Structural Design for a Hypersonic Research Airplane

L. Robert Jackson*
NASA Langley Research Center, Hampton, Va.
and
Allan H. Taylor†
Vought Corporation, Hampton, Va.

A research airplane was studied which has potential for large-scale demonstration of advanced propulsive, structural, and aerodynamic technologies for hypersonic application. Versatility is achieved through the selection of a configuration that provides for scramjet engine integration by the extensive performance envelope and by large removable sections of the airplane. After a screening of alternative structural concepts, design criteria were applied to an effective heat-sink structure of Lockalloy (Be-38 Al), wherein thermal stress alleviation is a prime consideration in the design. Structural analyses were performed with the SPAR computer program. Results indicate that no critical problems exist, and the resulting weight is within the initial estimate.

Nomenclature

C_p	= specific heat
E	= Young's modulus of elasticity
e	= strain
g	= acceleration due to gravity
k	= thermal conductivity
L	= length
M	= Mach number
N	= load factor
ΔP	= pressure difference
q	= dynamic pressure
ΔT	= temperature difference
ρ	= density
σ_{ty}	= tensile yield strength
σ_{ult}	= ultimate load stress
σ_{all}	= allowable stress
σ_{lim}	= limit load stress

Subscripts

x	= axial direction
y	= lateral direction
z	= vertical direction

Introduction

AS described in Ref. 1, the U.S. Air Force and NASA have been studying a B-52-launched hypersonic research airplane. One configuration concept, shown in Fig. 1, has a delta planform, is about 50 ft long, and is constrained by B-52 envelope and weight limitations. A fundamental objective of the research airplane is to provide versatile research capability with the least expenditure of resources for the life of the airplane. Research versatility is provided by the selection of a configuration that provides for scramjet integration, by the extent of the flight envelope, by a large replaceable payload bay with removable thermal protection, and by replaceable wings.

Factors considered to conserve resources include use of near-art construction for the airplane structure and thermal protection system (TPS) and use of available government-furnished equipment, such as the main propulsion rocket, landing gear, and avionics. Operating cost is minimized through selection of a reusable TPS that can withstand flight and ground-handling environments with negligible refurbishment.

Numerous structural/TPS concepts were studied, and results of screening which led to the one most attractive system are given. Material selection, structural arrangement, thermostructural design features, and wall construction are presented. The designs studied are supported by analysis wherein design criteria and trajectories were selected to determine temperatures and stresses. Weights were based on nonexceedance of material allowables. In particular, this paper presents the design of the heat-sink structure and addresses the principal thermal stress problem. A subsequent paper[2] details analytical results of the numerous thermostructural design options.

Research Airplane Versatility

Research versatility of interest for demonstration of advanced structures is provided by an extensive flight envelope and by replaceable sections of the research airplane. However, the configuration also provides research versatility for flight demonstration of advanced scramjet engines.

Configuration

The research airplane configuration considered in this structural study is shown in Fig. 1. The vehicle is 50 ft long and has a span of 24.2 ft and a depth of 8.67 ft, including an

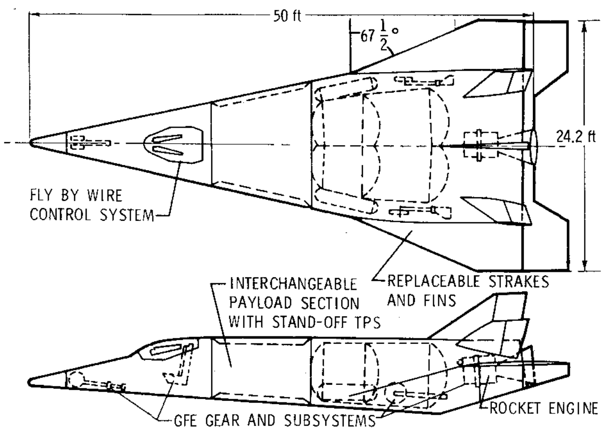


Fig. 1 Research airplane study configuration.

Received Sept. 13, 1976; presented as Paper 76-906 at the AIAA Aircraft Systems and Technology Meeting, Dallas, Texas, Sept. 27-29, 1976; revision received March 21, 1978.

Index categories: Structural Design; Thermal Stresses.

*Aero-Space Technologist, Hypersonic Aerodynamics Branch, High-Speed Aerodynamics Division. Member AIAA.

†Project Engineer, Hampton Technical Center. Member AIAA.

allowance for air-breathing research engines that may be mounted under the fuselage. The span, depth, and gross weight of about 60,000 lb are limits imposed by the B-52 launch airplane, which is available from the X-15 program. Of the 60,000 lb, about 10,300 lb were targeted for structure and thermal protection system weights from initial weight estimates used for performance analyses.

Further evolutionary changes are being studied for the configuration[3]; however, for moderate configurational changes, no significant changes in results of this structural study are anticipated. For the type of structure studied, thermal stress predominates. The principal cause of thermal stress is the temperature lag between the aerodynamic surface and the inner cap of the fuselage ring frames. This source of thermal stress is independent of vehicle configuration, since for any configuration the skin thickness is sized to result in a peak temperature of 600°F based on the local heat load, and the inner frame cap temperature is a function of the frame construction and the limited skin temperature. Therefore, the overall conclusions of this structural study are not sensitive to configuration changes.

