Alternative Designs of Integral Tanks for Advanced Space Transportation Systems

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Fully reusable launch vehicle concepts being studied for post-Shuttle era transports present major challenges for the structural design of large propellant tanks. The dominant structural elements are internal tanks for both cryogenic and noncryogenic propellants. These tanks must operate in a broad range of loading and thermal environments while meeting requirements for low weight and reusability. Several approaches to integral tank design are discussed and an analysis of a hot structure honeycomb sandwich tank for a circular body vehicle is presented.

Nomenclature

A,B = constants, R
T = temperature, R
x/L = Ratio of axial station (measured from nose) to vehicle reference length (the latter measured from nose to the hingeline of the body flap)
ϕ = angle measured circumferentially around vehicle body from the top centerline
σ/ρ = yield strength, psi divided by density, lb/in.^{3}

Introduction

ALTHOUGH the Space Shuttle has recently reached operational status and will have many useful years ahead, the study of more advanced systems is considered appropriate in view of the long lead times for the development of technology for future vehicles. Future fully reusable vehicles will probably use a winged configuration (like the Shuttle Orbiter) but (unlike the Orbiter) will carry all or most of its main rocket engine propellant internally in large tanks. The tanks on these vehicles is the dominant structural feature and typically constitutes from 20 to 30% of the dry weight (Fig. 1). Because tanks are a large portion of the weight, and because weight is critical to these systems, it is important to devise the lightest possible structure. The use of integral (load-carrying) tanks generally results in the lightest overall vehicle structure.

The objectives of this paper are twofold: 1) to examine several vehicles with differing design philosophies which have been proposed as Shuttle follow-on transportation systems and identify the approaches used for integral tanks, and 2) to describe some preliminary finite element stress analyses for the hydrogen (LH_2) tanks on one of the vehicles.

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Vehicle Configuration Alternatives

In order to provide some focus for ongoing research and technology, several studies have been made of advanced vehicle systems which could transport Shuttle-size payloads to and from Earth orbit.[1-5] Emphasis in these studies has been on full reusability with low operational costs. Both single- and two-stage systems in various launch configurations have been proposed, as illustrated in Fig. 2. These vehicles are essentially flying tanks with wings. In all systems, the vehicles are recovered by a horizontal landing. The boosters and the single-stage vehicle have integral circular forebody tanks and are similar to those utilized on the vehicles described in Refs. 3 and 4. The booster tanks all have maximum diameters of 25 ft, whereas the single stage has a diameter of 30 ft. (In reality, booster tank diameters would vary slightly when sized for the same mission because of slight system weight differences between the three launch vehicle arrangements.)

Piggyback and Tandem Concepts

For the tandem and piggyback launch configurations shown in Figs. 2b and 2c the effect of the Orbiter-induced structural loads on booster integral tank weights is substantial. Increases of 30-50% in the booster tank weight would not be unusual because of loads imposed on the booster by the attached Orbiter. For the tandem arrangement, Fig. 2b, the design load condition is associated with high dynamic pressures and a lateral wing gust (or wind shear) which applies incremental lift to the Orbiter wings and results in a bending moment on the booster tankage.

For the two-stage piggyback arrangement, Fig. 2c, the critical design condition occurs during peak acceleration when the booster is subjected to bending moments from the eccentrically located Orbiter mass.

Calculated maximum compressive loads in the LH_2 tank for the tank models just described and for an assumed 800,000-lb fully loaded Orbiter in the tandem and piggyback launch configurations are shown in Table 1 at −7980 and −8560 lb/in., respectively. These values represent the summation of all the compressive loads due to drag, inertia, and body bending. Internal tank pressure (normally relieving on the compression side of the tanks) is not included since the tank must be designed for the "loss-of-pressure case." From the individual values in Table 1, the major load for the tank walls for the configurations in Figs. 2b and 2c results from body bending.

Single-Stage and Base-to-Base Concepts

For the single-stage concept (Fig. 2a), the inertial load is much smaller than for the previous two-stage vehicles and is only that attributable to the weight of the nose section, nose gear, and weight of the tank itself ahead of the most highly loaded section (i.e., at a station near the aft dome of the LH_2 tank). Bending from asymmetric aerodynamic loads, compression from aerodynamic drag, and inertial loads constitute only a small portion of the tensile load from internal pressure (Table 1).

