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Metallic Thermal Protection System Requirements, Environments, and Integrated Concepts

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Achieving the ultimate goal of an economically viable reusable launch vehicle will eventually require developing Federal Aviation Regulation-type performance-based requirements and certification by the Federal Aviation Administration, as is currently done for commercial transports. Because the necessary requirements do not currently exist, there is no verifiable and traceable link between thermal protection system design implementation and resulting performance, safety, and cost. An initial attempt has been made to outline a set of performance-based thermal protection system design requirements. Critical requirements that will have a profound effect on the economic viability of a reusable launch vehicle, such as those for ground hail strike, lightning strike, bird strike, rain/rain erosion, and on-orbit debris/micrometeoroid hypervelocity impact have been proposed. In addition to design requirements, the importance of both compiling a comprehensive loads envelope and deriving time- and location-consistent loads for thermal protection system design and sizing is addressed. Including ascent abort trajectories as limit-load cases and on-orbit debris/micrometeoroid hypervelocity impact as one of the discrete-source-damage cases is imperative because of their significant impact on thermal protection system design and resulting performance, reliability, and operability. General features of a suite of integrated airframe concepts is summarized, and the specific details of a metallic thermal protection system concept having design flexibility that enables weight and operability to be traded and balanced is described.

Nomenclature

A = thermal protection system (TPS) panel area (Table 6)
E = impact energy; Eq. (1)
L = length of one TPS panel edge (Table 6)
P = TPS panel perimeter length (Table 6)
p = pressure; Eq. (2)
q = dynamic pressure (Table 3)
α = angle of attack (Table 3)
θ = angle between impacted surface and vertical plane; Eq. (1)

Subscripts

h = horizontal
v = vertical

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. For example, the proposed Lockheed Martin VentureStar™ shown in Fig. 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] Designing and manufacturing economically viable vehicles requires the successful balancing of performance, safety, and cost and is accomplished on a regular basis by commercial (aircraft) transport manufacturers (Fig. 2). However, safety, in the form of demonstrating compliance to applicable Federal Aviation Regulations[2] (FARs), is not negotiable for commercial transports, and successfully achieving economic viability requires the ability to trade cost and performance (while still meeting all safety requirements) as the design matures. The performance and costs associated with 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. To achieve economic viability, full life-cycle system costs, including vehicle operations and maintenance,[3] must now be considered. In addition, achieving commercial-transport-like rapid turnaround operations for commercial launch vehicles will most likely require establishing a vehicle certification process, including developing the necessary FARs (or their equivalent).

As an example (and point of departure), an RLV may be required to meet the structural design requirements of FAR Pt. 25,[2] Airworthiness Standards for Transport Category Aircraft, or a similar set of regulations developed specifically for RLVs. Because the requirements in FAR Pt. 25 are not sufficient by themselves, additional requirements and design objectives must usually be developed. The first objective of this paper is to illustrate the development of these design requirements and objectives for thermal protection systems (TPS) and decompose them into criteria that can be used for metallic TPS design. A second 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. One way to minimize total airframe weight is to develop innovative and integrated concepts for the major airframe structural components, including the TPS, the thermal protection support system (TPSS), and the cryogenic propellant tanks. The third objective of this paper is to introduce and review a suite of integrated airframe TPS/TPSS/tank concepts that

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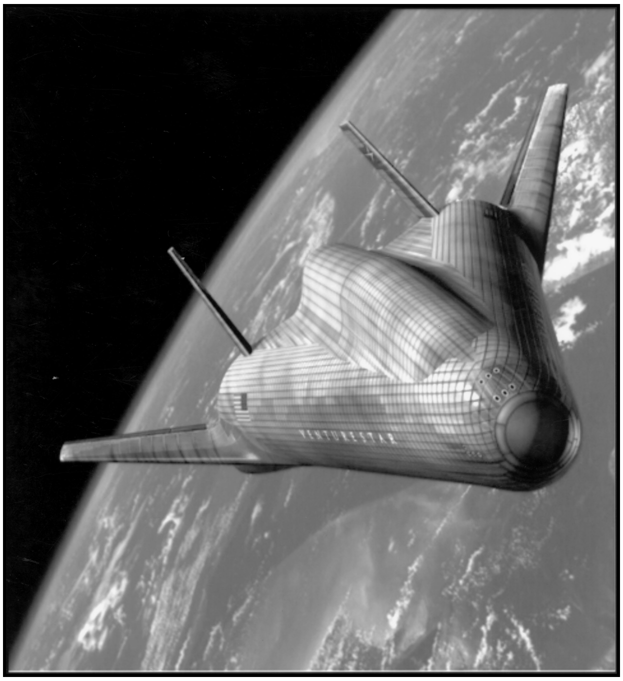


Fig. 1 Example of SSTO RLV with metallic TPS.

Fig. 2 Model for achieving economic viability.

were developed for different airframe geometries and tank materials and to illustrate the ease of adapting the TPS concepts to different VentureStar internal packaging configurations (Fig. 3).

