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NEW APPROACHES TO HYPERSONIC AIRCRAFT

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ABSTRACT

The strong interactions between the aerodynamic, structural, and propulsive systems of hypersonic air breathers offer important opportunities for achieving improved vehicles. One of the most promising is the use of the hydrogen fuel heat sink to provide cooling of major areas of the airframe. This possibility is explored in some detail, with considerations of the theoretical possibilities, engine designs for minimum cooling, comparative analysis of candidate high-level cooling systems, recent fluid-mechanical studies of slot cooling, structural designs compatible with practical cooling systems, and aerodynamic features made possible in actively cooled vehicles. The results suggest that hypersonic cruise vehicles constructed of largely unshielded aluminum or titanium alloys are feasible and offer a number of advantages. Further studies of the problems and possibilities of this category of hypersonic vehicles are suggested.

INTRODUCTION

The hypersonic cruise vehicles we are looking toward today will represent the ultimate achievement of aeronautics. Operationally, they will have the fastest atmospheric flight speeds useful to man in the limited confines of our planet, and technologically they will employ the ultimate in fuels, materials, and aircraft systems. It is safe to say that our present early concepts and technology levels for these vehicles fall far short of the final achievement. We are thus faced with long-term research and development before operationally justifiable systems are likely to emerge.

At this early stage it is appropriate to ask how we can most effectively work toward these ultimate vehicles. If past aeronautical developments provide a valid index to the next 20 years, we can not only count on continuing evolutionary technological growth but also we can confidently anticipate a number of those pivotal advances in technical thinking - the breakthroughs - which open new avenues for progress. A prime objective of all R and D programs should be the creation of a research environment which gives maximum encouragement to the generation of these breakthroughs. This reasoning might seem to be so virtuous that there could hardly be any argument about it, but actually there is considerable disagreement. A sizable group in the hypersonic area believes that the correct general approaches and the most promising concepts have already been identified and that all that remains is the detailed disciplinary R and D on the existing concepts to achieve their ultimate refinement. At best, they expect only secondary breakthroughs in the disciplines and they believe that we are already approaching with diminishing returns the final plateau of high-speed flight technology.

At the opposite extreme there is another prevalent view. This group considers that hypersonic vehicle concept development at this time is premature and should be deferred pending the establishment of "sound technology bases" in the separate supporting disciplines. This is a convenient philosophy for times of limited R and D funding, but there is little justification for it in previous aeronautical experience. Past vehicle developments in almost all cases were undertaken with large inadequacies in the supporting disciplines; and there is every reason to believe progress generally would have been retarded if these developments had been deferred until all was in good order in the disciplines. In our view, progress in hypersonic aircraft can best be served at the present point in history by imaginative flexible exploration of a variety of conceptual approaches. This requires equal imaginative flexibility in the pursuit of the related disciplines. A good case can be made for programs in which early concept R and D goes hand-in-hand with research in the supporting disciplines. This tends to bring into focus the real problems as opposed to the purely academic problems with which the disciplines otherwise sometimes get sidetracked. And it provides the elements of need and stimulation which give maximum encouragement to innovation and breakthrough.

Aircraft configurations based on the dictates of a single discipline such as aerodynamics are likely to be naive and impractical. In the early days of hypersonic research, the aerodynamics discipline was pursued very vigorously without benefit of equal effort in the propulsive and structure areas. As a consequence, the attainment of very high lift/drag ratios became an exaggerated goal attended by aerodynamic configurations having very large wings.[1,2] As more realistic structures, propulsion, and weight inputs became available, the wings began to shrink and the bodies became relatively larger and elliptical or flattened in cross section, until at this point in time we see the Mach 8 to 12 cruise vehicles of the future as lifting bodies with only rudimentary wings (Fig. 1) or perhaps with no wings at all.[3,4,5] Clearly, meaningful configurations are achieved only through a balanced pursuit of all of the relevant disciplines with free interplay of the varying pressures they generate.

In the course of this configuration work it became increasingly apparent that these interdisciplinary interactions play a far more dominant role in hypersonic aircraft than in previous aircraft developments. While it is true that they introduce new constraints and added problems, it is also true that they offer the designer a number of new opportunities. In fact, we see in these interactions a probable source of the hoped for improvements and innovations needed to achieve successful designs. Previous aircraft developments have, in the main, left it to the special genius and intuition of the "Chief Designer" and his team to pursue these complex problems of systems integration and optimization. We believe, however, that certain of the more fundamental interactions between structures, propulsion, and aerothermodynamics are so important to progress in hypersonic aircraft that they must be accepted as legitimate subjects for general research investigation in addition to continuing work in the disciplines.

About 3 years ago, following the general philosophy outlined above, we diverted a small part of our research group at Langley to explore a new vehicle concept exploiting one of these interdisciplinary interactions. We had noted with interest the coexistence in current designs of very hot airframes housing some 2 pounds of very cold hydrogen fuel for every pound of structure, with the designers doing their best to keep these elements entirely separated except in the engine itself. Our aim was to explore the possibilities of permitting the propulsive heat sink and the aerodynamic heat load to interact directly. Previous studies of heat-absorbing structures for cruise vehicles in which the heat sink was supplied by a fluid or gas carried as an added part of the structural system with no other useful function showed little promise.[6] Here, however, we propose to utilize the heat sink available in another aircraft system to that there is no weight penalty for the fluid itself, but there are, of course, possible weight penalties and other problems associated with tying the two systems together. This paper is a progress report on a continuing study of the problems and potentialities of these fuel-cooled aircraft.

The Problems of Uncooled Hypersonic Vehicle Structures

The incentive to study cooled vehicles stems from the many problems and restrictive features of the so-called "hot" structure, the approach which has so far received the principal attentions of the structures community. The basic virtue of this approach is that it disposes of the heat load largely by radiation, eliminating, for the most part, the necessity of absorbing the heat internally in a gas or fluid carried for cooling purposes. Early comparative studies[6] of the hot radiative structure showed it to be lighter than heat-absorbing systems for lengthy heating periods, hence the attention it has received. The feasibility of the "hot" structure rests on the happenstance that the radiation equilibrium temperatures of cruise vehicles tend to fall within the possible working temperature ranges of the so-called superalloys and the refractory metals. Figure 2 illustrates typical Mach 8 design surface temperatures calculated with maximum benefit from internal radiation between upper and lower surfaces, conduction, and so forth.[7] Virtually the entire structure exhibits varying degrees of red heat. Obviously, the development of long-lived lightweight structures for these conditions presents formidable problems.[8] Much ingenuity has been brought to bear in hot structures research in recent years and important progress has been made, resulting in the features illustrated in Figure 3 for the wing. The major problems of thermal stress are partially solved by deliberate design for thermal warpage through nonredundancy, corrugated shear webs, segmentation, and similar devices. In order to achieve lower operating temperatures and longer life for the Mach 8 primary structure visualized,[7] it must be largely covered by metal heat shields, reducing its temperature to about 1600° F. These shields, which are typically constructed of very thin (0.012 to 0.020 inch) superalloy metals and separated by slip joints, present difficult problems of their own. The principal questions relate to hot spot development and hot-air leakage at the joints, maintenance, inspection, rain leakage, icing, hail damage, and so forth. Much more work is necessary on thermal cycling, high-dynamic-pressure hot-air oxidation testing, and other problems, to determine whether long lifetimes and reusability in the sense of current commercial aircraft can be achieved with the hot structure. Naturally at this state, the structural weight fractions ultimately achievable for hypersonic aircraft are also quite uncertain.[8]

A basic problem of the hot structure which is not generally recognized in its full proportions is the need for special insulation of virtually every item of internal mechanical and electrical equipment, fuel, crew, passengers, and baggage from the hot environment surrounding them. Perhaps the most critical of these problems is protection of the liquid hydrogen fuel system which requires the special measures shown in Figure 4.[9] A purge gas must be employed to prevent cryo-pumping of air in the space between the bulk insulation and the hot structure. Several of the other critical subsystems can be adequately protected only by circulation of liquid coolants over large areas. Among these are the propulsion system, pilot and passenger cabins, avionics bays, landing gear bays, and so forth.

Actively Cooled Structures

Many of the problems of the hot structure could be greatly alleviated or eliminated altogether if the primary structure could be kept at the low temperatures of Mach 2 aluminum alloy or Mach 3 titanium alloy aircraft, and major new subsystem developments for the hot environment could be avoided. Let us now look at two schemes by which this might be accomplished. On the lower left of Figure 5 an aluminum alloy structure is kept within its temperature limit of 200° F by elaborate insulation. In general, three layers are required: the external metal heat shield, the thick bulk insulation, and, finally, the low-level coolant system to supplement the bulk insulation. The coolant system design may be either a passive water-wick approach, or a low-level water circulation system.[6,10,11] Typically, only 2 or 3 percent of the total heat load is absorbed by the coolant in this design. This is the type of "cooled" system* one is limited to in practice when the coolant weight must be charged against the system. Optimization studies[6] suggest that it may find application in space reentry vehicles where the heat pulse is of

*The term "actively cooled structure" in virtually all of the previous literature refers to low-level insulated systems of this kind.

short duration. However, for the cruise vehicle application, it tends to be heavier than other approaches; and because of its bulk, cannot be used on the tail surfaces, controls, or on thin regions of the wing.

The active high-level cooling system utilizing the fuel heat sink which we are proposing is shown in the upper sketch on the right of Figure 5. Here the coolant flow is in the exposed structure where it absorbs virtually the entire heat load. The nature of the cooling problem here is obviously quite different from the insulated case. Coolant capacities of the order of 20 to 100 times greater are involved. However, the structure and, as we shall see later, some of the fabrication technology developed for the insulated system(10) appear to be applicable to the high-level cooling case. In the sketch on the lower right, the local coolant heat-load rate is reduced by simple heat shields to about 15 percent of the total, in a combination of the radiative and heat-absorbing concepts.

The working temperatures for the actively cooled structure are the same as for current Mach 2 aluminum alloy or Mach 3 titanium alloy aircraft. Thus all of the extensive available structural and materials technology for such aircraft would be directly applicable. Equally long service lifetimes could be obtained and much of the subsystem technology for the lower speed aircraft could be used without further development. The problems of cooling the cabin and other critical volumes would be identical to the Mach 2 or 3 aircraft. In short, nearly all of the undesirable features and operational questions surrounding the hot structure could be avoided.

On the negative side, direct structural cooling involves the addition of a complex new mechanical system with major questions of weight, reliability, and fabrication. Structures specialists have a natural distaste for a structure which becomes intimately involved with and dependent upon a mechanical system.(6) And the propulsion specialists are not happy with our interdisciplinary tampering either. It was thus inevitable that we should receive discouraging advices from our friends in these quarters when we began our study. They had three main arguments:

1. Hypersonic engines would require virtually all of the fuel heat sink for engine cooling.
2. Airframe heat loads would prove to be excessive, probably greater than the total fuel-flow heat sink.
3. Active cooling systems for large structural areas would prove excessively heavy and otherwise impractical and unreliable.

As has often happened in past aeronautical developments, these intuitive early viewpoints, which seemed reasonable enough at first, have been shown to be unfounded as they were analyzed in more detail. We will now discuss each of these three problems, beginning with the propulsion system question.

Propulsion System Coolant Requirements

In conformance with traditional aeronautical practice, the hypersonic ramjet engine is regarded as a periodically refurbishable and replaceable system. It is not required to have the same lifetime as the airframe, it has much higher typical design temperatures, and it uses appropriate superalloy materials. Early estimates of engine coolant requirements centered on annular designs similar to the engine of Figure 6(a). The results were generally discouraging, especially with subsonic combustion, because these engines appeared to require more coolant than that obtainable from the normal fuel flow for hypersonic cruise. It was sometimes pointed out that the heat release in hydrogen combustion was 10 to 20 times larger than its heat capacity for cooling purposes, suggesting great difficulty (or perhaps impossibility) for the engine-cooling problem. It is unfortunate that these early assessments were interpreted in some quarters as ruling out any hope of hypersonic air-breathing propulsion with low coolant requirements and with a residual heat sink remaining for structural cooling.

Although the design art for engines having low coolant requirements with minimum sacrifice of performance has not progressed very far, one can readily identify many features which will reduce the internal heat loads, in particular the following:

- Supersonic combustion
- Nonannular ducts having low ratio of wetted area to flow area
- Short combustor lengths (efficient fuel injectors)
- Short cowl lengths
- Large combustor area ratios
- Reduced pressures and reduced fuel injection near the duct walls(12)
- Insulation, film cooling, and so forth, in the internal ducting(13)
- Use of the aircraft body for major part of inlet and nozzle functions

Using all of these features except insulation, we have designed and analyzed the scramjet concept shown in Figure 6(b). For comparison, the requirements of the annular engine (Fig. 6(a)) were also determined and the results are given in Figure 7, which shows the large reductions in cooling made possible in the three-dimensional design. These reductions were realized without any significant degradation of calculated specific impulse. It should be admitted, however, that the great complexity of the heat-transfer processes in the presence of combustion phenomena preclude, for the present at least, any really rigorous theoretical treatment. Thus any heat-load estimation method must be validated by comparison with experimental data. The boundary-layer method used here included consideration of profile distortions due to the large adverse pressure gradients associated with combustion, and its results agree well with recent supersonic combustor heating data obtained by Billig and Grenleski.(14)

Figure 7 clearly suggests that advanced engines can be found which will require only a minor fraction of the available fuel heat sink at Mach 6, leaving a large potential for airframe cooling. This finding removed the first of the supposed obstacles to structural cooling, and encouraged us to proceed with a general analysis of the airframe problem.