For the base-to-base configuration (Fig. 2d), engines on both the booster and the Orbiter are operated while the vehicles are mated. Therefore, the loads transferred to the booster from the Orbiter and vice versa are small, and loads on the integral tanks on the two vehicles are correspondingly small. The loading (in axial tension) from internal pressure is +3400 lb/in. and that for a single stage is +4100 lb/in.; the latter being larger because of the larger single-stage tank diameter. It is evident from Table 1 that, for the single-stage and base-to-base concepts (Figs. 2a and 2d), internal pressure is the design driver for the integral tranks.

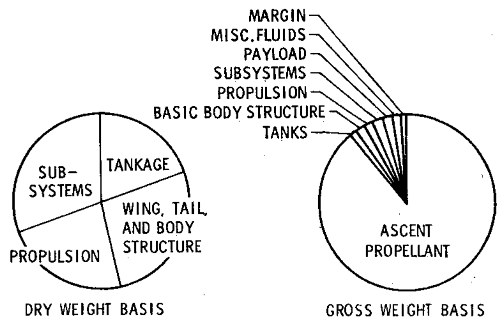

Fig. 1 Weight breakdowns for advanced Earth-to-orbit transports.

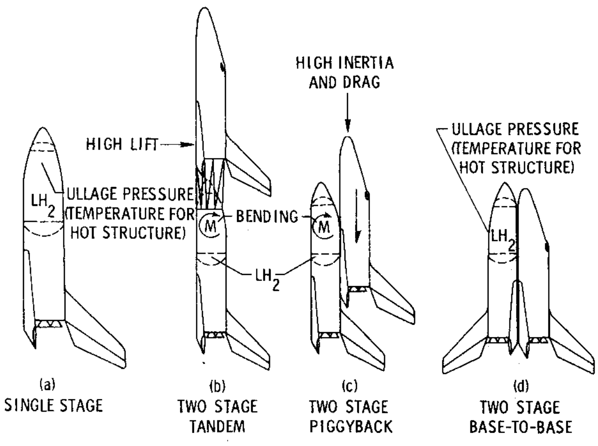

Fig. 2 Design drivers for integral tanks.

Single-Stage Advantages

As stated previously, from the standpoint of loads, the largest weight penalty occurs in the booster tanks in configurations of Figs. 2b and 2c because of the presence of the Orbiter mounted on the booster. Interactive loads should be moderate for the two-stage base-to-base system since both booster and Orbiter engines are operated in parallel; therefore, the full inertial weight of the Orbiter is not carried by the booster. Of all the launch configurations, the tank loads and, therefore, tank design and definition, should be the least difficult for the single stage because of the absence of interactive body and aero loads.

In addition, the somewhat larger single-stage vehicle tanks would be fabricated from thicker gage materials facilitating the fabrication of joints and attachment fittings. This size difference applies particularly when comparing Orbiter and single-stage tanks. By utilizing thicker gage materials, the influence of material flaws and optimums for joints and other structures would be reduced. (Note: The walls of a larger diameter tank for the same ullage pressure and pressure design criteria will be thicker, but the pounds of tank per pound of stored propellant will remain essentially constant. This can be shown through the relationship between tank wall thickness and radius for a given pressure; tank weight as the product of material density, thickness, and tank surface area; and tank volume as a function of tank dimension cubed.) In actual practice, the storage system weight, as a fraction of propellant stored, should decrease slightly because cryogenic insulation weight on the larger single-stage tank increases as a function of tank area, whereas weight of the propellant contained increases as a function of tank volume. All of the above attractive features of single-stage systems, as related to tank design, do not necessarily preclude the consideration of multistage systems, particularly for those missions requiring very high energies.

Integral Tankage Design Alternatives

A variety of material and configuration alternatives have been either used or proposed for tank wall construction. Materials, tank shape, and the Thermal Protection System (TPS), in particular, have a substantial impact on tank weight.

Material Alternatives

Similar to the current Shuttle external tank, most of the propellant tanks in the past have been fabricated from aluminum, exceptions were the stainless-steel Centaur LH_2 and LOX tanks.[6] A number of tanks on the Shuttle Orbiter itself are titanium. The only cryogenic tank, however, which is not aluminum is the LOX tank on the fuel cell system, and

Table 1 Examples for maximum loadings on forebody tanks (thermal stressed excluded)

Stage configuration	Design condition	Loads, lb/in. Pressure	Drag	Inertial	Bending	Design resultant
A. Single	Internal pressure	4100	−160	−50	−70	4100[a]
B. Tandem	Maximum dynamic pressure (q = 750 psf, 50 mph lateral gust, g's = 1.8)	3400	−300	−1530	−6150	−7980[b]
C. Piggyback	Maximum acceleration (q = 208 psf, g's = 3.0)	3400	−110	−2550	−5900	−8560[b]
D. Base-to-base	Internal pressure	3400	−80	−50	−70	3400[a]

[a] Internal pressure designs. [b] Design conditions associated with loss of pressure.

this tank is fabricated from Inconel 718. For future vehicles, however, aluminum must be well insulated for ascent and entry because of the relatively low tolerance of aluminum to elevated temperature (about 350°F max).

