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by

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PERFORMANCE TRADE-OFFS AND RESEARCH PROBLEMS FOR HYPERSONIC TRANSPORTS

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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. An indication of this capability can be seen in Fig. 1, which shows a comparison of the basic cruise efficiency (Breguet factor) of the 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. While 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 which pertain to its use in hypersonic aircraft are shown in Fig. 2. While 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 per unit weight (heat of combustion) which is roughly 2-1/2 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 one-tenth that of conventional 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° Rankine. 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 Fig. 2, it is apparent that hydrogen has physical and thermodynamic characteristics which differ greatly from hydrocarbon fuels. These differences have large effects on the performance and design trade-offs of transport vehicles. It is the purpose of this paper to explore some of these trade-offs and 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 trade-off studies will be presented.

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METHODS

A preliminary mission analysis of transport vehicles involves determining the values of vehicle parameters which 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 take-off 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. Fig. 3 indicates one arrangement considered. The aerodynamic lift and drag were estimated[1-4] primarily from the vehicle geometry with the friction drag including the effects of the trajectory. The structural weights and heat-protection weights were estimated[5,6] 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 in the wing compression field and was a variable-geometry, mixed-compression type. 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.[7]

Trajectory Considerations

Some of the considerations which affect vehicle trajectory are shown in Fig. 4. The desired cruise altitudes are indicated by the dashed curve in Fig. 4, 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 percent 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 contraints. 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. 4 is that due to sonic-boom limitations. Typical curves for overpressures of 2 and 3 pounds per square foot are shown. The exact location of these curves is affected by many factors;[8] for example, if the vehicle is shaped to minimize sonic boom, the lower curve can serve as an approximation of a 2 psf boundary.[9] 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 of the propulsion system increase rapidly causing a corresponding increase in the structural weight of the turboramjet engine. As a result, the trajectories usually are restricted by lines of constant internal pressure. Duct pressures are also dependent on inlet pressure recovery and 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. 4 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. 4, an optimum climb trajectory will depend upon numerous constraints and trade-offs involving considerations of structures, aerodynamics, and propulsion-system performance. The region bounded by the curves in Fig. 4 indicates reasonable operating conditions for the type of vehicles under consideration.

Considerations of Propulsion System Sizing

An indication of the propulsion-system sizing problem is given in Fig. 5, where typical vehicle acceleration characteristics are presented as a function of Mach number. Equivalent acceleration, as used here, combines the tangential acceleration and the climb rate into one term that denotes the net acceleration capability of a vehicle at any point in the trajectory. Equivalent acceleration is given by

$$\left(\frac{dV}{dt}\right)_{equiv} = \frac{dV}{dt} + g \sin \gamma$$

where dV/dt is the tangential acceleration, g is the acceleration due to gravity, and γ is the flight-path angle. The three curves in Fig. 5 show the acceleration of three similar vehicles designed for cruise at Mach numbers of 4, 6, and 8. The only difference between the vehicles is the size of the inlet, ramjet, and exhaust nozzle which were selected to give adequate thrust at the three different cruise conditions. Each vehicle has four turbojets which are all of equal size for comparative purposes. These comparisons indicate a unique propulsion-system sizing problem; there are essentially two types of engines with different sizing conditions that must operate with the same inlet and exhaust nozzle. The sizing condition for the turbojet is usually in transonic flight where a minimum value of acceleration margin must be maintained. In this study the cruise condition determined the size of the ramjet as well as the inlet and exhaust system. As seen in the figure, the vehicles with ramjets sized for higher Mach numbers have increased acceleration from the beginning of ramjet operation to the cruise speed; however, they have decreased acceleration at transonic speeds. This reduction is particularly evident in Fig. 5 for the vehicle designed for Mach 8 cruise. Although this vehicle has the same turbojet sizing as the other two, it has lower acceleration capability during turbojet operation due to the greater transonic drag caused by the use of a larger inlet. This situation is

alleviated somewhat for the Mach 6 cruise vehicle because the smaller inlet employed nearly matches the air-flow requirements of the turbojet at its maximum operating Mach number. However, transonic inlet drag is still significant for all vehicles. Some loss in the transonic acceleration capability of the Mach 8 vehicle is also due to the exhaust nozzle which is less effective at subsonic and transonic speeds than are those designed for a lower Mach number.

Another effect which becomes quite significant at hypersonic speeds is the influence of wing angle of attack on ramjet thrust. In Fig. 6(a), vehicle drag and thrust coefficients, both referenced to wing area, are shown as functions of angle of attack. Note that the thrust increases with cruise angle of attack almost as rapidly as drag. The higher thrust is due primarily to the increased air flow captured by the inlet as the wing angle of attack increases. As shown in Fig. 6(b), the use of the wing compression field allows the inlet system to be sized as much as 50 percent smaller than would be required if the inlet were placed outside of the flow field (shown by the results for zero angle of attack). It is also desirable to locate the inlet in the wing flow field to take advantage of the reduced Mach number, thereby decreasing the Mach number operating range of the inlet.

