Performance Tradeoffs and Research Problems for Hypersonic Transports

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The performance of hydrogen-fueled, hypersonic transports was analyzed in a computer study. The purpose of the study was to define the over-all mission capabilities of Mach 4 to 8 transports operating from conventional runways and operating within reasonable constraints set by structural, aerodynamic, and propulsive considerations. Tradeoff studies were conducted in an effort to define the vehicle characteristics which resulted in maximum payload fraction and also to indicate the areas in which additional research will be needed. The effects of variations in cruise Mach number, climb trajectory, engine sizing, inlet pressure recovery, regenerative cooling requirements, wing loading and aspect ratio, and fuselage fineness ratio were evaluated by relating them to the resulting changes in payload fractions. The results of the tradeoff studies indicated that hydrogen-fueled, hypersonic transports cruising at a Mach number of 6 can attain payload fractions (including reserve fuel) of about 20% of the vehicle's gross weight at ranges of about 5000 naut miles. However, before the technical feasibility of the hypersonic transports can be confirmed, additional analytical research and experimental investigations will be required in several areas, notably those involving the inlets, engines, and thermal protection systems.

Introduction

THE purpose of this paper is to examine the mission capability of hydrogen-fueled, hypersonic transports and to discuss some of the major technical approaches that may be used with such vehicles. It is the intention of this paper to indicate only the technical feasibility and capabilities of hypersonic transports. The equally important problems involving economic feasibility and operational problems have not been considered, since the study of these problems is thought to be premature until technical feasibility has been established.

Hydrogen-fueled hypersonic transports, besides having an obvious speed advantage, are also capable of long-range flight as can be shown with the aid of the Breguet equation. This equation may be written[1]

$$R = \overbrace{(IV\{L/D/[1 - (V/V_s)^2]\})}^{\text{Breguet factor}} \ln(W_i/W_f)$$

where R is the range; I, V, and L/D are the cruise values of specific impulse, velocity, and lift-drag ratio, respectively; W_i and W_f are the initial and final values of weight; and V_s is the local satellite speed. If structural considerations are neglected temporarily, an indication of cruise capability can be obtained from the factor contained in the boldface parentheses. This Breguet factor is shown in Fig. 1 for present-day subsonic jet aircraft, the supersonic transports currently proposed, and the type of vehicles considered in this study. The speed ranges and performance levels indicated in Fig. 1 are intended to serve only as guides for a general comparison and are not intended to represent limits. In this connection, the curve for supersonic transports could be extended to higher Mach numbers, but difficult propulsion-system cooling problems would arise. Likewise, the hydrogen-fueled ramjet systems assumed for the hypersonic-transport curve are also limited by cooling requirements at the higher speeds. With the subsonic burning ramjets considered here, it appears that cooling requirements may be excessive at Mach numbers above 8. For this reason, and since the current plans for the supersonic transport cover Mach numbers up to about 3, the Mach number range considered in this study is from 4 to 8. In this range, the cruise efficiencies indicated for the hypersonic transports are quite attractive. Although this type of comparison is by no means conclusive, it does indicate that hypersonic cruise vehicles do warrant analysis as long-range transports.

Since hydrogen is the key to the good efficiency shown in Fig. 1, the characteristics of this fuel that pertain to its use in hypersonic aircraft are shown in Table 1. Although these characteristics are well known, a comparison with conventional hydrocarbon fuels indicates the reasons that hydrogen is especially suited for hypersonic cruise vehicles. The first and most important property of hydrogen is its energy content/unit weight (heat of combustion), which is roughly $2\frac{1}{2}$ times that of hydrocarbon fuels. This factor accounts for most of the advantage in cruise efficiency shown in Fig. 1. A second important factor is the cooling capability (specific heat), since regenerative cooling of internal surfaces of the propulsion system is normally required at high speeds. The specific heat of hydrogen is about seven times that of hydrocarbons, and it is this factor that makes it possible for a hydrogen-fueled vehicle to operate at relatively high speeds and not require a coolant flow that exceeds the fuel flow required for propulsion. An important disadvantage of hydrogen is its liquid density, which is only about $\frac{1}{10}$ that of conventional

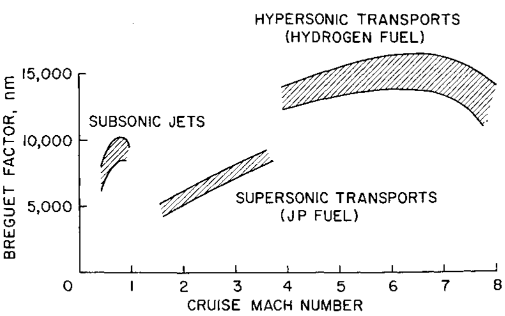

Fig. 1 Cruise efficiencies of subsonic, supersonic, and hypersonic transports.