Inconel 718 is relatively inexpensive, is easily brazed or welded, and can be field repaired. Compared to René 41, Inconel 718 has a greater yield strength (and a greater specific strength) up to about 1200°F (Fig. 3)—a desirable characteristic for the severe loading and thermal environment during ascent. For unprotected tank structure, inconel is suitable because of its good oxidation resistance and adequate strength up to 1600°F. This temperature is consistent with expected temperatures on the exposed portion of the tank during entry. (Note: Dynamic pressures are low during the peak heating entry period—only 90 psf at the end of the peak heating period vs 750 psf during ascent and 250 psf during subsonic flight.)

Titanium is ideal for structure and tanks on a supersonic airplane because it is compatible with hydrocarbon fuels and temperatures of 600-1000°F found in flight at about Mach 3. For tanks on advanced Earth-to-orbit transports, the material is also attractive because of its relatively high specific strength, low conductivity (for storage of cryogenic fuels), and compatibility with leeside temperatures anticipated on these vehicles. However, because titanium capability does not include the higher flight temperatures, it must be protected in the hotter regions of the vehicle.

Titanium, Inconel 718, and René 51 all embrittle to some extent in the presence of hydrogen[7,8]; but, apparently, most of the test data relate to much higher pressures than the anticipated ullage pressures of 15-35 psig. Salient properties of these materials as related to advanced space transportation systems are summarized in Table 2.

Tank Wall Alternatives

Many alternatives are available for the tank wall. Conventional approaches usually involve a facesheet stiffened with ring frames and stringers. Two extensively proposed alternatives include isogrid or honeycomb sandwich wall (Fig. 4). A blade-stiffened tank wall is still another alternative. In cross section, this design resembles a flat plate with integrally machined fins. The concept is inspectable from both sides (in the absence of cryogenic insulation) but only inspectable from the inside when the stiffeners are located inside and a cryogenic insulation is bonded to the outside. Tanks with metallic liners and composite overwraps may yield a weight advantage for the advanced transports but would have to be studied for this application.

The honeycomb sandwich is an attractive structural concept for a tank wall being one of the lightest types of construction for resisting inplane axial loads[9]—an important quality for integral tanks. As a bonus for cryogenic tanks, the construction also has some insulating qualities. Conveniently, at cryogenic temperatures the effective conductivity is low (about 0.5 Btu-in./ft^2-h-°F for René 41) and is high at elevated temperatures (about 17 Btu-in./ft^2-h – °F)[10]—the latter property tending to minimize thermal stresses during entry when the tank is depleted of cryogenic propellant.

Brazed stainless-steel honeycomb sandwich panels were used for skins on the XB-70 experimental aircraft on the body and wing and in tankage areas.[11] Presently, aluminum honeycomb sandwich panels are used extensively in the Shuttle Orbiter wing and control surfaces.[12] Other honeycomb sandwiches used on the Shuttle Orbiter include Inconel 718 and titanium (for elevon wing seals and rocket engine heat shields) and composites (for cargo bay doors).

More closely related proposed applications for honeycomb sandwich panels appear in Refs. 13 and 14 wherein titanium is used for the sandwich core and aluminum for the two facesheets. The system is fabricable, the titanium is light and has a low conductivity, and the aluminum is compatible with LOX or LH_2. The honeycomb sandwich combination was studied for the tank sections of a Mach 5 transport[13] and for an Earth-to-orbit transport with external heat shields.[14]

Thermal Protection Integration

Whether honeycomb, skin-stringer blade-stiffened, isogrid construction, or composite overwrap is used, a number of

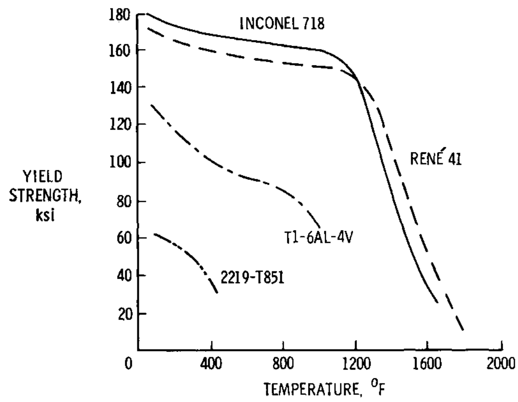

Fig. 3 Tank materials properties comparisons.