Representative Vehicle Mission and Service Objectives

The economic viability of a launch vehicle will depend on the total launch cost, which for an 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.[4] Cost is addressed with the VentureStar vehicle by defining operational goals in the form of a set of mission and service objectives. The magnitude of each individual cost component will then depend on the specific design implementation of the system that meets the mission and service objectives.

Certification standards and regulations developed for an RLV will be general, performance based, and apply to all vehicles of the applicable type. However, the flight envelope, critical load conditions, detailed load definitions, and design implementation all depend on

Table 1 VentureStar mission and operational requirements

Requirement	Specification
Functional life	20 yr (minimum)
Airframe design life	100 reference missions
Time to perform postflight maintenance and preflight operations	7 days (or less)
Time for rapid turnaround	48 h
Scheduled maintenance interval	20 flights (at least)
Time to perform scheduled maintenance	14 days (not to exceed)
Number of scheduled maintenance periods	3/yr (not to exceed)

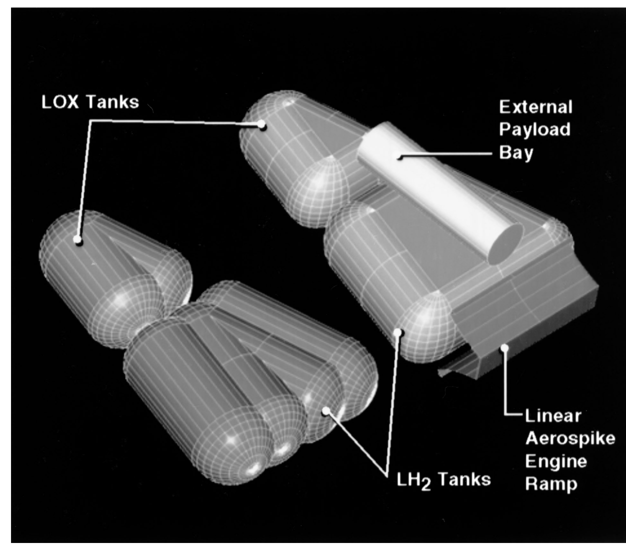


Fig. 3 Vehicle packaging and tank geometries.

details of the mission specifications and the vehicle configuration. The vehicle chosen to illustrate the development of airframe loads and integrated concepts is the SSTO Lockheed Martin VentureStar introduced earlier. For the VentureStar, only vehicle safing, propellant replenishment, and payload integration were deemed acceptable as routine operations to be performed after every flight.[5] Also, 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.[5] 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 are given in Table 1.

TPS Design Requirements

A critical link exists between vehicle cost components (Fig. 2) and airframe performance. This link establishes the traceability between operational goals and the design implementation while also ensuring that all safety requirements are met. For commercial (aircraft) transports, this is documented in FAR Pt. 25.[2] These standards have been established as performance-based requirements rather than design-specific requirements. Coverage includes 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.

Unfortunately, because of a lack of knowledge for this new class of vehicle (RLV), the critical link between cost and performance (needed to achieve significant life-cycle cost reductions) 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 to establish this critical link for TPS, based on FAR-type regulations, is made in this section of the paper. To achieve significant improvement in operational and cost goals, it will be desirable to obtain a type certification for an RLV from a regulatory agency [most likely the federal aviation administration (FAA)]. Thus, the requirements proposed in this paper are worded exactly, or similarly, as those for commercial transports.[2]

General Requirements

The TPS forms the external (aeroshell) surface of an RLV and is exposed to a wide variety of environments corresponding to all phases of flight. The type and nature of these environments will derive from the operational requirements that are established to meet the final set of mission and service objectives to make the vehicle economically viable. A set of general requirements that apply to any external vehicle airframe surface is proposed in this section. Although the specific content of the proposed requirements is open to debate, the final list of requirements should be comprehensive if the safety of the vehicle and ability to obtain certification is to be assured. Although this list can serve as an initial point of departure, it should not be considered complete at this early stage of RLV development.

Durability and Damage Tolerance Requirements

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 vehicle. This evaluation must be conducted in accordance with the provisions of paragraphs (b) and (e) of FAR Pt. 25, Sec. 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: 1) single-load path structure and 2) 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 a vehicle before failure of the remaining structure.

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

A durability analysis that takes into account stress levels and number of cycles (S–N based) 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 subcomponent, 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-in.-diam delamination, whichever is larger.

In addition to the ultimate load requirement in the preceding paragraph, an, 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 that are shown to contain growth of the damage, for example, 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, the 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:

1) For a tension structure, obvious damage is either a two-bay skin crack with central failed support member, for example, a stringer, or a 12-in. skin crack centered on a failed support member (if applicable), whichever is smaller.

2) For compression structure, it is a 6-in.-wide discrete damage penetration, except as noted later.

3) 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 through analysis and test.

4) For primary structure not meeting the preceding obvious damage objective, 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 that 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 just given.