The discussion to this point has indicated some of the methods that were used and some of the major problems that were considered in the present analysis. With this background covered, it is now appropriate to examine some of the quantitative results of the trade-off studies.

TRADE-OFF RESULTS

The results of the trade-off studies will cover some of the effects of thrust loading, inlet performance, fuselage fineness ratio, wing loading, aspect ratio, cruise Mach number, and range. In each case, the results will be presented in terms of fractions of the vehicle gross take-off weight with the objective being to obtain a maximum fraction for payload and fuel reserves. As noted earlier, operational problems are not considered; therefore, no attempt is made to separate payload and reserves. It should also be noted that small changes in the weight fraction available for payload can be relatively important. For example, if the payload is 10 to 20 percent of the gross take-off weight, then a change in payload represented as 2 to 3 percent of gross take-off weight is obviously significant. For this reason, the discussion will often recognize minimums and maximums of relatively flat curves.

For the trade-off results presented, the vehicles have gross take-off weights of 500,000 pounds and fuselage volumes of 71,500 cubic feet. The climb trajectory is the lower one presented in Fig. 4 except that the maximum internal pressure is 200 psi. Unless otherwise shown, the range is 5,600 nautical miles including climb, cruise, and descent at maximum lift-drag ratio and at minimum power. Most trade-offs are for a cruise Mach number of 6, except where noted.

Engine

The trade-off in sizing the turbojet, which was briefly mentioned earlier, is presented in more detail in Fig. 7. Vehicle thrust-to-weight ratio at sea-level static conditions is used as a measure of turbojet thrust loading. As this factor is decreased from 0.6, the engine weight decreases while the fuel consumption increases. This latter increase is gradual at first but becomes more rapid as the lower thrusts lead to long acceleration times at transonic speeds which, in turn, cause excessive fuel consumption. A reasonable minimum equivalent acceleration at transonic speeds might be about 2 ft/sec^2, and this limit tends to occur at thrust loadings of about 0.42. At a loading of about 0.32, the vehicles are unable to accelerate through the transonic region. Although a thrust loading between 0.4 and 0.5 gives maximum payload fraction, these results indicate that the vehicles are not particularly sensitive to turbojet thrust loading. This result differs from that encountered by most other aircraft, for example, the supersonic transports.[10] 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 can be met by adjusting only the turbojet size. The rest of the propulsion system is sized for the cruise condition. Consequently, the weight penalties incurred by increasing the transonic or take-off thrust loadings are not so severe as have been experienced in the past.

The results in Fig. 7 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 is 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 are not quite so important to the over-all mission performance for hypersonic transports as they are for supersonic transports.

Inlet

In the discussion of Fig. 4, a trade-off 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)[11] are shown in Fig. 8. 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 nautical 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 due to wing flow-field effects the Mach numbers at the inlet were about 5.0 and 6.3, respectively. The results presented in Fig. 9 show the effects on fuel-weight fraction and on propulsion-weight fraction. Although both effects are added together in Fig. 9(b), internal duct pressure, indicated by the symbols, is responsible for the effects on propulsion-system weight, and pressure recovery is responsible for the effects on fuel weight. For cruise at Mach number 6, the sum of the propulsion-system weight and the fuel weight is not very sensitive to the two parameters because the two effects tend to cancel each other over the ranges studied. At Mach number 8 the fuel consumption is not very sensitive to pressure recovery at levels above 0.15; therefore, structural weight becomes 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 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. 10 through 12. In Fig. 10 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 have the same fuselage volume, and the engine exhausts are assumed to exit at the rear of the fuselage. The appropriate exit-area requirements are included in the fuselage proportions. As noted in Fig. 10, the propulsion-system size (and also exhaust area) is reduced as the fuselage fineness ratio is increased. This trend results 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 cause drag and weight penalties. The payload fractions resulting from the indicated trade-offs suggest a maximum at a fineness ratio of 13. All the vehicle fuselages are 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 feet 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. 10, 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 substantially longer than current aircraft.