Performance

Research aircraft performance provides versatility through a wide range of Mach numbers and dynamic pressures. Figure 2 shows current performance objectives in terms of Mach number and altitude. As indicated, the research airplane flight envelope can include significant portions of ascent and entry vehicle trajectories. Virtually the entire trajectory of hypersonic cruise vehicles is included in the research airplane flight envelope, but the flight duration is less for the research airplane. However, since the greatest temperature difference within a structure occurs during transient heating, maximum thermal stresses may be generated during the relatively short flight time of the research airplane. An additional performance factor is that the research airplane can sustain higher maneuver loads than required for operational vehicles. Thus, realistic temperatures, gradients, and air loads are achievable with the research aircraft.

Replaceable Sections and Payloads

To fulfill the research functions of demonstrating advanced structures and propulsion systems for hypersonic application, large sections of the research airplane are replaceable. Figure 3 shows the replaceable sections, which include the nose section, payload bay, vertical stabilizer, wings, leading edges, heat shields on the payload bay, and heat-sink skin panels on remaining surfaces. Some panels may be replaced with advanced thermal protection systems for flight demonstration; others may be replaced to suit local structural and thermal requirements for flight tests of a scramjet engine, as discussed in more detail in Ref. 2. Versatility is a structural design consideration, so that numerous advanced technologies can be adequately demonstrated in flight.

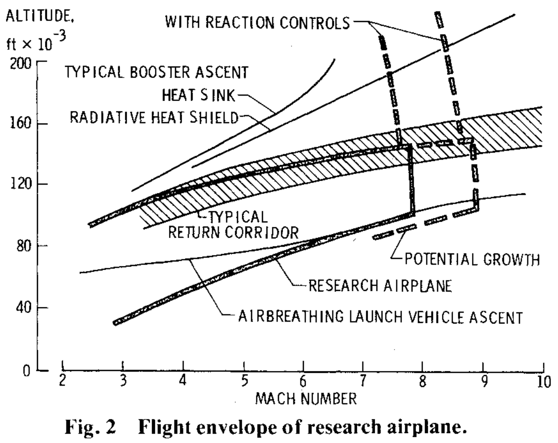


Fig. 2 Flight envelope of research airplane.

Structural developments such as integral tanks with advanced standoff thermal protection may be demonstrated. Such tests of either tank structures or advanced vehicle fuselage structures are fully representative of operational vehicles. The 10-ft-long payload bay provides a ratio of shell length to diameter of 2/1. Of particular significance, the general instability mode is made possible by the payload bay proportions. This is the common mode of failure for operational vehicles. This test feature complements ground testing. However, the research airplane provides the only test facility capable of simultaneously applying flight environment to a representative structure capable of general instability. Wind tunnels can duplicate the environment reasonably well, but test specimens are too small to achieve general instability with representative construction. Static laboratory test specimens can be large for verifying predicted general instability failure loads, but the environment is not realistic. The demonstration structure, having sustained flight limit loads and environment as an integral component of the research airplane, can be applied confidently to operational vehicles.

Structural Design

A number of structural solutions have been studied in-house and contractually.[4,5] A review of the in-house screening of structural/TPS concepts for research airplane application is given, followed by a more detailed discussion of the selected structural design.

Alternative Structural Concepts

Alternative structural concepts studied are shown in Fig. 4. They include two basic types of systems; the first is single-wall construction, and the second is double-wall construction. Generally the single-wall concepts are simpler. Referring to Fig. 4, these concepts include the direct bond insulator/ablator and reusable surface insulation (RSI) con-

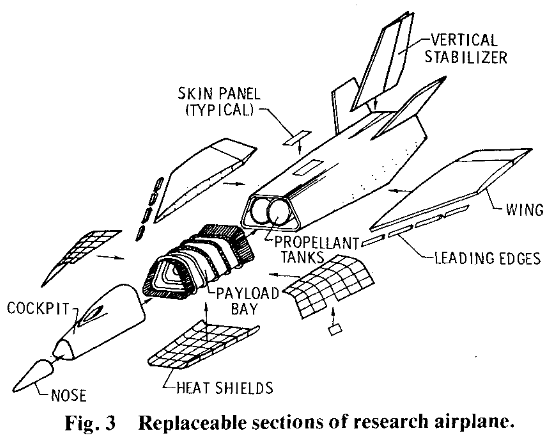


Fig. 3 Replaceable sections of research airplane.

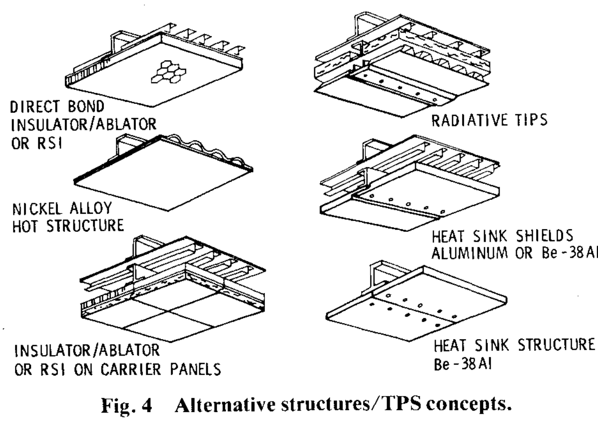


Fig. 4 Alternative structures/TPS concepts.

cepts, and the hot structure and heat-sink structure concepts. Each of these TPS concepts may also be applied to a double-wall construction wherein the thermal projection function is separated from the load-bearing structure. Slip-jointed shields on insulation or insulated carrier panels provide the TPS, whereas a conventional aluminum structure provides the load-bearing function. Estimated unit weights and costs are shown in Fig. 5 for the various structure/TPS concepts considered. Costs are based on data taken from Ref. 4 with in-house estimated refurbishment rates.