Table 2 Properties of tank materials

	Material			
Property	Aluminum 2219	Titanium 6AL4V	Inconel 718	René 41
Specific strength at design critical condition σ/ρ, in., (temp °F)	300	600	1400	1600[a]
Max structural utilization temperature, °F	570	644	593	456
Max short time operating temperature under light loading	350	1000	1800	1600
Hydrogen embrittlement	Negligible	Low	Moderate	High
LOX Compatibility	Excellent	Impact sensitive	Excellent	Excellent
Formability	Good	Satisfactory	Good	Satisfactory
Weldability	Good	Good	Good	Satisfactory

[a] Subject to alloy depletion and embrittlement in an oxidizing atmosphere above 1600°F.

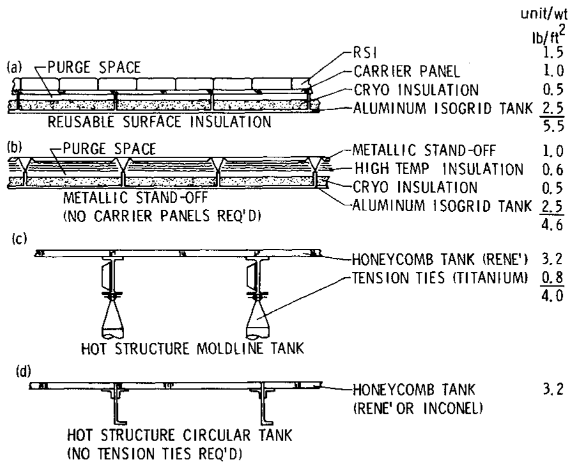

Fig. 4 Integral tank concepts and unit weight estimates.

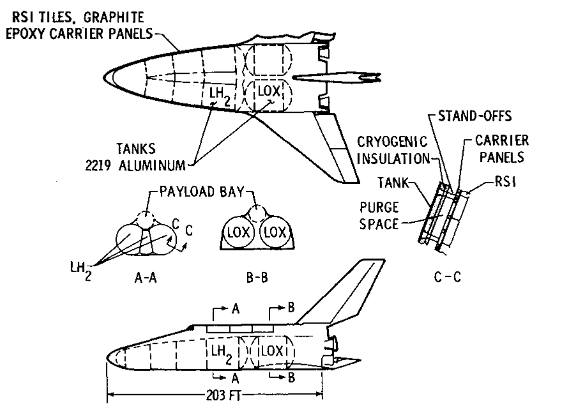

Fig. 5 Vertical takeoff single state with insulated structure.

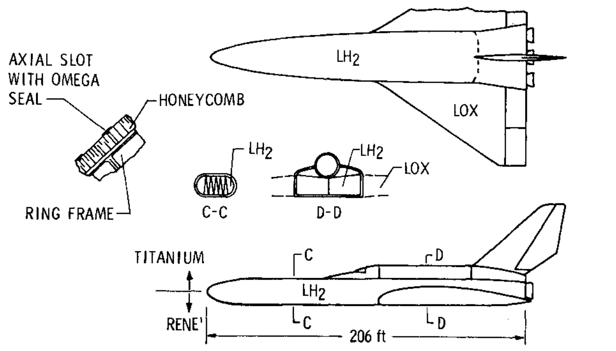

Fig. 6 Horizontal takeoff single stage with hot structure.

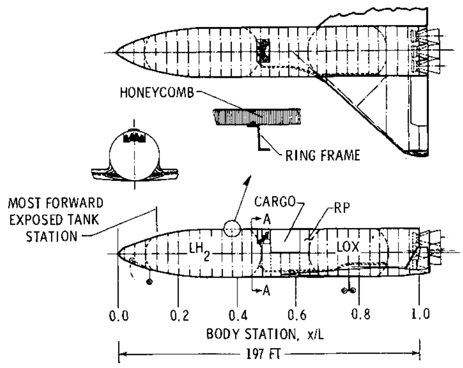

Fig. 7 Inboard profiles of circular body vehicle with hot structure LH_2 tank.

design options are available in regard to the method in which the thermal protection function is integrated into the tank wall. In Fig. 4, schematics of several different designs are shown; the outer surfaces of which are limited to about 1600°F. Direct comparisons are difficult to make, therefore, the unit weights shown are merely indicative.