Presented as Preprint 64-605 at the AIAA Transport Aircraft Design and Operations Meeting, Seattle, Wash., August 10–12, 1964; revision received November 20, 1964.

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Table 1 Comparison of fuel characteristics

Property	Hydrocarbon	Hydrogen
Heat of combustion, Btu/lb	19,000	51,000
Specific heat, Btu/lb/°F	0.46	2.7–3.7
Liquid density, lb/ft³	51	4.5
Boiling temperature, °R (1 atm)	820–915	36

fuels. As a result, hydrogen-fueled vehicles have large volumes, a characteristic that is particularly unattractive in hypersonic flight. Another disadvantage is the boiling point of liquid hydrogen which is 36°R. The fuel must therefore be carried in cryogenic tanks, and these tanks not only add weight to a vehicle but also increase its volume.

From the comparisons shown in Table 1, it is apparent that hydrogen has physical and thermodynamic characteristics that differ greatly from hydrocarbon fuels. These differences have large effects on the performance and design tradeoffs of transport vehicles. It is the purpose of this paper to explore some of these tradeoffs and to obtain some preliminary estimates of the performance and other characteristics of near-optimum vehicles. The methods and approaches used in the analysis will be discussed first, and then the results of the tradeoff studies will be presented.

Methods

A preliminary mission analysis of transport vehicles involves the determination of the values of vehicle parameters that yield maximum payload at a given range. Most parameters influence both the structural weight and the fuel weight of the vehicle, usually with opposing effects on the payload. To determine these effects, the present analysis utilized a mathematical-model technique in which computations of structural weight, aerodynamic performance, and propulsion-system performance were based on vehicle geometry, sizing, and trajectory parameters. These computations were performed on a digital computer in conjunction with a trajectory computation. The integrated computation simulated the flight of the vehicle and properly related the separate problems of structural weight and mission fuel consumption. The mission ground rules used in the study specified takeoff and landing from present-day runways and climb and descent within restrictions placed by noise and sonic-boom considerations.

Analysis Model

The vehicles chosen for analysis had large volume fuselages, triangular wings, and suitable tail surfaces. Figure 2 indicates one arrangement considered. The aerodynamic lift and drag were estimated[2–5] primarily from the vehicle geometry with the friction drag including the effects of the trajectory. The structural weights and heat-protection weights were estimated[6,7] from the vehicle geometry, imposed loads, and temperature environment. The airframe skin and leading edges were considered to be cooled by radiation, but regenerative cooling was considered for the internal surfaces of the propulsion system. The propulsion-system inlet was placed under the wing to take advantage of the higher density and lower Mach number of the flow in the wing compression field, and a variable-geometry, mixed-compression inlet was used. The pressure recovery was estimated but was also varied as a parameter. The engines were hydrogen-fueled turboramjets capable of conversion from turbojet to ramjet operation in the supersonic speed range. The engine and exhaust-nozzle performance and weight were derived from manufacturers' estimated data. The exhaust-nozzle flow was assumed to be in equilibrium.[8]

Tradeoff studies were performed to determine the effects of varying thrust loading, inlet performance, fuselage fineness

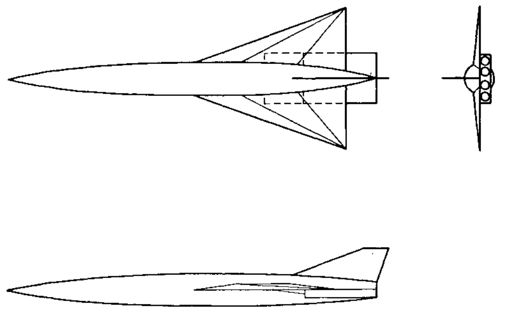

Fig. 2 Typical configuration.

ratio, wing loading, aspect ratio, cruise Mach number, and range. The results of these studies will be presented in terms of fractions of the vehicle gross takeoff weight with the objective being to obtain a maximum fraction for payload and fuel reserves. All vehicles had gross takeoff weights of 500,000 lb and fuselage volumes of 71,500 ft³. Unless otherwise shown, the cruise Mach number was 6, and the range was 5600 naut miles.