Theoretical Prospects for High-Level Active Airframe Cooling

The basic question here is whether the external heat loads of a cruising hypersonic aircraft are of such magnitude that they can be absorbed by the fuel heat sink, leaving enough residual cooling capacity for the engine ducting. An approximate general assessment of this problem is possible because the ratio of the airframe heat-load rate to the fuel heat-sink rate is a simple function of $\bar{C}_H/C_D$, the ratio of the average heat-transfer coefficient to the total drag coefficient. The external airframe heat-load rate is proportional to $\bar{C}_H$:

$$\dot{Q}_{airframe} = \bar{C}_H S c_p \rho g V\left(\bar{T}_{rec} - T_{wall}\right)$$

And the fuel-flow heat-sink rate is proportional to C_D:

$$\dot{Q}_{fuel} = \frac{1}{2}\rho V^2 C_D S_p \, \Delta T_{fuel}/I_{sp}$$

In these relations the coefficients are both referenced to the same area, and to free-stream conditions. Comparisons with a detailed heat-load assessment for a complete vehicle has shown that the above relation is a good approximation if the $\bar{C}_H/C_D$ value is estimated correctly. The desired ratio is

$$\frac{\dot{Q}_{airframe}}{\dot{Q}_{fuel}} = \frac{\bar{C}_H}{C_D}\frac{VI_{sp}}{\Delta T_{fuel}}\left(1 - \frac{T_w - T_\infty}{7 \times 10^{-5}V^2}\right) \times 3.1 \times 10^{-4}$$

in which ΔT_{fuel} is the fuel temperature rise utilized in the structural cooling process, with typical maximum permissible values indicated in Table II for materials of interest. The $\bar{C}_H/C_D$ ratio is configuration and L/D dependent, and can be estimated readily without elaborate calculations of local heat transfer from the usually available knowledge of the average skin-friction coefficient for the given configurations. That is,

$$\frac{\bar{C}_H}{C_D} = \frac{1}{2}\frac{\bar{C}_F}{C_D} \times \text{Reynolds analogy factor}$$

$$= 0.55\frac{\bar{C}_F}{C_D}, \text{ for cold-wall conditions}$$

Pertinent data on the average heat/transfer drag ratio are indicated in Figure 8(a), as functions primarily of L/D_{max} and configuration shape. Because of their lower lifting efficiency, circular bodies have much larger $\bar{C}_H/C_D$ factors than flat wings. Typical aircraft, which are characterized by features of both bodies and wings, fall between the theoretical extremes for the simple shapes, with the more bodylike configurations falling nearer to the bodies, and so forth. The $\bar{C}_H/C_D$ ratio for a given vehicle is strongly dependent on its flight attitude, Figure 8(b). Clearly cooled vehicle operations, in general, should favor the high angle-of-attack side of the flight polar for conditions of reduced L/D.

We are, of course, interested in whether the $\bar{C}_H/C_D$ values for the configurations of interest will produce $\dot{Q}/\dot{Q}_{fuel}$ values less than 1. The limit values of $\bar{C}_H/C_D$ for $\dot{Q}/\dot{Q}_{fuel} = 1$ are

$$\frac{\bar{C}_H}{C_D} = \frac{3.2 \times 10^3 \, \Delta T_{fuel}}{VI_{sp}\left(1 - \frac{T_w - T_\infty}{7 \times 10^{-5}V^2}\right)}$$

These limits are plotted as the shaded lines on Figure 9 using the appropriate engine and fuel data from Tables I and II. The limits shown are pertinent to 200° F cooled aluminum alloy and to 500° F cooled titanium alloy structures. These are the maximum permissible values of $\bar{C}_H/C_D$ for which the heat-load rate equals the fuel heat-sink rate. Minimums are seen to occur near Mach 8, primarily because the VI_{sp} product maximizes near this speed for the scramjet.

For comparison with the $\bar{C}_H/C_D$ fuel-cooling limit lines we have plotted on Figure 9 $\bar{C}_H/C_D$ values for typical winged configurations of various L/D, obtained from a mean of the cross-hatched region of Figure 8(a). We see at once that fully exposed, unprotected configurations in the range of main interest (L/D, 4 - 5) have the possibility of absorption of their total airframe heat loads by the available fuel-flow heat sink with titanium alloy. For aluminum alloy, an L/D of about 3.5 is the ideal limit for total absorption by the heat-sink fractions available for aluminum (Table II).

On the right of Figure 9 a similar comparison is made for the case where simple uninsulated metal heat shields are used over 33 percent of the surface as indicated for the winged Mach 6 vehicle* in Figure 10, reducing the total heat load to the aircraft by about 50 percent. Two-thirds of the vehicle surface including all of the leading edges (10-foot streamwise lengths) are completely exposed and about 92 percent of the vehicle structure is conventional alloy in this "partially shielded" case. We note from Figure 9, right side, very large margins for both alloys, even for the highest attainable L/D's. These margins are desirable because they permit simplifications in the design of the cooling system which usually are accompanied by overcooling in local areas and absorption of more than the ideal minimum heat loads.

It is important to note here that even 100 percent absorption of the heat sink available for 200°

*The vehicle design used in this illustration is a conventional wing-body arrangement selected some years ago for our coolant system study program because it had been the subject of a previous design study.[15] It is not a currently advocated concept.

or 500° F airframe cooling leaves a very large residual heat-sink capacity for engine cooling by further temperature increase of the hydrogen coolant up to the 1600° F value appropriate to cooling engine duct walls operating at 1800° F. (See Table II.) A breakdown of the engine and airframe requirements and comparisons with the various available coolant capacities at Mach 6 is shown in Table III.

It is of interest now to compare the magnitudes of typical engine and airframe cooling requirements throughout the range of Mach numbers of principal interest for cruise systems (Fig. 11). The engine assumed here is the three-dimensional low heat-transfer design of Figure 6(b), and it is seen to have linearly increasing coolant requirements with advancing speed in the range shown. The relative airframe requirements, as we have seen previously, peak near Mach 8 with values far below the engine requirement, contrary to early unfounded speculations. The Mach 12 airframe cooling problem is about the same as for Mach 6 in terms of the minor percentage of the fuel-flow heat capacity utilized. For this particular engine and airframe, the sum of the engine and partially shielded airframe values remains less than unity to Mach numbers in excess of 9, with comfortable margins for design purposes up to at least Mach 7. It might also be noted that there are excellent prospects for reducing the engine coolant requirements still further by the several measures previously mentioned. The engine concept of Figure 6(b) and its requirements (Fig. 11) are simply the fruits of a first attempt which did not by any means exploit fully the many possibilities for reducing the engine coolant demands. We believe that future engine developments will extend the flight speed for these cooled aircraft to at least Mach 12.

Studies of Active High-Level Airframe Cooling Systems

These promising theoretical possibilities for advanced fuel-cooled scramjet-powered cruise vehicles clearly justified preliminary engineering studies of possible cooling systems. The primary questions are:

Can a practical, reliable, lightweight means of heat transfer be devised for cooling very large surface areas?

Can the pumps, ducting, insulation, heat exchangers, controls, and so forth, associated with such a system have practical proportions, lightweight, reliability, and ability to copy with realistic off-design and emergency situations?

To obtain first-order answers we have been conducting a number of preliminary studies of possible airframe cooling systems, aided by personnel of the Bell Aerospace Company. A large number of cooling-system approaches have been assessed and compared.[16] Direct circulation of hydrogen fuel for purposes of regenerative cooling of engine combustor and ducting has been the subject of considerable recent research and development[8,17,18] but this approach was not found attractive for large airframe structural areas. In addition to the major concern for hydrogen leakage and safety, the pumping and distribution system was found to be heavier than for systems using secondary coolant liquids. The most promising airframe system utilizes a secondary coolant circulated internally to transfer heat from the structural surfaces to a central hydrogen-fuel-cooled heat exchanger (Fig. 12). A water-ethylene-glycol solution for aluminum alloy or a silicone base fluid such as Dow-Corning DC-331 for titanium alloy are suitable liquids. By circulating these liquids at moderate velocities and pressures, small coolant passages and low coolant liquid weights are achieved. Prevention of local freezing of these coolants in the hydrogen heat exchanger presents special design problems.

The application of internal fluid cooling to the very large areas involved in a hypersonic aircraft requires careful measures to minimize weight. The concept developed by Bell Aerospace and used in the study of reference 16 and for a number of subsequent studies which are still in progress is based on the use of conventional spar-stressed skin-stinger construction (Fig. 13). The coolant passage spacings indicated are typical of the design requirements for the pressure side of a Mach 6 cruise vehicle wing. Diffusion-bonded double skins are employed, with the coolant passages and headers die-formed into the inner layer after bonding.[19] The passages usually run in a direction which permits the passage walls to participate in the load-carrying process. An example of a small panel produced in this way is shown in Figure 14. There are no obvious obstacles to production of very large panels perhaps as long as 30 feet and as wide as 6 feet or more. These panels could readily be subjected to exhaustive tests as necessary to insure their integrity before installation in the aircraft structure. A similar approach is possible for the fuselage and, as shown on Figure 15, the typical required passage spacings are also practical. The close spacings for the top of the fuselage, which is a relatively cool region, were dictated by the special considerations of passenger cabin cooling.

An example of a large fuselage structural test model fabricated with integral coolant passages is shown in Figure 16. It must be acknowledged at once that this system[10,11] was designed only for low-level coolant capacity at the base of complex thermal insulation. (See "Insulated" case, Fig. 5.) Nevertheless, the fabrication problems and techniques were similar to what is proposed here.

These cooled structures developments have so far centered mainly around aluminum alloys. Corresponding technology has not yet been developed for titanium, but presumably it could be. Titanium is, of course, more difficult to work and has a disadvantage of lower thermal conductivity which leads to closer tube spacings than for aluminum. On the other hand, its much better high-temperature properties might make it the better choice when all factors are considered.

The wing leading edge is of particular interest and importance because the use of active cooling can eliminate the restrictions to high sweep and bluntness which have characterized previous "hot" leading-edge designs. To realize this inherent advantage and achieve a virtually sharp leading edge requires design features such as those proposed in Figure 17 where the forward section incorporates small machined passages covered by an outer skin. Radii as small as 0.06 inch appear practical. The remainder of the leading edge is similar in construction to the rest of the cooled wing. An exposed length of about 60 inches normal to the leading edge is desirable in most cases to extend back to where the wing is thick enough for the addition of heat shields.

In regard to the other elements required in a complete cooling system such as the hydrogen/glycol heat exchanger, pump, controls, ducting, and the like, Bell Aerospace personnel have conducted a preliminary design study with generally encouraging results. Their estimates indicate that the total weight of a cooled aluminum aircraft with its cooling system included is somewhat less than the same configuration with a current hot structure concept, assuming that no added hydrogen fuel is required specifically for cooling purposes. More will be said about this later.

A study of the effect of the inclusion of external radiation shields on the all-up weight of the cooling system revealed least weight for the completely unshielded vehicles and a rapid increase in weight when more than about 15 percent of the surface was protected by simple metal shields. It was assumed here that the fuel heat sink required did not involve any added weight. Of course, if the absence of partial shielding resulted in a requirement for an incremental supply of coolant, the optimum condition would shift in the direction of more extensive partial shielding. This is the basis on which the 33-percent partial shielding coverage was selected for the aluminum vehicle of Figure 10 which has an L/D of about 4.2, and if unshielded would have a coolant requirement somewhat larger than the nominal fuel flow heat sink.

At this stage, naturally, these cooled aircraft hardware concepts are defined only in broad outline and many questions of detail remain unanswered. The particular concepts so far developed are of interest mainly as illustrations of the possibilities. We believe the potential for actively cooled systems justifies more comprehensive systems studies and developmental research in the areas of cooled structures, their heat-transfer problems, and their fabrication problems. Only a small beginning has been made so far.

We would also like to emphasize the need for continued R and D to advance the hot structure for the numerous vehicle applications where fuel cooling cannot be considered. There are also possibilities that the hot and the cooled approaches can be combined to advantage in some cases. The possibilities which should be examined range all the way from hot airframes using only leading-edge cooling to the fully cooled case.

Slot-Cooled Systems

We will digress briefly at this point to review our studies of an alternate system which is inherently adaptable for cooling very large surface areas. The employment of a cooled gas either transpired or ejected through slots is interesting because, in addition to direct cooling, it provides aerodynamic blockage of part of the heat load, a skin-friction reduction, and a thrust force in the case of slots.[20] Unfortunately, the fluid mechanics of these coolant gas flows are not yet firmly enough established to permit rigorous design assessments. This situation has prompted several recent fluid mechanics investigations of slot cooling designed to answer some of the outstanding questions. We are concentrating on slot cooling rather than transpiration because it rather clearly seems to offer better overall performance for our application.* An indication of the present status can be given with the aid of Figure 18. In the region ahead of the impingement point of the inner mixing boundary the slot flow is perfectly effective, that is, the adiabatic wall temperature equals the recovery temperature of the coolant gas. After impingement, mixing with the hot external stream causes a gradual deterioration in effectiveness as shown. One of the reasons for interest is the greater slot effectiveness achieved at hypersonic conditions than at Mach 3, especially for multiple slots.[20,21,22] The skin-friction level in the zone before impingement is typically as little as 30 percent of the reference surface friction for Mach 6 flight conditions. Starting with reasonable assumptions for the issuing slot flow, Bushnell and Beckwith have applied their finite difference method of boundary-alyer analysis[23] with some success to predict the profile developments and the cooling effectiveness parameter. Our continuing program at Langley includes extended experiments at Mach 6 to assess the turbulent transport properties and the mixing processes more accurately and to verify the effects of multiple slots. One of the objects of the program is to determine the effects of the slot mass flow parameter. This has a critical influence in the extrapolation of wind-tunnel data, particularly in determining the slot spacing and mass flow requirements.

