NASA CONTRACTOR REPORT

NASA CR-2480

NASA CR-2480

INTERNAL CONVECTIVE COOLING SYSTEMS FOR HYPERSONIC AIRCRAFT

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • FEBRUARY 1975

1. Report No. NASA CR-2480	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle INTERNAL CONVECTIVE COOLING SYSTEMS FOR HYPERSONIC AIRCRAFT		5. Report Date February 1975
		6. Performing Organization Code
7. Author(s) F. M. Anthony, W. H. Dukes, and R. G. Helenbrook		8. Performing Organization Report No.
		10. Work Unit No. 760-66-01-02-00
9. Performing Organization Name and Address Bell Aerospace Post Office Box One Buffalo, NY 14240		11. Contract or Grant No. NAS1-11,357
		13. Type of Report and Period Covered Contractor Report
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

FINAL REPORT

16. Abstract

Parametric studies were conducted to investigate the relative merits of construction materials, coolants, and cooled panel concepts for internal convective cooling systems applied to airframe structures of hydrogen-fueled hypersonic aircraft. These parametric studies were then used as a means of comparing various cooled structural arrangements for a hypersonic transport and a hypersonic research airplane. The cooled airplane studies emphasized weight aspects as related to the choice of materials, structural arrangements, structural temperatures, and matching of the cooling system heat load to the available hydrogen fuel-flow heat sink. Consideration was given to reliability and to fatigue and fracture aspects, as well. Even when auxiliary thermal protection system items such as heat shielding, insulation, and excess hydrogen for cooling are considered the more attractive actively cooled airframe concepts indicated potential payload increases of from 40 percent to over 100 percent for the hypersonic transport as compared to the results of previous studies of the same vehicle configuration with an uncooled airframe.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Hypersonic Aircraft Actively Cooled Structure Cooling System Design Coolant Corrosion	Unclassified - Unlimited STAR Category 02

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	44	$3.75

*For sale by the National Technical Information Service, Springfield, Virginia 22151

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FOREWORD

This document summarizes results of design, analysis, and experimental studies relating to internal convective cooling systems for hypersonic aircraft. The work was performed under contract NAS 1-11-357 for the National Aeronautics and Space Administration, Langley Research Center, Hampton, Virginia by Bell Aerospace Company, Buffalo, New York. At Bell, W. H. Dukes was the Project Manager and F. M. Anthony was the Technical Director. In addition to the authors other personnel who made significant contributions to this program were G. G. Chormann, A. Krivetsky, W. N. Meholick and A. L. Mistretta (loads, structural design and analysis), D. A. Brzezinski, J. A. Giafaglione, and J. D. Witsil, Jr. (cooling system and thermal analysis), E. O. Allen, K. M. Cooper, J. J. Early, C. Rosini, and W. Yurkowsky (failure, hazard, and reliability analyses), and Dr. J. A. Davis, S. A. Long and Dr. A. A. Staklis (corrosion and compatibility studies).

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INTERNAL CONVECTIVE COOLING SYSTEMS FOR HYPERSONIC AIRCRAFT

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SUMMARY

Parametric studies were conducted to investigate the relative merits of construction materials, coolants and panel concepts for internal convective cooling systems applied to airframe structures of hypersonic aircraft. These parametric studies were then used as a means of comparing various cooled structural arrangements for hypersonic transport and a hypersonic research airplane. The cooled airplane studies emphasized weight aspects as related to the choice of materials, structural arrangements, and structural temperatures. Consideration was given to reliability and to fatigue and fracture aspects, as well.

Numerous candidate coolants were screened and coolant distribution systems sizing studies were conducted for the more promising coolants. Aqueous solutions, particularly ethylene glycol/water, were attractive for operation at temperatures below 394°K (250F). Operation at higher temperatures necessitated the use of new non-aqueous coolants whose lower specific heats required higher flow rates, but in some cases the pumping power penalty usually associated with a higher flow rate was compensated by a larger operating temperature range and the lower viscosity resulting from a higher maximum coolant temperature. Presently available coolants appear to be suitable for operation to 450°K (350F).

Because of the modest temperatures associated with actively cooled airframe structure the selection of construction materials closely parallels that for conventional aircraft structure except that consideration should be given to thermal conductivity as well as ratios of strength and stiffness to density. Beryllium and metal matrix composites are most attractive for minimizing weight in future applications while aluminum alloys are the most attractive of the materials commonly in use today for airframe structure.

Both tubular and plate-fin sandwich panels were attractive based on thermal design considerations. The former is attractive because of the relative ease of integration with the substructure; the latter is particularly well suited for regions of high heat flux. Both panel designs are adaptable to the incorporation of redundant coolant circuits. The choice of structural panel design was strongly influenced by the relatively low loading intensities associated with the types of hypersonic aircraft studied with highly swept delta wings and large fuselages required to house hydrogen fuel. Conventional stringer stiffened skin panels and honeycomb sandwich panels were attractive for fuselage and wing applications, respectively. Beaded skin designs showed lowest weight but the integration of coolant passages with such a structural configuration would pose significant problems.

Comparisons of non-redundant and redundant cooling systems indicated that the major weight increase associated with redundancy came from duplicating such items as the heat exchanger, the pump, and the coolant reservoir/accumulator; there was little weight penalty with respect to distribution

lines because of the design approach used. Minimization of the coolant distribution line weight for the redundant configuration was achieved by permitting half of the required flow through each of the redundant loops under normal operating conditions. With such a design it is necessary to increase coolant flow in the remaining loop in the event of a malfunction in one loop, or to accept higher operating temperatures after a malfunction.

For the Mach 6 hypersonic transport and the trajectory studied a surface temperature of about 544°K (575F) was required in order to have sufficient heat capacity in the fuel flow to absorb the heat loads for an unshielded airframe under all flight conditions including maneuvers. Operation at lower surface temperatures is possible if the heat load to the airframe is attenuated (by heat shielding or insulation) or if extra hydrogen is carried for cooling during specific flight conditions such as maneuvers. Combinations of these techniques were found to be attractive. Load factors near zero g are particularly adverse because the reduced drag reduces power requirements and fuel flow while heat loads are reduced to a lesser degree. For the configuration and trajectory studied, a structure operating at a maximum temperature of 394°K (250F) would require heat shielding and hydrogen flow in excess of fuel needs for maneuver involving load factors of between ± 0.8 g.

Even when auxiliary thermal protection system items, such as heat shielding, insulation, and excess hydrogen for cooling, are considered the more attractive actively cooled airframe concepts indicated potential payload increases of from 40% to over 100% as compared to the results of previous studies of the same vehicle configuration with an uncooled airframe. For the actively cooled hypersonic transport the use of a redundant cooling system reduced the indicated failure rate by two orders of magnitude, and added about 23% to the cooling system weight, (less than 0.7% of the aircraft gross weight).

Because of the nearer term aspects of a hypersonic research airplane as compared to the hypersonic transport, the active cooling studies for this application focused on aluminum alloy construction and aqueous coolants. Ethylene glycol/water was selected over methanol/water primarily on the basis of operational considerations of volatility, flammability, and toxicity. For the various cooled concepts examined weights of the airframe structure and total thermal protection system ranged from 30 to 40% of the launch weight. The smaller size of the research airplane necessitated fewer cooled panels and connectors than for the transport; this resulted in relatively higher reliability. Redundancy reduced the indicated failure rate by two orders of magnitude and added about 26% to the weight of the cooling system (about 2% of the aircraft gross weight).

INTRODUCTION

Preliminary studies in Ref 1-3 of a cooled structure for a hydrogen fueled Mach 6 transport indicated potential advantages of reduced thermal distortion and stresses, use of state-of-the-art materials and subsystems, and significant payload improvements when compared to hot structures. The cooling system found to be most attractive was an internal convective system which used a cooling fluid circulated through integral surface cooling passages to transfer the structural heat load to a centrally located hydrogen-fuel-cooled heat exchanger. A preliminary design for such a system was developed in Ref 3; however that study was limited in scope to two materials and coolants and a single cooled panel concept. Therefore, the purpose of this contract was to investigate a wide range of materials, coolants and panel concepts, such that optimum design concepts for cooled hypersonic structures could be more accurately defined.

An extensive survey of current and future airframe construction materials and coolants was conducted, so that the most promising candidates for cooled-panel, cooling-system and airframe concepts could be examined. Consideration was given to a wide range of structural materials, coolants, and structural panel concepts, several thermal panel concepts, and 3 cooled airframe design approaches, including unshielded, shielded, and dual temperature types. As an adjunct to the studies of materials and coolants,tests of corrosion potential and stress corrosion were carried out to investigate the compatibility of several promising structural materials and coolant fluids. The concept identification and parametric comparison phase of the study examined all major elements of the convectively cooled airframe, including the differing requirements at various locations on the aircraft.

The parametric results were used for the investigation of two separate vehicles, a hypersonic transport with a length of 96m (314 ft) and a weight of 240,000 kg (528,600 lb) and a hypersonic research airplane, with a length of 25m (80 ft) and a weight of 20,300 kg (44,700 lb). On the basis of NASA supplied trajectories, the heat loads and structural loads for both of these aircraft were predicted and used as baseline values for the comparative studies of different coolants and different cooling system concepts. In addition, consideration was given to cooling system concepts for selected regions of the vehicle, such as leading edges, integral tankage, and control surfaces. Fault hazard and reliability analyses were made to define critical cooling system components and to study the effects of redundancy on system weight and reliability.

Since the primary objective of this project was to compare materials and thermal/structural design approaches for cooled aircraft, emphasis was placed on cooling systems and primary load carrying airframe structure. Simplifying assumptions were used where it was felt that they would not influence comparisons seriously. Heat load calculations neglected control surface deflections and temperature variations over the vehicle surface. Approximations were used to account for secondary structural items such as leading edges, and control surfaces. With respect to the primary airframe structure, the analytical efforts were focused on the stiffened skin panels; allowances were made for frames, hard points, and non-optimum configurational features such as doors, based on prior experience, with an attempt to be conservative.

This report presents the highlights of the work performed during this contract. More complete details of the study are presented in Ref 4 which includes the results of the corrosion testing, in addition to the various structural and thermal analyses, design studies and discussion of concepts for selected regions.

All computations were performed in the English system of units and then converted to SI units.

VEHICLE DATA

Two specific vehicle configurations were studied, a hypersonic transport and a hypersonic research airplane. The 65° delta wing transport (Figure 1) had a takeoff gross weight of 236,000 kg (520,652 lb), a wing span of 32.9 m (108 ft) a wing area of approximately 650 m^2 (7000 ft^2) and a total wetted area of 3200 m^2 (34,000 ft^2). The Mach 6 cruise speed is reached at approximately 30,400 meters (100,000 ft) about 1400 sec after takeoff. By this time approximately 40% of the fuel has been used. The descent begins at 31,920 meters (105,000 ft) about 4500 sec after takeoff. Fuel flow is stopped when aerodynamic heating is no longer significant, at about 5700 sec after 80,000 kg (177,000 lb) of hydrogen fuel have been consumed. This particular vehicle was studied during the work reported in References 1 through 3 and Reference 5. As compared to the original trajectory of Reference 5, the transition from ascent to cruise was modified to reduce heating conditions thereby reducing the size of the cooling system, and a powered descent was started somewhat earlier to provide a fuel flow for cooling purposes during the descent.

For nominal and maneuver flight conditions heat loads for aircraft with and without heatshields were computed assuming three average structural temperatures 366°K (200F), 477°K (400 F) and 589°K (600F). This range was consistent with the desire to investigate a variety of construction materials. The data permitted comparisons with fuel flow so that the question of matching airplane cooling requirements and normal fuel flow scheduling could be considered.

The hypersonic research airplane shown in Figure 2 resulted from in-house studies at NASA and featured integral hydrogen tanks. The highly swept wings, with a span of 9.9m (32.5 ft) and an area of 38 m^2 (410 ft^2), have tip mounted fins and rudders. The fuselage is 26.4m (80 ft) long with a wetted area of 196 m^2 (2110 ft^2). The total wetted area of the research airplane is 305 m^2 (3290 ft^2), approximately 1/10 that of the transport. The takeoff gross weight is approximately 20,000 kg (43,875 lb). Flight trajectories with cruise speeds of Mach 8 and 10 were used to calculate heat loads corresponding to nominal and maneuver conditions. Minimum and maximum load factors were -1.0 g and +3.0 g. Heat loads were computed for three average structural temperatures, 366°K (200F), 477°K (400 F), and 588°K (600 F). The structural loadings included landing, taxiing, captive and flight conditions. In addition to the design static loads defined in the conventional manner a loading spectrum was established, based on fighter, fighter-bomber and trainer aircraft experience, to permit the analysis of fatigue and fracture characteristics.

By defining the local heating intensities, integrated heat loads, and structural loadings for these two different airplanes, it was possible to establish bounds for the parametric analysis. In addition to establishing numerical levels,the ranges of magnitudes of heating and loading intensities provided guidance as to the thermal and structural concepts likely to be of interest.

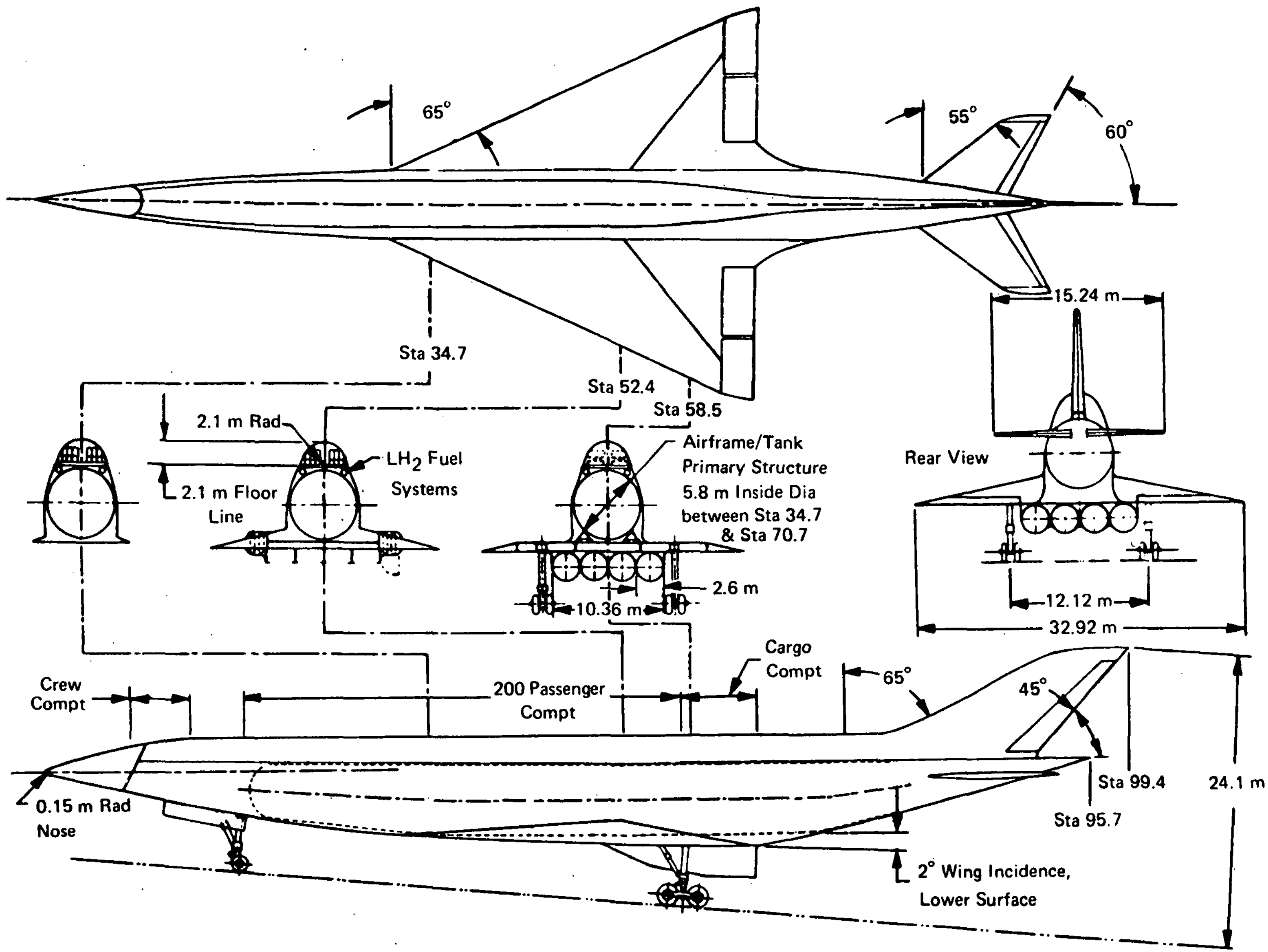


Figure 1. Delta Wing Configuration (from Reference 4)