Ground Hail Zone and Requirements

Exposed/external structure (such as TPS) shall meet ultimate load requirements and not require immediate repair following hail impacts spaced 12 in. apart, as specified in Table 2. Impact energies for surfaces between vertical and horizontal may be calculated as

$$E = E_v + (E_h - E_v)\sin\theta \tag{1}$$

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

Table 2 Ground hail requirements (commercial transports)

Structure type	Surface position	Hail diameter, in.	Impact energy, in. · lb[a]
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

[a]Based on the lesser of 500 in. · lb for horizontal surfaces, 300 in. · lb for vertical surfaces, or the terminal velocity of an iceball, at standard temperature and pressure, assuming a drag coefficient of 0.4.

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 inches per hour 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 before the structure degrades 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: 1) impact with a 4-lb bird, at likely operational speed and altitudes up to 8000 ft (FAR 25.571) during ascent and entry; 2) 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; 3) uncontained engine failure and engine disk component or rotating machinery penetration of any structural element in its possible path, with the assumption that a failed rotor disk or other fragment will have infinite energy and will pass through anything in its path; 4) a loose, or thrown tire tread from a tire rotating at landing speed, with the assumption of a loose tread size of 10 in. long by 15 in. wide and a thrown tread size of 7 in. square; and 5) tire burst in the wheel well at any flight altitude.

Hypervelocity Impact

The vehicle shall be capable of successfully completing the flight during which likely structural damage occurs due to hypervelocity impact from on-orbit debris and/or micrometeoroid particles. A maximum particle size shall be calculated according to models and methods that have been approved by the regulating authority. The damaged structure must be able to withstand the static loads that are reasonably expected to occur on the flight, defined herein as 70% of design limit loads, in addition to any associated consistent aerothermal environments. Dynamic effects on these static loads need not be considered. Corrective action to be taken by the pilot following the incident, such as limiting maneuvers, avoiding turbulence, and modifying entry trajectories, must be considered. If significant changes in structural stiffness or geometry, or both, follow from a structural failure or partial failure, the effect on damage tolerance must be further investigated.

Analyses shall be performed to determine the size of resulting damage zones in the TPS 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, that is, 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.[6] Possible effects to be evaluated include (but are not limited to) 1) material degradation due to exposure (metal corrosion, for example), 2) catalytic decomposition of propellants, 3) hydrogen embrittlement, 4) material contamination, 5) stress corrosion, 6) galvanic corrosion, and 7) ignition of materials (in presence of oxidizers).[6]

Metallic TPS: Specific Requirements

The requirements in the preceding section pertain to all RLV primary airframe structures. In this section, additional requirements that are specific to metallic TPS systems are proposed. It is anticipated that specific requirements will also need to be derived for other TPS material systems (such as ceramic tiles and blankets, carbon–carbon, ceramic–matrix composites, etc.).

Minimum Gauge

In addition to manufacturing considerations, minimum gauge for metallic TPS sandwich panel facesheet and core material must be sufficient such that panels also satisfy hail, lightning strike, bird strike, etc., requirements.

Deflection

Deflection limits must be established for the following conditions (may be vehicle and/or configuration specific) and used when demonstrating compliance with requirements.

Boundary layer transition. During entry flight above Mach 5.0, following the initial thermal transient, the surface deflection (achieved at thermal equilibrium) for an individual panel, as well as the gaps between adjacent TPS panels, shall be no greater than the limit values required to prevent transition of the boundary layer from laminar to turbulent flow conditions. 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 the established limit values, additional local heating due to panel deflection is assumed to be negligible.

Insulation. If fiberous insulation is susceptible to permanent compaction due to the specific design of the TPS, the deflection of the upper TPS panel shall be less than or equal to the limit deflection value (may be a percentage of the total TPS panel thickness) at all times to prevent such compaction from occurring.

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

A comprehensive flight envelope that includes all nominal ascent and entry trajectories, abort trajectories, engine and aerodynamic acoustic environments, etc., must be developed to derive critical loads for TPS sizing.

Configuration Geometry and Packaging

The Lockheed Martin VentureStar lifting-body configuration was used as an example to generate integrated airframe (with accompanying TPS) concepts and to illustrate the development of loads for TPS design. The vehicle is a 2.62×10^6 lb gross liftoff weight class SSTO RLV with linear aerospike engines (Fig. 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 external to the tanks (as opposed to located between the two LH2 tanks for the X-33 configuration) to take advantage of more efficient structural load paths and reduce tank weight.[7] 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 (Fig. 3). Because of the potential for accommodating more efficient integrated TPS/TPSS/tank concepts,[8] the semiconformal tank geometry is being considered along with the lobed-tank geometry (Fig. 3) for generating TPS concepts. The tank planform is generated by a 15-deg included angle for this particular lifting body shape, and the tank depth (through the vehicle thickness) is 157 in. (Ref. 7).