A current estimate of the net drag cost of slot cooling the wing of the Mach 6 cruise vehicle of Figure 10 is given in Figure 19. The prime difficulty is the very large ram drag penalty associated with taking the coolant air aboard the aircraft and losses in the distribution system. This is only partially offset by the thrust and friction reduction components. There seems to be little prospect for cooling at Mach 6 down to aluminum temperatures with this approach, and even 800° F temperatures would involve a 15-percent drag penalty (equivalent to some 36,000 pounds of added fuel and loss of most of the payload for this vehicle). Thus at this stage, slot cooling with the ram air/fuel heat-sink/heat-exchanger system is not competitive with the internal liquid convective system.

*The brief preliminary assessment of slot cooling given in reference 16 has been superseded by more comprehensive unpublished studies in our program and by the recent work of Ferri, Zakkay, and Fox, and others at New York University who are associated with the Langley program.

There is an interesting possibility for making the slot-cooling technique workable without a large net drag penalty and without basic dependence on the fuel heat sink. Ferri, Fox, and Hoydysh[24] have proposed a "turbocooler device," Figure 20, in which a supersonic impulse turbine is used to extract enthalpy and reduce the temperature of the cooling airflow. The turbine shaft power is absorbed by a supersonic compressor which adds the energy to the propulsive airflow, thereby canceling a significant part of the ram drag. Direct fuel cooling is employed only for the turbocooler hardware itself and possibly for local areas such as the leading edges which may be difficult to handle with slot cooling. Preliminary estimates of the overall performance of the turbocooler system appear promising, although, naturally, at this early stage a truly rigorous systems analysis is not possible.

It is intriguing to speculate on future directions in which the turbocooler might evolve. As presently conceived, these devices would be carried beneath the aircraft in addition to the propulsion system. One can speculate, however, that ultimately the turbocooler might be combined with rotating elements of the propulsion system to provide in a single device, take-off, acceleration, and cruise propulsion, and low-cost coolant air as well. In essence, we are suggesting the ultimate fuel-cooled turbo-ramjet with added cold-gas generation capability for the higher speeds. We offer this as an example opportunity for the kind of "broad look" or "far-out" probing of fresh approaches which are needed at this point in hypersonic technology development.

Operational Considerations for Actively Cooled Aircraft

After we had satisfied ourselves with the general feasibility of high-level active cooling for the Mach 6 cruise condition, we turned our attention to the flight maneuver situations which usually prove critical in aircraft design. Our principal finding can be briefly stated: all possible steady powered flight conditions have fuel flows greater than those required for airframe and engine cooling. This implies an operating corridor unrestricted by the type of thermal limitations imposed on hot structures (see Table IV), except for possible heat-shield limits if partial shielding is used. As the design operating altitude is reduced the coolant flows will generally have to be increased and the tube spacings decreased, and eventually, of course, physical limits will be reached. A cursory evaluation, however, suggests that the limits will correspond to very high dynamic pressures beyond the range of practical interest.

Conditions of thrust/drag ratio greater than unity obviously also present no problem. However, for thrust/drag below unity the fuel flow can be reduced only to those values required for cooling purposes. This latter condition is important only in the final deceleration from cruising flight. In one extreme type of descent the engine may be cut off completely, in which case we find that the excess hydrogen usage for deceleration to Mach 2 is of the order of 1 percent of the gross weight, provided the descent is made at an optimum reduced L/D rather than at L/D_{max} for reasons that will be apparent from inspection of Figure 8(b) and Figure 9. If the descent is made with power on at the fuel-flow levels necessary for cooling, no extra hydrogen is required but the descent must be started earlier than for an uncooled aircraft and the average flight speed is, of course, slightly reduced. However, only some 5 minutes is added to the duration of a typical 5000-mile trip.

The reliability problem of actively cooled vehicles is much more serious than for nominally uncooled aircraft because a failure of the cooling system could cause a structural failure. In this area the radiation-cooled hot structure appears to have an important inherent advantage. We tend to forget, however, the fact that the so-called "uncooled" vehicles will also have to have elaborate circulating coolants over large areas. Considering the extensive water-glycol (or similar) circulating coolant required for the passenger and crew cabins, avionics, landing gear, and so forth, and particularly the direct use of circulating hydrogen over the large hot areas of the propulsion system ducting, we find that a total area as large as 20 percent of the external surface area must be actively cooled in any case. Thus a major commitment to reliable active cooling must be made for all hypersonic aircraft regardless of structural philosophy. On the positive side, an actively cooled Mach 7 to 8 aluminum alloy aircraft will generally require less heat-shield coverage than an "uncooled" vehicle, and with cooling the heat shields need not extend into the most hazardous locations near the leading edges. The use of dual pumps, controls, cooling passages, and other devices will have to be considered for the actively cooled systems, and reliability will have to be accepted as a prime problem for R and D for these vehicles.

Configuration Design for Aircraft With High-Level Active Cooling

High-level active cooling frees the aircraft designer from many of the hypersonic shape restrictions which have characterized previous vehicle concepts (see Table IV). The wing loading limits imposed by corridor thermal considerations are eliminated for all practical purposes, and the wing leading edge can be sharp and unswept if desired. A vast complex of interrelated aero-thermo-structural-elastic and insulation problems are either entirely or largely eliminated. The exploitation of these freedoms is a large new subject in itself and we will attempt here only a few simple illustrations and comparisons.

We consider first the aerodynamic improvements accruing from the "sharp" leading edge which is made possible by internal cooling. The improvements here go beyond the drag saving, which is typically 1000 pounds or more and which is, of course, important in itself. The highly swept wing depends on leading-edge sharpness to insure the controlled separated vortex flow which is responsible for most of its favorable low-speed characteristics. As the radius is increased into the range of values necessary for radiation-cooled long-lived metallic leading edges, the desired controlled flow separation and vortex formation is delayed to higher angles of attack and the aerodynamic characteristics deteriorate with erratic behavior of the forces, moments, and load distributions.[25] Let us examine a particular case - the delta wing of the Mach 6 transport analyzed in references 15 and 16. The uncooled, blunt-leading-edge (3/4-inch radius) superalloy version of this wing was estimated to weigh 54,100 pounds.[16] If a sharp-edged aluminum alloy cooled wing together with its cooling system is designed for equal lift at low speed it can be smaller (Fig. 21) because its lift is increased due to increased vortex action engendered by the sharp leading edge. This smaller aluminum wing is estimated to be as much as one-third lighter than the hot wing. A further reduction in wing weight can be made by reducing the sweep of the maximum thickness line to zero* (Fig. 21), and this cooled wing, if the areas are held constant, would produce an estimated 27-percent increase in low-speed lift-curve slope over the highly swept blunt-edged hot wing. The low leading-edge sweep for this wing places it in the category of designs where special devices, such as leading-edge flaps, must be employed to achieve acceptable low-speed aerodynamic properties.

Many other interacting factors, of course, influence the final choice of a wing in a real aircraft development[31] and it remains to be seen how the cooled and hot wings would compare if other critical factors could be evaluated.

Our second illustration exploits the possibility of locating the vertical fins in a high dynamic pressure region previously forbidden for hot structures, namely, the high-pressure, high-heating environment beneath the wing. Using the weight estimation methods previously referred to, we see in Figure 22 that small dual fins located under the wing could be over 40 percent lighter than the conventional hot fin for equal directional stability.

Finally, let us attempt a first-order overall comparison of cooled and hot versions of the Mach 6 transport configuration of reference 14. For a simple comparison we will not take advantage of any configuration changes other than the substitution of a sharp leading edge for the cooled aircraft. The hot structure analyzed is based on the recent recommendations of Plank, Sakata, Davis, and Ritchie[7] rather than on the less advanced earlier assumptions in the original design.[15] The cooling-system weights are based on the Bell Aerospace work[16] for the wing, together with their current cooling studies of the remainder of the aircraft. The results of these comparisons are shown on Table V. All of the items favor the actively cooled approach except for the weight penalty of the cooling system itself which is more than nullified by the other reductions.

It should be noted that many items of insulation weight could not be evaluated meaningfully in this first look, all of which would favor the cooled structure. One very large potential item noted by Bell Aerospace is the possibility of a further reduction of as much as 10,000 pounds in tankage insulation through development of vacuum insulation to replace the heavy system considered here (Fig. 4). It seems probable that this development can be realized more readily for the cooled aluminum structure than for the hot structure. Inclusion of this item would bring the payload advantage of the cooled airplane up from the 70-percent shown in Table V to almost 100 percent. It will also be noted that added weight savings accruing from configurational changes such as those of Figures 21 and 22 are not included in Table V and would, in large measure, be additive.

While one cannot claim high accuracy for this kind of preliminary estimate, there is little doubt that the cooled airplane offers potential weight and performance improvements.

CONCLUSION

We have explored one area of strong interaction between the propulsive, structural, and aerodynamic systems of hypersonic cruise aircraft, and have found that the use of the fuel heat sink for active high-level cooling of all or part of the airframe is feasible and offers many potential advantages. The spectrum of interesting possibilities ranges from the completely cooled Mach 6 to 8 airframe, as we have considered it, to hot airframes employing active cooling only in selected critical areas such as the leading edge.

The hypersonic engine with minimal internal coolant requirement is at the heart of these cooled aircraft. A number of possibilities for achieving such engines exist, and this is a fertile field for research. Progress here will benefit propulsion systems installations even where active airframe cooling is not involved.

The difficult questions of practicality and reliability of cooled aircraft structures have been given encouraging answers to the extent that this is possible in an exploratory study. Naturally, these questions cannot be answered with real finality without a great deal of further research and development. The skeptics will doubtless make the most of these new problems. We would urge, however, that they not lose sight of the enormous potential advantages offered by these cooled vehicles, including major alleviations of the insulation and subsystem design problems, reduced thermal warpage and thermal stress, and new freedom for improved aerodynamic design. These important potentialities justify more comprehensive systems studies and developmental research for these new vehicles.

*Obtained from methods outlined in an unpublished working document entitled, "Derivation of Structural Weight Estimation Techniques." Proceedings of Second Weight Prediction Workshop for Aerospace Design Projects at Wright-Patterson Air Force Base, Ohio, Oct. 31-Nov. 1966, pp. 126-150 (compiled by W. K. Smith). Use was made of references 26 through 30 to estimate the aerodynamic properties of these wings.

ACKNOWLEDGMENTS

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REFERENCES

1. Eggers, A. J., Jr., and C. H. Syvertson, "Aircraft Configurations Developing High Lift-Drag Ratios at High Supersonic Speeds," NACA RM A55L05, Mar. 1956.

2. Becker, J. V., "Studies of High Lift/Drag Ratio Hypersonic Configurations," Proceedings, Fourth Congress of ICAS, Palais de l'UNESCO, Paris, France, Aug. 24-28, 1965; Robert R. Dexter, editor, Spartan Books, 1965.

3. Brewer, G. D., "Manned Hypersonic Vehicles," Lockheed Horizons, p. 16, Issue 4, First Quarter, 1966.

4. Gregory, Thomas J., Williams, Louis J., and Wilcox, Darrell E., "The Air-Breathing Launch Vehicle for Earth Orbit Shuttle - Performance and Operation," AIAA Advanced Space Transportation Meeting, Feb. 4-6, 1970.

5. Henry, J. R., and McLellan, C. H., "The Air-Breathing Launch Vehicle for Earth-Orbit Shuttle - New Technology and Development Approach," AIAA Advanced Space Transportation Meeting, Feb. 4-6, 1970

6. Dukes, W. H., "Protection of Aircraft Structures Against High Temperatures," Proceedings, Second International Congress in the Aeronautical Sciences, Zurich, Sept. 1960, A Pergamon Press Book, the MacMillan Co., New York.

7. Plank, P. P., Sakata, I, F., Davis, G. W., and Ritchie, C. C., "Hypersonic Cruise Vehicle Wing Structure Evaluation," Lockheed Missile and Space Company, NASA CR 1568, May 1970.

8. Heldenfels, R. R., "Structural Prospects for Hypersonic Air Vehicles," Paper No. 66-31 presented at Vth ICAS Congress, London, England, Sept. 12-16, 1966.

9. Jackson, R., Davis, J. G., Jr., and Wichorek, G. R., "Structural Concepts for Hydrogen Fueled Hypersonic Airplanes," NASA TN D-3162, 1966.

10. Dukes, W. H., Gosden, C. E., Kappelt, G. F., and Mirti, A. E., "Manufacturing Methods for Insulated and Cooled Double-Wall Structures," ASD Tech. Rept. 61-7-799, Sec. 1, Vol. 1, May 1961.

11. Goesch, W. H., Boensch, F. D., and Phillips, L. R., "Structural Design Considerations for Cryogenic Fueled Hypersonic Cruise Vehicles," AIAA Paper No. 1088, Oct. 1968.