In what might be called a "conventional approach," an aluminum tank serves for propellant containment and as the body structure; and separate insulations are used for the cryogenic tank and for protection against entry heating (Fig. 4a). Composite honeycomb sandwich panels are used on standoffs to support a Shuttle-type TPS. A unit weight estimate for this cross section is 5.5 psf. This approach was proposed for the vertical takeoff single-stage vehicle shown in Fig. 5 and described in Ref. 1. A variation of this concept shown in Fig. 4b uses a corrugated metallic panel. Essentially because no carrier panel is needed in this latter concept, a weight savings of about 16% is estimated over the concept shown in Fig. 4a. The corrugated metallic panel[15] could be applied to this vehicle in lieu of reusable surface insulation (RSI) in areas where temperatures are below 1600°F.

When a single honeycomb moldline shell is used for the integral tank, an estimated savings of about 11% is realized (Fig. 4c). This approach was proposed for the vehicle shown in Fig. 6. Analysis[2] has shown that the critical design case for this all LOX/LH_2 horizontal takeoff vehicle occurred during ascent when external aerodynamic heating combined with the cryogenic temperature of the hydrogen internally gave a temperature drop across the honeycomb sandwich of approximately 800°F. Because the tank is noncircular, internal bracing is required to maintain tank shape under internal pressure. A René 41 honeycomb sandwich was used for the lower half of the vehicle and titanium for the upper (cooler) side. To bring the René 41 within design allowables, axial slots were provided in the external skin at a spacing of approximately every 5 in. Details of tests on a section of René 41 honeycomb sandwich panel for this vehicle are given in Ref. 16.

Rationale for Honeycomb Sandwich Wall Selection

In order to understand the motivation for selecting the honeycomb sandwich hot structure design, some of the weight, design, and operational implications of a more conventional construction are reviewed.

For a multilayered conventional tank wall, the purge space must be kept at a temperature above the liquefaction temperature of O_2 and N_2 to avoid cryopumping of air; and, further, a dry gas must be circulated to avoid condensation or ice formation. Also, because closed cell foams are limited to 175-200°F for continuous use, the aluminum tank underneath cannot be allowed to reach its limit of 350°F. Therefore, the high temperature insulation must be increased in weight to protect the intermediate insulation material over that normally required just to protect the aluminum tank. Some effort is being directed toward development of graphite polyimide foams that can be used up to 600°F for this application, but no directly applicable data are available.

The hot structure honeycomb sandwich construction is not without concerns. A major concern is the intrusion of the contained liquid, ambient air, or moisture into the honeycomb cells through a hairline crack, followed by heating and expansion of the liquid or gas during entry. By careful inspection between launches for localized core-to-facesheet separations and/or hairline cracks, this type of degradation could be minimized. Such an inspection could be made utilizing infrared imaging cameras in a process referred to as thermography.[17]

For any design, operational considerations are extremely important; for example, inspection and repair of the tankage. If a conventional approach is used, detecting a flaw or leak in the tank wall and repairing it from the exterior may be difficult when confronted with several layers of insulation and a heat shield. For the honeycomb sandwich shell, however, the tank can be inspected and repaired from both sides.

Circular Body Vehicle and Integral Tank Description

After considering the materials and configuration options as outlined in the preceding sections, integral LH_2 forebody tank design using a circular cross section and a hot structure honeycomb sandwich construction was established—the integral tank forming the basis for a finite element stress analysis later in this paper. (Note: The LOX and hydrocarbon tanks are not dealt with herein, but are described in Ref. 4.)

Vehicle Description

By making the tank circular (Fig. 7), essentially all internal pressure-induced loads are taken out as membrane stresses (as opposed to panel bending). The circular body vehicle takes off vertically, lands horizontally, and utilizes LOX and two fuels, LH_2 and a hydrocarbon[4]—the hydrocarbon being depleted early in the flight. The vehicle is approximately 200 ft long and, like the preceding designs in Figs. 5, 6, and 2a, is a single stage designed for a 65,000-lb payload. The cargo bay is 30 ft in diameter by 17 ft long and extends through the body cross section with cargo bay door hingelines located at the maximum body diameter. By making the vehicle circular and confining the cargo compartment to a space between propellant tanks, relatively simple load paths are established, and large savings are realized in structural weight.