Direct bond concepts each have low initial cost, with the insulator/ablator having the lowest initial cost of all concepts. The reusable surface insulation is potentially the lightest concept. Consequently, these direct bond concepts were retained for further study. The insulator/ablator is a primary candidate and, with the shuttle sponsoring the development of RSI, this form of insulator is an alternate to the insulator/ablator for the research airplane. Refurbishment requirements of the external insulation concepts have to be determined for the research airplane.

Nickel alloy hot structure has the highest initial cost, about 19,000 $\$/ft^2$ or 21.6 M$ more vehicle cost than the direct bond/insulator concept. Although the weight is competitive, this high initial cost is excessive. Consequently, the hot structure has been screened from this study.

Carrier panels with either insulator/ablator or RSI are more expensive and heavier than the direct bond concepts. However, the versatility offered by the standoff feature of these systems is needed for research. Consequently, these systems are candidates for the payload bay.

Radiative TPS has high initial cost and potentially high RDT&E costs. It is conceivable that an elementary radiative TPS concept could be constructed economically at an acceptable weight penalty. However, such a concept has been neither defined nor developed. A further effect on research airplane cost is tank volumetric efficiency. Radiative TPS with nonintegral tanks requires thicker walls than either direct bond or heat-sink structure concepts. Consequently, radiative TPS results in lower volumetric efficiency, requiring a larger vehicle for a given performance. Therefore, radiative TPS has been dropped from this study for the basic structure. Other standoff TPS are more suited to the initial payload bay construction, and so radiative TPS is also dropped for this area. Radiative TPS is a candidate for research on the payload bay. However, radiative TPS could be a candidate for the initial TPS of the payload bay should the radiative TPS be man-rated by current research prior to final design of the research airplane.

Heat-sink TPS on standoff shields offers the same versatility as external insulators on carrier panels, but heat-sink TPS is heavier. Aluminum heat sink is too heavy, whereas Lockalloy is more weight-competitive. Lockalloy standoff TPS is a candidate for the payload bay.

Heat-sink structure offers a significant weight savings over the heat-sink shield concept. A heat-sink structure of Lockalloy is weight-competitive with the direct bonded insulator concepts. An aluminum heat-sink structure is too heavy and has been dropped from this study. In addition, with the Lockalloy plates bolted to the substructure as proposed by Lockheed,[5] internal access is provided at all locations, and advanced TPS can be tested at all locations. This versatility is possible by replacing a structural panel, approximately 20 × 24 in., with an equivalent recessed structure with advanced TPS for research.

Heat-sink structure is also cost-competitive with the direct bond concepts. As indicated, initial vehicle cost may be increased by about 1 M$ for Lockalloy over the insulator/ablator concept. However, the heat-sink structure requires negligible refurbishment, and so its life-cycle cost should be less. Moreover, the heat-sink structure is more readily disassembled for installation of large demonstration structures. For these reasons, the Lockalloy heat-sink structure was selected for this structural study.

Heat-Sink Material

Several heat-sink materials are candidates for a heat-sink structure. In addition to the aluminum alloys and Lockalloy, titanium, beryllium, and super alloys may be used for heat-sink structure. Reference 6 shows that the heat-sink weight is competitive for the Lockalloy, titanium, and beryllium materials. However, for weight efficiency the titanium and beryllium operate to higher peak temperatures than the Lockalloy. Sealing out hot boundary-layer air may be achieved by silicone adhesives with the 600°F Lockalloy, but the 900°F titanium and 1200°F beryllium each require further study of sealants. Moreover, since these latter two systems operate at higher temperatures, each requires more internal insulation than the Lockalloy system, which, when added to the heat-sink weight, may result in greater total weight than the Lockalloy system. Further study may indicate that a mixture of materials offers the best heat-sink structure.

Lockalloy has a high specific heat and specific stiffness at low density, as seen in Table 1 and Fig. 6. Be-38Al has higher

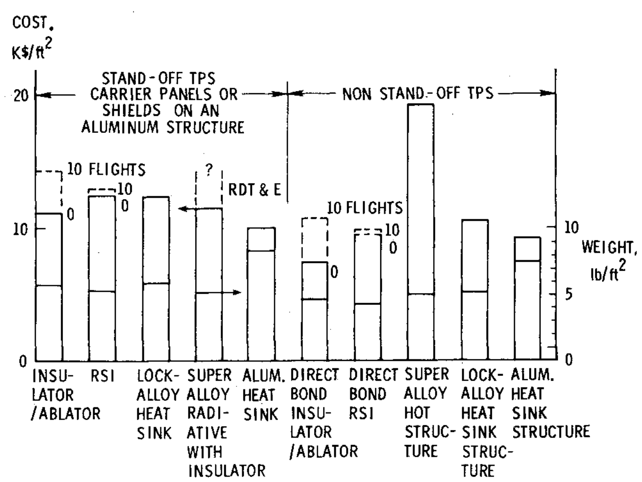
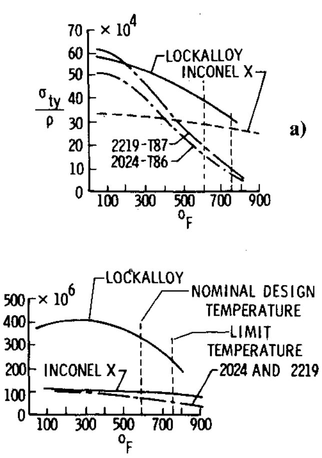


Fig. 5 Cost and weight comparisons of structure/TPS concepts.



a) Specific strength

b) Specific stiffness

Fig. 6 Comparative heat-sink material properties vs temperature.