12. Ferri, A., "Review of Scramjet Propulsion Technology," Journal of Aircraft, Vol. 5, No. 1, Jan./Feb. 1968, p. 3.

13. Pagel, L. L., and Wanubold, W. R., "Active Cooling of a Hydrogen Fueled Scramjet Engine," Journal of Aircraft, Vol. 6, No. 5, Sept./Oct. 1969, p. 472.

14. Billig, F. S., and Grenleski, S. E., "Heat Transfer in Supersonic Combustion Processes," Fourth International Heat Transfer Conference, Versailles, France, Aug. 1970.

15. Jarlett, F. E., "The Hydrogen Fueled Hypersonic Transport," Paper presented at ASME Annual Aviation and Space Conference, Beverly Hills, California, June 16, 1968.

16. McConarty, W. A., and Anthony, F. M., "Design and Evaluation of Active Cooling Systems for Mach 6 Cruise Vehicle Wings," Bell Aerosystems Company, Report No. 7305-901001, Dec. 1968.

17. Kelly, H. N., Gellersen, E. W., and Richard, C. E., "Development of Fabrication Techniques and Performance Tests of Hydrogen-Cooled Structural Panels," Aerospace Structures Design Conference, Aug. 4-5, 1969. Published by Seattle Professional Engineering Employees Assn., Seattle, Washington.

18. Buchmann, O. A., and Walters, F. M., "Heat Transfer and Fluid Flow Analysis of Hydrogen Cooled Panels and Manifold Systems," NASA Contractor Report No. CR 66925, 1970.

19. Krusos, J. N., McConarty, W. A., and Anthony, F. M., "Fluid-Cooled Cabin Wall Design for Hypersonic Aircraft," Technical Report AFFDL-TR-68-109, Aug. 1968.

20. Goldstein, R. J., Eckert, E. R. G., Tsour, F. K., and Haji-Sheikh, A., "Film Cooling With Air and Helium Injection Through a Rearward-Facing Slot Into a Supersonic Air Flow," University of Minnesota, Heat Transfer Lab., TR No. 60, Feb. 1965.

21. Parthasarathy, K., and Zakkay, V., "Turbulent Slot Injection Studies at Mach 6," ARL 69-0066, April 1968, Aerospace Research Labs., Wright-Patterson Air Force Base, Ohio.

22. Cary, A. M., Jr., and Hefner, J. N., "Film Cooling Effectiveness in Hypersonic Turbulent Flow," Proposed Technical Note for AIAA Journal with probable publication in 1970.

23. Bushnell, D. M., and Beckwith, I. E., "Calculation of Nonequilibrium Hypersonic Turbulent Boundary Layers and Comparison With Experimental Data," Presented at the AIAA Second Fluid and Plasma Dynamics Conference, San Francisco, California, June 16-18, 1969.

24. Ferri, A., Fox, H., and Hoydysh, W., "Active Cooling of Hypersonic Airplanes," NASA CR-66930, Jan. 1970.

25. Maltby, R. L., "The Development of the Slender Delta Concept," Aircraft Engineering, Mar. 1968, pp. 14-17.

26. Henderson, William P., "Studies of Various Factors Affecting Drag Due to Lift at Subsonic Speeds," NASA TN D-3584, Oct. 1966.

27. Polhamus, Edward C., "A Concept of the Vortex Lift of Sharp-Edge Delta Wings Based on a Leading-Edge-Suction Analogy," NASA TN D-3767, Dec. 1966.

28. "USAF Stability and Control Datcom," Douglas Aircraft Company, Inc., Oct. 1960, revised Nov. 1965.

29. Polhamus, Edward C., "Predictions of Vortex-Lift Characteristics Based on a Leading-Edge Suction Analogy," presented at the AIAA Sixth Annual Meeting and Technical Display, Anaheim, California, Oct. 1969.

30. Wentz, William H., Jr., and Kohlman, David L., "Wind Tunnel Investigations of Vortex Breakdown on Slender Sharp-Edged Wings," University of Kansas Center for Research, Inc., Report FRL 68-013, Nov. 1968.

31. Wilde, M. G., and Cormerg, G., "The Aerodynamic Derivation of the Concorde Wing," RAS/CASI/AIAA, 11th Anglo American Aeronautical Conference, Sept. 8-12, 1968, London, England.

TABLE I.- ESTIMATED SPECIFIC IMPULSE FOR ENGINE CONCEPT OF FIGURE 6(b)

$\eta_{inlet, K.E.} = 0.98$, $\eta_{combustion} = 0.95$,
$\eta_{nozzle, K.E.} = 0.98$, $q_\infty = 350$ lb/sq ft,
$\varphi = 0.75$

V, ft/sec	I_{sp}, seconds
4000	3530
5000	3260
6000	2850
7000	2440
8000	2005
9000	1710
10000	1410

TABLE II.- HEAT CAPACITY AVAILABLE IN HYDROGEN FUEL FOR TYPICAL COOLED MATERIALS AND ASSUMED WORKING TEMPERATURES

(Assumed H_2 Inlet Temperature to Heat Exchanger, -360° F)

Material	Working temp., T_w	H_2 outlet temp.	Maximum available heat capacity	% of Max. avail. for engine cooling
Aluminum alloy	200° F	150° F	1785 Btu/lb	26
Titanium alloy	500° F	400° F	2660	39
Rene '41 (engine ducting)	1800° F	1600° F	6850	100

TABLE III.- FUEL HEAT SINK UTILIZATION FOR MACH 6 CRUISE, L/D = 4.2

Vehicle of Figure 10

	Titanium alloy (unshielded)	Aluminum alloy (33 percent shielding)
Airframe		
Required	21 percent	11 percent
Available	39	26
Engine		
Required	32	32
Available	61 to 79	74 to 89
Total		
Required	53	43
Available	100	100

TABLE IV.- COMPARISON OF THERMAL RESTRICTIONS FOR COOLED AND UNCOOLED HYPERSONIC AIRCRAFT

	Uncooled (hot structure)	Cooled (exposed structure)
Wing planform	Highly swept delta	Unrestricted
Wing loading	Corridor	Unrestricted*
Leading edges	Blunt	Unrestricted, can be sharp
Surface	Thermal warpage Heat shield joints and irregularities over major fraction of vehicle at Mach 8	No warpage. Smoothness otherwise comparable to today's aircraft
Vertical tail locations	Low heating locations only (e.g., upper surface wing tip region)	Unrestricted
Maneuver restrictions		
Power on	Radiation-cooled structural temperature limits	Unrestricted
Power off	Ditto	Unrestricted if incremental fuel consumption for cooling is permitted

*Partial heat shielding would impose some restrictions.

TABLE V.- WEIGHT, DRAG, AND PAYLOAD CHANGES FOR COOLED AIRCRAFT

(Config. of Fig. 10 with assumed gross weight = 600,000 lb, payload = 42,000 lb)

	$W_{cooled} - W_{hot}$ (lb)	Equivalent payload change (lb)
Basic structure	-22,270	22,270
Cooling system	16,100	-16,100
Heat shields	-3,912	3,912
Insulation	-10,532	10,532
Cabin, -5270		
Landing gear, -2422		
Tanks, -2840		
Other items not estimated, - ?		
	$D_{cooled} - D_{hot}$	
Leading-edge drag	-1,000	2,000
Surface-irregularity drag	-1,000	2,000
Specific impulse increase due to regenerative heat (2 percent)		4,800
	Net increase	29,414 lb
	(A 70-percent increase in payload)	

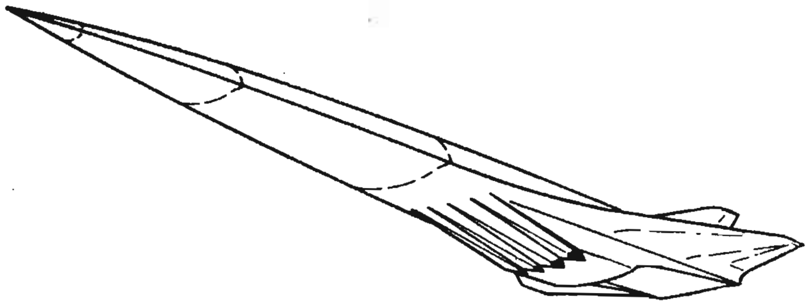


Figure 1.- Cruise vehicle concept.

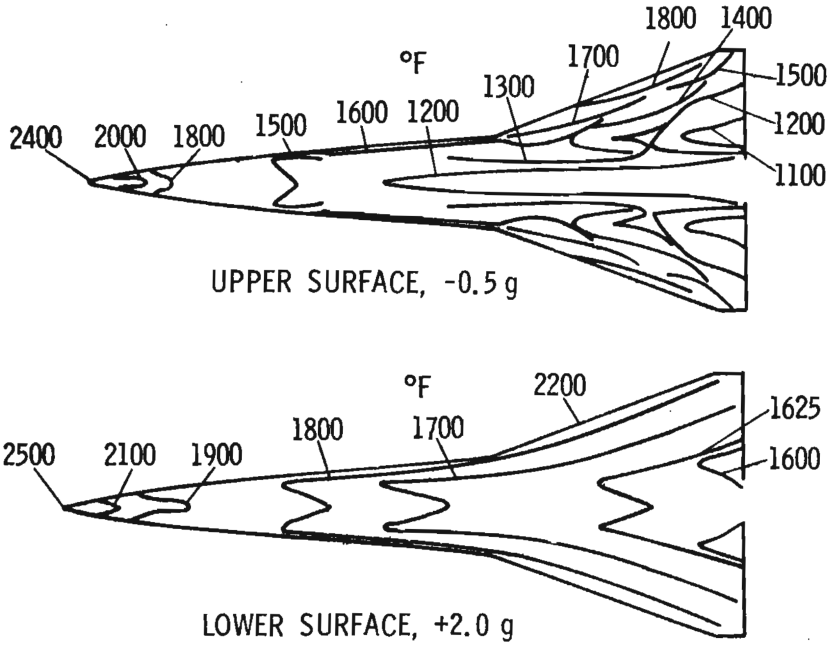


Figure 2.- Typical temperature profiles for radiation-cooled hot structure without heat shields. Internal radiation, external radiation, and leading-edge conduction and heat-sink effects are included. Mach 8, altitude 90,000 ft.

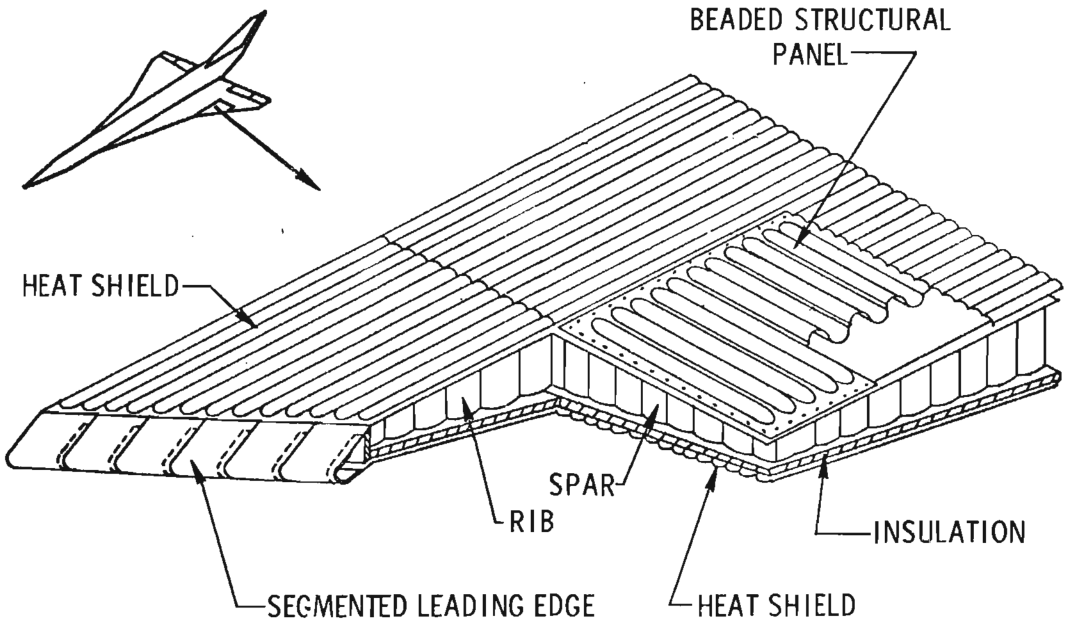


Figure 3.- Typical radiation-cooled hot wing structure.