The round forebody integral tank meets two needs: 1) low structural mass for overall vehicle efficiency as a rocket, and 2) decreased lift forward to faciltate hypersonic aerodynamic trim for these vehicles which typically have aft c.g.'s because of the presence of rocket engines in the base. For the circular body vehicle, the entry planform loading is 37 psf (26% less than the Shuttle) and is an order of magnitude less than the ballistic capsules such as Mercury, Gemini, and Apollo. In comparing recent unpublished experimental test data for the circular body vehicle with the Shuttle Orbiter,[18] it appears that heating rates may be slightly higher in the region of 30-60% of reference body length on the circular vehicle than the Shuttle, but will be actually lower forward and aft of this region. (Laminar flow is assumed.) Further, the heating rates tend to be much lower in the circumferential direction from the bottom centerline if comparisons are made at the chine (or glove) radius on the Shuttle. Overall the average TPS unit weight should be considerably less on a circular body vehicle.

Tank Description

The tank has an overall length of 74.3 ft and a maximum diameter of 32.8 ft. When installed on the circular body vehicle, the exposed portion of the tank extends from the station 290 to 965 in. or, as a percent of body length, from $x/L = 12.6$ to 40.8%. The shell of the tank is fabricated from a 1.25-in.-thick Inconel 718 honeycomb sandwich. The gage of the facesheets is 0.025 in. and the sandwich core density is 7 lb/ft^3. The core is sealed and evacuated; a high-temperature nickel braze alloy is assumed in the fabrication process. A typical frame is 1.75 in. deep. Spacing of the frames varies from 86.5 in. in the ogive section to 77.5 in. in the barrel section. This weight (and the frame spacing and sizing) is the result of calculations using methods given in Ref. 19. Thermal stresses were not included in this analysis.

Tank Finite Element Analysis

After sizing the ring frames as previously described, a finite element analysis was used to determine stresses with temperature gradients included. In the following sections, temperature distributions, loads, simplifying assumptions, and stress analysis results are presented.

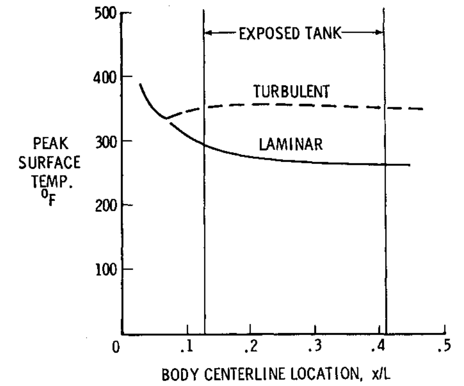

Fig. 8 Predicted bottom centerline temperatures (ascent).

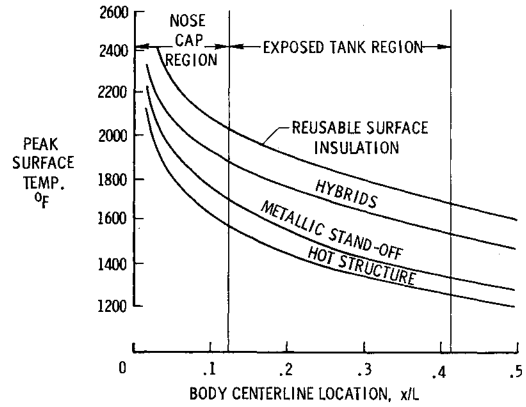

Fig. 9 Predicted bottom centerline temperatures (entry).

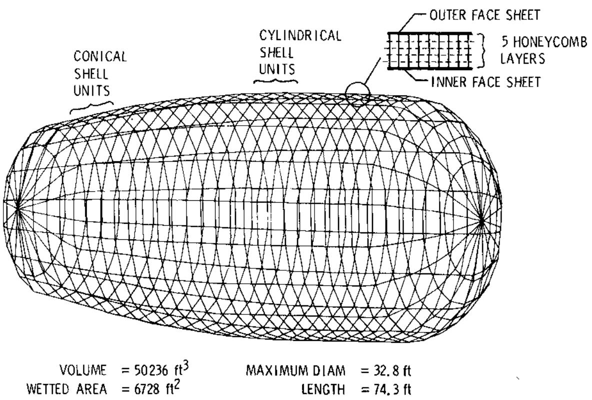

Fig. 10 Finite element model of tank.

Tank Temperature Distribution

In order to determine stresses in the tank wall, the axial and circumferential temperature distributions were estimated. For the bottom centerline distributions for ascent and entry, the MINIVER program was employed.[20] These results are shown in Figs. 8 and 9. For the top centerline, estimates were made for the axial distribution based on similar vehicles and trajectories, some information being obtained from Ref. 2. For the ascent case, the temperature over the top centerline region of the exposed tank was essentially constant at 150°F for the design case. This condition occurs approximately 130 s after liftoff. For the top centerline during entry, the design temperature was estimated to be 660 and 510°F at the forward and aft most stations of the exposed tank, respectively.