Table 1 Comparative heat-sink material properties

		Lockalloy	Aluminum	Inconel X
K	=	1440 Btu-in./h-ft^2-°F	889	202
C_P	=	0.45 Btu/lb-°F	0.23	0.12
ρ	=	0.075 lb/in.^{3}	0.102	0.300
e	=	8%	5%	20%

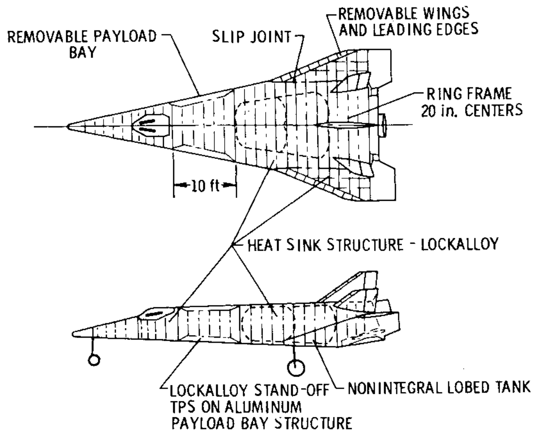


Fig. 7 Structural arrangement.

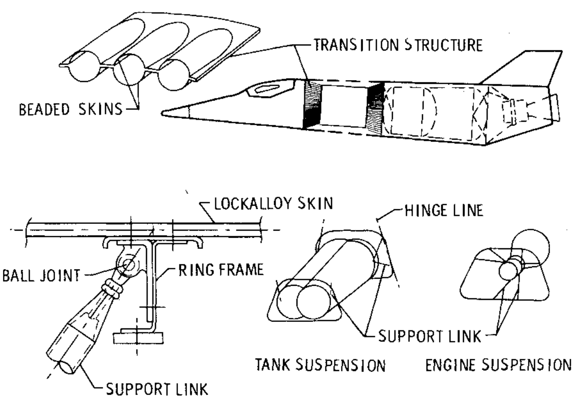


Fig. 8 Thermostructural design features.

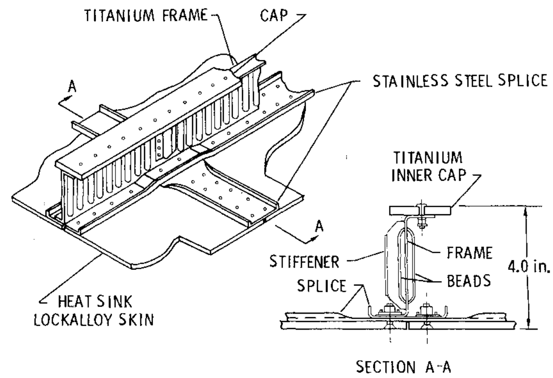


Fig. 9 Fuselage heat-sink structural design.

specific strength and stiffness than either the Inconel-X used as heat-sink material for the X-15 or aluminum-alloy heat-sink material considered for the flyback booster in early shuttle studies. Moreover, to enable fail-safe and other desirable operational features, the Be-38A1 has excellent ductility, greater than that of structural aluminum. Further characterization of Be-38A1 is reported in Ref. 7, which resulted from a study that successfully fabricated a Be-38A1 ventral fin for the YF-12 Mach 3 + aircraft.

Structural Arrangement

The structural arrangement is shown in Fig. 7. The shell structure is stabilized by ring frames spaced on 20-in. centers. A slip joint is provided at the wing root to permit testing advanced hot structure wings by attachment to the fuselage fittings without structural modification. The primary structure in the payload bay is recessed 6 in. on the lower surfaces and 4 in. on the sides and top. The recess is provided to enable testing advanced radiative TPS consisting of shields and insulation packages. The radiative TPS replaces the BE-38A1 heat-sink shields for flight testing.

As indicated in Fig. 7, a lobed nonintegral propellant tank was selected to provide high volumetric efficiency with state-of-the-art construction. A double-wall bulkhead, actually two separate tanks, are nested at what appears to be a common bulkhead. Foam-filled nonmetallic honeycomb core is bonded between mating domes, and a splice plate joins tank walls at Y rings at the dome peripheries.

Thermostructural Design Features

Special features of the structural arrangement are necessary to accommodate unrestrained thermal growth of the shell with simultaneous support of inertial or thrust loads. Figure 8 shows the thermostructural design features for the tank suspension, engine suspension, and payload bay transition structure. A six-link suspension system for the tank provides simple support for vertical loads and cantilever support at the aft end for lateral loads. The aft end of the tank is not free to swing because the hinge lines provided by the pivoted attachment of the links to the body are not parallel on opposite sides of the tank. Thus, the tank is free to contract when filled, and the body is free to expand on heating without restraint by the tank, while simultaneously the tank inertial loads are supported without detrimental tank motions. Engine suspension is similar to the tank suspension; however, three points on the engine are identically attached to the structure, providing a cantilever support in all planes. The pivoted links permit the fuselage shell to grow without restraint by the engine thrust structure. Similar link-type attachments are required for other items, such as the landing gear and crew compartment. A detail of the link joint is shown in Fig. 8. The centerline of the link intersects the centerline of the frame and the centroid of the Lockalloy skin.

Transition structures at the ends of the payload bay permit radial expansion of the heat-sink structures without restraint by the relatively cold payload bay structure. Simultaneously, axial and shear loads are transmitted through the transition structures, providing continuous load support. The transition structure consists of back-to-back beaded sheets, forming a multitubular cross section. This construction is very efficient for support of loads, as shown in Ref. 8. One end of the transition is hot, whereas the other end is cold. The beaded skins open on the hot end, similar to a fan, offering little restraint to thermal growth of the heat-sink structure. Flat ends flex where attached to ring frames to accommodate the slight angular change in the transition structure as the heat-sink structure expands on heating. Consequently, several sources of thermal stress are avoided as a result of the thermal-structural design features studied. However, not all sources of thermal stress are avoidable, but these may be reduced and sustained by further application of effective thermostructural design.