Trajectory Considerations

Some of the considerations that affect vehicle trajectory are shown in Fig. 3. The desired cruise altitudes are indicated by the dashed curve in Fig. 3, and the problem is to select a climb trajectory to the cruise altitude that maximizes the payload weight for the specified mission. Since many components of the vehicle are sized during climb, and since up to 40% of the fuel is consumed in this phase of flight, selection of the best available climb trajectory is important. Climb trajectories are affected by a variety of constraints. The trajectories used in this study followed the lines defining the various constraints, since trajectories at higher altitudes resulted in increased fuel consumption and correspondingly lower payloads. The first constraint indicated in Fig. 3 is that due to sonic-boom limitations. Typical curves for overpressures of 2 and 3 psf are shown. The exact location of these curves is affected by many factors[9]; for example, if the vehicle is shaped to minimize sonic boom, the lower curve can serve as an approximation of a 2-psf boundary.[10] As the airplane accelerates to high supersonic speeds along a line of constant overpressure, the dynamic pressure increases until structural considerations dictate some limiting value. The trajectory then follows this maximum allowable dynamic-pressure line until another consideration becomes important. At flight Mach numbers of approximately 5, the internal pressures in the propulsion system increase rapidly causing a corresponding increase in the structural weight of the turboramjet engines. As a result, the trajectories usually are restricted by lines of constant internal pressure. Duct pressures are also dependent on inlet pressure recovery and

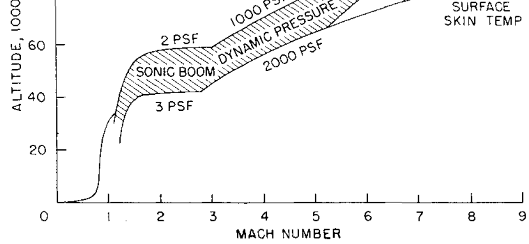

Fig. 3 Constraints on climb trajectories.

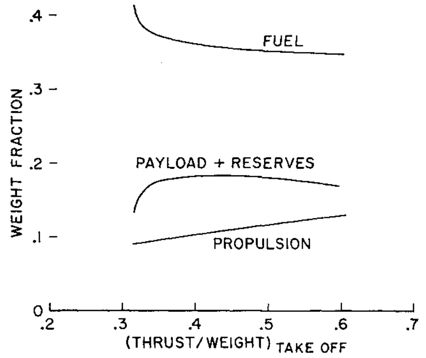

Fig. 4 Effect of turbojet size on mission performance.

on wing angle of attack which affects both the Mach number and pressure field in which the inlet is located. Thus the constant duct-pressure lines shown in Fig. 3 should be considered only as examples. The next constraint is encountered at higher Mach numbers where aerodynamic heating becomes important. Typical curves for equilibrium temperatures of 1200° and 1600°F are shown. These temperatures are, of course, dependent on such things as angle of attack and distance from the leading edge. In summary of Fig. 3, an optimum climb trajectory will depend upon numerous constraints and tradeoffs involving considerations of structures, aerodynamics, and propulsion-system performance. The region bounded by the curves in Fig. 3 indicates reasonable operating conditions for the type of vehicles under consideration. Except where noted, all vehicles in the present studies followed the trajectory defined by a sonic-boom overpressure of 3 psf, a dynamic pressure of 2000 psf, and a duct pressure of 200 psi.

Engine

The sizing criteria for turboramjet engines are somewhat different from those for turbojet engines since each of their two components, the turbojet and the ramjet, can be sized independently. In the present studies the inlet, ramjet, and exhaust system were sized to allow stoichiometric ramjet operation at the cruise Mach number. The turbojet was then sized independently to provide adequate transonic thrust.