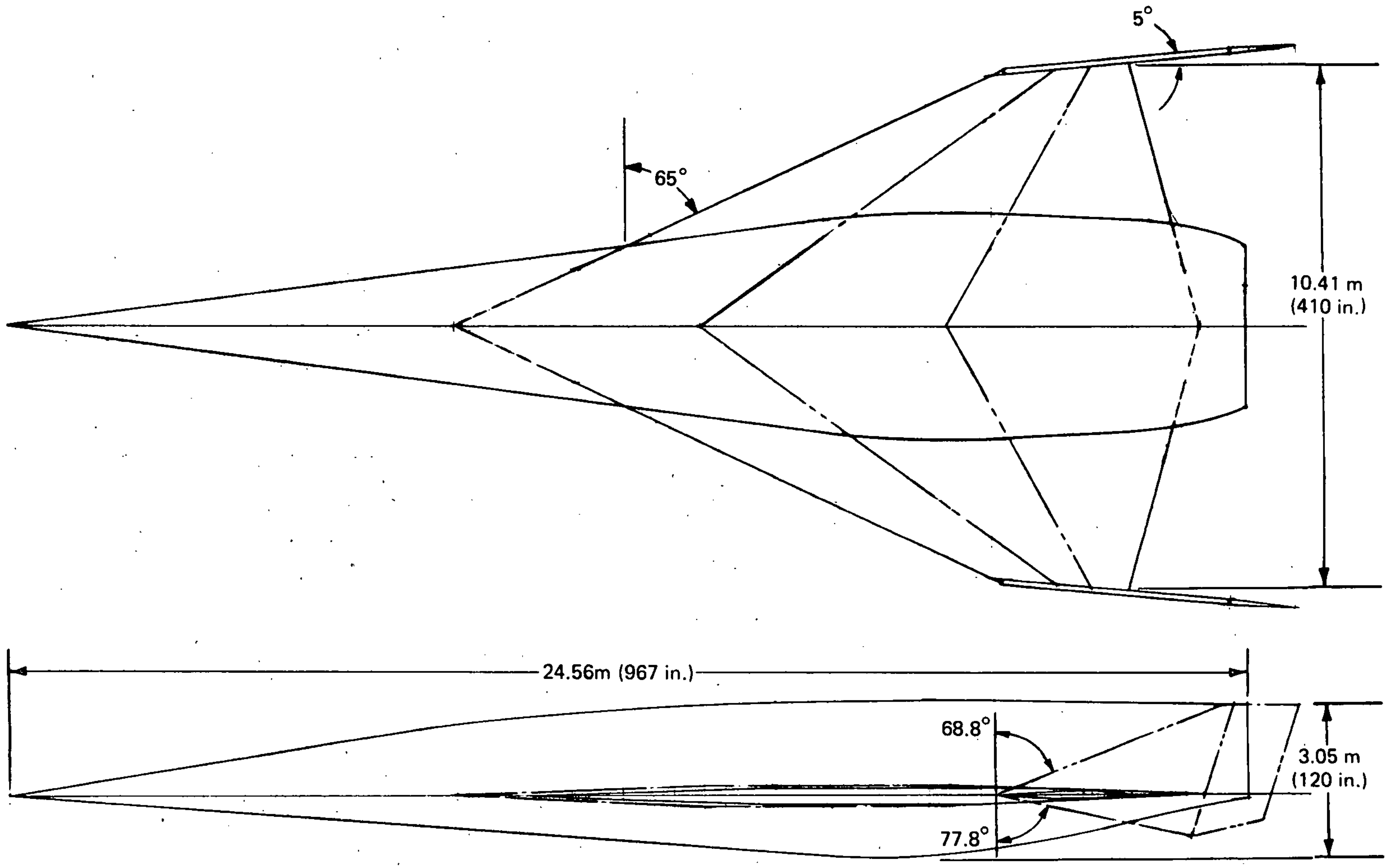


Figure 2. Hypersonic Research Aircraft Configuration

PARAMETRIC STUDIES

Material Selection

Prior studies of cooled airframe structures emphasized ethelyne glycol/water as the coolant that absorbed aerodynamic heat input from the structure and transported it to a heat exchanger where it was rejected to the hydrogen fuel. One of the objectives of the study was to investigate the relative merits of other coolants. Over 50 candidates were screened and more detailed sizing analyses were conducted for the 17 more promising candidates. Comparisons utilizing the better of the coolants in each of the several generic classes was dictated by the desire for a broad data base that would be useful after corrosion data was obtained for candidate construction material and coolant combinations. As a result of these screening analyses coolants from 3 classes were selected for two temperature ranges. For operation at temperatures below 394°K (250F), ethylene glycol/water was the aqueous solution chosen, FC-43 was the prefluorinated coolant, and Coolanol 45 was the representative silicate ester. For operation at maximum outlet temperatures in the 450°K (350F) range a silicate ester, (Coolanol 45), a silicone (Dow Corning XF-1-3755), and a prefluorinated type (Freon E-5) were selected. It should be noted that the choice of specific coolants was based on the desire to include generically different types and to provide system weight and performance estimates that would be conservative. For example, at temperatures below 366°K (200F) Coolanol 15 is preferable to Coolanol 45 and in a temperature range to 450°K (350F) Freon E-3 is superior to E-5. The more conservative choices were made to provide a slightly pessimistic weight estimate that could be reduced as more refined studies dictated requirements for specific applications. For the research airplane detailed comparisons were provided for ethylene glycol/water, methanol/water, Coolanol 20, and Coolanol 40.

The consideration of construction materials for the hypersonic transport recognized the fact that this vehicle is unlikely to be constructed until about the turn of the century, and as a result, present day developmental materials may be commonplace. However, the comparisons did not project property improvements; presently available property data was used to be conservative. Over 100 specific structural materials were reviewed and the 33 more applicable were compared in greater detail. This comparison led to initial parameteric analysis of structural efficiencies for fifteen materials assuming a variety of structural configurations. The parametric results indicated relatively small differences within a material class but large differences between classes. Therefore, subsequent structural concept comparisons considered materials that were representative of a class rather than the best material from each class. The selected material classes included alloys of aluminum, magnesium, titanium, steel, superalloy, beryllium and metal matrix composites.

In addition to an interest in coolants and construction materials in their separate functions related to the cooling system and the airframe, their chemical compatibility is also important. Since little definitive data was available regarding compatibility of the coolants and alloys of interest, corrosion and stress-corrosion testing was conducted at elevated temperatures with several materials and coolants. Within the scope of the studies, no problem was indicated in finding compatible combinations of coolants and construction materials. The non-aqueous coolants were quite inert. While some aqueous solutions attacked particular construction material more readily than others, it appears possible to select a particular aqueous solution that would be satisfactory for long life at temperatures up to 366°K (200F) for all construction materials tested, except magnesium. Since no attempt was made to precoat the metal specimens or to use special inhibitors for the various metals (standard coolants were used), it may be possible to develop a satisfactory system for magnesium.

Panel Concepts - Thermal

Four basic skin panel thermal design concepts were considered including: 1) tubular, 2) plate-fin sandwich, 3) sphere-core sandwich, and 4) plain skin/cooled substructure. These are illustrated in Figure 3 which shows some of the variations of each concept. The tubular design minimizes the coolant retained in the passage network while the space between discrete passages facilitates structure assembly and integration with the substructure. The specific tubular arrangement used will depend upon the particular material of construction and the types of joining processes that are most appropriate for the construction material. The formed skin approach is relatively simple but high peel stresses can exist at the joint between the sheets and it is necessary that the construction be highly efficient and compatible with the coolant. Installation of tubing onto the structural skin separates the structural efficiency and chemical compatibility considerations but necessitates metallurgical joining which might limit or restrict material choices. By sandwiching the coolant passage tubing between two formed skins, the advantages of both concepts can be achieved while the high stresses in the joints near the coolant passage are reduced significantly. For adhesively bonded aluminum alloy tubular panels in-house experimental evaluations demonstrated an increase in pressurization capability of nearly an order of magnitude when the sandwiched tube design was used. This concept also appears to be well suited to incorporation of crack arrestors in the form of wires or filaments adjacent to the tubes which should enhance the damage-tolerance of the cooled panel.

To provide redundancy with the tubular concept two separate cooling systems with independent coolant passages can be used. The two sets of passages can be spaced alternately (a and b of Concept A2, Figure 3) or located together in a divided passageway (a and b of Concept A3). In the alternately spaced arrangement there is some separation of the two sets of passages in case of localized damage to the skin; the side-by-side arrangement results in fewer passageways with wider spacing. Under operating conditions 50% of the total requirement circulates through each of the adjacent networks. If one of the redundant networks should fail, the flow in the other is doubled. Even if two malfunctions occur such that the flow of coolant in the remaining loop is not increased maximum structural temperatures will not reach catastrophic levels, although maneuver capability will be reduced and local permanent deformation may be experienced.

The stacked plate-fin concept, B3, can be used to obtain redundancy with this concept. However, it suffers a weight penalty as a result of the center sheet if the additional material is not required for structural purposes. Since temperature gradients between coolant passages are eliminated with the sandwich panels the allowable temperature rise in the coolant can be higher for a specified maximum structural temperature than is possible for the tubular panel concept and for relatively high heating conditions the cooling system weight is less. Therefore the coolant flowrate is less than for the tubular skin panel designs. Of the cooled panel concepts considered, the plain skin/cooled stringer design appears most tolerant of skin cracks that might occur in service but because of limitations due to stringer size and thermal resistance at the skin-stringer joint, the concept can be used at only low heat fluxes, (less than 2 w/cm^2 (2 BTU/ft^2 sec) for aluminum and beryllium material, and even lower levels with materials of lower conductivity). Metallurgical joining will reduce the thermal resistance but damage tolerance is likely to be reduced as well.

As compared to the plate-fin concept, the sphere-core results in higher weights due to higher pressure drop and a greater coolant content despite the fact that the hollow spheres constitute about 50% of the volume between the face skins. However, the sphere-core concept is more adaptable to

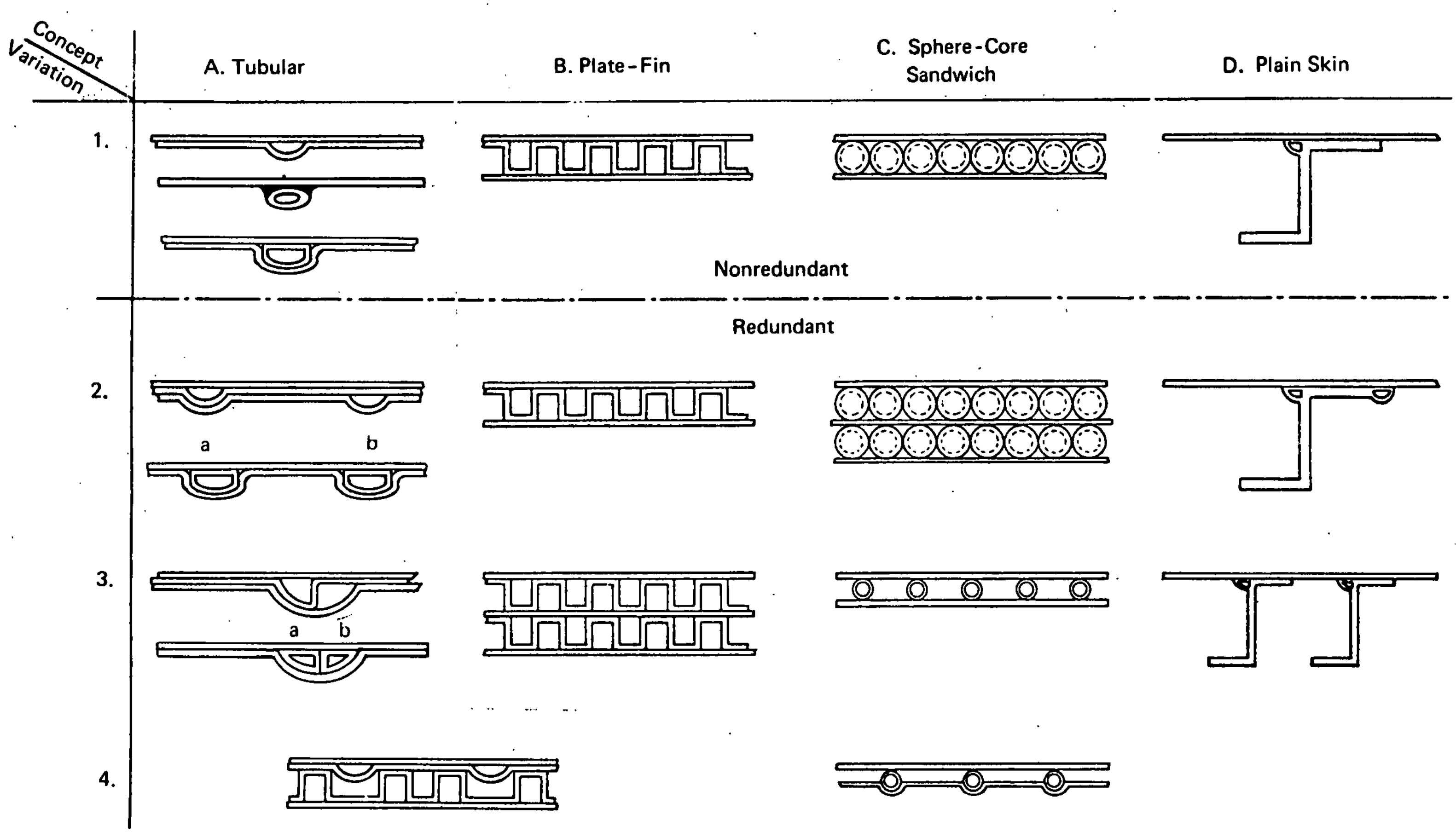


Figure 3. Nonredundant and Redundant Skin Panel Concepts

structures involving significant double curvature such as nose caps. In some instances the added thickness of the sphere-core panel may be advantageous from a structural point of view by increasing the spacing between substructural elements thereby simplifying the construction which tends to reduce costs. However, for the two hypersonic aircraft considered, structural loading intensities were low and very little weight is saved by increasing the distance between stiffening members. Both sandwich concepts present problems with respect to accommodating mechanical fasteners, cutouts, and small doors. Where large doors are involved they can be treated like separate panels. Other potential disadvantages include sensitivity to skin cracks that induce leakage, and the relatively long heat flow path associated with the stacked configurations in the event of a malfunction of the outer cooling loop. If brazing is used for assembly purposes,the choice of construction alloy is limited to candidates whose strength properties are not as high as some of the more conventional airframe structural alloy that are not brazeable or weldable.