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 boundary of the flight envelope must be investigated to ensure that the critical load condition that sizes 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 an ultimate load case, was used for illustrating the TPS-sizing loads-development effort. The trajectory was generated for a specific lifting-body configuration designated 003c (Ref. 9). More severe loads, due to launch abort scenarios for example, could also be generated to represent limit-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, the development of which was proposed in an earlier section of this paper.

The ascent trajectory was designed (Table 3) to maximize the payload weight inserted into the target International Space Station orbit.[10] The trajectory was optimized by adjusting the pitch attitude, engine power level, and engine mixture ratio flight profiles, subject to the parameters and constraints listed in Table 3. An additional constraint required that the mixture ratio profile be consistent with and satisfy the overall fuel to oxidizer weight ratio value (of 6.0) used to size the tank volumes.

Table 3 Ascent trajectory design parameters

Parameter	Specification
Orbital altitude	50 × 248 n mile (International Space Station)
Orbital inclination	51.6 deg
Angle of attack	$-4 \leq \alpha \leq 12$ deg
Engine operating limit	$0.2 \leq$ engine power level ≤ 1.0
Engine mixture ratio	$5.5 \leq$ mixture ratio ≤ 7.0
Vehicle axial acceleration	$\leq 3.0\ g$
Dynamic pressure	$q \leq 600$ lb/ft^2
$\lvert q \times \alpha \rvert$	≤ 1500 lb · deg/ft^2
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 boundary-layer transition such that turbulent heating levels would not exceed those experienced during the initial laminar phase of the entry. A reference heating rate limit of 45 Btu/ft^2 · s was used in the trajectory optimization process to constrain the radiation equilibrium temperatures to roughly 1800°F on the windward side metallic TPS and 2000°F on the nosecap and chine regions. The entry trajectory was also designed to meet a 750 n mile crossrange constraint. Transition onset, angle-of-attack, bank-angle, and control surface deflection limits were also included in the trajectory optimization process.[10]

The loads described in this section are derived from a single nominal ascent and entry trajectory (as described earlier). 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 loads would include 1) aerodynamic pressure and shear forces, 2) inertial loads, 3) heating rate, 4) acoustic pressure, 5) vibration, and 6) shocks (especially on seals). Derived loads needed to size the TPS[11] 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.

Vehicle Surface Aerothermal Loading

Aerothermal loads were derived based on the ascent and entry trajectory designs described earlier. The loads, consisting of transient heating rates 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 that 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[12]) is shown in Fig. 4 for the peak laminar heating condition, which occurs at Mach 20 and 237,000-ft 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 (as opposed to a ceramic) TPS. A value of 225 for the transition parameter, Re_θ/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 ascent and entry transient aerothermal environments were generated using the engineering analysis code MINIVER.[13]

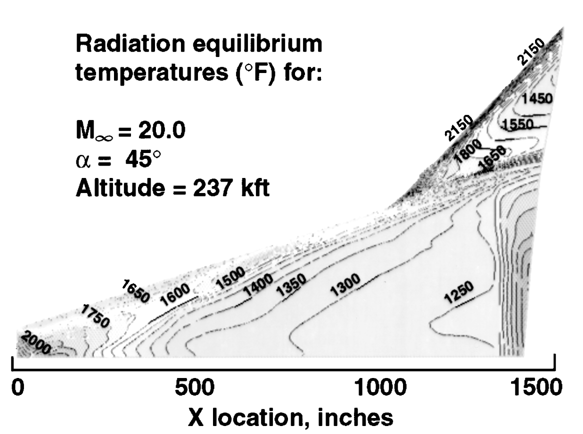
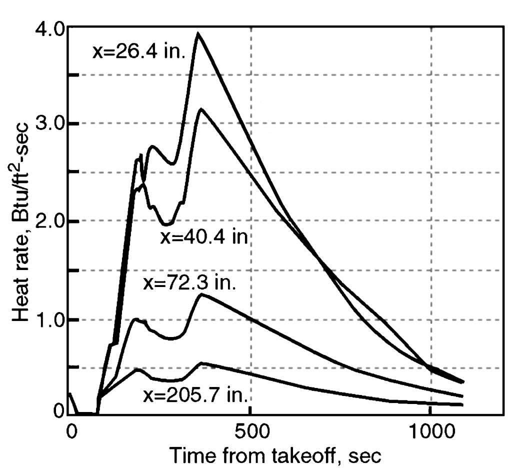


Fig. 4 Windward temperature distribution for peak laminar heating condition.