The tradeoff in sizing the turbojet is presented in Fig. 4. Vehicle thrust-to-weight ratio at sea-level static conditions was used as a measure of turbojet thrust loading, and the total mission fuel consumption, total propulsion-system weight (inlet, engine, and exhaust nozzle), and resulting payload and reserves weight are shown as functions of that parameter. As the turbojet thrust loading was decreased, the engine weight decreased while the fuel consumption increased. This latter increase was gradual at first but became more rapid as the lower thrusts led to long acceleration times at transonic speeds which, in turn, caused excessive fuel consumption. At a loading of about 0.32, the vehicles were unable to accelerate through the transonic region. Although a thrust loading between 0.4 and 0.5 gave maximum payload fraction, these results indicate that the vehicles were not particularly sensitive to turbojet thrust loading. This result differs from that encountered by most other aircraft, for example, the supersonic transports.[11] For most aircraft, the critical sizing condition (namely, when the acceleration margin is least) determines the complete propulsion-system size, including that of the inlet and exhaust system. In comparison, the sizing requirement encountered at transonic speeds by the vehicles considered here was met by adjusting only the turbojet size. The rest of the propulsion system was sized for the cruise condition. Consequently, the weight penalties incurred by increasing the transonic or takeoff thrust loadings were not so severe as have been experienced in the past.

The results in Fig. 4 are for a sonic-boom overpressure of 3 psf. The use of a lower limit on sonic-boom overpressure has also been studied. To maintain a lower overpressure, of course, it was necessary to increase the turbojet engine size and weight at the expense of payload. In this respect, hypersonic transports share a common problem with the supersonic transports. For the reasons just cited, however, the effects of sonic-boom restrictions were not quite as important to the over-all mission performance for hypersonic transports as they are for supersonic transports. For example, the effect of altering the climb trajectory to correspond to a maximum overpressure of 2 psf, rather than to 3 psf, was investigated. It was found that, if the turbojet was properly sized in both cases, the weight fraction for payload and reserves was reduced from 0.18 to 0.15. The sonic-boom overpressures generated during cruise were also estimated and found to be approximately 1 psf or less.

Inlet

In the discussion of Fig. 3, a tradeoff was indicated as a result of the opposing effects of inlet pressure recovery on engine performance and of internal pressure on propulsion-system weight. In order to examine the effects on the transport mission more closely, several pressure-recovery schedules were used with a trajectory having a maximum dynamic pressure of 1500 psf. These schedules and the standard pressure-recovery schedule suggested by the military (MIL-E-5008B)[12] are shown in Fig. 5. The recovery schedules were designed so that the maximum internal pressure in the propulsion system was held constant at each of four levels from 150 to 300 psi. Transport missions (5600-naut-mile range) were analyzed for each pressure-recovery schedule, and weight fractions were determined. The computations were made for cruise Mach numbers of 6 and 8, but, because of wing flow-field effects, the Mach numbers at the inlet were about 5.0 and 6.3, respectively. The results presented in Fig. 6 show the effects on fuel-weight fraction and on propulsion-weight fraction. Although both effects are added together in Fig. 6b, internal duct pressure, indicated by the symbols, was responsible for the effects on propulsion-system weight, and pressure recovery was responsible for the effects on fuel weight. For cruise at Mach number 6, the sum of the propulsion-system weight and the fuel weight was not very sensitive to the two parameters because the two effects tended to cancel each other over the ranges studied. At Mach number 8, the fuel consumption was not very sensitive to pressure recovery at levels above 0.15; therefore, structural weight became the dominant factor. These trends are dependent on the structural-weight estimates used, and a different relationship between structural weight and internal pressure could influence the results. For this reason, the generality of these results must be examined for other engine systems. It does appear, however, that the attainment of high pressure recovery at high Mach numbers may not be so important as it

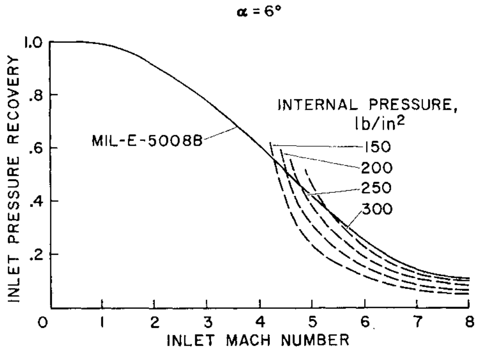

Fig. 5 Reduced pressure-recovery schedules.

is at lower speeds. This trend could be expected, since at hypersonic speeds the pressure ratio across the exhaust nozzle is already quite high, and further increases in pressure ratio do not result in greatly increased thermodynamic efficiency. In addition, to take advantage of the increased thermodynamic performance, the exhaust nozzle must provide a wider range of throat area variation, which usually results in some weight penalty.