Based on the thermal analyses conducted, it appears desirable to utilize plate-fin construction in stagnation regions and to use the tubular panel designs for the major portion of the airframe. The heat flux at which a transition should be made depends upon the particular application of interest but is expected to range between 22 and 45 w/cm^2 (20 and 40 BTU/ft^2 sec).

In addition to design variations applicable to the cooling of the skin, it is possible to consider techniques for shielding the skin panel from direct contact with the hot boundary layer thereby attenuating the heat load that must be absorbed by the cooling system. Three approaches were considered: 1) a relatively dense ceramic spray coat, 2) metallic heating shields, 3) resuable surface insulation of the type being developed for the space shuttle. For practical ceramic coating densities, the weight of the coating was greater than the savings in cooling system weight due to heat load attenuation. Metallic heat shields without intermediate insulation are promising for both aircraft types. The RSI concept was attractive for the research airplane, but since it does not appear to be practical for the extended, all-weather operation of a hypersonic transport because of its fragility and consequent high replacement cost, it cannot be recommended as a prime candidate for the HRA. Rather, it should be considered as a means of extending the HRA flight speed or other parameters that increase heating beyond those operating conditions for which verification of an unprotected cooled aircraft is to be demonstrated for subsequent transport application.

Panel Concepts - Structural

Prior studies of cooled airframe structure utilized concentional construction, skin/stinger/frame for the fuselage and skin/stringer/spar and rib for the wing. The promising potential of these earlier cooled airframe studies warranted more detailed examination of structural concepts to define a more nearly optimum structure. On the basis of structural efficiency and relative complexity, six internal stiffening concepts were chosen for study as applied to the fuselage: 1) ring-stiffened monocoque, 2) sandwich monocoque, 3) skin/stringer/frame, 4) ring and corrugated skin, 5) ring and symmetrical double-beaded skin, and 6) ring and unsymmetrical double-beaded skin. Wing cover candidate types compared in a preliminary manner as wide column included: 1) honeycomb sandwich, 2) stringer stiffened, 3) symmetrical double-beaded and 4) unsymmetrical double-beaded. Representative arrangements are illustrated in Figure 4.

Analyses of candidate constructions were conducted for the range of structural loading intensities of interest for the two airplanes being studied, assuming representative members of the seven

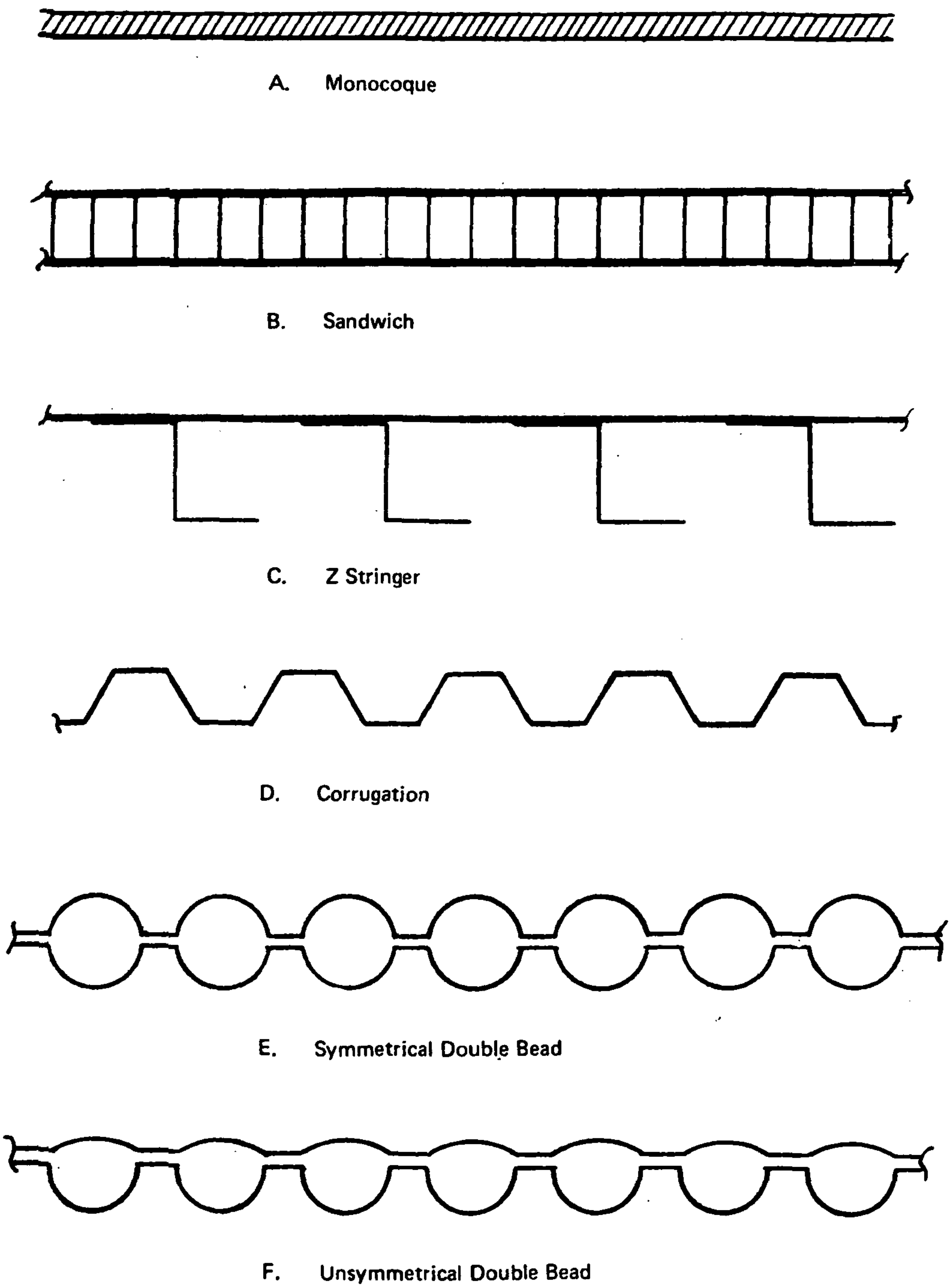


Figure 4. Candidate Constructions

promising material classes, to permit the computation of airframe weights. Initially minimum gage constraints were not considered but were added later when the panel data was integrated to determine airframe weights. For a particular construction material the choice of the structural panel concept resulted in weight differences of between 50% and 100% from the lightest panel concept to the heaviest over the range of loading intensities of interest. Since minimum gage constraints were not imposed, there was little change in the ratio of maximum to minimum weight as the loading intensity was varied. While the lightest structural weights were associated with the symmetrical double-beaded skin the incorporation of coolant passages with such a design would be very difficult. The relatively conventional skin/stringer/frame and honeycomb sandwich approaches were most attractive for fuselages and wings respectively. The parametric studies showed the significant advantage of beryllium and metal matrix composites in reducing skin panel weights. Aluminum was generally more attractive than magnesium or titanium. High density alloys of steel and nickel were not attractive for the skin panels but might be useful for fittings, cryogenic tanks, and heat shields.

A primary purpose of actively cooling the airframe structure is to reduce temperature levels such that conventional construction material can be used and thermal stresses can be minimized. In the case of the tubular coolant passage designs heat is removed along discrete lines while the heat input is applied uniformily. This generally results in modest gradients normal to the passages with thermal stresses which are likely to range between 10 and 20% of the allowable yield strength of the construction material. While such stress levels must be considered in design they are not particularly high so that they are not a primary design parameter, per se. However, local areas of higher temperature gradients and thermal stress could occur near edge attachments, inserts, manifolds, etc. with either tubular or sandwich panel design and must be identified by detailed thermal analyses.

Cooling System Concepts

Although the scope of the project was limited to internal convective cooling systems several variations are possible within this single type. Redundant concepts were examined for both aircraft on the basis of the cooling system arrangements shown in Figures 5 and 6 and were found to increase system reliability very significantly with modest increases in weight. (Weight and reliability results for each aircraft are discussed later.) The redundant concept studied in most detail consists of two independent cooling systems each with a heat exchanger and a pump designed for the full heat load, but with distribution lines and panel coolant passages sizes optimized for one-half the required coolant flow rate. In normal operation, each system carries half the heat load. In case of failure of one system, the coolant flow rate is increased in the other to accommodate the full heat load. Separate identity of each cooling system is provided but the close proximity of coolant passages in the skin panels (such that an incident likely to damage one system may damage the other) raises a question as to whether the systems are truly redundant.

In addition to systems based on a single construction material and a single coolant, dual temperature convective cooling systems employing different coolants and materials in different areas were examined as a means of enhancing the cooling capacity of the system. Since the stagnation regions are likely to have a different type of structure than that used for the major proportion of the aircraft, it might be practical to use different materials and coolants in stagnation and non-stagnation regions. Systems of this type appeared to offer significant advantages by reducing the amount of shielding needed to match aerodynamic heat absorption and fuel flow characteristics for the lower temperature airframe designs.

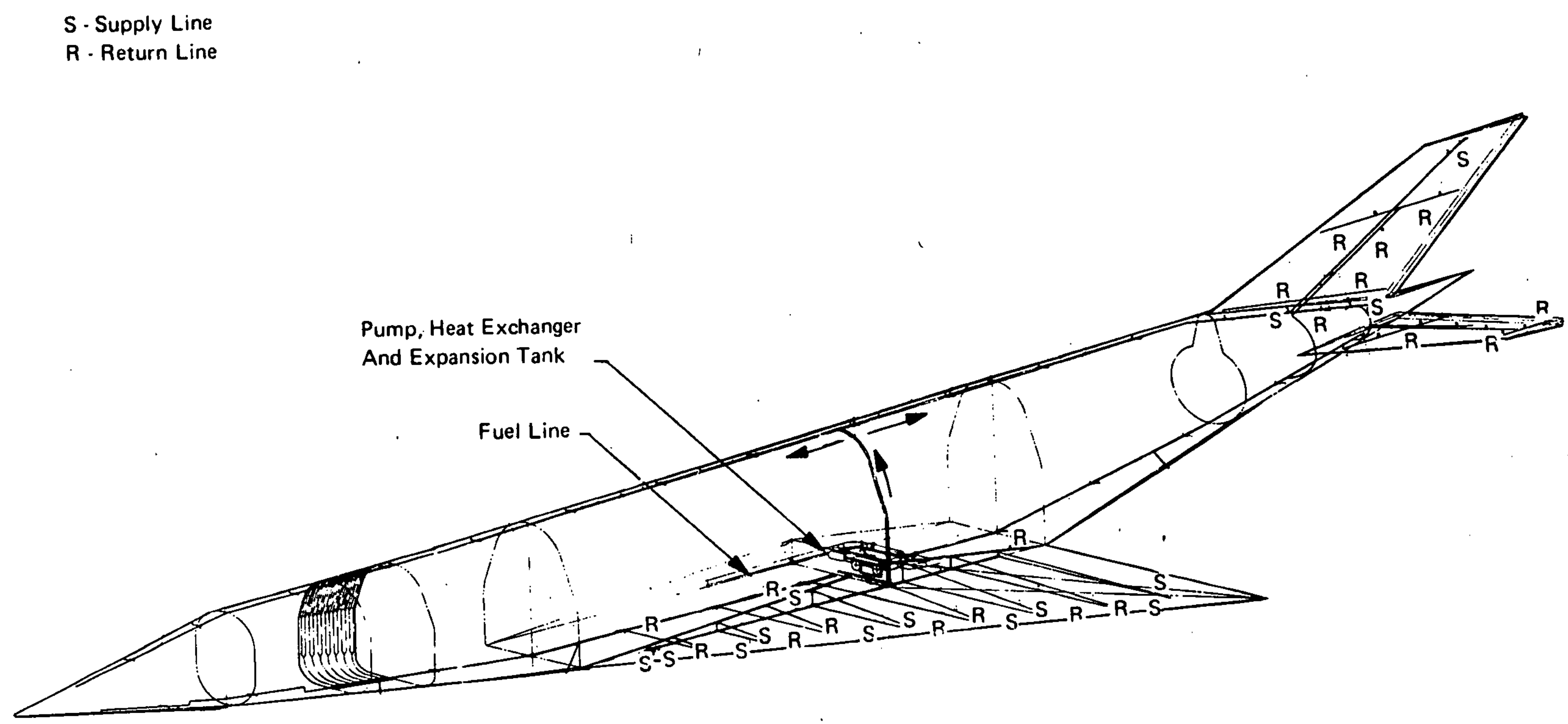


Figure 5. Distribution System Layout, Hypersonic Transport

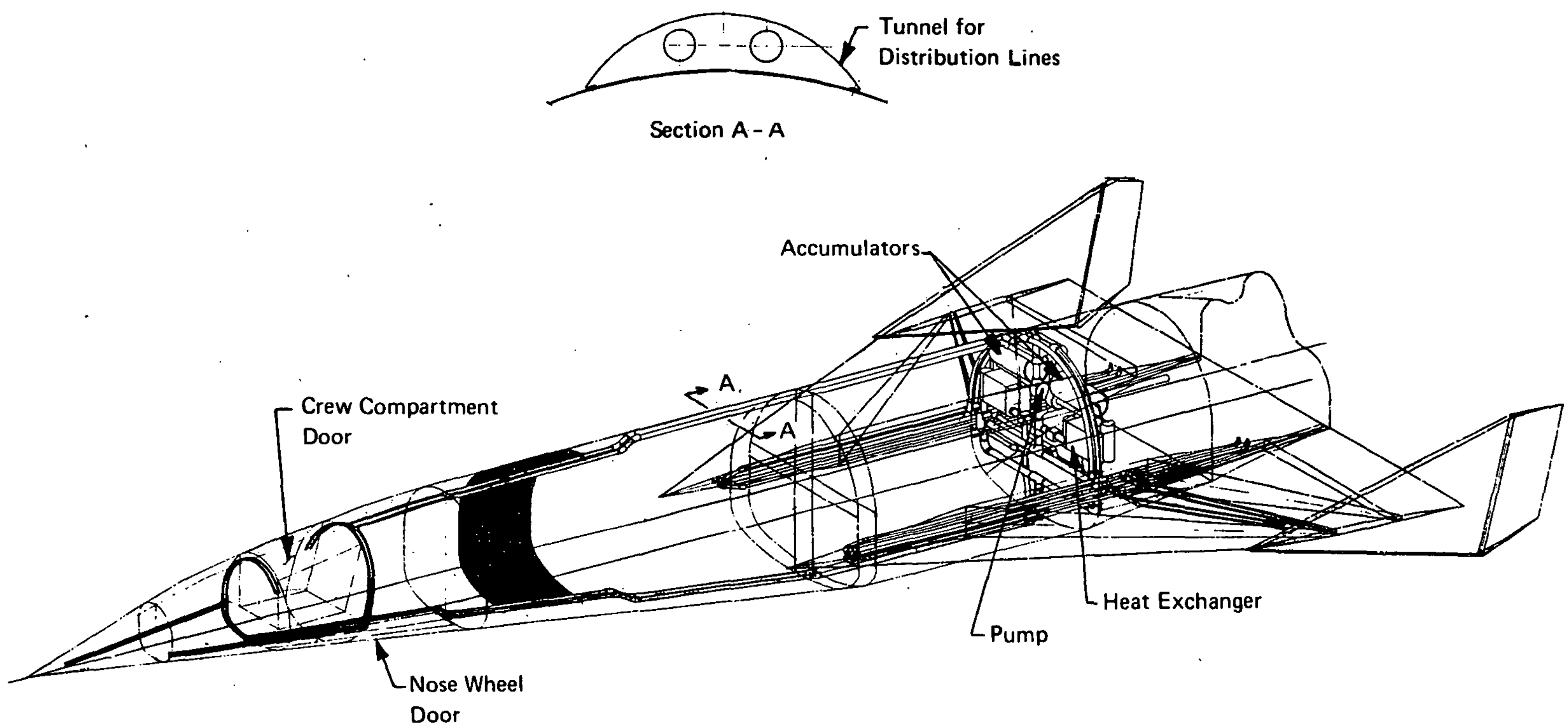


Figure 6. Distribution System Layout, Hypersonic Research Aircraft

As part of the cooling system design studies parametric anlayses were conducted to define characteristics, particularly weights, of the cooling system distribution lines, heat exchangers, pump, and panel elements for the various coolants considered.