Wall Constructions

The fuselage heat-sink structural wall is shown in Fig. 9. Ring frames, shown on 20-in. centers in Fig. 7, consist of an outer cap, a Z-section ring, stiffeners on the ring web, and an inner cap. The outer cap serves as a splice plate for the Lockalloy skins. Longitudinal splices are provided to enable removal of the skins for selective and occasional access to the interior. The Lockalloy skins are fastened by snug-fitting (class 4) fasteners, since the skins are the primary load-bearing structure. The ring frames stabilize the Lockalloy shell and provide support of transverse bending resulting from aerodynamic lift and vent pressure loads. Since the frames are essential and since the inner cap lags in temperature from the shell temperature, the frames introduce an unavoidable source of thermal stress. However, as will be shown in a later discussion, effective choice of materials, frame proportions, and design can yield acceptable thermal

Table 2 Selected design criteria[a]

Limit load factors	N_z	N_x	N_y	Propellant
B-52 taxi	2.0	±0.25	0	Full
Postlaunch pull-up	2.5	±1.5 −0.5	±0.5	Full
Max. Mach pull-up	4.0	±0.5	±0.5	Residual
Speed brakes	0	−2.0	0	Residual
Subsonic flight	3.5 −1.0	±0.5	±0.5	Residual
Landing (10 ft/s)	2.0	−1.0	±0.5	Residual

[a] Dynamic pressure limit $q_{max} = 1250$ psf; vehicle flutter $q = 1.32q_{max}$; panel flutter $q = 1.5q_{local}$; laminar heating factor = 1.10; turbulent heating factor = 1.25; interference heating factor = 1.50; noise 160 dB takeoff, 158 dB climb; boundary-layer noise $\Delta P = 0.007q_{local\ max}$, $0.022q_{local\ max}$ (trans.), and $0.10q_{local\ max}$ (separated); permanent strain maximum 0.2% (creep or stress); maximum mismatch at gaps = 0.050 in. for aft-facing steps only; exclude inflow of boundary-layer air.

stresses. One design feature for alleviation of thermal stress is shown in Fig. 9; that is, the frame webs are beaded on close centers to reduce extensional stiffness of the web. Thus, little restraint is offered by the web to differential expansion caused by the nonlinear temperature distribution through the depth of the web.

Wing structure is similar to the basic heat-sink structure; however, rib and spar caps serve as longitudinal and transverse splices for the skins. Webs have vertical beads between upper and lower caps on close centers to alleviate thermal stress resulting from the lag in web temperatures and the nonlinear temperature distributions in the rib and spar webs.

Wall construction in the payload bay consists of a conventional aluminum-alloy skin-stringer structure as shown in Fig. 10. Stiffeners external to the skin are used for simplicity and less weight. Heat shields of Lockalloy are attached to the structure at the ring frames by standoff frames of low-conductivity metal. Slip joints and slotted holes permit the shields to expand relative to the primary structure at little restraint to avoid thermal stress. Thermal stress alleviation of the standoff frames is provided by a bead-stiffened truss web, as shown in Fig. 10.

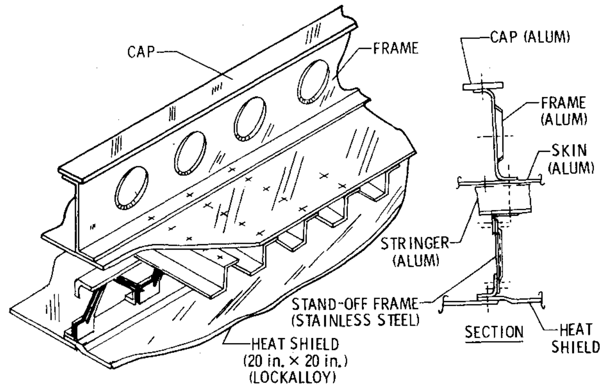


Fig. 10 Payload bay wall construction.

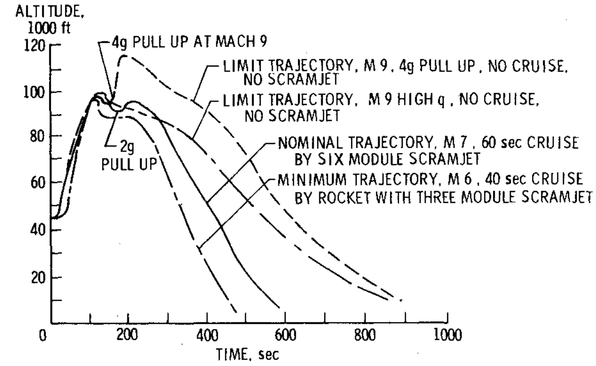


Fig. 11 Design trajectories showing altitude vs time.

Structural Analysis

To perform the structural analyses, loads and temperatures were determined from design criteria and design trajectories.

Design Criteria

Design criteria were selected and are listed in Table 2. Some criteria result from X-15 experience and B-52 launch vehicle limits. Space Shuttle design criteria[9] offer a wide range of general criteria, some of which have been selected for this study. Other criteria are those judged necessary for the research airplane under study. Normally, load factors are applied to the 1-g load when a specific trajectory has not been defined. However, when aerodynamic heating is significant, specific trajectories are essential to determine temperatures, local heat loads, and the resulting thicknesses of Lockalloy.