Certainly the analysis of propulsion systems for hypersonic transports is a study in itself. Since the present study is intended to have a broader objective, attention will now be directed toward some of the parameters affecting the airframe.

Fuselage

The effects of variations in vehicle shape parameters are shown in Figs. 7–9. In Fig. 7, the effects of fuselage fineness ratio are presented. The weights of the fuselage, propulsion system, fuel, and payload plus reserves were computed for vehicles with fuselage fineness ratios from 8 to 16. All vehicles had the same fuselage volume, and the engine exhausts were assumed to exit at the rear of the fuselage. The appropriate exit-area requirements were included in the fuselage proportions. As noted in Fig. 7, the propulsion-system size (and also exhaust area) was reduced as the fuselage fineness ratio was increased. This trend resulted from the decreased wave drag of the fuselage, which also accounts for the decrease in fuel-weight fraction. At higher fineness ratios, increased fuselage surface area, friction drag, and bending moments caused drag and weight penalties. The payload fractions resulting from the indicated tradeoffs suggested a maximum at a fineness ratio of 13. All of the vehicle fuselages were considerably longer than those of present-day subsonic jets, which are depicted by the cross-hatched sketch. An attractive fineness ratio might be 12, but the fuselage length of 285 ft could well present a serious landing problem. The very large and long fuselages may also present problems in achieving adequate stability at hypersonic speeds.

In summary of Fig. 7, then, the optimum fineness ratio based on mission performance may be longer than practical considerations will allow, but the performance penalties associated with shorter vehicles probably will dictate vehicles that are subtantially longer than current aircraft.

Wing

The effects of thickness ratio, wing loading, and aspect ratio for a simple delta wing were evaluated in tradeoff studies of wing weight and mission fuel; fuel was affected primarily by drag changes. Thickness ratios from 0.02 to 0.06 were examined, and the results, which are not presented in detail herein, indicated a maximum payload ratio occurred for thickness ratios between 0.04 and 0.05. These results also indicated that some latitude was available in choosing the thickness ratio, since it was not a sensitive parameter. The effect of varying wing loading on several weight fractions is shown in Fig. 8. Changes in wing loading caused a number of other changes in the vehicle, but only the more important ones were isolated for presentation in Fig. 8. As wing loading increased, wing weight decreased, as expected from the reduction in wing size. The decrease in wing size also resulted in a decrease in cruise lift-drag ratio as shown at the top of Fig. 8. Fuel consumption, being related to lift-drag ratio, increased. These relationships are straightforward, but the change in propulsion-system weight is somewhat obscure. It might be expected that, with lower lift-drag ratios, more thrust and hence more propulsion-system weight would be required. More thrust was required, but it was obtained by two factors that are not immediately apparent in Fig. 8. First, the available thrust increased at the higher wing loadings, since the cruise altitude was lower. Second, the cruise angle of attack (at maximum lift-drag ratio) was greater for vehicles with high wing loading. Higher angles of attack increased the thrust coefficient because the wing shock wave provided greater compression. The net result was that, even though the required thrust increased with higher wing loadings, the increased performance of the propulsion system actually allowed a reduction in its size and weight.

In summary of this particular tradeoff, the takeoff wing loading giving highest payload fraction was about 90 psf. According to the present results, however, wing loadings from 70 to 120 psf were nearly as attractive.