VEHICLE STUDIES

The parametric studies summarized in the preceding discussions provided data for evaluating various tradeoffs for cooled airframe structure and cooling systems for the two aircraft of interest. This discussion emphasizes structural concepts and materials that would allow operating temperature to 589°K (600F), refinement of weights for convective cooling systems that employ a variety of coolant types, and the integration of structural concept/material/panel design with the cooling systems for various operating temperatures to obtain total airframe systems weights.

In reviewing the application of the parametric studies to the two airplanes of interest, it is desirable to keep in mind the comparative nature of the effort and the use of various simplifying assumptions. It is felt that these assumptions have little influence on comparisons that are made among the systems considered but may influence the absolute magnitude of the weights discussed.

Hypersonic Transport

Promising Structural Concepts - As an aid in clarifying the relative merits of candidate approaches, minimum skin gages were defined for candidate materials and various forms of construction to obtain minimum equivalent thicknesses. In turn, these were incorporated with the parametric sizing results so that integration provided the weight of the covers for the fuselage, to which were added a weight increment for frames and a weight estimate for the passenger compartment floor to obtain a subtotal for the fuselage weight. Weight estimates for the wings were obtained in a similar manner. Unit tail weights were assumed to be the same as for the wing. Nonoptimum weight allowances are included to account for doors, inability to taper stringers, practical constraints on stiffener spacing, etc; the allowance was 15% for sandwich construction and 10% for other constructions. No weight allowance was included for major concentrated load points such as landing gear, wing, and tail attachments. It is expected that the weights of such items will be influenced more by the material of construction than by the type of construction and might add about 5% to the weight of this airframe structure.

Aluminum alloys were superior to magnesium or titanium for near term applications, from 1360 to 2720 kg (3,000 to 6,000 lb) for sandwich construction and about 1820 kg (4,000 lb) for skin/stringer/frame construction. The use of beryllium or boron/aluminum would provide a very substantial reduction in fuselage weight, about 4540 kg (10,000 lb) which is about 20% of the aircraft payload. However, it should be noted that these trends are based on structural consideration and will be combined with thermal design considerations later. As in the case of the fuselage, aluminum alloy construction appears to be more desirable than other conventional materials for the wing. Significant weight benefits are possible through the use of beryllium or boron/aluminum, although the magnitude of the benefit depends on the type of construction employed.

Total weights for the airframe structure are summarized in Figure 7 based on an assumed structural temperature of 315°K (100F). The combination of a skin/stringer/frame fuselage, combined with a sandwich construction wing yields lightest weight for any particular material except beryllium for which a stringer-stiffened wing is slightly lighter. The second most promising combination depends upon the particular construction material. Based on these lightest weight combinations, and taking the aluminum alloy construction as a reference, the use of titanium would add about 8% or 2950 kg (6500 lb), the use of boron/aluminum would reduce airframe weight by 14%, 5000 kg (11,000 lb), while beryllium would reduce weight by 23% or 8,600 kg (19,000 lb). The weight

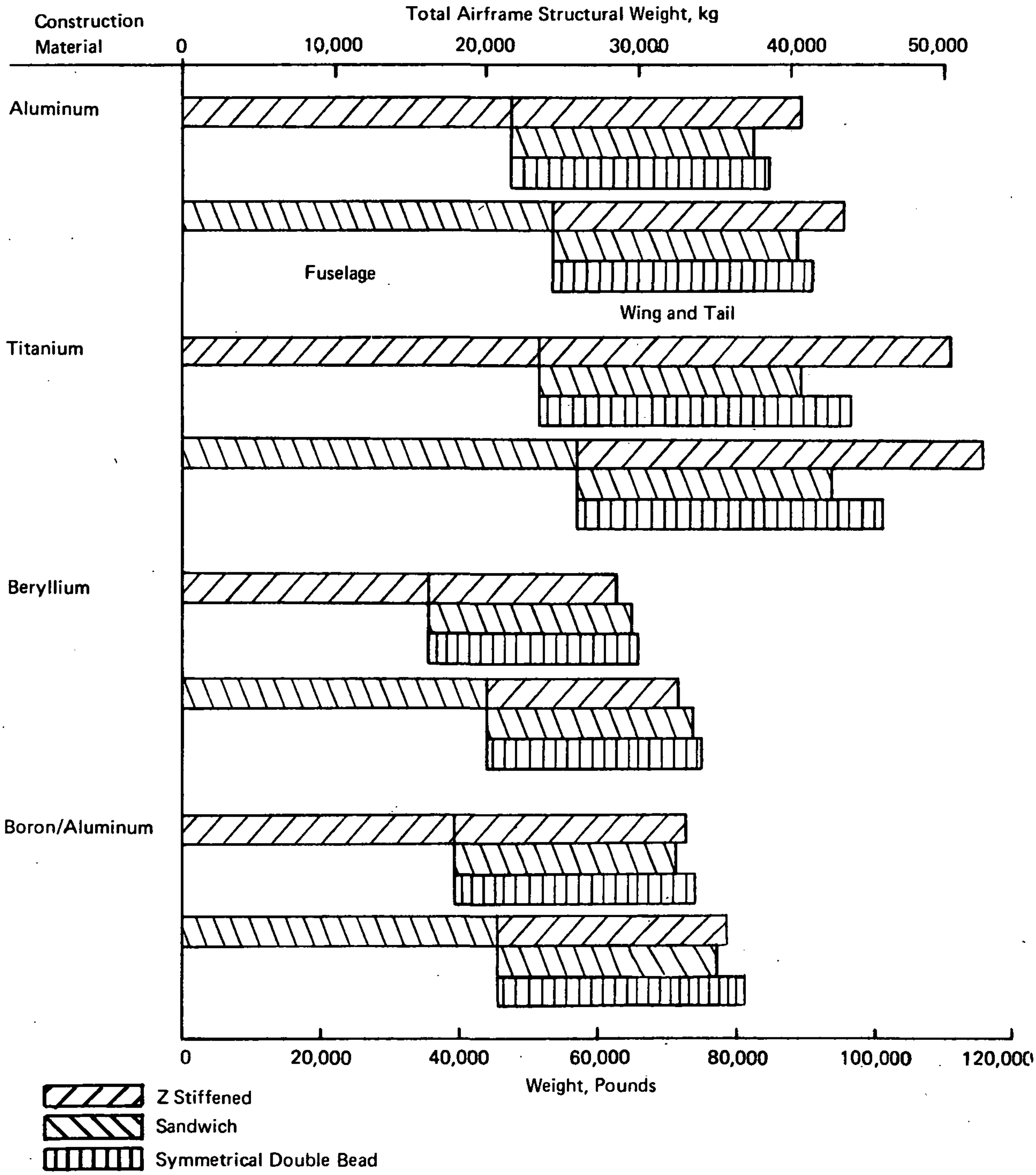


Figure 7. HST Airframe Structural Weight, 100F Reference Temperature

saving potential of the advanced materials is quite significant in view of the 21,800 kg (48,000 lb) payload of the baseline uncooled airplane.

Promising Thermal Concepts - According to Reference 3, the most promising cooled airframe approach for the hypersonic transport, using essentially current structures technology, was aluminum alloy construction partially protected by superalloy heat shields to reduce the heat load to the glycol/water cooling system to acceptable levels with respect to aircraft fuel flow. While the use of a single conventional construction material and a single cooling system simplifies fabrication and development efforts, it does not necessarily lead to an optimum aircraft. For example, when low cost aluminum alloy construction is used, heat shields are required, whereas, if the maximum coolant temperature is increased, it should be possible to eliminate heat shielding. As an aid in establishing trends, heat load data was assembled in the form of the ratio of hydrogen required for airframe cooling, divided by the hydrogen required for thrust, and studied in relation to load factor at the most critical times during cruise and descent for shielded and unshielded transports with assumed wall temperatures of 366°K (200F), 478°K (400F), and 589°K (600F). The heat shields were assumed to cover those portions of the aircraft where the equilibrium radiation temperature exceeds 811°K (1000F), approximately 1/3 of the wetted surface.

For the shielded vehicle, the hydrogen fuel flow is more than required for cooling purposes when the wall temperature is 588°K (600F) and is sufficient for all flight conditions when the wall temperature is 477°K (400F). For the 366°K (200F) wall the fuel flow is sufficient for nominal flight conditions and all maneuver conditions except those between about ±0.8g. For the unshielded vehicle the fuel flow is adequate for cooling during all flight conditions if the wall temperature is above 544°K (575F). For a 477°K (400F) wall it is adequate for nominal flight conditions and for manuever conditions except those near zero-g where power requirements and fuel flow are reduced to levels below structural cooling needs. (About 34 kg (75 lb) of excess hydrogen would be required during a zero g maneuver lasting 10 sec.) The situation becomes substantially worse as the airframe wall temperature is decreased further. With an unshielded 366°K (200 F) vehicle, fuel flow is inadequate for cooling at nominal flight conditions and a zero g manuever lasting 10 seconds would require more than 115 kg (250 lb) of excess hydrogen.

Figure 8 relates coolant temperature and the extent of metal heat shields required so that all cooling system heat loads can be absorbed by the fuel flow without exceeding specified temperatures. Coolant temperature is plotted rather than structural temperature because the latter depends upon the type of cooled skin panel used and its material of construction. If the discrete tubular panel design and materials of high thermal conductivity, like aluminum or beryllium, are used for the structure. the maximum structural temperature would be about 28 to 56°K (50 to 100F) higher than the outlet coolant temperature indicated. If the construction material has low thermal conductivity, like titanium, the maximum structural temperature is likely to be from 56 to 112°K (100 to 200F) higher than the coolant temperature. Use of the plate-fin concept would practically eliminate thermal conductivity effects so that maximum structural temperatures would be only 14 to 28°K (25 to 50F) higher than the outlet coolant temperature indicated on Figure 8. As the coolant temperature is allowed to increase, more heat can be rejected to the hydrogen fuel and less shielding is needed. With about 1/3 of the airfrace covered with shields, 930 m^2 (10,000 ft^2) a coolant temperature of 366°K (200F) is required; the maximum structural temperature of an aluminum structure will be about 394°K (250F).

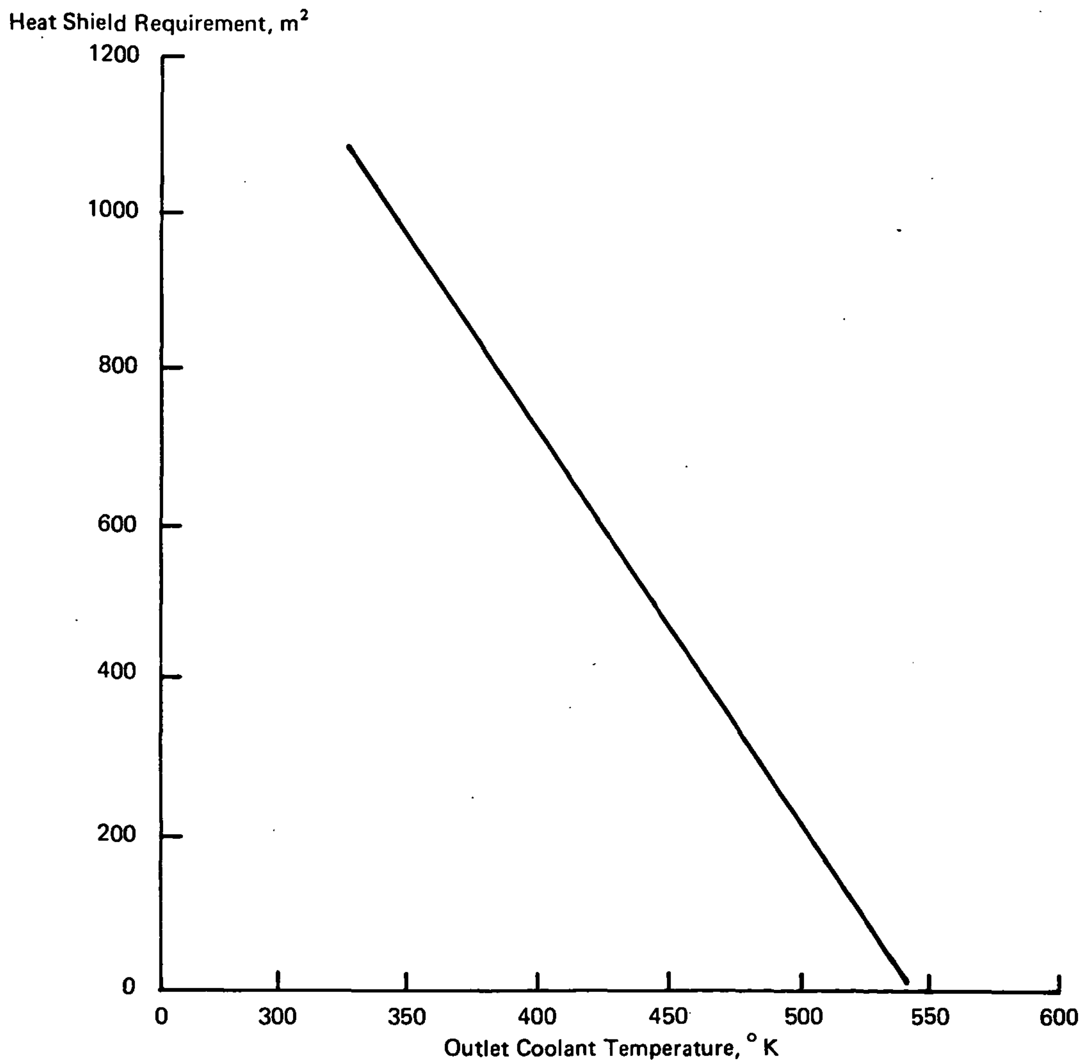


Figure 8a. Heat Shield Requirement as a Function of Coolant Temperature, Hypersonic Transport