Design Trajectories

Four design trajectories are given in Fig. 11 in terms of altitude and flight time. The trajectory analysis is a computerized integration of the equations of motion using time as a variable, with design constraints imposed. The minimum trajectory satisfies the minimum scramjet experiment of 40 s rocket-powered cruise at Mach 6 with a small three-module test engine. However, the vehicle must satisfy more severe trajectories. A nominal trajectory was selected consisting of 60 s of scramjet-powered cruise at Mach 7 with six larger scramjet modules. This trajectory provides the design heat load for heat-sink weight determination, with the operating temperature held to a peak value of 600°F.

Two limit trajectories are defined. One consists of a 4-g pullup maneuver at the maximum speed of Mach 9. The other limit trajectory is also at Mach 9, but no pullup is performed. In the latter trajectory, high-dynamic-pressure flight is maintained for 60 s after burnout. Thus, a higher heat load is achieved for the high-dynamic-pressure trajectory than for the nominal Mach 7 trajectory. This limit trajectory is the most severe heat load and results in a 750°F temperature, which is the selected limit temperature for Lockalloy. Each of the Mach 9 limit trajectories requires study. The 4-g pullup maneuver trajectory adds higher air loads to the 600°F temperature and corresponding thermal stresses; whereas the high-dynamic-pressure trajectory has higher thermal stresses and higher Lockalloy temperatures, which lower the allowable stresses.

Several trajectories are needed to determine the required structural weight, because different parts of the structure are designed by local limit loads, which depend on the specific load condition. For instance, the wings experience limit bending loads during the 2.5-g pullup maneuver occurring after launch from the B-52, and the landing gear area is designed by the 2-g landing limit load. However, in this study, design effort is primarily concerned with overall structural response rather than localized areas. Thus, air, inertial, venting, and thermal loads are of principal interest.

Structural Temperatures

Computer analyses were used to determine heat loads and resulting heat-sink skin thicknesses and transient temperatures of the individual structural components. The transient heating program is given in Ref. 10.

Fuselage structural temperatures for a point on the lower surface centerline 100 in. aft of the nose are shown in Fig. 12. The skin thickness at this location is 0.431 in. Local skin thicknesses are determined by the local heat load calculated for the nominal trajectory and a 600°F design temperature for the heat-sink material.

The Mach 9, 4-g pullup trajectory has about the same heat load as the nominal trajectory, as indicated by the same 600°F maximum skin temperature. However, the Mach 9, high-

Table 3 Limit loads and allowables

Loads

1) Mach 9, 4-g pull-up trajectory:

$\Delta T_{max} = 525°F$: $T_{lower\ surface} = 600°F$, $T_{sides\ and\ upper} = 400°F$
$\Delta P = 1.0$ psi (venting acting inward) all surfaces
$\Delta P = 1.13$ psi (lift at 4-g inertial load) lower surface

2) Mach 9, high-dynamic-pressure trajectory:

$\Delta T_{max} = 650°F$, $T_{lower\ surface} = 750°F$, $T_{sides\ and\ upper} = 725°F$
$\Delta P = 1.0$ psi (venting acting inward) all surfaces
$\Delta P = 0.3$ psi (lift at 1-g inertial loading)

Allowables (tension yield)

Lockalloy	= 20 ksi at 750°F (limit temperature)
	= 22 ksi at 600°F (nominal temperature)
Aluminum	= 41 ksi (2024-T4) and 61 ksi (7075-T6) at 70°F
Titanium	= 145 ksi at 70°F, 109 ksi at 600°F, 100 ksi at 750°F
301 stainless steel	= 93 ksi at 70°F, 78 ksi at 600°F, 75.3 ksi at 750°F
E-glass epoxy	= 167 ksi at 70°F
Kevlar	= 133 ksi at 70°F

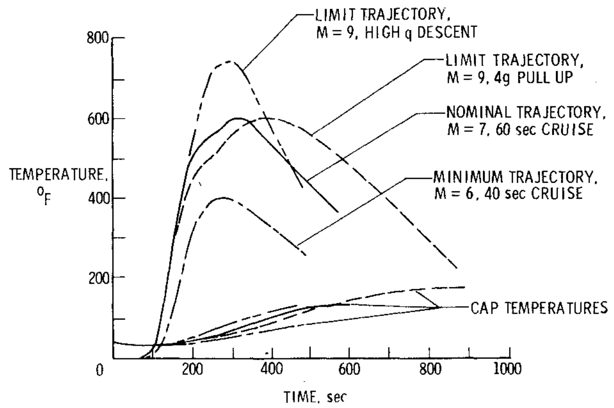


Fig. 12 Fuselage structural temperatures at 100-in. station on lower centerline with constant 0.431-in.-thick skin.

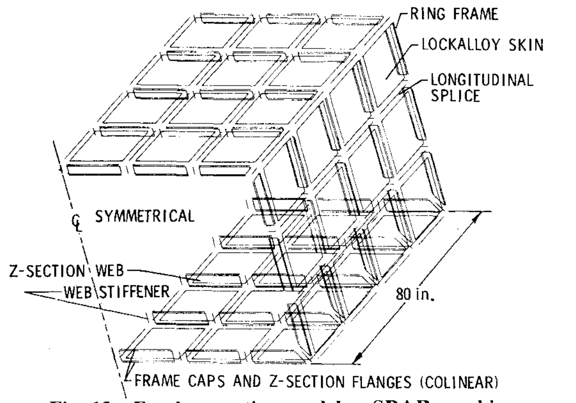


Fig. 13 Fuselage section model — SPAR graphics.

dynamic-pressure trajectory has about 25% greater heat load than nominal. Since the local skin thickness is not changed, temperatures of 750°F are attained for this trajectory which is the limit temperature for the heat-sink material.