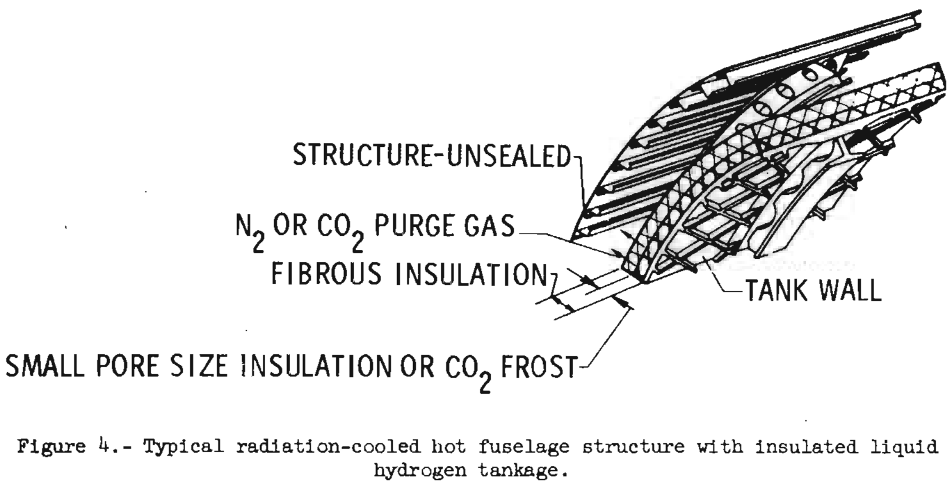


Figure 4.- Typical radiation-cooled hot fuselage structure with insulated liquid hydrogen tankage.

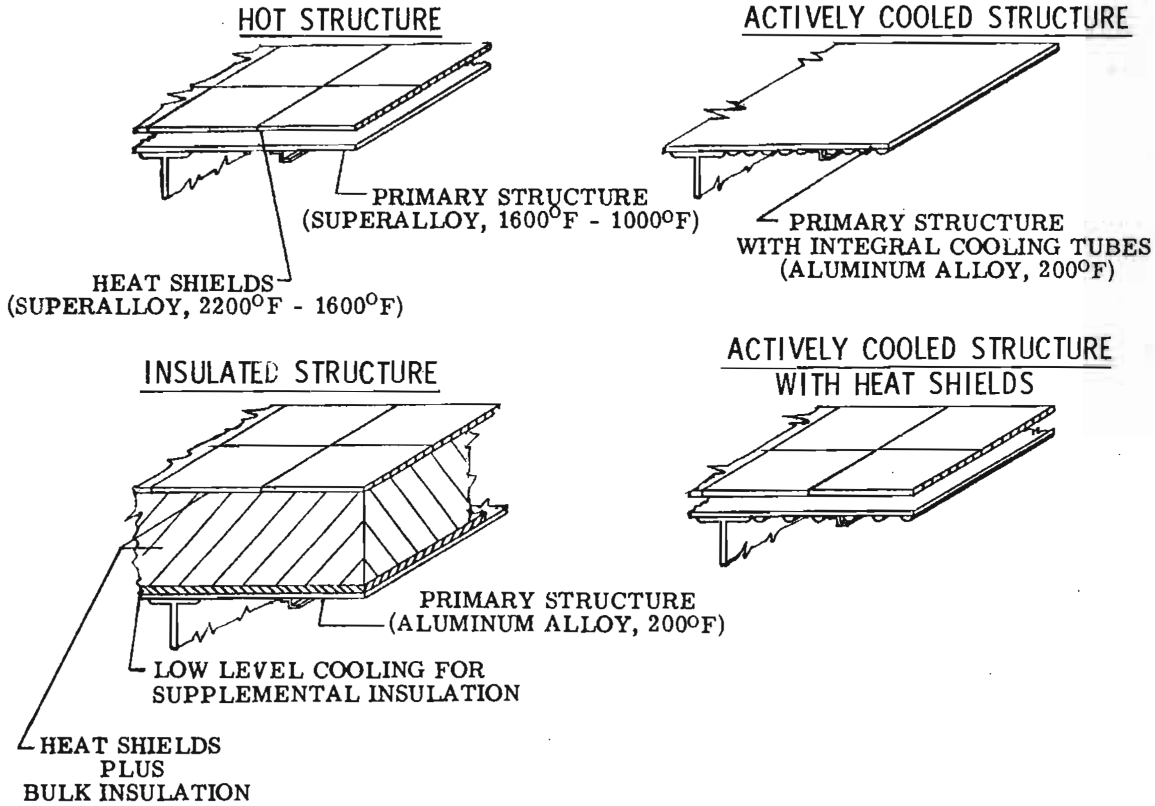
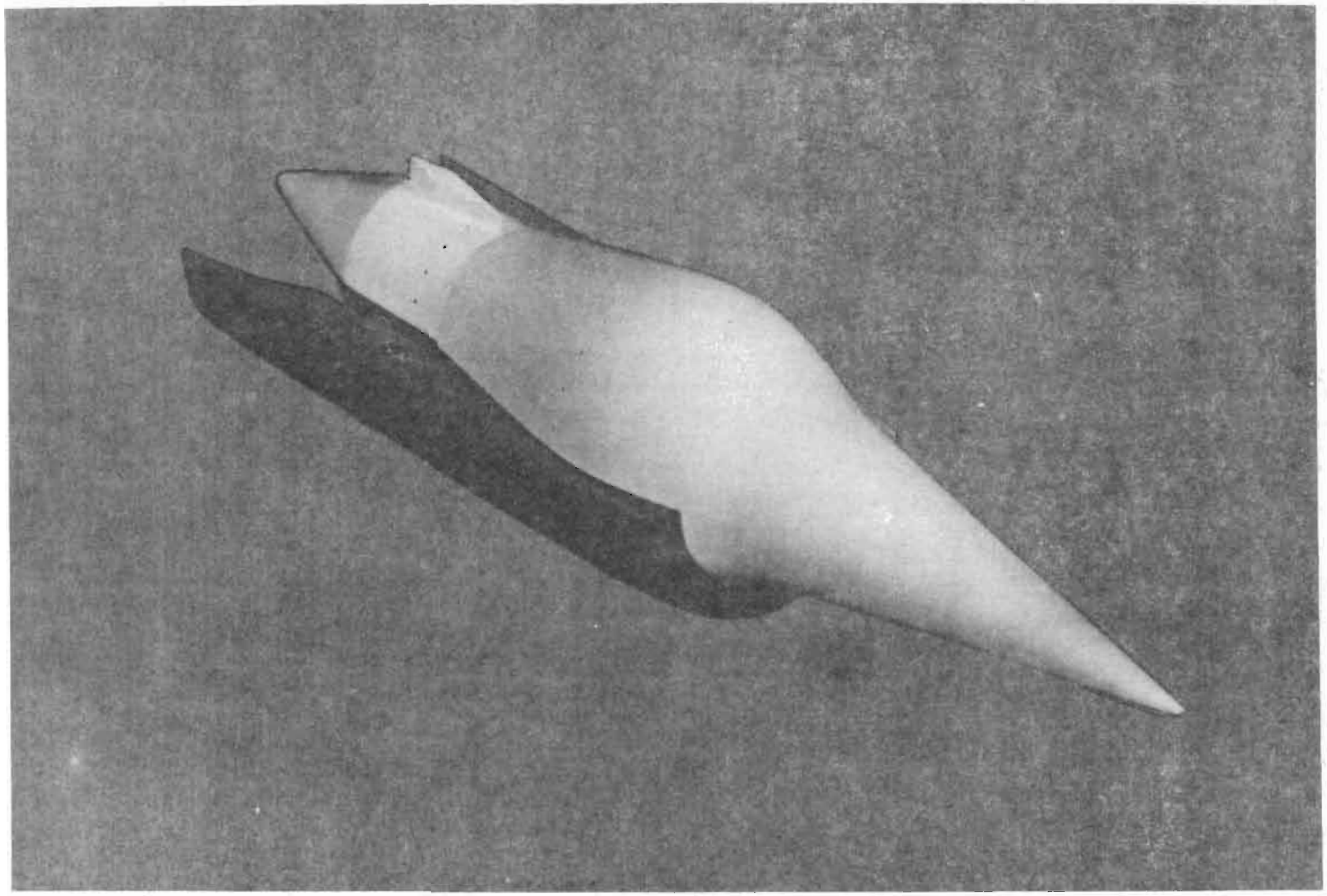
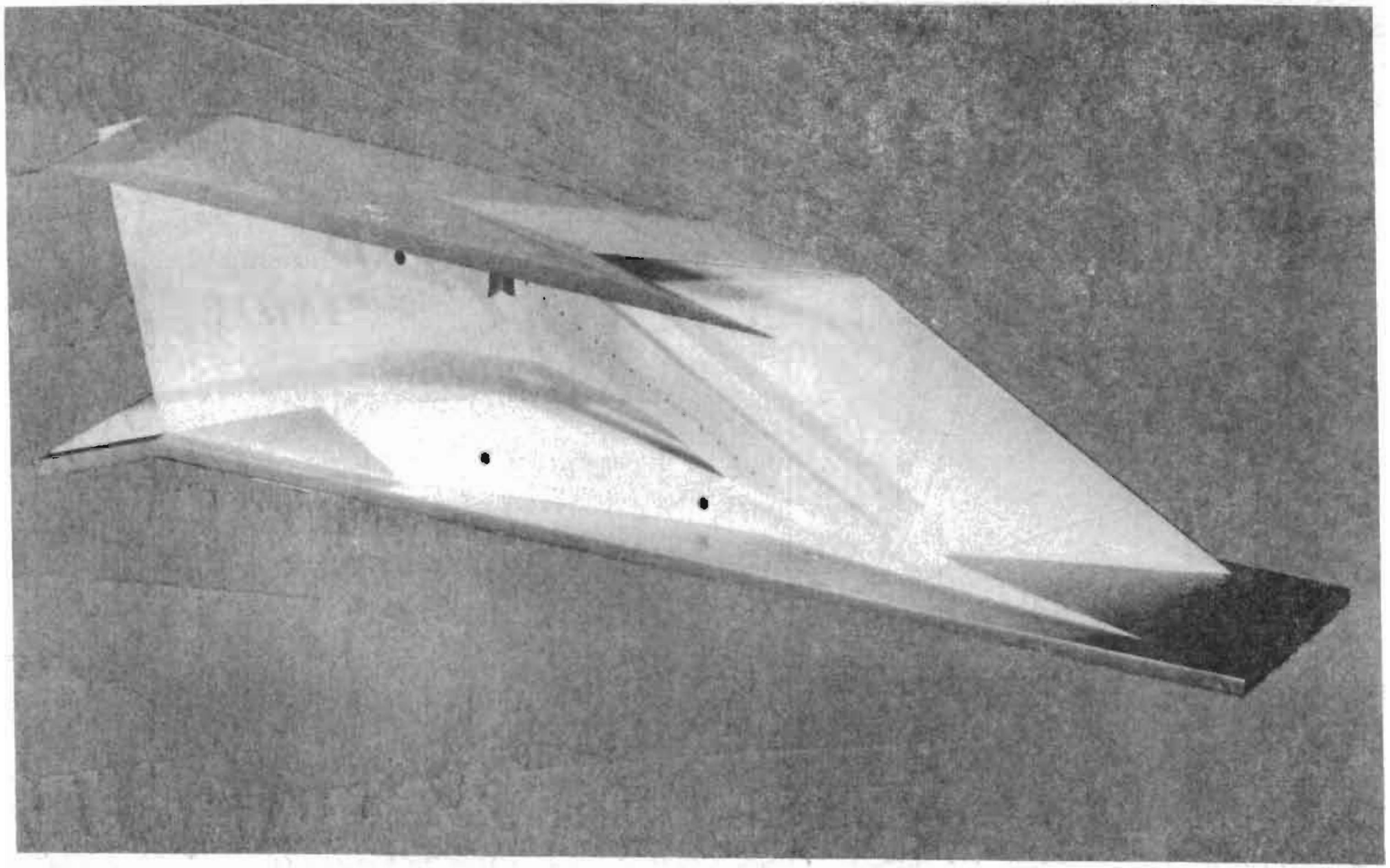


Figure 5.- Comparison of hot and cooled structural concepts.



(a) Typical axisymmetric design with annular flow passage.

Figure 6.- Scramjet engine concepts.



(b) Three-dimensional design.

Figure 6.- Concluded.