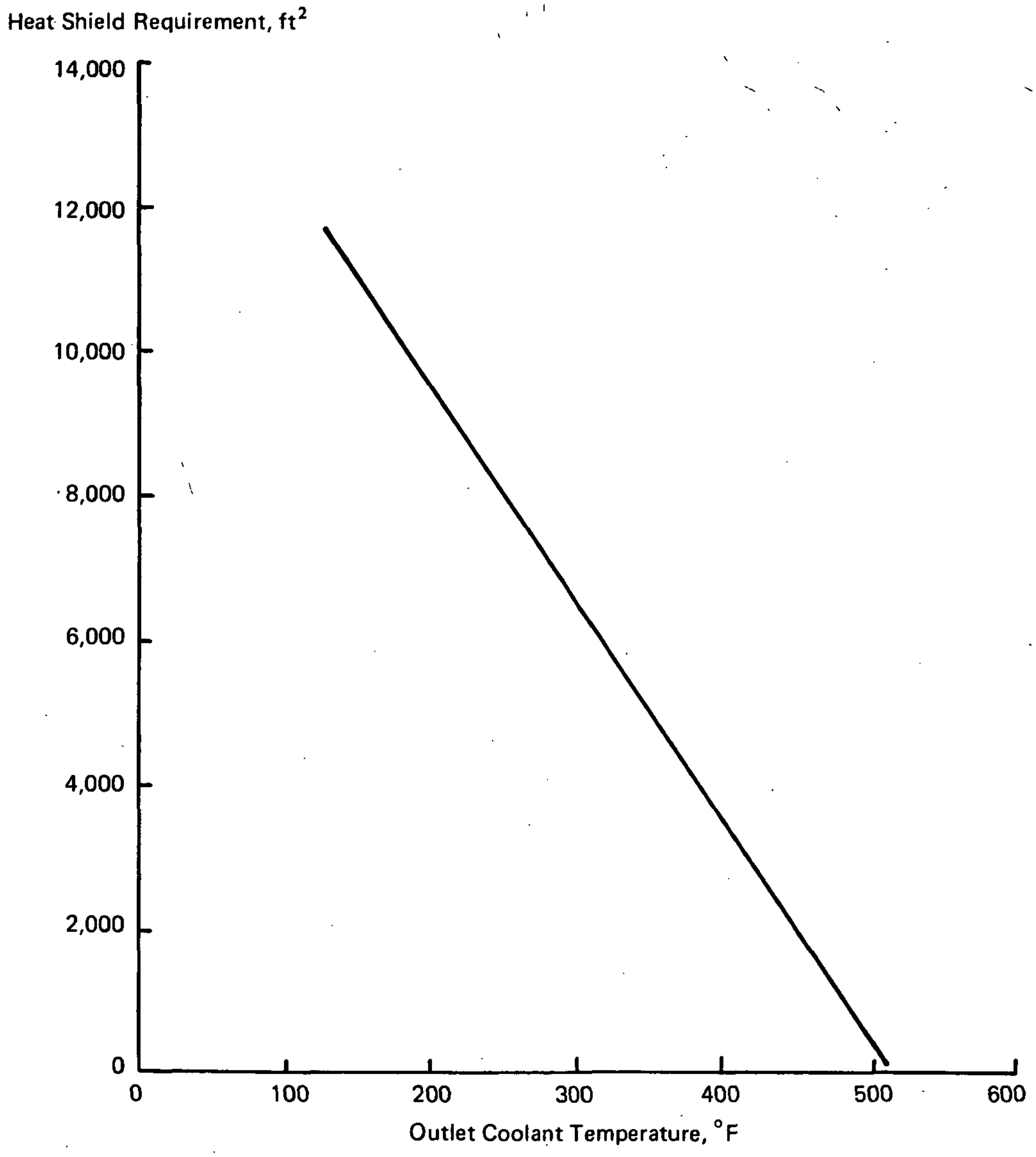


Figure 8b. Heat Shield Requirement as a Function of Coolant Temperature, Hypersonic Transport

A prohibitive increase in hydrogen quantity would be required for an unshielded 366°K (200F) structure during the cruise and descent portions of flight, about 10,500 kg (23,000 lb) compared to the approximately 4,500 kg (10,000 lb) of heat shielding which is partially compensated for by a reduction in cooling system weight as indicated by References 2 and 3. When the average wall temperature is increased to about 478°K (400F), there is no need for shielding except for maneuvers that induce load factors between about ±0.6g; maneuvers within this range are infrequent and could be accommodated by carrying about 227 kg (500 lb) of hydrogen for cooling purposes; although controlling the flow of this excess fuel to match heat load requirements may be difficult.

In considering cooling system weights, both shielded and unshielded structures of different average operating temperatures should be compared. Heat load data and parametric results were used to assemble such comparative data for the system of Figure 5 as shown in Table I. Included in the comparison is a dual temperature cooling system concept that utilizes an aluminum alloy structure in all areas but those of leading edges, where a separate higher temperature cooling system is used to permit a maximum transport coolant temperature of 533°K (500F) so that the maximum hydrogen temperature can be increased along with the quantity of heat that can be absorbed from the airframe. The higher temperature system utilizes beryllium or boron/aluminum structure and covers the first 1.5m (5 ft) of the wing surface (top and bottom as measured perpendicular to the leading edge), the first 0.6m (2 ft) of the horizontal and vertical tails, and a small portion of the fuselage nose.

Both nonredundant and completely redundant system concepts are indicated in Table I; such small differences as 865 to 1590 kg (1900 to 3500 lb) result from the assumption of 50% flow in each of the redundant loops under normal operating condition. Regardless of whether a nonredundant or redundant system is used, the ranking of the cooling systems concepts is unchanged (not considering weights of shields or excess hydrogen). The unshielded 589°K (600F) and the shielded 366°K (200F) systems are essentially comparable in weight followed by the shielded dual temperature system, the shielded 478°K (400F) system and the unshielded dual temperature system which are of comparable weight, and the unshielded systems for 478°K (400F) and 366°K (200F), which are essentially the same in weight. Note that the lowest ranking systems are about 50% heavier than the highest ranking systems. It is also interesting to note the significant reduction in cooling system weight between the nonredundant shielded 366°K (200F) system presented in Table I, 4750 kg (10,640 lb), and the weight projected for the same type of system in Reference 3, 5900 kg (13,000 lb). This weight reduction resulted from a slight modification of the end of ascent, a more accurate optimization of cooling system distribution lines and a more detailed passage sizing in various panels over the airframe structure.

While the cooling system weights (which are directly related to the design heat loads) are of interest by themselves, the weights of associated heat shielding and/or excess hydrogen which are listed in the lower part of Table I must also be considered in the comparison of cooling system concepts. The 366°K (200F) shielded airframe concept assumes that all regions of the aircraft that would have radiation equilibrium temperatures in excess of 810°K (1000F) are protected by metallic heat shields with only a modest interchange of radiant energy between these shields and the cooled structure that they protect from the hot boundary layer. This represents about 1/3 of the total wetted area of the hypersonic transport, as shown in Figure 9. The 478°K (400F) shielded concept is somewhat similar to the system just described, but shielding is eliminated from the lower surface of the wing and from the horizontal and vertical tails leaving only that portion of the fuselage forward of the wing, about 15% of the total wetted area. The shielding employed for the dual temperature 366°K/589°K (200F/600F) shielded concept is the same as the 477°K (400F) shielded concept. For the two lower temperature unshielded airframes the amount of hydrogen shown corresponds to

TABLE IA
COMPARISON OF COOLING SYSTEM CONCEPTS AND ASSOCIATED HEAT SHIELD OR EXCESS HYDROGEN REQUIREMENTS, HYPERSONIC TRANSPORT(9)

Item	Weights For Cooling System For Average Airframe Temperature Indicated, Kilograms						
	Unshielded			Shielded		Dual Temp	
	366°K	477°K (6)	588°K (6)	366°K	477°K (6)	Unshielded(4)	Shielded(5)
Design Heat Load, Mw	104.5	88.3	69	68	82	92.6	73.5
NONREDUNDANT							
Distribution Lines(1)	3,746	3,632	2,592	2,588	2,860	3,405	2,951
Heat Exchanger	944	636	477	613	577	863	749
Pump	120	114	79	73	82	120	120
Panel Residual(2)	1,503	1,934;2,293	1,112;1,317	1,117	1,430;1,698	1,335	1,044
Miscellaneous(3)	633	631; 667	416; 447	440	499; 522	574	488
Total	6,946	6,946;7,341	4,676;4,912	4,831	5,448;5,739	6,297	5,353
REDUNDANT							
Distribution Lines(1)	4,122	3,995	2,851	2,847	3,146	3,746	3,246
Heat Exchangers, Two	1,889	1,271	953	1,226	1,153	1,589	1,362
Pumps, Two	240	227	159	145	163	240	222
Panel Residual(2)	1,503	1,934;2,293	1,112;1,317	1,117	1,430;1,698	1,335	1,044
Miscellaneous(3)	772	774; 781	508; 530	422	586; 613	690	586
Subtotal	8,526	8,172;8,567	5,584;5,811	5,684	6,433;6,683	7,600	6,460
Heat Shields	0	0	0	4,540	1,270	0	1,270
Excess Hydrogen(7)	10,440	230	0	182	0	3,180	0
Total(8)	18,966	8,402;8,797	5,584;5,811	10,406	7,703;7,953	10,780	7,730

(1) Including piping, contained coolant, and APS fuel to drive pump (3 Step Flowrate Schedule as in Ref. 3)
(2) Redundant and nonredundant entries are the same because half flow is in each of the redundant sets of passages
(3) 10% to account for valves, controls, connectors, supports, etc.
(4) Aluminum/Beryllium Structure, Glycol/Water and Coolanol 45; 366/588°K
(5) Aluminum/Beryllium Structure, Glycol/Water and Coolanol 45; 366/588°K
(6) Double entry for panel residual and miscellaneous result from assuming beryllium and titanium structures where the differences in thermal conductivity dictate different passage spacings; Coolanol 20 used at 477K, Coolanol 45 used at 588K
(7) Required for cooling, does not include containment
(8) Redundant cooling system plus heat shields and excess hydrogen
(9) Glycol/Water Coolant Inlet/Outlet Temperatures of 283K/366K, Coolanol 45 Inlet/Outlet Temperatures of 283K/450K.

TABLE IB
COMPARISON OF COOLING SYSTEM CONCEPTS AND ASSOCIATED HEAT SHIELD OR EXCESS HYDROGEN REQUIREMENTS, HYPERSONIC TRANSPORT[9]

Item	Weights For Cooling System For Average Airframe Temperature Indicated, Pounds						
	Unshielded			Shielded		Dual Temp	
	200F	400F (6)	600F (6)	200F	400F (6)	Unshielded (4)	Shielded (5)
Design Heat Load, 10^6 BTU/Hr	357	301	235	232	280	316	250
NONREDUNDANT							
Distribution Lines(1)	8,250	8,000	5,710	5,700	6,300	7,500	6,500
Heat Exchanger	2,080	1,400	1,050	1,350	1,270	1,900	1,650
Pump	265	250	175	160	180	265	265
Panel Residual(2)	3,310	4,260; 5,050	2,450; 2,900	2,460	3,150; 3,740	2,940	2,300
Miscellaneous(3)	1,395	1,390; 1,470	915; 985	970	1,100; 1,150	1,265	1,075
Total	15,300	15,300;16,170	10,300;10,820	10,640	12,000;12,640	13,870	11,790
REDUNDANT							
Distribution Lines(1)	9,080	8,800	6,280	6,270	6,930	8,250	7,150
Heat Exchangers, Two	4,160	2,800	2,100	2,700	2,540	3,500	3,000
Pumps, Two	530	500	350	320	360	530	490
Panel Residual(2)	3,310	4,260; 5,050	2,450; 2,900	2,460	3,150; 3,740	2,940	2,300
Miscellaneous(3)	1,700	1,640; 1,720	1,120; 1,170	930	1,290; 1,350	1,520	1,290
Subtotal	18,780	18,000;18,870	12,300;12,800	12,520	14,170;14,720	16,740	14,230
Heat Shields	0	0	0	10,000	2,800	0	2,800
Excess Hydrogen(7)	23,000	500	0	400	0	7,000	0
Total(8)	41,780	18,500;19,370	12,300;12,800	22,920	16,970;17,520	23,740	17,030

(1) Including piping, contained coolant, and APS fuel to drive pump. (3 Step Flowrate Schedule as in Ref. 3)

(2) Redundant and nonredundant entries are the same because half flow is in each of the redundant sets of passages.

(3) 10% to account for valves, controls, connectors, supports, etc.

(4) Aluminum/Beryllium Structure, Glyco/Water and Coolanol 45; 200F/600F.

(5) Aluminum/Beryllium Structure, Glycol/Water and Coolanol 45; 200F/600F.

(6) Double entry for panel residual and miscellaneous result from assuming Beryllium and Titanium structures where the differences in thermal conductivity dictate different Passage Spacings; Coolanol 20 used at 400F, Coolanol 45 used at 600F.

(7) Required for Cooling, does not include Containment.

(8) Redundant Cooling System plus Heat Shields and Excess Hydrogen.

(9) Glycol/Water Coolant Inlet/Outlet Temperatures of 50F/ 200F, Coolanol 45 Inlet/Outlet Temperatures of 50F/350F.