Circumferential temperatures were estimated from the top and bottom centerline temperatures using an equation of the form,

$$T = [1 - \cos(\phi/2)]A + B$$

where A and B are constants determined from temperatures and ϕ values at top and bottom (ϕ is taken as 0 deg at the top centerline and 180 deg at the bottom centerline). The above equation is an approximate representation for the temperature distributions between the top and bottom centerlines of the vehicle. (For more exact solutions, see heating rate distributions in Ref. 21 which can be converted to temperatures.)

Loads

In the analysis of the single-stage tank, only tank ullage (gage) pressures and thermal loads were considered. This simplification is considered reasonable by virtue of the low magnitude of the stresses caused by drag, inertial, and bending loads, as indicated in Table 1. The stresses induced by thermal gradients and tank ullage pressure of course, are additive, putting the inner facesheet of the honeycomb sandwich in tension. The relatively small stresses from inertial loading and drag on the forebody are relieving for tensile stresses on the inner facesheet.

The bending stresses from aerodynamic lift are relieving for the inner facesheet on the compression side of the tank when the tank is pressurized, but are additive on the tension side. This latter stress negates some of the stress relief from inertial and drag forces. Slapdown loads at landing on the forward end of the tank are expected to design some portions of the tank, but the weight differences are expected to be small and were not considered in this analysis.

Finite Element Program

After establishing the design drivers, an analysis was undertaken of the circular body hot structure using the STAGS program (Structural Analysis of General Shells).[22] This program has been in existence for about 11 yr with several updates resulting in increased capability and efficiency. The structure may be modeled using up to 30 different simple shell units such as cones, cylinders, paraboloids, etc., and one set of various finite elements. The element units include linear axial and torsional springs, nonlinear beams, and nonlinear quadrilateral plate elements.

Subroutines were used to input the temperature distributions as a function of the spatial coordinates and vary material properties as a function of temperature. One unique feature of STAGS is that the temperature can vary radially through the shell elements. This feature does not exist in conventional plate finite elements which are utilized in programs such as SPAR[23] and NASTRAN.[24]

In the finite element model, 17 different shell units were used to simulate the tank (Fig. 10). In turn, each unit consists of two to four frustrums of a cone or sections of a cylinder. This modeling technique is made possible by the STAGS program which geometrically describes axisymmetric bodies of revolution. One ring frame was assigned to each shell unit. For purposes of modeling, seven structural layers were utilized to simulate the shell wall—a layer for the inner and outer skin and five intermediate layers for the honeycomb. Each layer was assigned a temperature and the material properties associated with that temperature. The STAGS program output consists of the stresses and forces at the inner and outer surfaces of each layer as well as the deformation of the midsurface of the element.

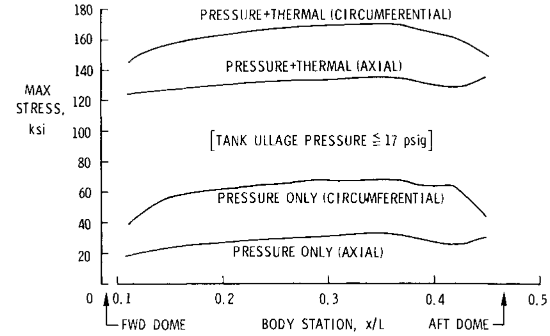

Fig. 11 Maximum stresses for ascent (inner facesheet).

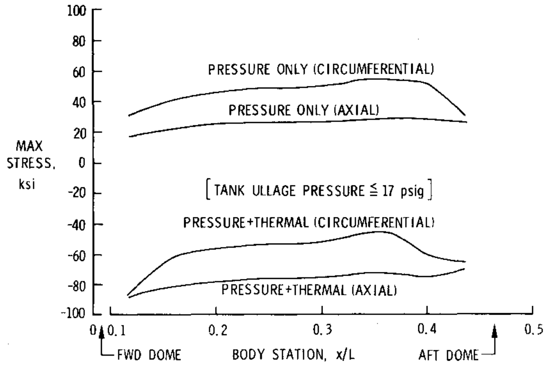

Fig. 12 Maximum stresses for ascent (outer facesheet).