Corresponding internal cap temperatures are shown in Fig. 12 for the trajectories. At the time of peak skin temperature, the cap temperature is only about 100°F. Consequently, a temperature difference of over 500°F exists through the frame depth. This temperature difference is responsible for the principal thermal stresses in the structure.

Limit Loads and Allowables

With the maneuver and heating conditions determined, temperature differences and air pressures provide limit loads when combined with inertial loads. Limit loads are given in Table 3 for the Mach 9 trajectories. As indicated, the air loads are greater for the 4-g pullup trajectory, and the temperature difference is greater for the high-dynamic-pressure trajectory. Limit air, inertial, and vent loads are increased by a factor of 1.5 and then combined with limit thermal loads to yield ultimate loads.

Material allowables are listed for the Lockalloy at both nominal design and limit temperatures. Various materials were studied for the ring frames. Aluminum was considered for low cost. Stainless steel was considered to reduce thermal stress, because it has the same coefficient of thermal expansion as Lockalloy. Titanium was considered for its high strength, and the near-art composites were studied for the inner cap to reduce thermal stress through their low modulus of elasticity. Allowable stresses are based on yield strength rather than ultimate strength, because elements loaded in compression generally buckle at yield if not at lower stresses. Moreover, thermal buckling is the principal failure mode of concern. To assist in the material selection, a fuselage section consisting of several ring frames was analyzed.

Fuselage Section Analysis

A fuselage section was modeled to obtain experience with a new computer program, SPAR (structural performance and resize),[11] before attempting to model the entire research airplane. The SPAR program is based on small deflection theory solved by a system of computer programs consisting of linear finite-element components. SPAR graphics produced the exploded view of the fuselage section in Fig. 13. A primary function of the section model was to study alternative designs of ring frames. As see in Fig. 13, the model consists of five ring frames and numerous structural elements. Elements consist of Lockalloy skins, longitudinal splices, outer and inner frame caps, outer and inner flanges on the Z-section frames, the Z-section web, and web stiffeners.

Results of the fuselage section analysis for the Mach 9, 4-g limit load condition are given in Tables 4a and 4b. Five frame designs are shown for fuselage areas where the Lockalloy skin is thin, which occurs where the heat load is relatively low.

Thermal buckling of the thin skins is a design consideration. The first frame design given in Table 4 consists of a 0.15-in.-thick unstiffened skin with a titanium ring frame. As indicated, the Lockalloy skin compressive stress is 18.4 ksi; however, the critical buckling stress is only 5.7 ksi; consequently, a large negative margin results. One source of this excessive compressive stress is the use of titanium for the frame. Titanium has only half the coefficient of expansion of Lockalloy; thus the outer cap of the frame, which is heated to about the temperature of the skin, is resisting the thermal growth of the Lockalloy.

In the second frame design, stainless steel is selected for the web and outer cap of the frame. Stainless steel has the same coefficient of expansion as Lockalloy. Consequently, the compressive stress in the Lockalloy is greatly reduced. However, the Lockalloy skin is still subject to thermal buckling.

Stiffening the Lockalloy skins with Lockalloy extrusions riveted to the inner surface between ring frames prevents the thermal buckling while preserving the individual panel removal feature. However, with titanium outer caps on the frames, the hoop stress in the skins exceeds the yield stress, as seen in Table 4 for frame design 3. Thus interfastener buckling and low cycle fatigue are possible failure modes. By making the stainless-steel substitution for titanium, the hoop stress in the skins is less than yield, resulting in positive margins, as seen in frame design 4. One additional beneficial

Table 4a Results of fuselage section analysis for thin skins: effect of splice material

Case	Element	Size	Material	$\sigma_{\lim}$ (thermal), ksi	σ_{ult} (combined), ksi	$(\sigma_{\mathrm{all}}/\sigma_{\mathrm{ult}})-1$
1)	Skins	0.15 in.	Lockalloy	−18.4	−21.4	−0.7
	Splices	0.6 in.^{2}	Titanium	34.5	34.5	3.2
	Caps	0.6 in.^{2}	Titanium	71.2	74.1	0.9
2)	Skins	0.15 in.	Lockalloy	− 9.9	−10.8	−0.5
	Splices	0.3 in.^{2}	301 − 1/2H cres	−17.2	−15.1	5.0
	Caps	0.6 in.^{2}	2024-T4 al.	41.3	58.7	−0.3
			7075-T6 al.			0.04

Table 4b Results of fuselage section analysis for thin skins: effect of skin-stiffening and cap material

Case	Element	Size	Material	$\sigma_{\lim}$ (thermal), ksi	σ_{ult} (combined), ksi	$(\sigma_{\mathrm{all}}/\sigma_{\mathrm{ult}})-1$
3)	Skins with stiffeners	0.15 in.	Lockalloy	−40.6	−57.1	−0.5
	Splices	0.6 in.^{2}	Titanium	33.4	33.5	3.3
	Caps	0.6 in.^{2}	7075-T6 al.	32.7	50.7	0.2
4)	Skins with stiffeners	0.15 in.	Lockalloy	−14.6	−23.1	0.2
	Splices	1.2 in.^{2}	301 − 1/2H cres	20.0	−19.8	4.4
	Caps	0.6 in.^{2}	E-glass epoxy	37.9	49.6	2.4
5)	Skins with stiffeners	0.15 in.	Lockalloy	−22.5	−31.5	−0.1
	Splices	1.2 in.^{2}	301 − 1/2H cres	−28.8	−27.2	1.9
	Caps	0.6 in.^{2}	Titanium	67.9	77.0	0.4

design change, however, was the increase in outer cap area. This change shifted the centroid of the frame-skin combination outward, thereby reducing the hoop stress in the skin.