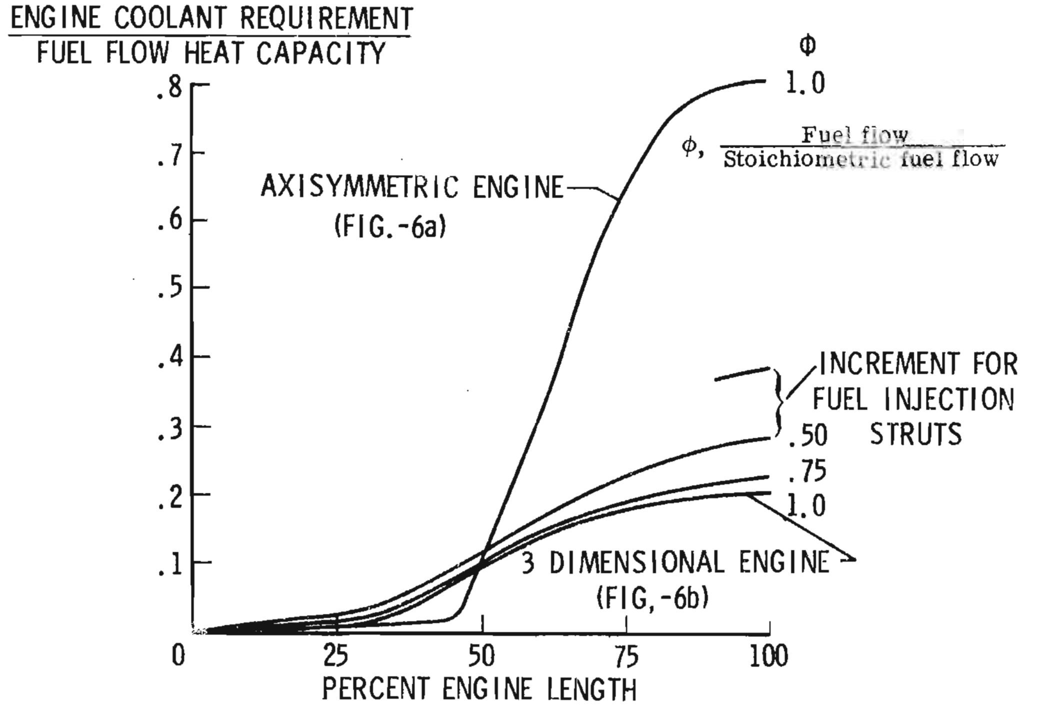
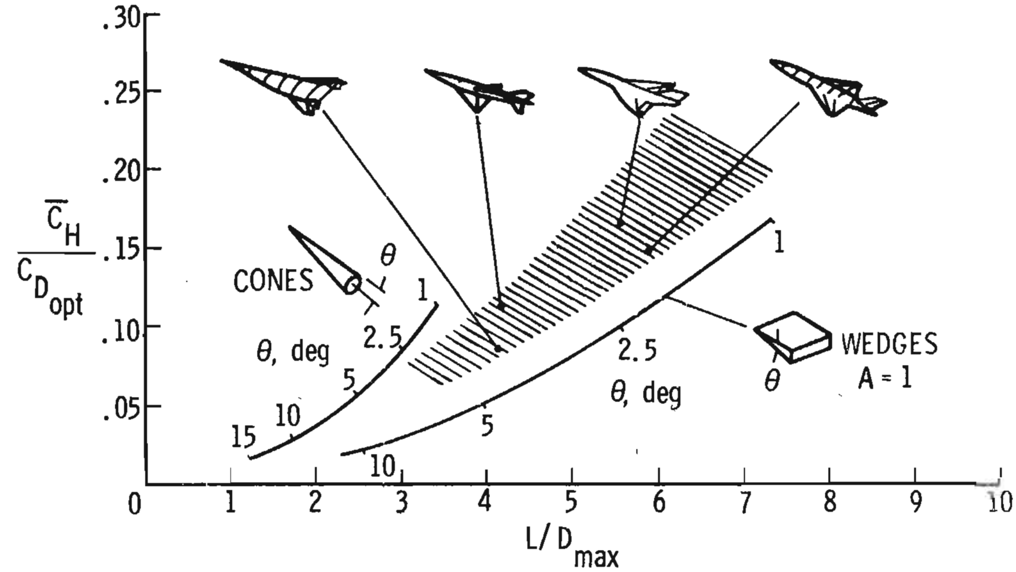
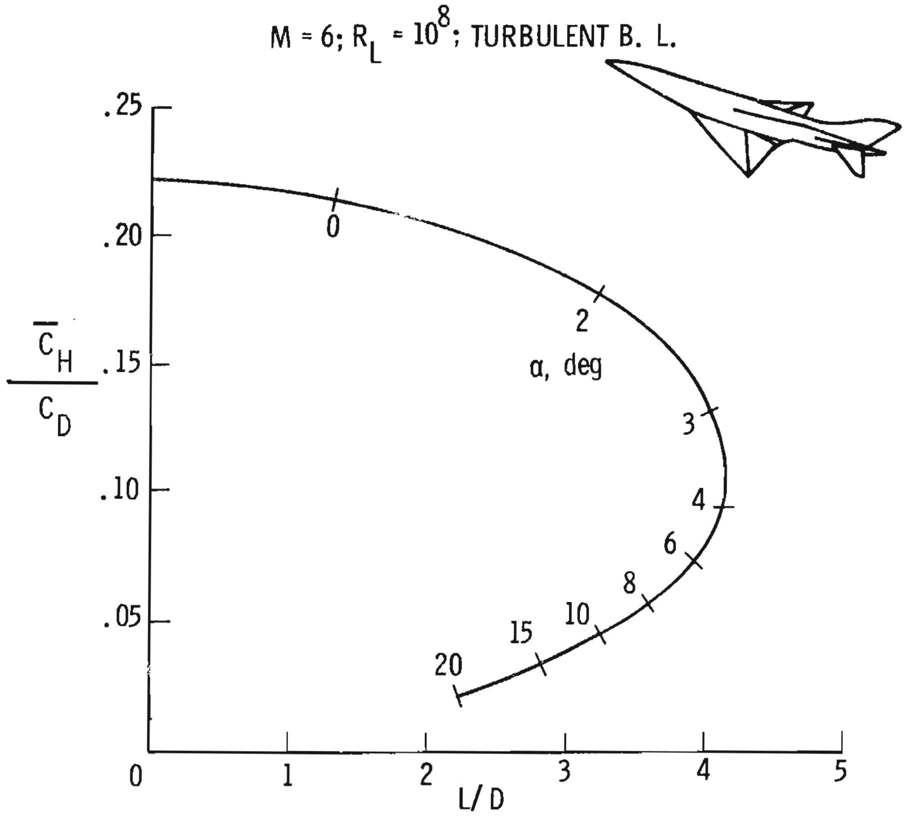


Figure 7.- Comparison of coolant flow requirements for engines of Figures 6(a) and 6(b). Mach 6, capture area 39.1 sq ft, altitude 112,000 ft, supersonic combustion.



(a) Effect of shape and L/D.

Figure 8.- Heat-transfer/drag-ratio evaluations. Mach 6; R_L, 140×10^6; turbulent boundary layer.



(b) Effect of vehicle attitude.

Figure 8.- Concluded.

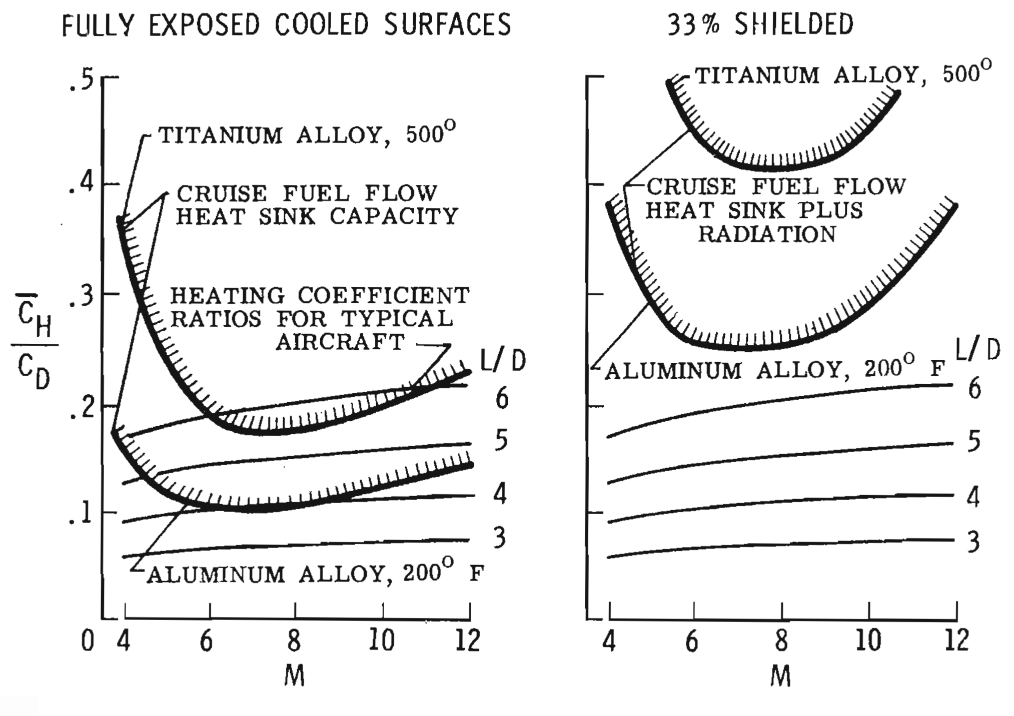


Figure 9.- Prospects for fuel-cooled aircraft. Maximum permissible heat-transfer/drag ratios compared with values for typical aircraft.

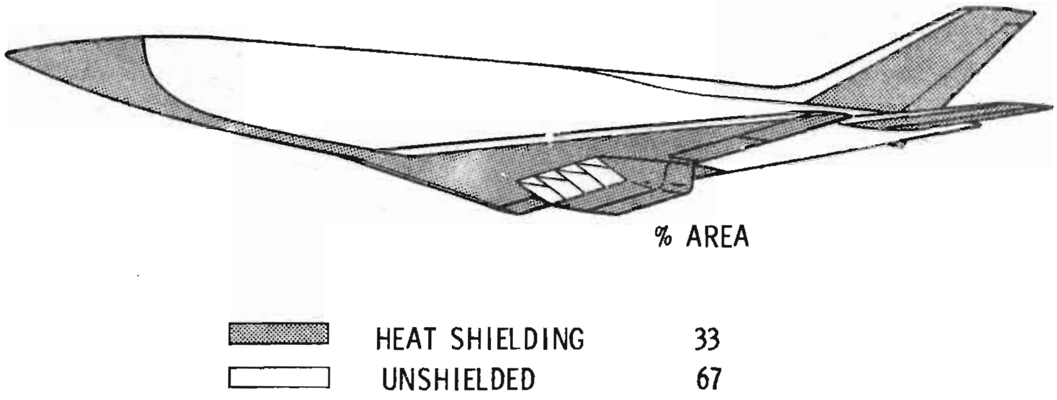


Figure 10.- Cooled Mach 6 hypersonic transport used for study purposes.(15) "Heat shielding" case has simple shields on all areas where radiation equilibrium temperature exceeds 1000° F except for 5 ft-wide exposed leading edges.

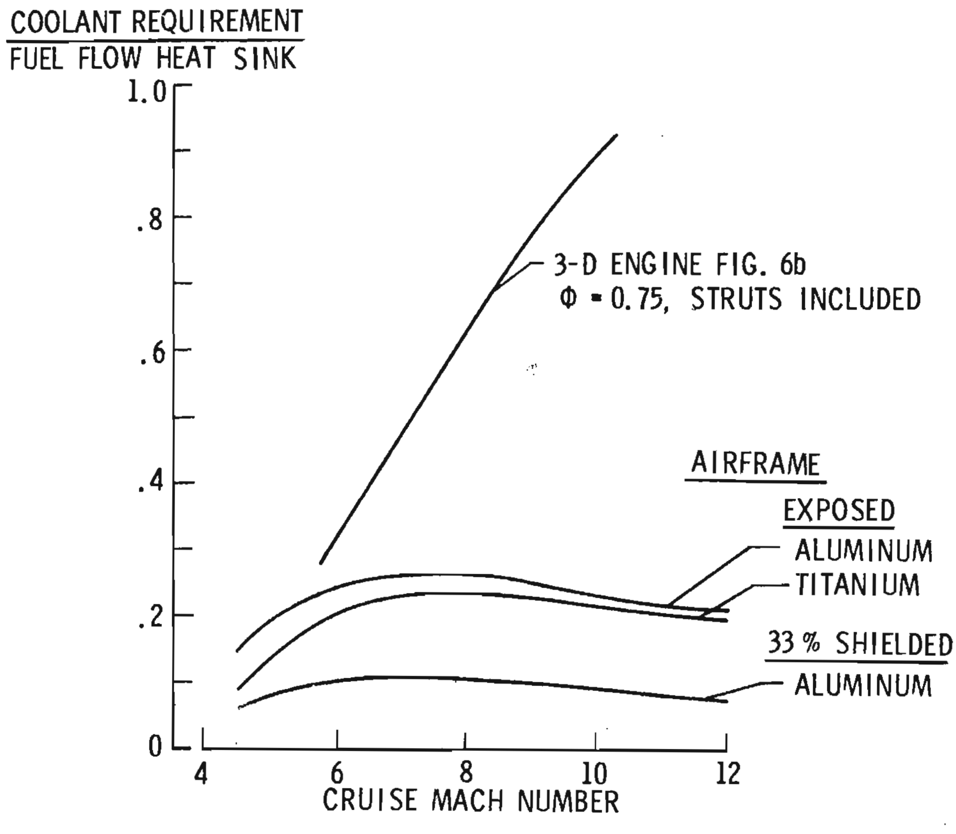


Figure 11.- Comparison of engine and airframe coolant requirements.