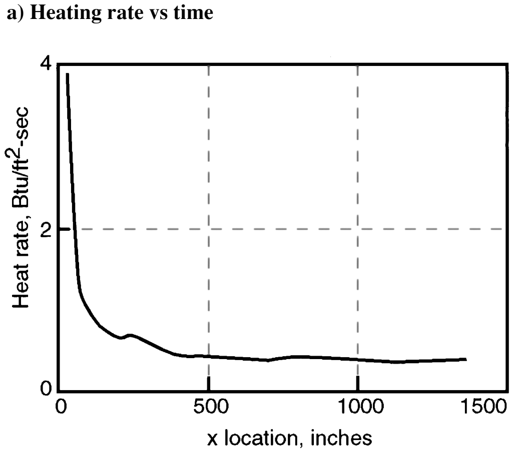
Radiation equilibrium temperatures were assumed to approximate the surface temperatures for determining the applied heating rates. Because the local pressures generated by MINIVER are based on compressibility effects, these results were not used at transonic and subsonic conditions. Thus, a value of 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 Fig. 5a. The rates are shown for a number of vehicle stations (STA, or x locations), starting near the nose (STA 0.0, or $x = 0.0$ in.), with x locations given in Fig. 4. The peak heating rate along the vehicle centerline location for ascent is shown in Fig. 5b. Other than near the vehicle nose (a stagnation point), the heating rate is relatively low (less than 0.5 Btu/ft$^2 \cdot$ s) on the vehicle surface.

Heating rates and peak heating for the vehicle windward side during entry are shown in Figs. 6a and 6b and for the vehicle leeward side in Figs. 7a and 7b. The entry simulation is initiated at an altitude of 400,000 ft and Mach = 28 (relative orbital velocity of 25,000 ft/s) and terminated approximately 2100 s later at an altitude of 86,000 ft and Mach = 2.55 (2500 ft/s). 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 and 500 s after entry, plateaus at the peak rate for approximately 1000 s, and then begins decreasing through the end of flight (Figs. 6a and 7a). On the windward side, the peak heating rate drops rapidly with distance from the vehicle nose, decreasing by 50% (from 18 to 9 Btu/ft$^2 \cdot$ s) in the first 120 in. and below 25% at 500 in. (Fig. 6b). An even quicker reduction takes place on the leeward side, with the heating rate asymptoting to approximately 0.5 Btu/ft$^2 \cdot$ s beyond 500 in. (a level very similar to that experienced during ascent).

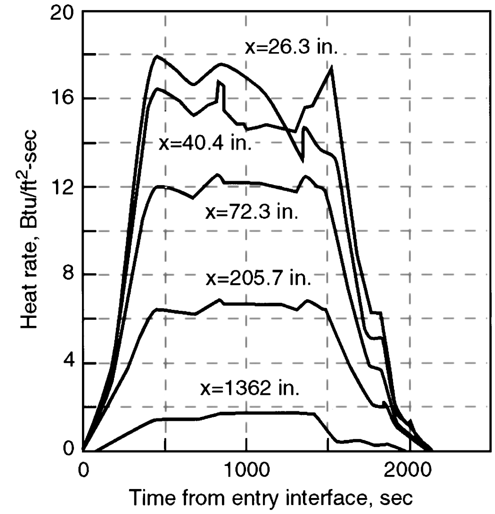


a) Heating rate vs time

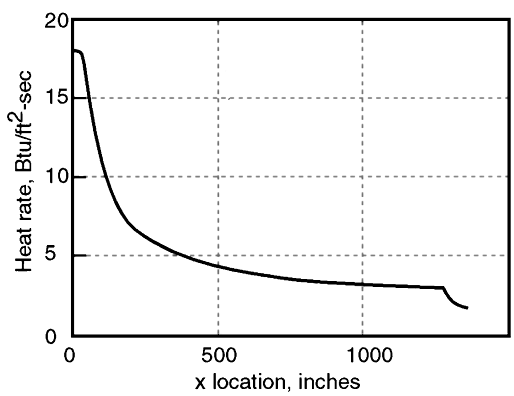


b) Peak heating rate vs vehicle station

Fig. 5 Ascent heating conditions on vehicle windward centerline.



a) Heating rate vs time



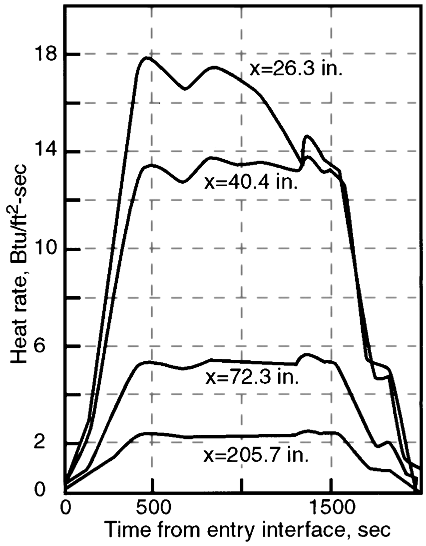
b) Peak heating rate vs vehicle station

Fig. 6 Entry heating conditions on vehicle windward centerline.