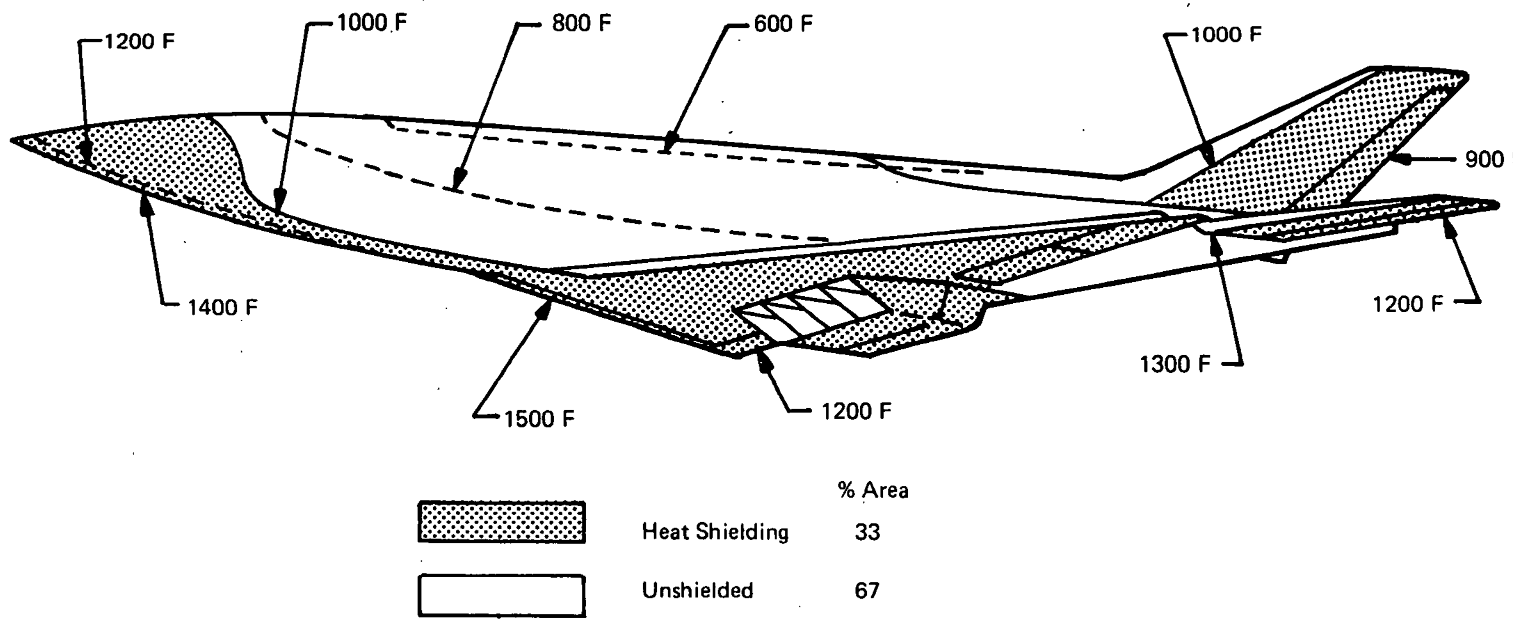


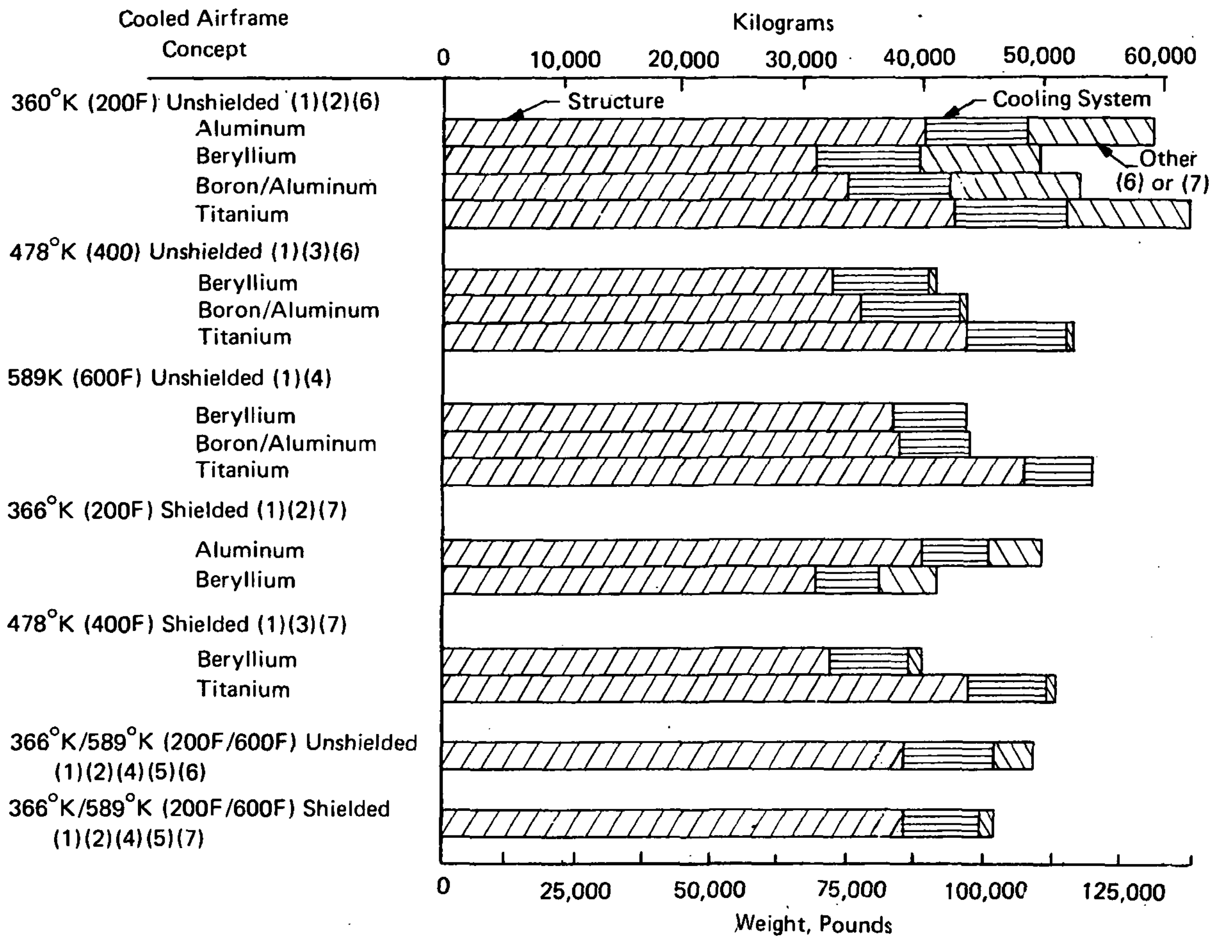
Figure 9. Shielded Hypersonic Transport Concept 1,000F Shields

the excess needed for nominal flight. The weight of containment (insulated tankage and additional fuselage length) is not included but would be relatively small, except for the unshielded 366°K (200F) case, since the total fuel load is approximately 82,000 kg (180,000 lb). For the 366°K (200F) shielded airframe a conservative heat shield weight of 4.9 kg/m^2 (1.0 psf) was assumed giving a total heat shield weight of 4,550 kg (10,000 lb). For maneuver load factors between about ± 0.8 g some excess hydrogen is needed for cooling even though shields are used. The estimated hydrogen requirement for maneuvers was based on a stepped flight load spectrum, with 0.2g increments; maneuvers that prorated to less than 1.0 per flight were assumed to occur once during the flight. On this basis a total weight of excess hydrogen was computed to be 182 kg (400 lb). In the case of the 477°K (400F) shielded airframe, the addition of 1270 kg (2800 lb) of heat shielding attenuated heat loads sufficiently so that no excess hydrogen is needed even during maneuvers. With the two dual temperature systems it is seen that shielding is more efficient than using excess hydrogen, the former requiring an additional weight increment of 1270 kg (2800 lb) and the latter 3180 kg (7000 lb).

Higher structural temperatures decrease heat loads and cooling system weights. However, structural weight increases with temperature for a specific material and may increase or decrease when materials are changed, i.e., aluminum to titanium and aluminum to beryllium.

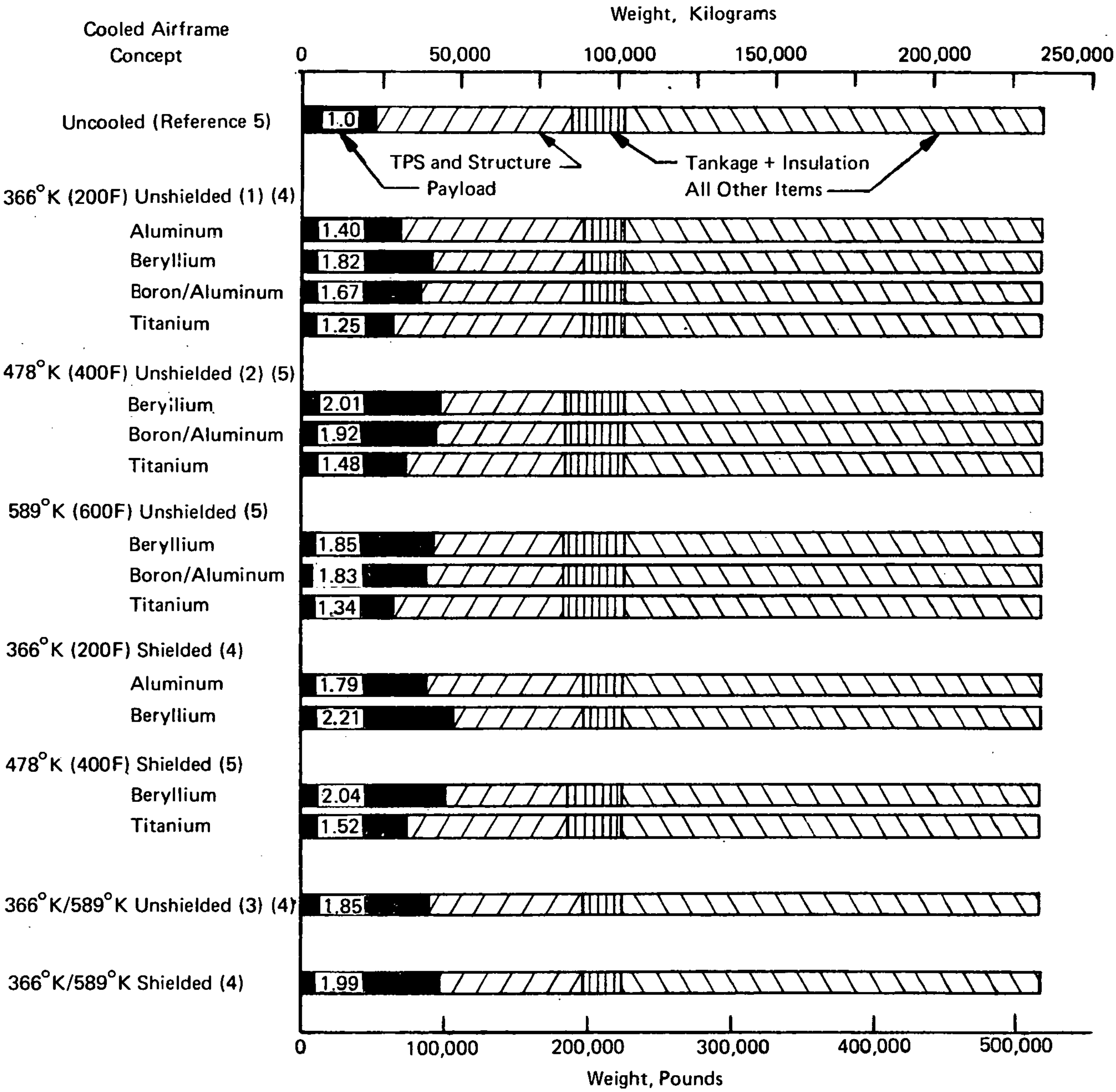
Cooled Airframe Weights - Having considered the weight of various structural concepts/material combinations and of a variety of cooling system concepts that operate at different temperature levels, it is appropriate to consider integrated cooled airframe combinations in order to obtain estimates of weight and improvements in payload capability. In Figure 10,the structural weights from Figure 7 as corrected for operating temperature levels are added to the weights of the cooling systems and associated heat shields or excess hydrogen from Table I. A review of the structural weights clearly indicates the advantage of using advanced types of construction material such as beryllium and metal matrix composites, of which boron/aluminum is typical. The slight differences in weight for these two advanced materials are not considered to be particularly significant. For the unshielded vehicles minimum weight will be found near a structural temperature of 478°K (400F). The titanium systems are always relatively heavier than others in a particular class. The systems that use advanced structural materials (beryllium and boron/aluminum) result in lightest weights, but the various aluminum alloy structural approaches are attractive from a weight point of view, particularly the dual temperature concepts and the 366°K (200F) shielded approach.

While trends of cooled airframe weight are indicative of merit, a clearer picture of the ranking of the various cooled airframe structural concepts is provided by the data of Figure 11. Here, the various systems are compared for a takeoff gross weight of 232,000 kg (520,652 lb), the weight of an uncooled aircraft studied in Reference 5, of which the payload constitutes 21,800 kg (48,000 lb). Differences from this uncooled baseline aircraft are in the areas of the structure and thermal protection, the weight of tankage and insulation, and the payload. Other items were assumed to be of identical weight despite the fact that some subsystem benefits are likely with the cooled concepts. Some of the cooled airframe concepts require hydrogen fuel to be carried specifically for cooling purposes, and while the liquid hydrogen weight is included, the weights associated with its containment and with the additional fuselage volume required have not been subtracted from the payload gain. This weight increment is particularly detrimental to the 366°K (200F) unshielded aircraft category where it could amount to about 4540 kg (10,000 lb). Similarly, for those instances where payload gains are indicated for the cooled aircraft concepts there has been no resizing of the airplane to accommodate the volume of this additional payload. All comparisons are based on vehicles of the same gross weight and physical size so that the payload improvements represent the net weight saved in



(1) Structural Weights of Figure 7 Corrected for effect of average temperature
(2) Glycol/Water Coolant
(3) Coolanol 20 coolant
(4) Coolanol 45 Coolant
(5) Aluminum and Beryllium Construction at 366°K and 589°K (200F and 600F) Respectively
(6) Excess Hydrogen for cooling, does not include containment
(7) Metallic Heat Shields

Figure 10. Cooled Airframe Weight Summary Hypersonic Transport



(1) Weights do not include additional tankage, insulation, and fuselage (about 4,540 kilograms) (10,000 lb) to carry additional LH_2, thereby reducing payload.

(2) Weights do not include additional tankage, insulation, and fuselage (about 90 kilograms) (200 lb) to carry additional LH_2, thereby reducing payload.

(3) Weights do not include additional tankage, insulation, and fuselage (about 1,362 kilograms) (3,000 lb) to carry additional LH_2, thereby reducing payload.

(4) From Reference 2, Inconel 718 tanks and sealed foam insulation.

(5) From Reference 2, Inconel 718 tanks and CO_2 frost insulation.

Figure 11. Impact of Airframe Cooling on Payload, Hypersonic Transport

areas analyzed. In most instances it will not be possible to achieve the total gain indicated because of the necessity to provide additional fuselage volume for the added payload.

Examination of Figure 11 suggests that payload improvements from 50% to over 100% can be expected if cooled airframe technology is exploited in conjunction with advanced structural materials such as beryllium and metal matrix composite. The most attractive system is the 366°K (200F) shielded beryllium structure where the payload increase is about 120% of that for the uncooled baseline aircraft. However, it is rather surprising that a relatively state-of-the-art concept, the 366°K (200F) shielded aluminum alloy aircraft should permit a payload increase of about 80%.

Reliability - Reliability analyses were performed for a nonredundant system and then for a redundant system comprised of two completely independent systems operating in parallel with each accommodating half of the aircraft heat load. In the reliability model for the nonredundant system all components are in series (except for dual pumps); failure of any component except a pump was considered to fail the system even though many of the anticipated failure modes would result in only degraded performance rather than complete system failure. For the redundant arrangement a failure that renders one loop inoperative does not constitute a cooling system failure since each loop is designed to accommodate the full heat load by increasing its coolant flow rate. (The reliability of a fault detection system to indicate the need for increased flow rate was not considered in the analysis.) A necessary assumption was that the environment causing failure of one loop does not cause simultaneous failure of the second loop. The studies indicated failure rates of 7 per 1,000 flights and 4 per 100,000 flights for nonredundant and redundant concepts respectively for a glycol/water system and aluminum alloy construction. A very small weight penalty of 860 to 1600 kg (1,900 and 3,500 lb) seems worth the substantial increase in reliability. The reliability analysis indicated that the skin panels and flexible hoses constitute the highest failure rates. Since the reliability analyses were based on panels having a width of 3m (10 ft) and a length of up to 15m (50 ft), the use of smaller panels would increase the failure rate because additional connections would be required between the coolant distribution lines and the cooled skin panels. To a first approximation, the failure rate of the total skin panel area would not increase; the increase in number of panels would be offset by a decrease in failure rate per panel because of reduced area of each panel. In actuality, a larger number of panels tends to mean an increased number of mechanical fasteners which tend to reduce reliability.

Fatique and Fracture Considerations - In the design of airframe structures, allowable stress levels must be established. This involves consideration of static and fatigue strengths, as well as fracture toughness, and crack growth characteristics for various sizes of initial defects. As an aid to future design efforts, fatigue and fracture analyses were conducted for 2024-T3 using the loading spectrum. The fatigue life estimates considered theoretical stress concentration factors from 1.0 to 5.0 and design allowable ultimate stress levels from 27.5 to 45 kN/cm² (40,000 to 65,000 psi). Based on fatigue considerations alone, and assuming a conservative stress concentration factor of 5.0 and a life scatter factor of 4.0, it is possible to estimate design allowable ultimate stress levels for various vehicle lives, as shown below.

Service Life hours/flights	Design Life hours	Design Allowable Ultimate Stress, kN/cm² (psi)
10,000/5,000	40,000	45 (65,000)
30,000/15,000	120,000	40 (58,000)
50,000/25,000	200,000	37 (54,000)

If the stress concentration factor is reduced from 5.0 to 4.0, the design allowable ultimate stress level increases by 2.06 kN/cm^2 (3,000 psi). In reviewing the results, it appears that modest decreases in stress levels produce substantial increases in life and that realistic service lives can be expected at stress levels of the same magnitude as those that dictate the design of compressively loaded sheet metal structure. Significant benefits are obtained by minimizing stress concentrations.