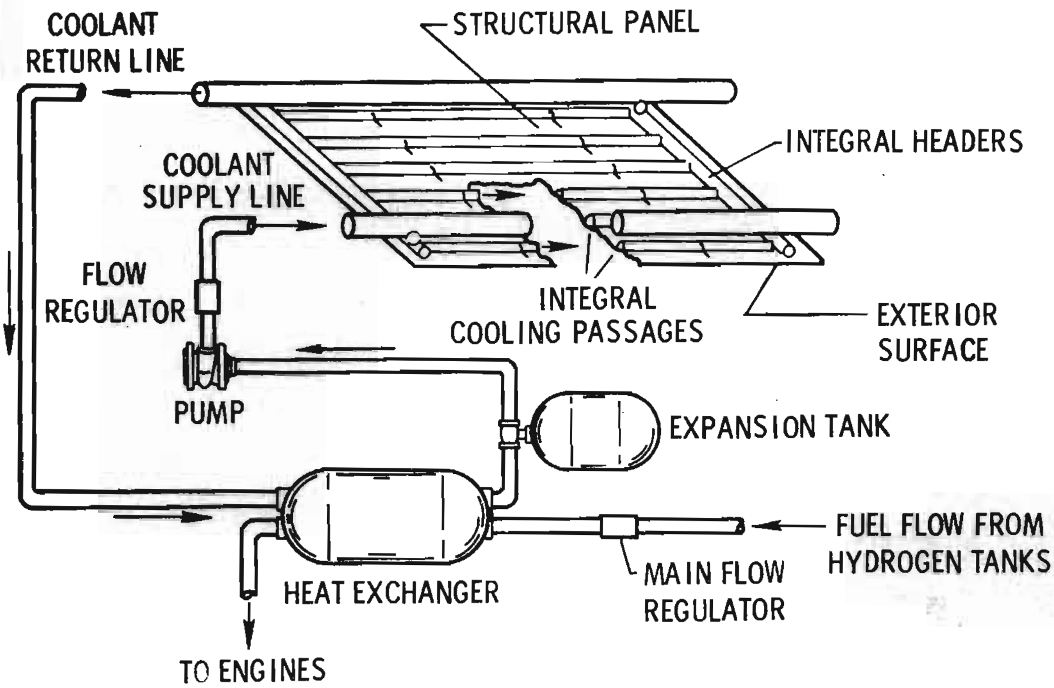


Figure 12.- Scheme of liquid convective cooling system.

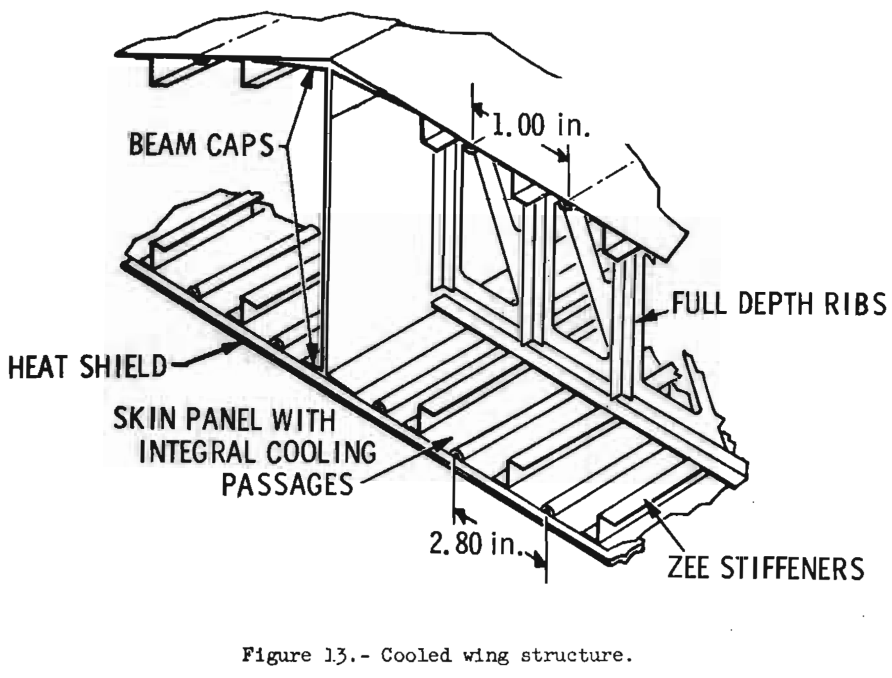


Figure 13.- Cooled wing structure.

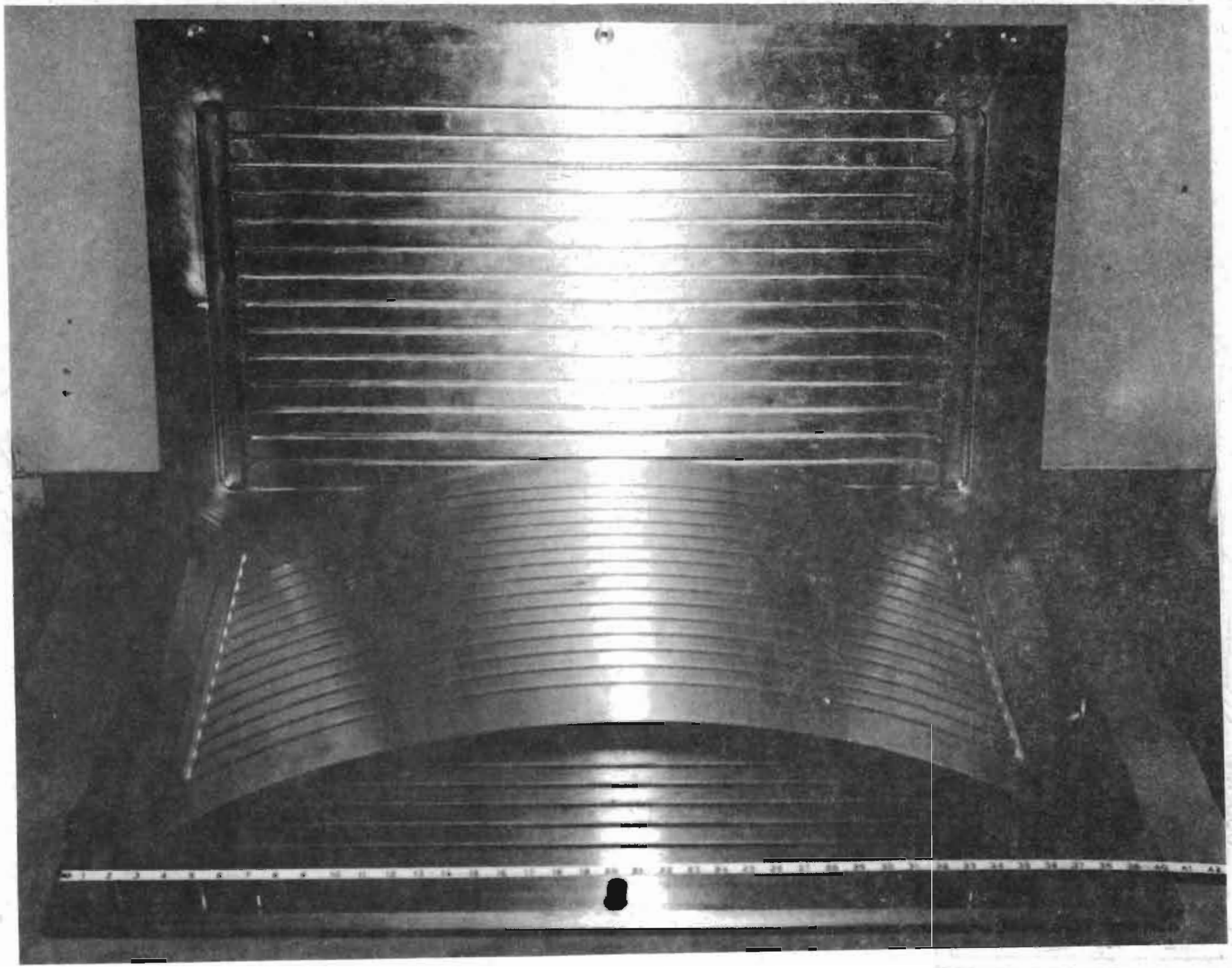


Figure 14.- Fabrication technique for cooled wing skin structure. (From U.S. Air Force-sponsored research, reference 10.)

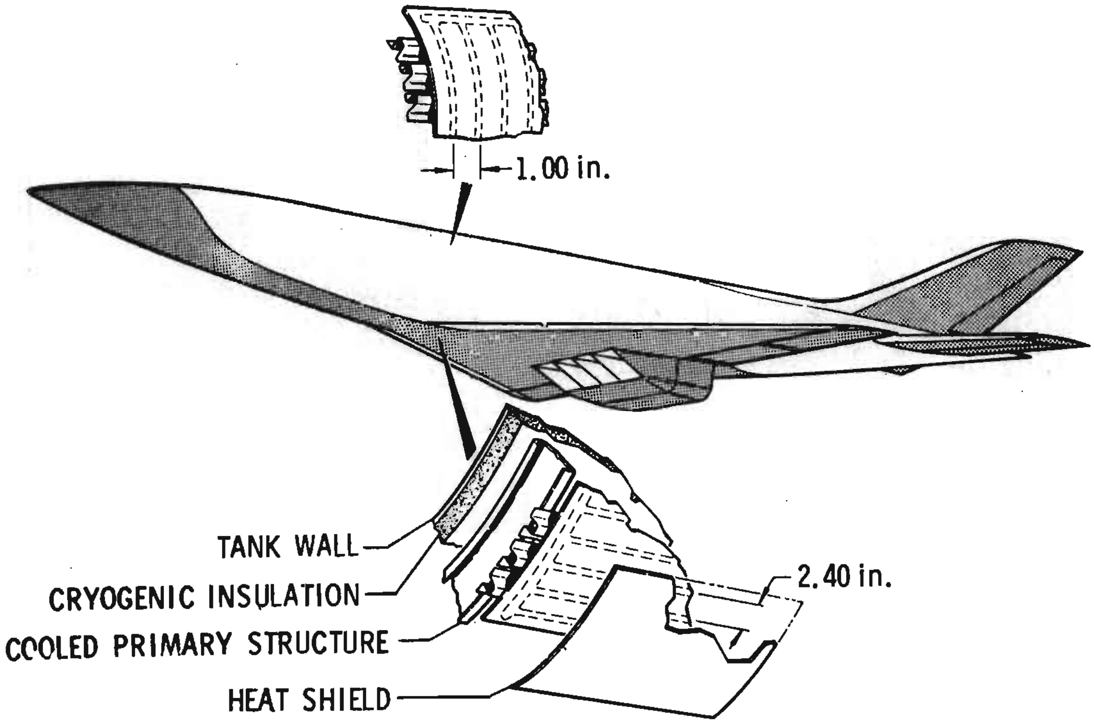


Figure 15.- Cooled fuselage structure.

EXAMPLE OF LARGE STRUCTURAL TEST

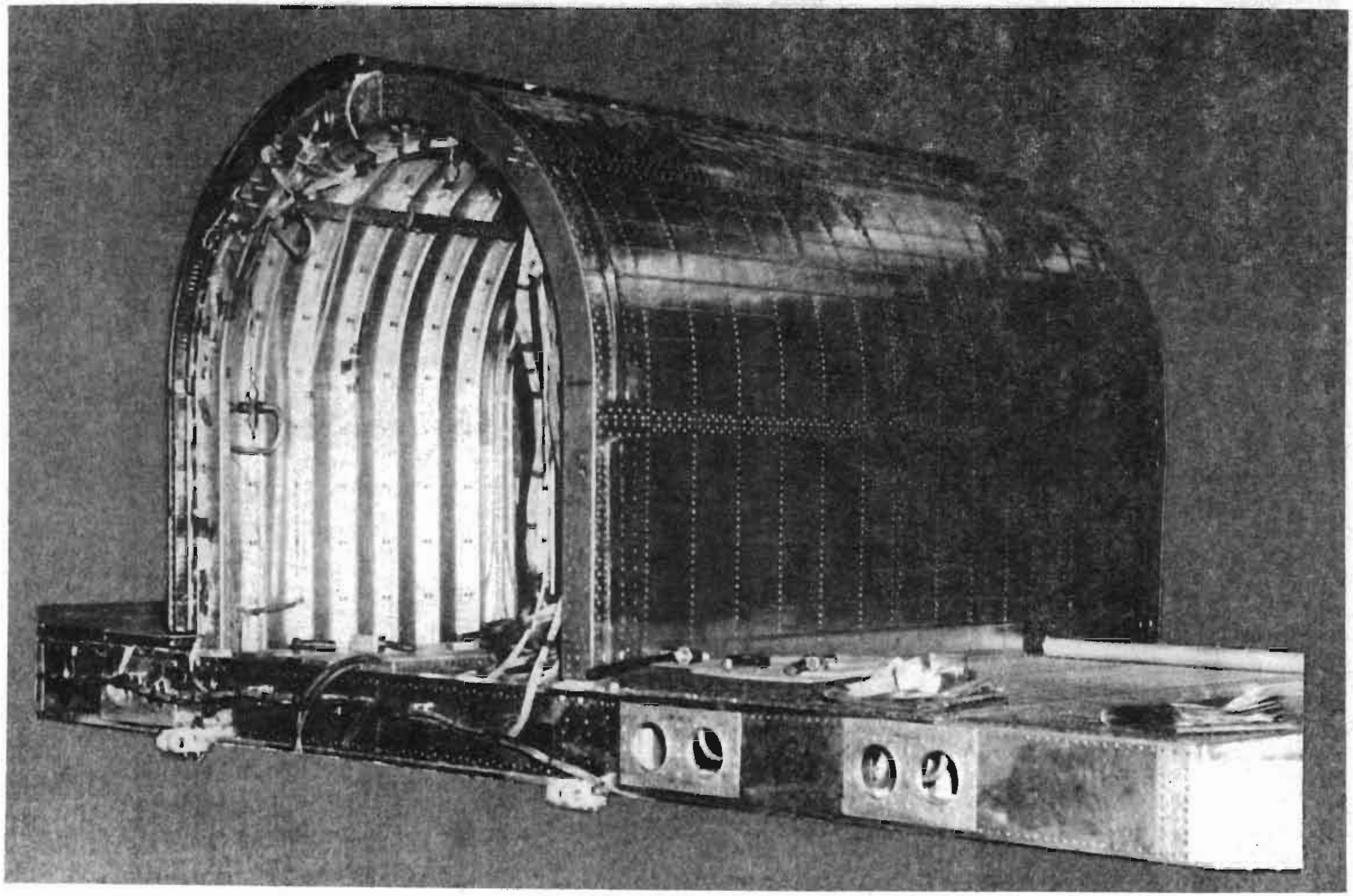


Figure 16.- Example of large structural test model with integral cooling tubes. (See refs. 10,11.)