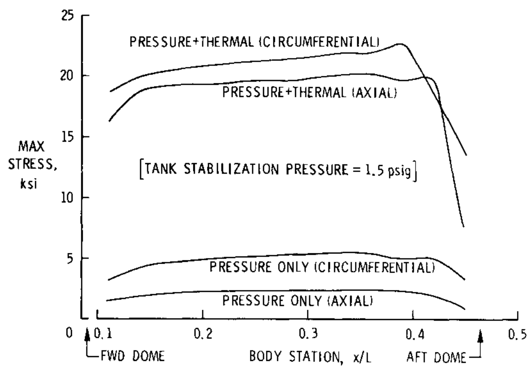

Fig. 13 Maximum stresses for entry (inner facesheet).

Analysis Results

The results of the stress analysis for ascent using the STAGS program are summarized in Figs. 11 and 12. The maximum tensile stress found for the ascent case is approximately 170 ksi in the circumferential direction and occurs on the inner facesheet at the bottom centerline near x/L station 0.35 (Fig. 11). The results are based on a purely elastic analysis. The inner facesheet at this time is at −420°F. At this temperature, the yield stress for the Inconel 718 is over 200 ksi or about 18% higher than room temperature values. Also at this temperature, notch toughness when compared to room temperature values is up from 170 to over 200 ksi.[25]

The maximum compressive stresses occur on the outer facesheet of the sandwich during ascent (Fig. 12). These maximums are in the axial direction and occur near ring frames. The maximum compressive stress is 85 ksi and occurs at the forward end of the tank at the bottom centerline. The outer facesheet is at 350°F at this station for this point in time in the trajectory. Circumferential and axial stresses for the tank under pressure without thermal gradients are shown for purposes of comparison.

For entry, the maximum stress is 23 ksi from circumferential tension on the inner facesheet near the aft end of the exposed tank on the bottom centerline (Fig. 13) For this condition, the yield strength of the inconel sheet has dropped to about 120 ksi, whereas the ultimate stength has dropped from over 200 to 140 ksi. These stength values are considered to be more than adequate and are those associated with an estimated 1300-1400°F temperature of the inner facesheet. Compression stresses (not shown) are equally low (in absolute value) for the outside facesheet at entry and are not considered to be design drivers. A tank stabiliztion pressure of 1.5 psig was assumed for the entry case.

Since a gage thickness of 0.025 in. for the honeycomb sandwich facesheets was used throughout, optimization could be used to reduce peak stresses and tank weight. For example, skin thicknesses could be varied to reduce stress levels. One approach might be to increase the inner facesheet thickness relative to the outer facesheet thickness.[2] A number of computer runs were made in order to optimize the ring frames—these structural elements being designed by both thermal and aerodynamic buckling loads. However, aerodynamic loads are relatively small compared to thermally induced loads. Therefore, for the preliminary analysis, the ring frames were sized using only thermally induced loads. On this basis, a ring fame of relatively small dimensions was found to be adequate. The resultant tank weighs only 26,000 lb for the storage of 216,000 lb of LH_2 and is 11% lighter than the Shuttle external LH_2 tank of skin-stringer aluminum structure when compared on the basis of pounds of tank per pound of propellant stored.

Summary Remarks

Previous studies of various overall launch systems and individual vehicle reveal substantial differences in integral tank weights depending upon the design approach taken. For example, booster tanks in a two-stage system may be penalized 30-50% in weight because of the loads imposed by an attached Orbiter.

In this study, an analysis of an LH_2 integral forebody tank on a single-stage Earth-to-orbit transport is presented as an example. The analysis could equally apply to an LH_2 tank in the Orbiter of a two-stage system. A honeycomb sandwich of Inconel 718 was selected for the shell of the tank—the single shell serving as thermal protection, structure, and tank. The principal design driver is the large temperature gradient experienced across the tank wall during ascent created by aerodynamic heating on the outisde of the shell and cryogenic LH_2 inside.

The honeycomb sandwich shell approach was selected because of its simplicity and potential light weight when compared to an integral tank having multilayers consisting of tank wall, cryogenic insulation, high-temperature insulation, and heat shield. Major differences in this study over a previous study is that a simple round shell was assumed, and Inconel 718 (instead of René 41 or titanium) was used for the honeycomb sandwich wall.[2] The stresses predicted during ascent and entry appear to be within the capabilities of the inconel material selected. Further, these stresses should be even lower when plasticity of the material is taken into consideration.

Whatever the approach to tank design for these future space transports, more research and development will be needed to establish the most viable concepts. Based on the considerations discussed in this paper and the results of a preliminary structural analysis, the honeycomb sandwich circular tank design without insulation looks very promising.

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