Crack growth computations for the 2024-T3 alloys were made using the CRACKS computer code. Conservatively, retardation effects were neglected. An infinite plate was assumed for the particular analyses conducted. Stress levels for the loading spectrum were obtained in the same manner as for the fatigue analyses. Defects of two types were considered: surface scratches having a depth of 0.05 mm (2 mils) and through cracks having a half length of 0.25 mm (10 mils). For these conditions, the design allowable ultimate stresses listed below were predicted using the Forman equation for representative life times to failure. Failure was defined as "break thru" for the surface scratch and unstable growth for the through crack.

Service Life hours/flights	Design Life hours	Design Allowable Ultimate Stress, Scratch/Through Crack, kN/cm^2 (psi)	
10,000/5,000	40,000	45/45	(65,000/65,000)
30,000/15,000	120,000	35/32	(50,000/47,000)
50,000/25,000	200,000	30/28	(43,000/40,000)

For a design life of 5,000 flights and the assumed initial defect sizes, fracture mechanics considerations do not influence the design. For the more realistic design lives between 15,000 and 25,000 flights, fatigue and fracture considerations impose constraints of significant magnitude upon the design, with the crack growth being more restrictive. The results predicted for the surface scratches are considered to be conservative. The stress concentrating effects of a shallow surface scratch are relatively unknown; behavior may be more similar to fatigue than to fracture mechanics. For a laminated skin, the nature of the bond may modify the growth rate through the thickness, such that the progress of crack growth is retarded. This type of behavior has been observed for laminated aluminum alloy and titanium alloy. Experimental evaluations are needed to clarify behavior characteristics for cooled panels.

Although fatigue and crack growth considerations can be expected to have an influence on the design allowable strength levels for hypersonic transports with service lives in excess of 5000 flights, the decrease in the design allowable ultimate stress level should not be used as a measure of the weight increase that will result. For this particular hypersonic transport a substantial portion of the fuselage and the outer wing panels are designed by minimum gage constraints rather than by stress limits. Furthermore, other portions of the aircraft are designed by buckling considerations; here too, the decrease in the design allowable ultimate stress would have little influence on weight. An additional consideration is the fact that a relatively small amount of the total airframe structure is actually designed to the allowable ultimate stress level because of variations in loading intensities and practical considerations. These comments are not meant to minimize the importance of fatigue and fracture considerations but rather to caution against pessimistic weight estimates. In many areas such considerations will dictate the design allowable and significant local weight increases will result. However, each aircraft must be considered in detail before defining the impact on structural weight.

Hypersonic Research Airplane

Structural Concepts - Because of the nearer term potential of this aircraft as compared to the transport, structural considerations were limited to aluminum alloys. Based on the results obtained from comparing numerous structural concepts for the hypersonic transport over a range of loading intensities, the structural configuration for the HRA fuselage was assumed to be skin/stringer/frame construction with Z-stiffners. The structure was sized conservatively to preclude buckling under ultimate load conditions so that there would be no distortions that might influence the cross-sectional shape of any of the coolant passages. The low limit loadings for the fuselage 1.0 kN/cm (560 lb/in) maximum and for the wing, 1.1 kN/cm (630 lb/in), resulted in minimum gage design for most of the airframe $w = 4.97\ kg/m^2$ (1.02 psf). In addition, sizing studies were conducted for the propellant tankage which was assumed to be integral. Such analyses were not required for the transport because the baseline configuration employed nonintegral tanks. For both the fuselage and the wings, the weight estimation procedure involved sizing the covers for the loading intensity or the minimum equivalent thickness, whichever was greater, and using design experience factors to account for substructure, fittings, and nonoptimum considerations. Structural weight of the covers, skins plus stringers, were computed at various fuselage locations and were integrated circumferentially and axially to determine the cover weights.

No detailed analyses were performed for the wing covers since the low loading intensities suggested the use of lighter structure than dictated on the basis of minimum gage design which resulted in a cover weight of 365 kg (800 lb). Prior studies of low aspect ratio highly swept wings suggest that the wing covers constitute 60% to 65% of the wing weight. Thus, a weight of 227 kg (500 lb) was assumed for the substructure. A relatively high fitting weight, 137 kg (300 lb), was assumed since the main landing gear is mounted in the wing near the wing/fuselage intersection. A nonoptimal weight penalty of 91 kg (200 lb) was also included to account for integration of the substructure with the cooled skin panels, access doors and the penalties involved with the large landing gear doors. The total wing weight was estimated to be 820 kg (1800 lb).

Since the loads on the vertical tails should be lighter than on the wings by virtue of the shorter tail spans, the tail weights were estimated on the basis of minimum gage design for the covers which resulted in 330 kg (730 lb). Weight increments for the ribs and spars, the fittings, and the nonoptimum allowance were 160 kg (350 lb), 50 kg (110 lb), and 50 kg (110 lb) respectively. Thus, the total tail weight was 590 kg (1300 lb).

In sizing the integral propellant tankage analytical emphasis was placed on the tank frames and heads since the skin is relatively easy to size. The frames were sized initially for representative conditions of captive and free-flight landing, and taxiing; internal pressure loads were combined with other structural loads. Initial analysis assumed constant frame depth and EI but after these initial solutions were obtained, the frame characteristics were refined but were not fully optimized. Thus, the weight results should be slightly conservative. The tank heads were also sized in a somewhat conservative manner by assuming the heads to be flat plates, computing the unit weight of the head, and applying this unit weight to the actual head area. The results of the fuselage sizing studies yield a fuselage weight of 2220 kg (4900 lb), 1740 kg (3825 lb) for covers and 480 kg (1055 lb) for frames. To this is added 600 kg (1320 lb) for fittings and the nonoptimum penalty bringing the total fuselage structural weight to 2850 kg (6220 lb). The wing and tail structure adds an additional 1410 kg (3100 lb) bringing the airframe weight to 4250 kg (9320 lb).

Cooling System Concepts - One potential objective of a hypersonic research airplane is to evaluate an actively cooled structure. The relatively near term of such a research airplane is likely to focus attention on conventional materials, with aluminum being the most promising. Because of this, the primary effort with respect to cooling system comparisons involved consideration of the most likely coolant choice. However, since it might be desirable to utilize the same construction material, coolant, and operating temperature level for the HRA as those expected to be used in a hypersonic transport, consideration was given also to other coolants which would permit structural temperatures of up to 589°K (600F). Consideration of heat load matching to fuel flow heat capacity indicated that either some form of heat load attenuation or enhancement of available heat capacity of the fuel would be required after the initial acceleration phase or else extra hydrogen would be required, especially for cooling purposes. This approach involves about 1530 kg (3400 lb) of hydrogen, (somewhat more than that required to accelerate to Mach 8) with an additional 10% to 20% required to deal with maneuver heat loads.

The heat load attenuation techniques considered included the use of a higher average structural operating temperature to reduce the heat load and to simultaneously increase the available heat capacity of the hydrogen fuel through higher coolant temperatures. For the Mach 8 trajectory, an increase in the average structural temperature, from 366°K (200F) to 589°K (600F) reduces the heat load by about 16% and increases the fuel heat capacity that is available for structural cooling by about 35%, reducing the amount of hydrogen that would have to be carried for cooling purposes by about 455 kg (1000 lb). The dual temperature cooling system also provides a means of enhancing the available heat capacity of the fuel flow. Another means of attenuating the heat load to the cooling system is to use external heat shielding or insulation. In fact, the plan for this particular HRA vehicle was to employ RSI to extend flight speed capability from Mach 8 to Mach 10. Several types of insulation systems will be discussed later.

Weights for convective cooling systems based on the configuration of Figure 6 are presented in the upper part of Table II for various coolants and airframe thermal protection concepts. Nonredundant systems tend to be about 20 to 25% lighter than redundant systems. The use of the methanol/water coolant results in a system that is about 8% lighter than for glycol/water. Although not shown in the table, other coolants were considered for the 366°K (200F) average temperature aluminum alloy structure; Coolanol 20, Coolanol 40 and FC-43 yielded systems that were significantly heavier than those based on aqueous solution. (The lower weight of the aqueous systems can be seen by comparing the unshielded 366°K (200F) systems to the unshielded 478°K (400F) system.) Because of the benefits of higher operating temperature, the weight of the nonaqueous 589°K (600F) system is approximately equal to that of an unshielded 366°K (200F) ethylene-glycol/water system. A comparable weight can also be obtained with the dual temperature system concept. When shielding is added to the aircraft, the aerodynamic heat load to the cooling system is attenuated and cooling system weights decrease as shown. Although weight considerations favor the use of methanol/water with aluminum alloy construction, other considerations are involved in the selection between aqueous solutions of glycol and methanol as coolants. Both are electrically conductive and rely on inhibitors for corrosion resistance. Neither coolant type shows any decomposition below 394°K (250F). Primary disadvantages of methanol/water solutions are their high vapor pressure, flammability, and toxicity. The major advantages of methanol/water over water/glycol are lower freezing points and lower viscosity at temperatures below about 255°K (0F); at temperatures above 225°K (0F) viscosity characteristics are about the same for each.

The weights in the upper part of Table II represent only cooling system elements. Auxiliary items required for proper functioning of the thermal protection system in a total sense are shown in

TABLE IIA
COMPARISON OF COOLING SYSTEM CONCEPTS AND ASSOCIATED HEAT SHIELD OR EXCESS HYDROGEN REQUIREMENTS, HYPERSONIC RESEARCH AIRPLANE BASELINE PANEL

Item	Weights for Cooling System for Average Airframe Temperature Indicated, Kilograms						
	Unshielded; M = 8				366°K, Shielded		366°K, RSI
	366°K[4]	477°K[5]	588°K[6]	366°K/ 588°K[7]	M = 8.0	M = 10.0	Optimized for M = 10.0
Design Heat Load, Mw	28.1	26.9	23.4	26.9	10.5	18.8	5.3
NONREDUNDANT							
Distribution Lines[1]	667; 622	972	690	667	281	436	168
Heat Exchanger	250; 234	327	177	232	91	163	41
Pump	18; 16	32	18	18	14	14	7
Panel Residual[2]	268; 241	345	400	295	204	295	109
Miscellaneous[3]	123; 114	168	127	120	59	91	32
Total	1,326; 1,227	1,844	1,412	1,332	649	999	357
REDUNDANT							
Distribution Lines[1]	717; 663	1,053	745	722	327	490	172
Heat Exchangers, Two	499; 468	654	354	413	177	327	81
Pumps, Two	36; 32	64	36	36	23	23	14
Panel Residual[2]	268; 241	345	400	295	204	295	109
Miscellaneous[3]	154; 140	213	154	159	73	114	39
Subtotal	1,674; 1,544	2,329	1,689	1,675	804	1,249	415
Heat Shields or RSI	0	0	0	0	1,189	1,189	1,071
Extra Hydrogen[8]	1,679	NC	1,220	NC	681	1,090	82
Total[9]	3,353; 3,223	NC	2,909	NC	2,674	3,528	1,568

(1) Lines, contained Coolant, and APS fuel to drive pump.
(2) Redundant and nonredundant entries are the same because half flow is in each of the two redundant sets of passages.
(3) 10% to account for valves, controls, supports, etc.
(4) Glycol/Water; Methanol/Water Weights Account for Double Entries, Coolant Inlet/Outlet 283°K/360°K.
(5) Titanium Structure, Coolanol 20, Coolant Inlet/Outlet 283°K/394°K.
(6) Titanium Structure, Coolanol 40, Coolant Inlet/Outlet 283°K/450°K.
(7) Dual Temperature System, Glycol/Water and Coolanol 40.
(8) Required for Cooling, does not include Containment.
(9) Redundant Cooling System plus Heat Shields and Excess Hydrogen.

TABLE IIB
COMPARISON OF COOLING SYSTEM CONCEPTS AND ASSOCIATED HEAT SHIELD OR EXCESS HYDROGEN REQUIREMENTS, HYPERSONIC RESEARCH AIRPLANE BASELINE PANEL

Item	Weights For Cooling System For Average Airframe Temperature Indicated, Pounds						
	Unshielded, M = 8				200F, Shielded		200F, RSI
	200F(4)	400F(5)	600F(6)	200F/600F(7)	M = 8.0	M = 10.0	Optimized for M = 10.0
Design Heat Load 10^6 BTU/Hr	96	92	80	92	36	64	18
NONREDUNDANT							
Distribution Lines(1)	1,470;1,370	2,140	1,520	1,470	620	960	370
Heat Exchanger	550; 515	720	390	510	200	360	90
Pump	40; 35	70	40	40	30	30	15
Panel Residual(2)	590; 530	760	880	650	450	650	240
Miscellaneous(3)	270; 250	370	280		130	200	70
Total	2,920;2,700	4,060	3,110	270	1,430	2,240	785
REDUNDANT							
Distribution Lines(1)	1,580;1,460	2,320	1,640	1,590	720	1,080	380
Heat Exchangers, Two	1,100,1,030	1,440	780	1,020	390	720	180
Pumps, Two	80; 70	140	80	80	50	50	30
Panel Residual(2)	590; 530	760	880	650	450	650	240
Miscellaneous(3)	340; 310	470	340	350	160	250	85
Subtotal	3,690;3,400	5,130	3,720	3,790	1,780	2,750	915
Heat Shields or RSI	0	0	0	0	2,620	2,620	2,360
Extra Hydrogen(8)	3,700	NC	2,700	NC	1,500	2,400	150
Total(9)	7,390;7,100	NC	6,420	NC	5,900	7,770	3,425

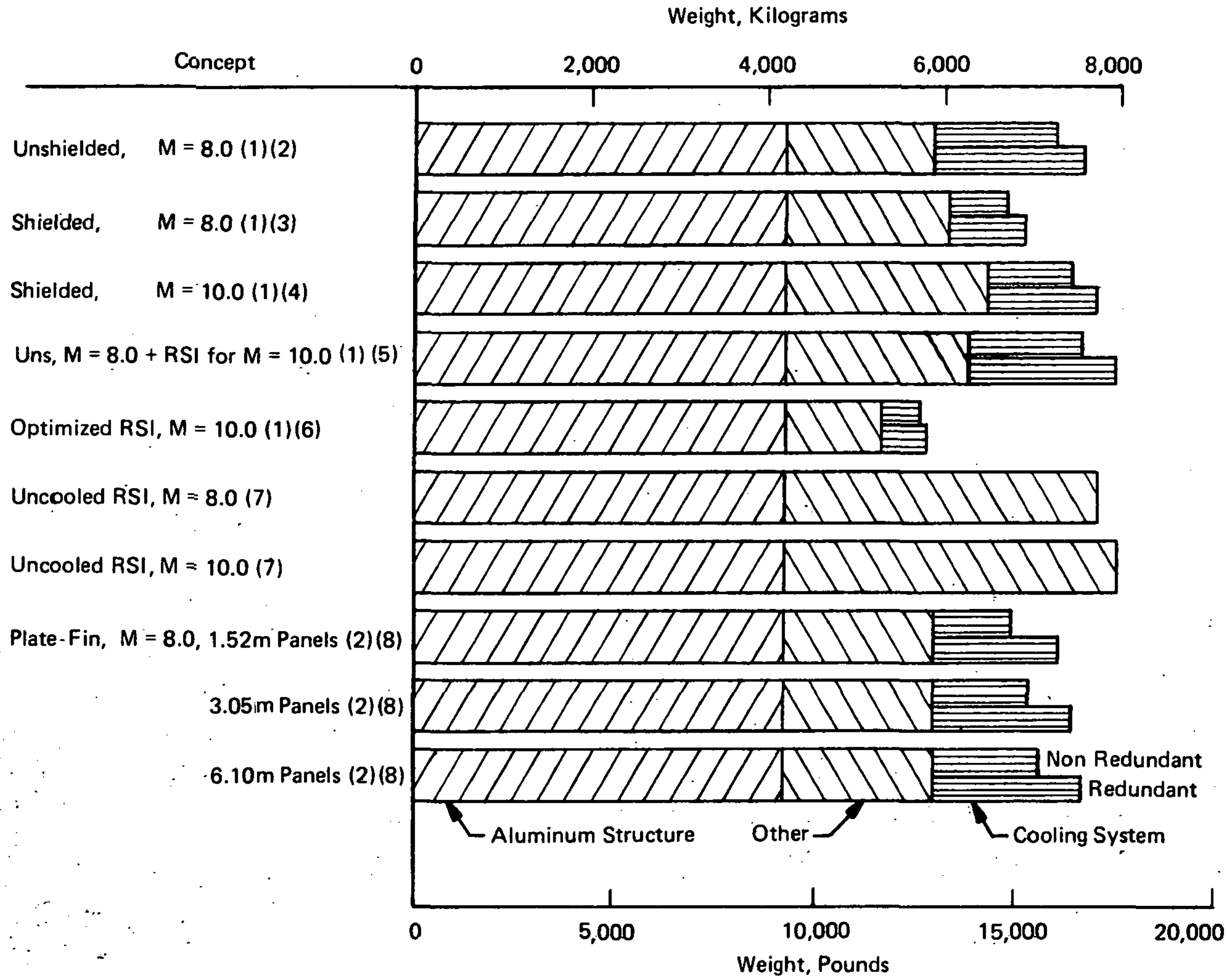
(1) Lines, contained coolant, and APS fuel to drive pump
(2) Redundant and nonredundant entries are the same because half flow is in each of the two redundant sets of passages
(3) 10% to account for valves, controls, supports, etc.
(4) Glycol/Water; Methanol/Water Weights Account for Double Entries, Coolant Inlet/Outlet 50F/200F
(5) Titanium Structure, Coolanol 20, Coolant Inlet/Outlet 50F/250F
(6) Titanium Structure, Coolanol 40 Coolant Inlet/Outlet 50F/350F
(7) Dual Temperature System, Glycol/Water and Coolanol 40
(8) Required for cooling, does not include containment
(9) Redundant cooling system plus heat shields and excess hydrogen.