Wing aspect ratio was also varied in the present study. The range covered was from 1.0 to 2.0, and the effects on wing and fuel weights are illustrated in Fig. 9. Through the range covered, both of these weights increased continuously with increasing aspect ratio. This trend indicates that the wing aspect ratio of interest will probably be governed by other considerations, such as landing or takeoff. For example, in Fig. 9 the minimum aspect ratios that can be used for takeoff runs of 8000 ft or landing approach speeds of 160 knots are indicated for two maximum allowable angles of attack. Based on preliminary evaluations, it is indicated that conventional takeoff and landing requirements can be met with the vehicles studied, but, as noted earlier, the

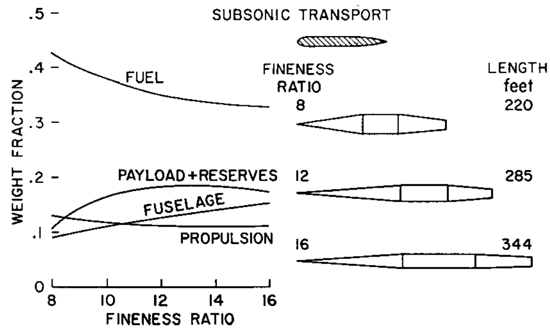

Fig. 7 Effect of fuselage fineness ratio on mission performance.

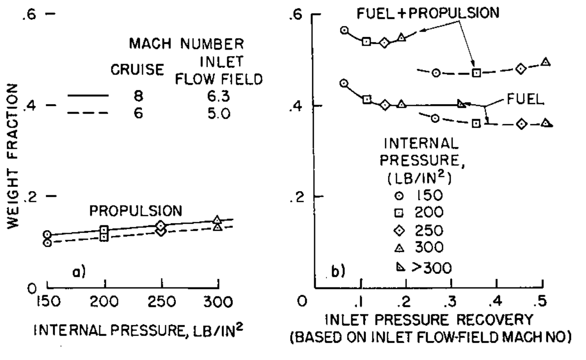

Fig. 6 Effect of inlet internal pressure and pressure recovery on mission performance.

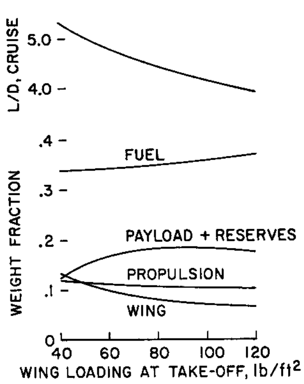

Fig. 8 Effect of wing loading on mission performance.

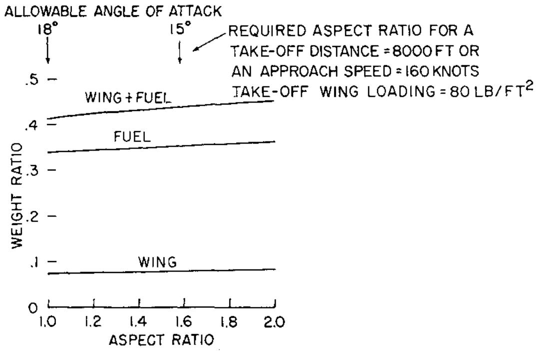

Fig. 9 Effect of aspect ratio on mission performance.

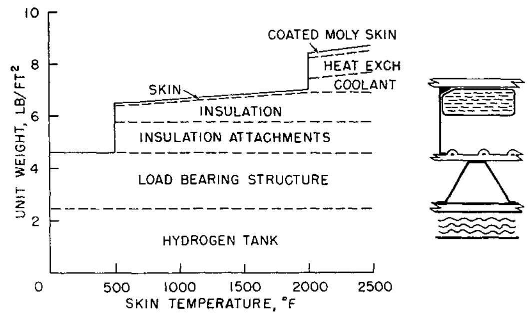

Fig. 11 Unit weight of typical fuselage structure.

problems associated with the long fuselage may aggravate these operations.

The results presented thus far indicate some of the effects due to variations in the parameters that influence engine and airframe sizing. The performance parameters of cruise Mach number and range are also of interest and will be examined next.