In addition to stress in the skin, stress in the inner cap is of interest for various cap materials. Aluminum alloys were studied as caps for cost savings. The more ductile 2024-T4 alloy is stressed beyond yield, whereas the less ductile 7075-T6 offers a positive margin. Near-art composites show large positive margins and therefore warrant further study. However, the deflections were greatest with a composite for inner caps. Titanium 6Al-4V alloy is suited for cap material because of its high strength, and titanium has a positive margin, as seen for frame design 5. However, the high elastic modulus, relative to either aluminum or E-glass epoxy, results in a higher hoop compressive stress in the skin and causes a small negative margin. Further iteration of cap areas, however, can eliminate this negative margin.

In all five frame designs, thermal stress is a large fraction of the ultimate stress, justifying designs that alleviate thermal stress. A fully metallic ring frame design based on the preceding results was selected for the analysis of the entire research airplane structure. This frame consists of a 301 stainless-steel outer cap (with stainless-steel longitudinal splices), a titanium Z-section frame with titanium stiffeners on the web, and a titanium inner cap.

Research Airplane Analysis

Although much was learned from the fuselage section model, a complete research airplane analysis was needed to verify vehicle weight estimates. Vehicle structural analysis with the SPAR program was performed through a finite-element model of the entire structure, shown in Ref. 2.

Both the Mach 9, 4-g loads and the Mach 9, high-dynamic-pressure loads were analyzed. As noted for the fuselage section analysis, thermal stress predominates. Although resizing steps were not completed, few elements had stresses exceeding the allowables, requiring no increase in weight from that initially modeled. With each iteration, weight reduced. The present weight is less than the initial weight estimate made for performance and cost analyses. However, all load conditions have not been analyzed, but indications are that the weight would continue to reduce at a diminishing rate. The initial weight estimated is 10,265 lb. The present weight defined by the SPAR analysis is 10,080 lb, of which 5280 lb is required heat-sink weight, not controlled by SPAR analysis.

Concluding Remarks

A structural design for a high-speed research airplane consisting of an efficient heat sink was studied. The emerging technology of Be-38Al (Lockalloy) offers an effective heat sink and structural material and is therefore ideally suited to the hypersonic research airplane application. Studies have shown Lockalloy to be weight- and cost-competitive with competing thermal protection and structural systems. Moreover, Lockalloy offers a durable removable surface, which is operationally serviceable and eases the performance of numerous research functions.

Thermal stresses predominate in the heat-sink structure. Thermal stress can be sustained within material and structural allowables by logical design. For instance, stainless steel is well suited to ring frame outer caps and skin splices because of its matching thermal expansion with Lockalloy. Heavier outer caps on the frames are beneficial, as well as beading the frame webs of titanium. Ball-jointed links enable support of internal items such as the rocket engine, tanks, and landing gear without generating thermal stress. Other features to alleviate thermal stress are described and are essential to a successful heat-sink structural design.

Thin-skin areas require special attention to avoid thermal buckling. A combination of skin-stiffening and ring frame proportions satisfies this design consideration. No insurmountable structural problems were encountered. However, other load cases, resizing cycles, and structural analyses such as general instability and vehicle flutter are required to verify the design.

Weight, as required by the computer analysis, is less than the initially estimated weight. Further design and analysis with supportive tests may lead to an efficient, fully reusable structure/TPS for the hypersonic research airplane.

References

[1] Hearth, D.A. and Preyss, A.E., "Hypersonic Technology—Approach to an Expanded Program," *Aeronautics and Astronautics*, Vol. 14, Dec. 1976, pp. 20-37.

[2]Taylor, A.H. and Jackson, L.R., "Heat Sink Structural Design Concepts for a Hypersonic Research Airplane," *Journal of Aircraft*, Vol. 15, April 1977, pp. 199-205.

[3]Weidner, J.P., Small, W.J., and Penland, J.A., "Scramjet Integration on Hypersonic Research Aircraft Concepts," AIAA/SAE 12th Propulsion Conference, Palo Alto, Calif., July 26-28, 1976.

[4]Williams, E.T. and Coakley, R.C., "Costing Methodology for High-Speed Research Airplanes Including Parametric Cost Trend Evaluations," Space Div., Rockwell International Corp., Summary, NASA CR-132658, 1974.

[5]Combs, H.G., et al; "Configuration Development Study of the X-24C Hypersonic Research Airplane—Phase III," Lockheed-California Co., Advanced Development Projects, NASA CR-145103, 1977.

[6]Vahl, W.A. and Edwards, C.L.W., "Study of Heat-Sink Thermal Protection Systems for Hypersonic Research Aircraft," AIAA 16th Aerospace Sciences Meeting, Huntsville, Ala., Jan. 16-18, 1978.

[7]Duba, R.J., Haramis, R.C., Marks, R.F., Payne, L., and Sessing, R.C., "YF-12 Lockalloy Ventral Fin Program Final Report," Vol. I, Lockheed-California Co., Advanced Development Projects, NASA CR-144971, Jan. 1976.

[8]Musgrove, M.D., and Geene, B.E., "Advanced Beaded and Tubular Structural Panels," The Boeing Co., NASA CR-2514, Sept. 1975.

[9]"Structural Design Criteria Applicable to a Space Shuttle," NASA SP-8057, Jan. 1971.

[10]Garrett, L.B. and Pitts, J.I., "A General Transient Heat-Transfer Computer Program for Thermally Thick Walls," NASA TM X-2058, Aug. 1970.

[11]Whetstone, W.D., "SPAR Reference Manual," NASA CR-120504, 1975.