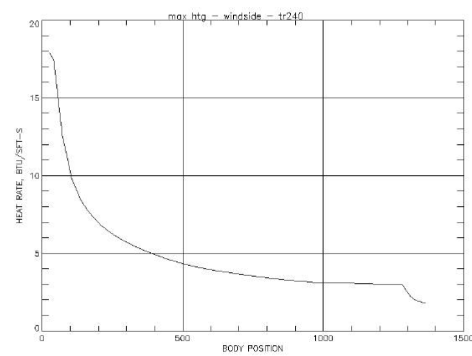


9b. Peak heating rate versus vehicle station.

Figure 9. Ascent heating conditions on vehicle windward centerline.

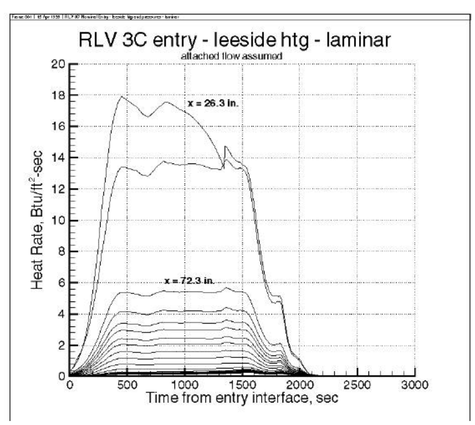


10a. Heating rate versus time.

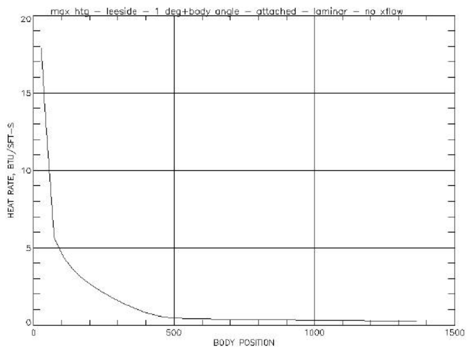


10b. Peak heating rate versus vehicle station.

Figure 10. Entry heating conditions on vehicle windward centerline.

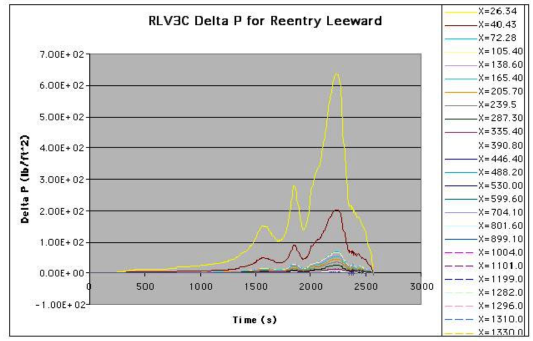


11a. Heating rate versus time.



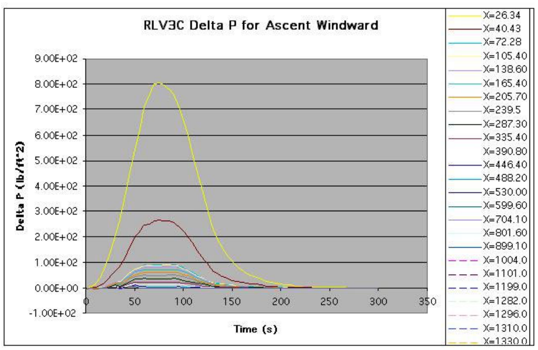
11b. Peak heating rate versus vehicle station.

Figure 11. Entry heating conditions on vehicle leeward centerline.

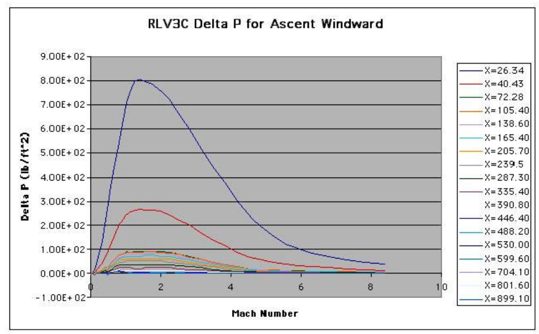


12c. Entry leeward Δp.

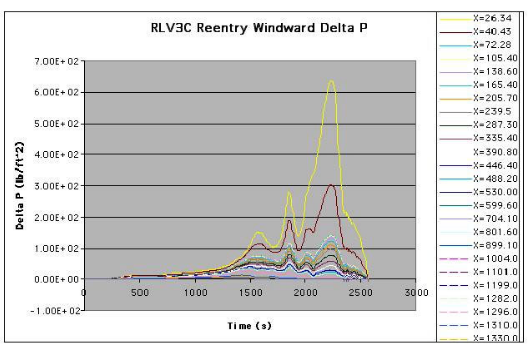
Figure 12. Pressure differential versus time.



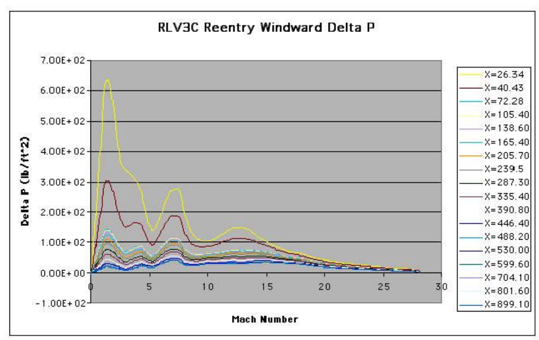
12a. Ascent windward Δp.



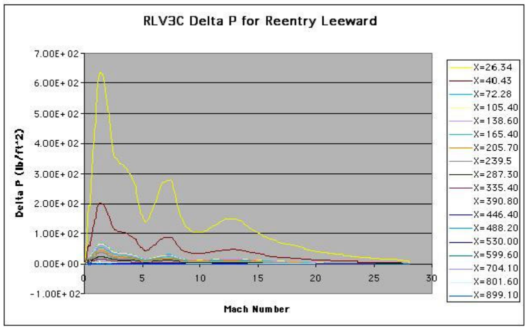
13a. Ascent windward Δp.



12b. Entry windward Δp.



13b. Entry windward Δp.



13c. Entry leeward Δp.

Figure 13. Pressure differential versus Mach number.

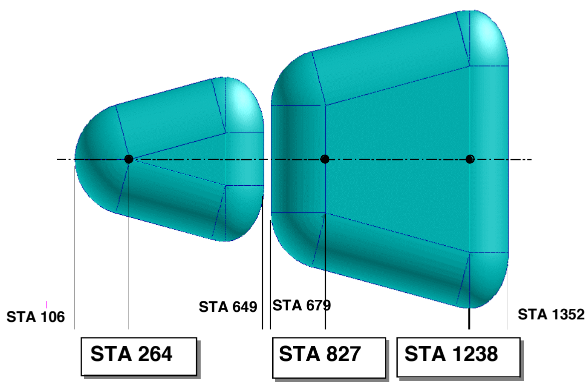


Figure 14. Initial vehicle locations chosen for TPS sizing.