Cruise Mach Number

Information derived from tradeoff studies, such as the foregoing, was used in the definition of several vehicles suitable for cruise at Mach numbers between 4 and 8. The primary difference between these vehicles was the sizing of the propulsion system, mainly in the sizes of the inlet, ramjet, and exhaust nozzle. The results of this particular phase of the present study are shown in Fig. 10. Weight fractions calculated for the airframe, propulsion system, fuel, and payload plus reserves are shown as functions of cruise Mach number. Block times for the 5600-naut-mile range are also indicated at the top of Fig. 10. The calculations were made both for stoichiometric fuel-air ratio and, at higher speeds, for a higher ratio, which would provide the added fuel flow estimated to be required for inlet and engine cooling. Results for stoichiometric fuel-air ratio will be discussed first. These results indicate that a maximum weight fraction for payload and reserves occurred for a Mach number of 6. The difference for a Mach number of 4 was relatively small, about 2%. The loss for a Mach number of 8 was greater, about 4%, and of this only 1% was due to increased airframe weight. In fact, the airframe weight was relatively invariant with Mach numbers from 4 to 8, even though the wing skin temperature increased from below 800° to above 1500°F as indicated at the top of Fig. 10. This point can be clarified by examination of Fig. 11, which shows a schematic cross section of the structural concept considered in this study. The figure indicates a typical structure with insulation protecting a hydrogen tank and a cold load-bearing structure. The skin temperatures noted in Fig. 10 required the use of insulation in the structure, and Fig. 11 indicates the corresponding unit weight penalty. It also shows that the actual weight of insulation added is less than half of this penalty, the other part being attachments and supports for the insulation. For the temperature range indicated in Fig. 10, the insulation weight does not increase greatly. Major unit weight changes occur at about 500° and 2000°F. Within the range of cruise Mach numbers from 4 to 8, it does not appear that one of these major changes is encountered, except at the fuselage nose and wing leading edges.

A major effect of heating due to increased flight Mach number was encountered in the inlet and engine, however. The dashed curve in Fig. 10 shows the result of increased fuel flow required for inlet and engine cooling. Based on present estimates, this requirement became important at a Mach number of about 6.7 and, at a Mach number of 8, had made a serious inroad into the payload, causing a reduction in the weight fraction for payload and reserves from about 14 to about 6%. It should be emphasized, however, that the exact magnitude of this effect is strongly dependent on the details of the calculations. In the present estimates, for instance, it was assumed that the inlet wall temperature was 1500°F, that the turbulent boundary layer was removed at several points in the inlet, and that 60% of the heat capacity available, when the fuel is heated from a liquid to a gas at 1500°F, was available for cooling. It is believed that the current estimates tend to be somewhat conservative. The primary point to be noted here is that the speed at which inlet and engine cooling requirements dictate fuel-rich operation may very well be the maximum attractive cruise Mach number.

Range

The same vehicles sized for a 5600-naut-mile range and for cruise at Mach numbers between 4 and 8 were studied at ranges other than the 5600 naut miles. The results are presented in Fig. 12, where the weight fractions for payload plus reserves are shown as functions of range. Curves for cruise at stoichiometric fuel-air ratio for Mach numbers of 4, 6, and 8 are presented along with a curve for Mach number 8

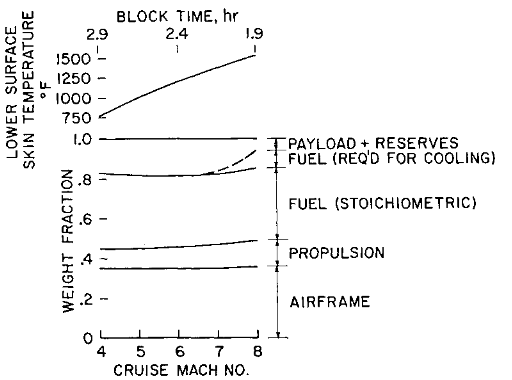

Fig. 10 Effect of cruise Mach number on mission performance.

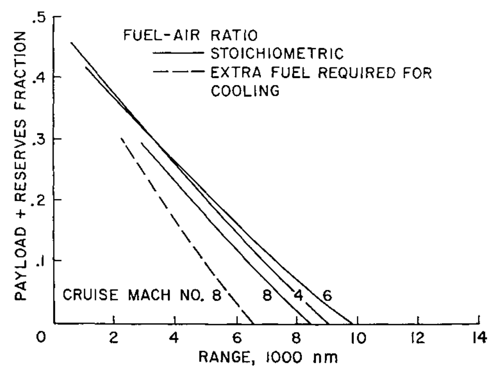

Fig. 12 Range-payload capability of hypersonic transports.

with the fuel-air ratio dictated by the present estimates of inlet and engine cooling. The results for stoichiometric cruise in particular indicated that these vehicles were all capable of very long-range flight with respectable payloads. At shorter ranges, the vehicles carried quite substantial payloads, even though some penalty was paid since the fuselage volume was still based on a range of 5600 naut miles. For cruise at Mach number 8, inlet and engine coolant requirements tended to reduce significantly the payloads at longer ranges. The effects of coolant requirements at shorter ranges were less severe, since the vehicle spent less time at a Mach number of 8. In fact, the vehicle covered a total of about 3000 naut miles during the two phases of acceleration to Mach number 8 and descent to the landing point.