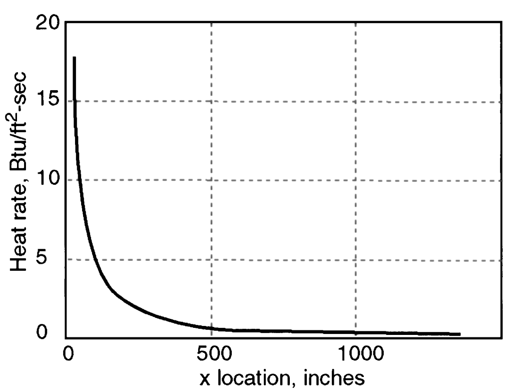
The transient nature of the heating response suggests that thermal gradients will occur in the metallic TPS, which should be evaluated during the entire entry and, thus, considered when compiling load cases for sizing (especially with respect to any requirements or limitations on deflections). The large 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. To size the adaptable robust metallic operable reusable (ARMOR) TPS system (recently developed at NASA Langley Research Center and described in Ref. 14), the applied pressure and element temperatures are needed at the time of occurrence of critical ascent and entry flight conditions. When the ARMOR TPS is integrated with a stiffened-skin tank structure, a cavity exists between the base of the metallic TPS panels and the outer mold line (OML) of the tank wall. This cavity is assumed to be vented to the atmosphere to allow pressure equalization on ascent (pressure transitions from 1 atm to vacuum) and entry (pressure transitions from vacuum to 1 atm). 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



a) Heating rate vs time

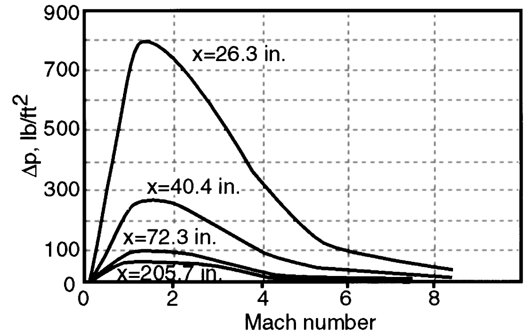


b) Peak heating rate vs vehicle station

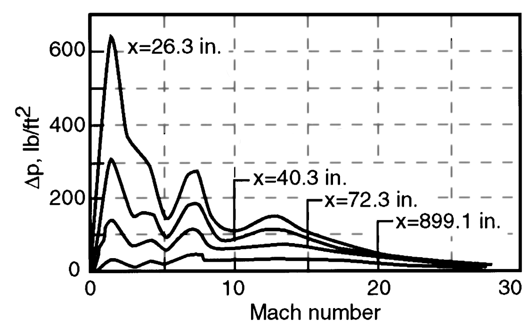
Fig. 7 Entry heating conditions on vehicle leeward centerline.

Table 4 Engine and aerodynamic acoustic pressure summary

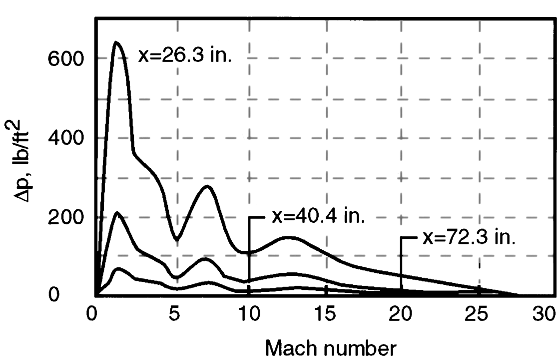
Parameter	Nose STA 264	Middle STA 827	Bottom STA 1238
SPL, dB	154.7	159.6	170.7
Engine p_{rms}, psf	22.7	39.9	143.1
Aerodynamic p_{rms}, psf	5.0	1.3	1.3



a) Ascent windward Δp



b) Entry windward Δp



c) Entry leeward Δp

Fig. 8 Pressure differential vs Mach number.

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

$$\Delta p_{\mathrm{aerodynamic}} = p_{\mathrm{local\,static}} - p_{\mathrm{atmospheric}} \tag{2}$$

Values of Eq. (2) vs flight Mach number are given for ascent and entry on the windward and leeward centerlines in Figs. 8a–8c. Because of the limitations of the analysis methods in MINIVER to calculate the local static pressures cited earlier, the pressures for Mach numbers below 1.5 are not accurate. This leads to very low Δp values at the beginning of ascent and the end of entry, as shown in Figs. 8a–8c. Alternate methods should be used to generate Δp values for Mach numbers below 1.5 because the maximum values occur at those speeds during both ascent and entry, as shown in Figs. 8a–8c.

TPS Design Pressure

Equivalent static pressures induced by engine acoustics at liftoff and aerodynamic acoustics during ascent and entry were calculated.[15] The resulting pressures at three locations on the vehicle, summarized in Table 4, were derived from the sound pressure levels (SPL) also listed. 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. Because the gross panel (not the seal) sizing only is being addressed, the aerodynamic pressure is always assumed to be acting inward, whereas the acoustic pressure is oscillating and can be acting inward or outward. Thus, the following two total pressure values are calculated:

$$\Delta p_{\mathrm{ultimate,TPS+}} = [\Delta p_{\mathrm{aerodynamic}} + 3\Delta p_{\mathrm{rms,acoustic}}]1.4 \tag{3}$$

$$\Delta p_{\mathrm{ultimate,TPS-}} = [\Delta p_{\mathrm{aerodynamic}} - 3\Delta p_{\mathrm{rms,acoustic}}]1.4 \tag{4}$$

where 1.4 is the factor of safety, $\Delta p_{\mathrm{aerodynamic}}$ is from Eq. (2), and $\Delta p_{\mathrm{rms,acoustic}}$ is the equivalent rms pressure due to acoustics.