the lower portion of the Table, in order to give a clearer picture of thermal protection system weights. For any of the cooled systems, hydrogen must be carried specifically for cooling during descent since the vehicle as presently configured uses all of its fuel by the end of cruise and unless hydrogen flow is continued for airframe cooling, the structure will overheat. Obviously, the radiation shields and RSI incur weight penalties. Total airframe weights will be discussed later.

The cooling system weights presented in Table II were based on the use of the baseline discrete tubular panel concept. The choice of panel concept does influence the weight of the distribution lines and in particular the weight of the residual coolant within the cooled panels. To assess weight trends in this regard, analyses were conducted for plate-fin and sphere-core panels of three lengths. Since these sandwich concepts are less sensitive to the viscosity effects associated with low coolant temperatures, it is possible to employ a lower inlet temperature than can be used for the discrete passage concept. Furthermore, the film temperature drop constitutes the only temperature gradient at the outlet end of the coolant flow path so that the coolant outlet temperature can be higher for the plate-fin concept than for the discrete passage concept. Thus, a total coolant temperature rise of 129°K (230F), (from 255°K to 383°K (0F to 230F)) was used as compared to 83°K (150F) (from 283°K to 366°K (50°K to 200F)) for the discrete passage approach. Although the same quantity of heat must be removed in either case, for the sandwich approaches the heat exchanger weight decreased due to the larger log mean temperature difference and the weight (of the pump and fuel to drive it were reduced because of the lower flow rate due to a larger coolant temperature rise. As a result, weights for both nonredundant and redundant plate-fin arrangements were less than for tubular designs. The weight of the sphere-core concept depends upon the manner of packing the spheres but was always much heavier than the plate-fin design. When a tight packing is used, relatively high pressure drops are encountered even though the sandwich thickness is increased relative to the plate-fin. The weight penalty associated with the pumping power and the thickness increase is not offset by the fact that about 50% of the sandwich volume is free of coolant because of the hollow spheres. When a square array of spheres is used, weight of the system is reduced somewhat because the decreased pressure drop permits a thinner sandwich with less residual coolant and less pumping power penalty, but the weight is still nearly twice that of the plate fin system.

Cooled Airframe Weights - Weights of the cooled airframe for various thermal protection systems concepts based on active cooling are presented in Figure 12 for the hypersonic research airplane. Aluminum alloy construction is assumed in all cases; maximum structural temperatures would be approximately 394°K (250F) under nominal flight conditions and 422°K (300F) under maneuver conditions. For comparison purposes, optimized uncooled RSI protected systems are included; weights were computed using the data of Reference 7 and the techniques of Reference 6. These uncooled concepts are comparable to the heaviest of the cooled approaches.

While the optimized cooled RSI concept is the lightest (less than 5600 kg (13,000 lb) thermal protection system for the hypersonic research airplane, it is not representative of concepts likely to be used for a hypersonic transport. In its present form, reusable surface insulation is too fragile for all weather operation and is probably too expensive in view of the high replacement and repair to be expected because of its frailty. Weight differences among the other four candidates employing the baseline tubular skin panel approach are comparable for similar operating conditions with weight differences of less than 10%. The similarity of cooled airframe weights suggests the possiblility of designing the basic airframe for an unshielded Mach 8 capability to maximize the speed regime over which an unprotected airframe could be operated, and then evaluating both RSI and shielded concepts for the higher operating capabilities desired.



(1) Glycol/water coolant 283°K (50F) inlet/360°K (200F) Outlet

(2) "Other" is the LH_2 for cooling during descent, including maneuver allowance

(3) "Other" is 1189 kg (2620 lb) for metallic heat shields over 80% of the surfaces + 861 kg (1500 lb) for LH_2 for cooling during descent

(4) "Other" is 1189 kg (2620 lb) for metallic heat shields over 80% of the surface + 1090 kg (2400 lb) for LH_2 for cooling during descent

(5) "Other" is 672 kg (1480 lb) of RSI over 80% of the surfaces + 1362 kg (3000 lb) of LH_2 for cooling during descent

(6) "Other" is 1072 kg (2360 lb) for RSI + 68 kg (150 lb) for LH_2 for cooling during descent

(7) Computed using Reference 6

(8) Glycol/Water coolant 273°K (OF) inlet/383°K (230F) outlet

Figure 12. Cooled Airframe Weight Summary, Hypersonic Research Airplane, Aluminum Alloy Construction

As compared to the baseline panel concept for the unshielded Mach 8 concept, the plate-fin design approach is expected to be slightly lighter, or slightly heavier, depending upon the size of the skin panels that are used. The larger size leads to higher weight but less system complexity. The use of smaller panel sizes will reduce weight but will decrease reliability as indicated below.

Reliability - Studies similar to those for the hypersonic transport were conducted to examine the reliability of the HRA. They were based on the cooling system shown in Figure 6. For a nonredundant system, about 7 failures are indicated per 10,000 flights, while for a completely redundant system the rate is approximately 1 failure per million flights. This large reduction in failure rate is obtained by a weight increase of only 370 kg (800 lb). Other reliability/weight trade studies indicated that while some weight can be saved by relaxing reliability requirements and using a larger number of panels the weight reductions seem to be quite small in comparison to the reliability that is lost; three times the failure rate to save 78 kg (170 lb) in the most favorable case. Various pump arrangements were also considered; approximately 18 kg (40 lb) is saved by using only one pump in each coolant transport loop of the redundant system with a very small change in failure rate.

Fatigue and Fracture Considerations - The fatigue and fracture analyses conducted for the hypersonic research airplane paralleled those of the hypersonic transport. For this research airplane a service life of 250 hr was assumed (250 flights) so that with a scatter factor of 4.0 the design life was 1000 hours. The significance of fatigue and fracture considerations on the design of the 2024-T3 cooled airframe can be summarized in relation to the design allowable ultimate stress for two representative lives as follows:

Design Consideration	1000 hr. Design Life kN/cm^2 (psi)	2000 hr. Design Life kN/cm^2 (psi)
Fatigue, K_t = 5.0	33.2 (48,000)	29.8 (43,000)
Fatigue, K_t = 4.0	35.3 (51,000)	31.8 (46,000)
Fatigue, K_t = 3.0	45.0 (65,000)	39.4 (57,000)
Surface Scratch 0.06 mm (2 mils)	36.0 (52,000)	30.5 (44,000)
Through Crack, 0.25 mm (10 mils)	31.8 (46,000)	27.0 (39,000)

Based on these considerations, it appears that the fracture mechanics considerations are of primary importance particularly with respect to the presence of through cracks. The likelihood of such cracks being present is extremely remote and deserves further consideration before the design is penalized to the degree indicated. Surface scratches do not appear to be quite as significant as fatigue considerations. The above comments pertain to the airframe structure specifically and not to the propellant tankage. Because of the combined consideration of airframe loads and internal pressurization experienced by integral propellant tanks, the design allowable stress levels are likely to be somewhat lower for the tankage than for the airframe structure. In addition, the fatigue and fracture characteristics of the 2219 alloy are not quite as good as those for the 2024 alloy.

The adverse influence of fatigue and crack growth considerations on the design allowable ultimate stress should not be equated to an increase in structural weight. While some weight increases may be incurred in local areas of high stress, most of the HRA structure was designed by minimum gage considerations, rather than by stress limits. Therefore, a substantial reduction in the design allowable stress can be expected to cause a relatively small increase in structural weight.

CONCLUSIONS

Emphasis in this study was placed on convective cooling systems and primary load carrying structure with the objective of comparing materials and concepts. Because of this emphasis on concept examination and parametric comparisons, the efforts devoted to definition of the characteristics of the hypersonic transport and the hypersonic research airplane were limited to a level less than that associated with a preliminary design study. The net effect of the simpifying assumptions used is considered to be small with regard to comparisons, but may be somewhat larger when absolute magnitudes are considered.

The comparisons of the various cooled structural concepts for the hypersonic transport are made by comparing payloads for airplanes of the same gross weight, geometric configuration and dimensions. That is, the aircraft was not resized but rather it was assumed that the additional payload weight made possible by the lighter, cooled structure could be carried within the original volume. In some instances additional hydrogen was assumed to be carried specifically for cooling purposes, but the vehicle configuration was not altered to accommodate the increase in fuel volume associated with this additional hydrogen weight. These simplifications lead to optimistic estimates of payload increases.

The primary conclusion reached as a result of these extensive parametric and aircraft system analyses is that the potential benefit to be derived from actively cooled airframe structure may be greater than anticipated in earlier studies. When advanced structural materials are considered an actively cooled hypersonic transport could carry approximately 200% of the payload that could be carried by an uncooled vehicle of the same gross weight if the additional payload could be accommodated within the original vehicle configuration.

From a weight/payload point of view, the most attractive design utilized a beryllium airframe structure maintained at less than 394°K (250F) by a glycol/water cooling system. Approximately 30% of the external surface of the aircraft was shielded with superalloy panels to permit matching of the airframe heat load to the capacity available from the normal fuel flow schedule. The same basic concept, but with an aluminum alloy structure, served as the baseline system and indicated a payload of 180% of that for an uncooled structure. Both cooled designs employed completely redundant cooling systems.

In addition to the primary conclusion of the superior payload weight potential for the cooled airframe concept, a number of more detailed conclusions were reached with regard to various aspects of the total design picture; they are grouped into the following categories: (1) airframe concepts, (2) materials, (3) structural concepts for panels, (4) thermal concepts for panels, (5) reliability and safety, and (6) local areas requiring attention. Specific conclusions with respect to airframe concepts are as follows:

1. Matching the heat load absorbed by the cooling system to the heat capacity of the fuel flow schedule is a principle consideration in minimizing the weight of an actively cooled airframe structure. Several approaches including trajectory tailoring, external shielding and/or dual temperature airframe structures can be used to match cooling system heat load to fuel heat capacity.

2. To eliminate any form of external shielding or the carrying of excess hydrogen specifically for cooling purposes requires a structural operating temperature capability of about 575°K (575F) for the specific hypersonic transport studied.

3. For the hypersonic transport, a beryllium structure resulted in the lightest weight for all concept variations considered, with boron/aluminum almost as attractive. Payload increase varied for specific cooled airframe concept with the shielded 366°K (200F) system most beneficial, and the unshielded 366°K (200F) system least attractive.

With respect to construction materials and coolants, the following conclusions can be drawn from the results of the various studies:

1. As compared to aluminum alloy construction advanced materials might increase payload by more than 25%. Beryllium is most attractive with metal matrix composites only slightly less attractive.

2. There appears to be no major problem in finding compatible combinations of attractive construction materials and promising coolants based on the corrosion and stress corrosion tests conducted.

3. There are potential advantages for mixed material structures when this allows the hydrogen fuel to be heated to a higher temperature than possible when a single material is used.

4. Aqueous coolants are best for temperatures below about 394°K (250F), nonaqueous solutions are attractive when maximum coolant temperatures exceed about 394°K (250F).

While a large number of structural panel concepts were considered and compared, the relatively conventional skin/stringer/frame and the honeycomb sandwich approaches were most attractive for the fuselage and wings respectively. Lighter weights were indicated for symmetrically double beaded fuselage skin panels but this design poses major problems of integrating coolant passages with the structural arrangement.

With regard to the weight of panel designs as influenced by thermal factors, the following conclusions may be drawn:

1. The baseline tubular panel concept is most attractive from weight and assembly points of view for regions of modest heat flux and areas where minimum gage considerations set skin thickness requirements. The sandwiched tube concept appears particularly attractive.

2. The plate-fin panel concept is most attractive for high heat flux levels where the use of the tubular concept would impose weight penalties associated with high coolant flow rates, or preclude practical assembly because of very close passage spacing. The heat flux for changing of the panel concept will depend upon specific vehicle design requirements and requires further study.

3. While much heavier than the plate-fin concept, the sphere-core panel concept may be useful in regions requiring double curvature.

4. The plain skin with cooled substructure appears to be attractive only for low heat flux levels unless metallurgical joining is used, in which case its primary advantage of damage tolerance is lost.

5. When aerodynamic heating is to be attenuated, metallic heat shields appear more attractive than high density ceramic external surface insulation.

Reliability considerations appear to dictate the use of the completely redundant cooling system designs for the hypersonic transport. While the weight penalty involved is about 1370 kg (3000 pounds) this is only about 0.6% of the takeoff gross weight, and the added safety appears to warrant such an approach. About half of the weight increment is due to a second heat exchanger. Therefore, the possibility of not doubling up on this item may warrant further investigation. The relatively small penalty associated with complete redundancy is due to an improved redundancy concept defined during the course of the project, flowing 50% of the required coolant in each of two adjacent coolant passage networks. Even if the flow rate in the single operating loop was not doubled under shutdown of one loop, structural temperatures would not rise catastrophically but the 1.5 design ultimate factor of safety will be reduced as a result of the temperature increase and strength reduction. The reduced size and shorter design life for the hypersonic research airplane made the need for complete redundancy uncertain. However, it may be desirable to include a redundant system as a means of testing the concept for later use on advanced transports. If redundancy is not used, a means of avoiding excessive temperatures of the load carrying structure will be needed to deal with possible emergencies associated with cooling system malfunction. Other conclusions reached with regard to reliability and safety include:

1. Redundant coolant passage networks can be provided in both the discrete tubular and plate-fin concepts with relatively small weight penalties by using concepts defined by this study.

2. The discrete tubular passage concept appears well suited to the incorporation of crack arrestors which should enhance the damage tolerance of actively cooled skin panels.

3. Failure mode and effects analyses indicate that the panel and connections should receive special attention during detail design.

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