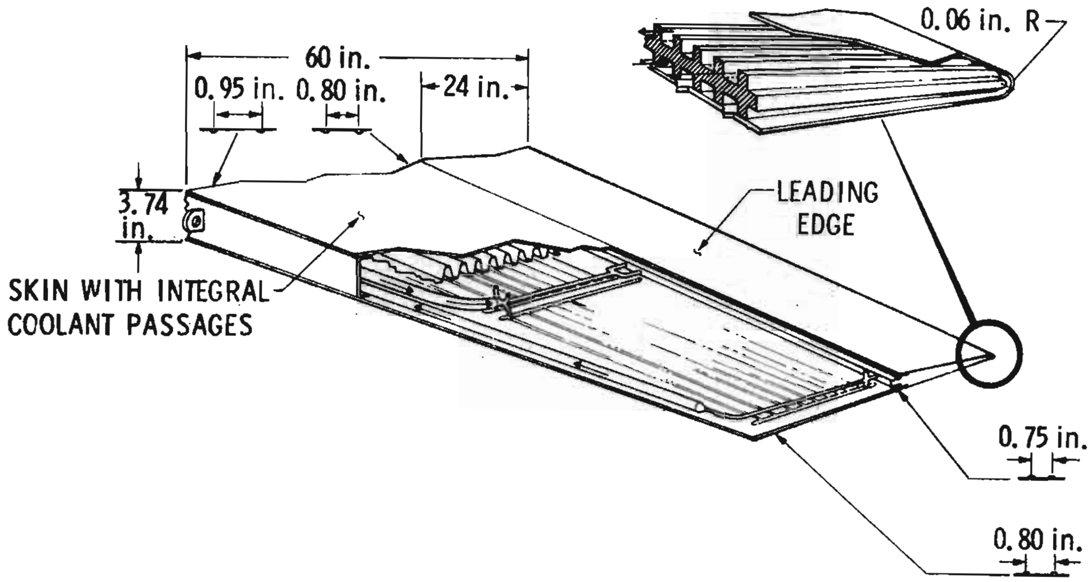


Figure 17.- Cooled leading-edge structure.

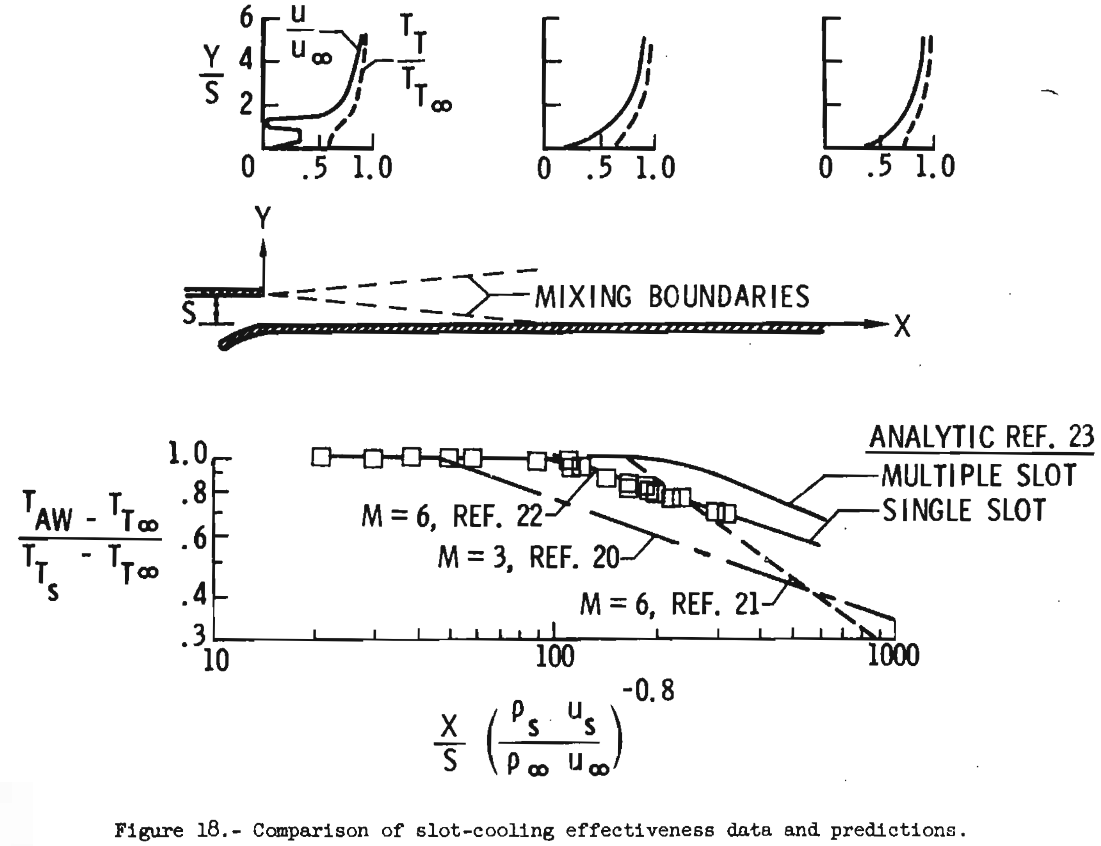


Figure 18.- Comparison of slot-cooling effectiveness data and predictions.

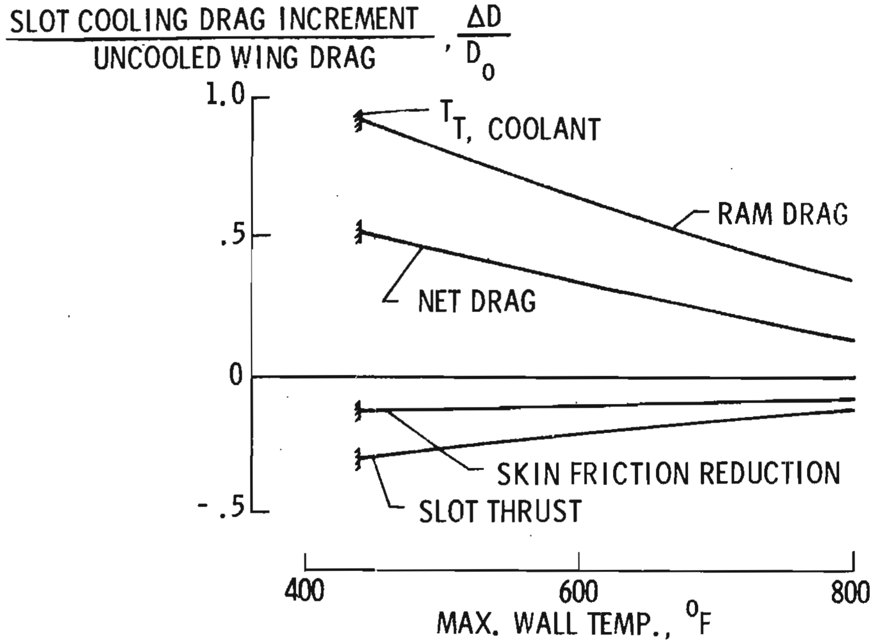


Figure 19.- Drag and thrust components associated with slot cooling. Fuel cooled heat exchanger system. Mach 6, $\alpha = 7.4^{\circ}$, altitude 105,000 ft.

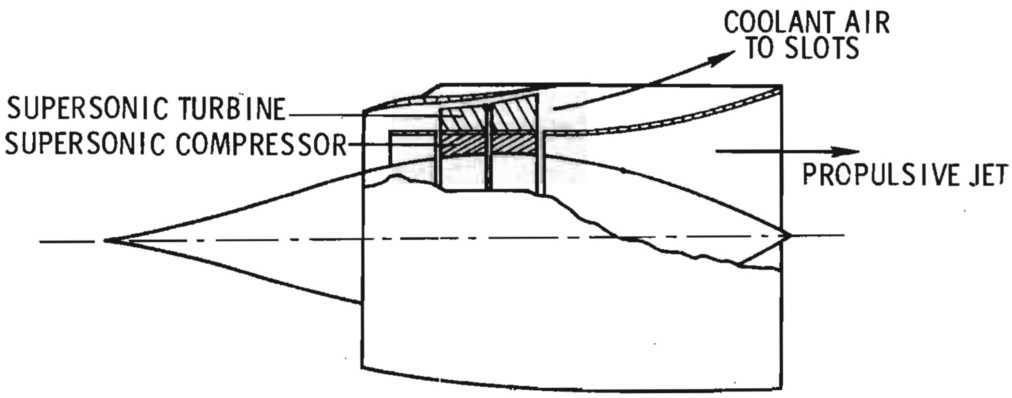


Figure 20.- Scheme of turbocooler (ref. 24).

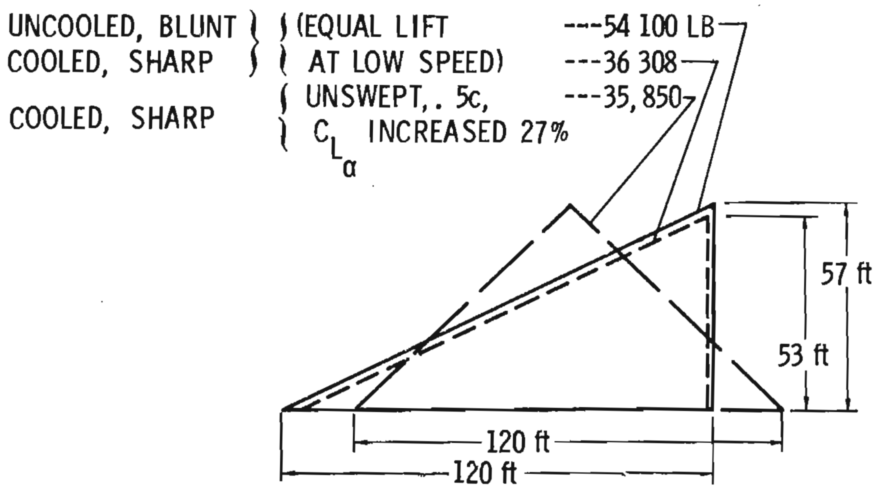


Figure 21.- Comparison of blunt uncooled superalloy and sharp-edged cooled aluminum alloy wings. (Wing weights include heat shields and wing cooling distribution system.)

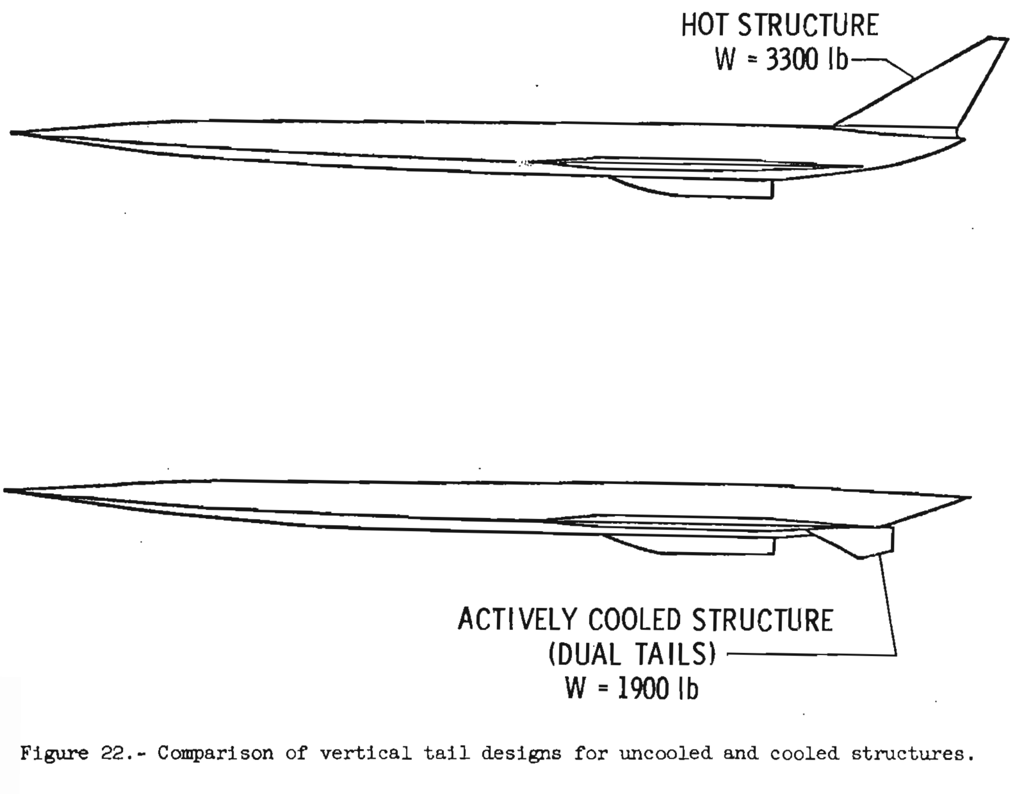


Figure 22.- Comparison of vertical tail designs for uncooled and cooled structures.