Concluding Remarks

In the present study, some performance and weight tradeoffs have been examined for hypersonic transports propelled by hydrogen-fueled turboramjets. Results of this study have confirmed that these vehicles are capable of sustained flight to ranges approaching one-half the earth's circumference. The results indicate that, at ranges in the neighborhood of 5000 naut miles, 15 to 20% of the gross takeoff weight was available for payload and fuel reserves. The key to this excellent performance was the hydrogen fuel; the cruise speed was of secondary importance. The best performance was obtained at a cruise Mach number of 6, where a stoichiometric fuel-air ratio was possible. At Mach 8, fuel-rich operation, required for cooling purposes, resulted in significantly reduced performance. It was indicated that the maximum desirable cruise speed may well be dictated by cooling requirements of the inlet and engine. Based on the current approximate analysis of these coolant requirements, this maximum speed would appear to be near a Mach number of 6.7.

The current study indicated that hypersonic transports can operate within the usual restrictions imposed by takeoff, landing, sonic-boom overpressures, and aerodynamic heating. It was also indicated, however, that such restrictions often impose constraints on the vehicles or their operation. This study also indicated several specific areas where additional research and study would appear to be useful. These areas are 1) inlet and engine cooling, 2) landing problems for aircraft with very long fuselages, and 3) high-speed stability characteristics of aircraft with large fuselages.

In addition, it should be recognized that the quantitative results obtained in the present study are only as valid as the approximations upon which they are based. Further research in a number of areas, such as configuration studies, engine and exhaust-nozzle development and structural and tankage problems, will be required before the payload levels and trends indicated herein can be confirmed.

References

[1] Eggers, A. J., Jr., Allen, H. J., and Neice, S. E., "A comparative analysis of the performance of long-range hypervelocity vehicles," NACA Rept. 1382 (1958).

[2] Bergesen, A. J. and Potter, J. D., "An investigation of the flow around slender delta wings with leading edge separation," Princeton Univ., Dept. of Aeronautical Engineering, Rept. 510 (May 1960).

[3] Kelly, M. W., "Wind tunnel investigation of the low-speed aerodynamic characteristics of a hypersonic glider configuration," NACA RM A58F03 (1958).

[4] Koelle, H. H., *Handbook of Astronautical Engineering* (McGraw-Hill Book Co., Inc., New York, 1961).

[5] Ulmann, E. F. and Bertram, M. H., "Aerodynamic characteristics of low-aspect-ratio wings at high supersonic Mach numbers," NACA RM L53I23 (1953).

[6] Shanley, F. R., *Weight-Strength Analysis of Aircraft Structures* (Dover Publications, Inc., New York, 1960).

[7] Dickson, J. A., "Thermal protection with a temperature capability to 2500° F, for cool structures," *Proceedings of the Conference on Aerodynamically Heated Structures, Cambridge, Mass., July 25, 1961*, edited by P. E. Glasser (Prentice Hall Inc., Engelwood Cliffs, N. J., 1962).

[8] Fransiscus, L. C. and Lezberg, E. A., "Effects of exhaust nozzle recombination on hypersonic ramjet performance: II. Analytical investigation," AIAA J. **1**, 2077–2083 (1963).

[9] Carlson, H. W., "The lower bound of attainable sonic boom overpressure and design methods of approaching this limit," NASA TN D-1494 (1962).

[10] Hutchinson, H. A., "Defining the sonic boom problem," Astronaut. Aerospace Eng., **1**, 55–61 (December 1963).

[11] Jones, J. L., Hunton, L. W., Gregory, T. J., and Nelms, W. P., "Delta wing configurations for the supersonic transport," Astronaut. Aerospace Eng. **1**, 74–81 (July 1963).

[12] "Specification for engines, aircraft, turbojet, model," Military Specification MIL-E-5008B (January 1959).