For acoustic pressure, two conditions are defined: 1) liftoff where induced pressure is due to engine acoustics and aerodynamic pressure is zero, and 2) ascent/entry where induced pressure is due to aerodynamic noise.

For the preliminary TPS sizing reported in Ref. 11, any additional pressure induced by oscillating acoustic shocks is ignored because 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.[16]

Load Conditions for TPS Sizing

Knowledge of the ascent and entry flight envelopes, 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. In addition, methods for obtaining accurate pressures during the transonic portions of ascent and entry would need to be exercised. However, for the preliminary ARMOR TPS sizing described in Ref. 11, some general observations, based on the information from this study were used to guide compilation of a small subset of loads that are representative (but by no means considered to be comprehensive) of critical 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. Because 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 its peak temperature. Because 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 associated induced stresses are 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.

Integrated Airframe Concepts

For an RLV, the integrated airframe is composed of the TPS, TPSS, and tank components. Many design options are possible for each component, allowing for a wide variety of combinations. Optimizing the integrated system as a whole has the best potential for meeting safety, cost, and performance objectives.

Integrated Concept: Motivation and Definition

To define integrated concepts successfully, one must understand and consider the parameters that most significantly impact the design, performance, and cost of each subcomponent; the interactions (and associated sensitivities) between each of the subcomponents; and the 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. For example, there is a strong interaction between the total thickness of the integrated airframe concept (measured inward from the vehicle outer mold line) and the vehicle volumetric efficiency, with the vehicle weight significantly impacted by this interaction.[8] Thus, an integrated airframe concept that is thinner, but heavier than a thicker concept, may be desirable from a system perspective because the net effect of additional airframe weight, when coupled with a decrease in vehicle weight (due to the increase in volumetric efficiency), could result in a (net) lighter weight vehicle.

Strong interactions may also occur between the definition and design of an integrated concept and safety or cost. As another example, the ARMOR TPS (introduced earlier) is capable of protecting the tank structure from on-orbit debris and micrometeoroid impact.[17] Improving the degree of protection provided by the ARMOR TPS might incur an increased weight for the TPS, but could lead to an increase in vehicle safety and a decrease in operations costs (by reducing or eliminating the need for tank inspection/repair). Thus, if done at the outset, defining and optimizing an integrated airframe concept, while also taking into account its context and interactions within the entire system, can lead to a better product when compared to optimizing each individual subcomponent.

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 (Table 5) 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 intended to perform 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 shown in Table 5. Other features will also have an impact on an integrated airframe concept definition. For example, differences 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 semiconformal tank geometry (Fig. 3). The lobed tank geometry, at a minimum, would require bridging structure to span the valleys between the lobes, whereas the semiconformal tank geometry allows for the possibility of completely eliminating bridging structure.

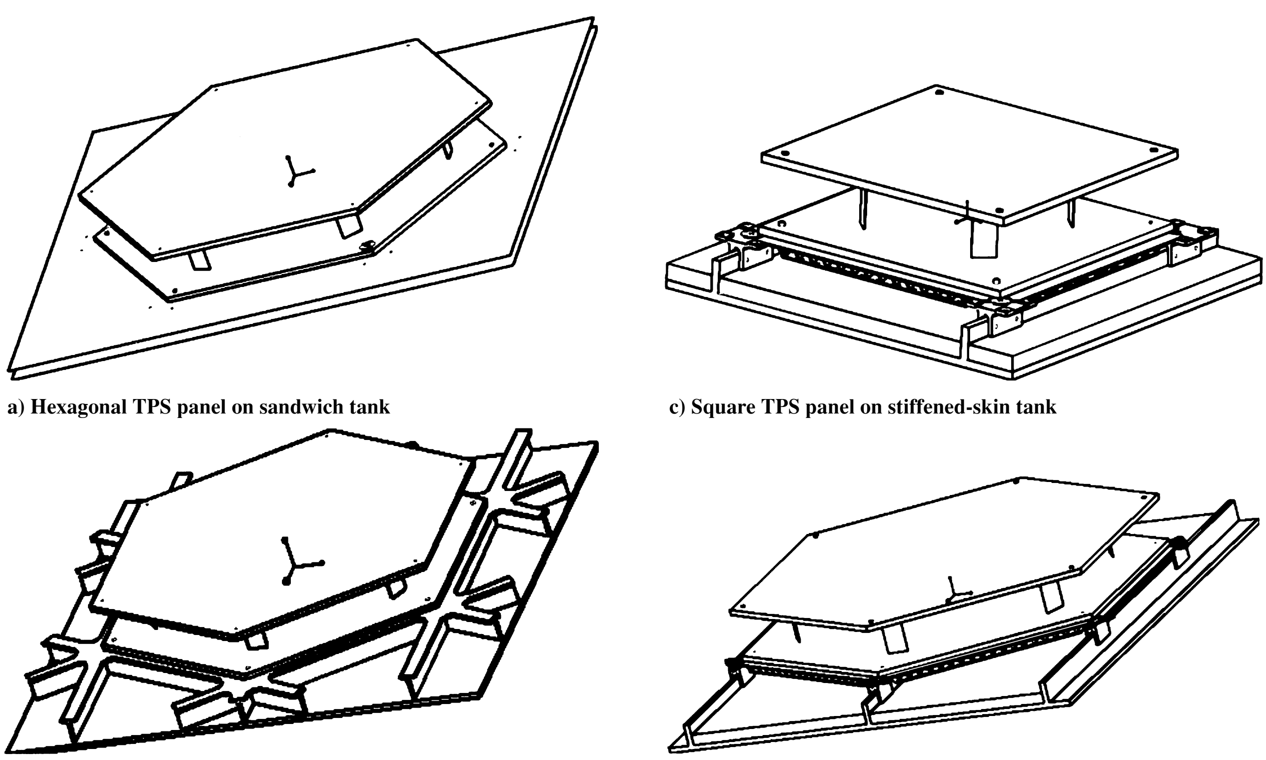
Two key geometry parameters that affect the definition of an integrated airframe concept are the shape and size of the TPS panel. Both parameters impact and can be traded against performance and

Table 5 Potential options for developing integrated airframe concepts

Functions	Examples
TPS	
React aerodynamic pressure	Metallic panels
React induced heating	Ceramic tiles
Reradiate stored thermal energy	Ceramic blankets
Protect/isolate inner subcomponents from high temperature	Carbon–carbon structure
	Carbon–silicon carbide structure
Accommodate thermal gradients (in-plane)	
Store thermal energy	
Bridging structure	
Isolate inner subcomponents from high temperature	Standoffs
	Lattice work
Accommodate temperature and strain differences between layers	Carrier panel
	Frames/stiffeners
Transmit loads between subcomponents	
Support another layer	
Cryogenic insulation	
Isolate inner subcomponents from high temperature	Foam
	Honeycomb core
Accommodate thermal gradients	Foam-filled honeycomb core
Reduce ground-hold heat flux	Gaseous purge
Prevent air liquifaction	
Tank structure	
React vehicle flight loads	Stiffened skin
Accommodate thermal gradients	Integrally stiffened (isogrid, orthogrid)
Store thermal energy	Sandwich
Contain propellant (under pressure)	

Table 6 Panel size and geometry summary

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



a) Hexagonal TPS panel on sandwich tank

c) Square TPS panel on stiffened-skin tank

b) Hexagonal TPS panel on isogrid tank

d) Hexagonal TPS panel on stiffened-skin tank

Fig. 9 Integrated tank/TPSS/TPS concepts.

weight, operations and cost, and safety and reliability, as summarized in Table 6. In general, 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) for all panel shapes.

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 reaction surface for the seals. The TPS panel size and geometry must be selected with consideration for providing these sealing surfaces, as well as 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 semiconformal tanks, where the sandwich tank walls develop adequate stiffness and strength (such that no additional stiffening is needed), TPS panels can be attached directly to the tank wall, thus, eliminating any need for TPSS, as shown in Fig. 9a. If an integrally stiffened concept is used for semiconformal tanks, 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

Table 7 Integrated concept classes and associated pros and cons

Integrated concept classes	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 mismatch 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 mismatch	Discrete hard points difficult to incorporate in sandwich structure, more so for polymeric composite 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 mismatch 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 mismatch	Discrete heat shorts due to TPS fasteners
Stiffened-skin tank wall TPS mechanically Attached to TPSS or Bridging structure	Purge gap easily accommodated Required for nonconformal 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 mismatch	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

must be compatible with the geometry and spacing of the stiffeners (Fig. 9b). For general stiffened-skin tank walls, whether semiconformal 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 (Figs. 9c and 9d, respectively). Additional features and the pros and cons associated with each of the three concept classes are summarized in Table 7.

Conclusions

Achieving the ultimate goal of an economically viable RLV will eventually require developing FAR-type performance-based requirements and certification by the FAA, as is currently done for commercial transports. Because the necessary requirements do not currently exist, there is 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 remain to be defined, 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 Pt. 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 debris/micrometeoroid 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 an ultimate-load case. Including ascent abort trajectories as limit-load cases and on-orbit debris/micrometeoroid 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 during 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.

During entry, the leeward side of the RLV generally experiences a much more benign aerothermal environment than 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.

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 gauges 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.

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