Wing

The effects of thickness ratio, wing loading, and aspect ratio for a simple delta wing were evaluated in trade-off 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. 11. 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. 11. As wing loading increases, wing weight decreases, as expected from the reduction in wing size. The decrease in wing size also results in a decrease in cruise lift-drag ratio as shown at the top of Fig. 11. Fuel consumption, being related to lift-drag ratio, increases. 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 is required, but it is obtained by two factors that are not immediately apparent in Fig. 11. First, the available thrust increases at the higher wing loadings, since the cruise altitude is lower. Second, the cruise angle of attack (at maximum lift-drag ratio) is greater for vehicles with high wing loading. As indicated previously, higher angles of attack increase the thrust coefficient because the wing shock wave provides greater compression. The net result is that even though the required thrust increases with higher wing loadings, the increased performance of the propulsion system actually allows a reduction in its size and weight.

In summary of this particular trade-off, the take-off wing loading giving highest payload fraction is about 90 psf. According to the present results, however, wing loading from 70 to 120 psf are 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. 12. 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 take-off. For example, on Fig. 12 the minimum aspect ratios that can be used for take-off runs of 8000 feet 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 take-off and landing requirements can be met with the vehicles studied, but as noted earlier, the 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 which 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 trade-off 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 is 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. 13. 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 nautical mile range are also indicated at the top of Fig. 13. 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 occurs for a Mach number of 6. The difference for a Mach number of 4 is relatively small, about 2 percent. The loss for a Mach number of 8 is greater, about 4 percent, and of this only 1 percent is due to increased airframe weight. In fact, the airframe weight is relatively invariant with Mach numbers from 4 to 8, even though the wing skin temperature increases from below 800° F to above 1500° F as indicated at the top of Fig. 13. This point can be clarified by examination of Fig. 14 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 on Fig. 13 require the use of insulation in the structure, and Fig. 14 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. 13, 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 is encountered in the inlet and engine, however. The dashed curve in Fig. 13 shows the result of increased fuel flow required for inlet and engine cooling. Based on present estimates, this requirement becomes important at a Mach number of about 6.7, and at a Mach number of 8, has made a serious inroad into the payload, causing a reduction in the weight fraction for payload plus reserves from about 14 percent to about 6 percent. 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 percent 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 nautical mile range and for cruise at Mach numbers between 4 and 8 were studied at ranges other than the 5600 nautical miles. The results are presented in Fig. 15, 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 with the fuel-air ratio dictated by the present estimates of inlet and engine cooling. The results for stoichiometric cruise in particular indicate that these vehicles are all capable of very long-range flight with respectable payloads. At shorter ranges the vehicles carry quite substantial payloads, even though some penalty is paid since the fuselage volume is still based on a range of 5600 nautical miles. For cruise at Mach number 8, inlet and engine coolant requirements tend to reduce significantly the payloads at longer ranges. The effects of coolant requirements at shorter ranges are less severe, since the vehicle spends less time at a Mach number of 8. In fact, the vehicle covers a total of about 3000 nautical 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 trade-offs 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 nautical miles, 15 to 20 percent of the gross take-off weight is available for payload and fuel reserves. The key to this excellent performance is the hydrogen fuel; the cruise speed is 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 take-off, 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,

(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 on 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.

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CRUISE EFFICIENCIES

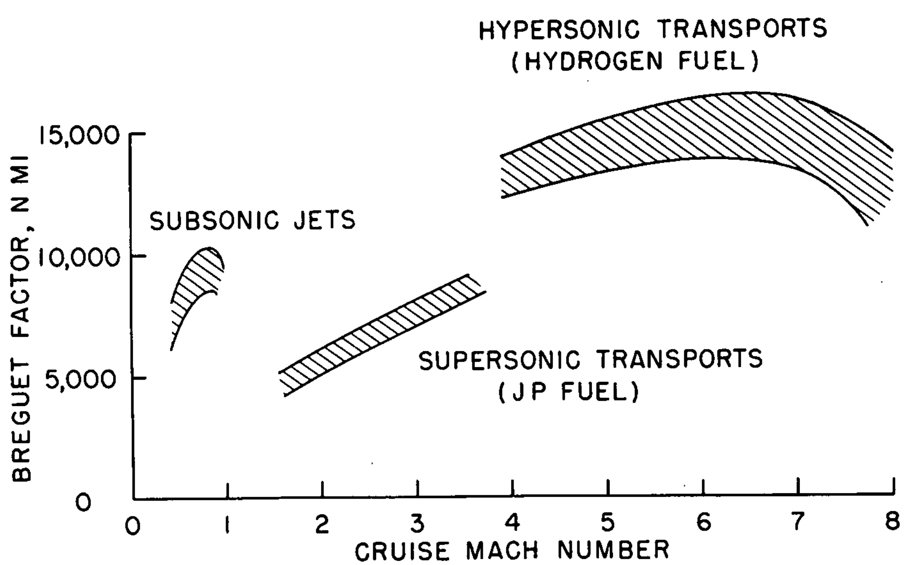


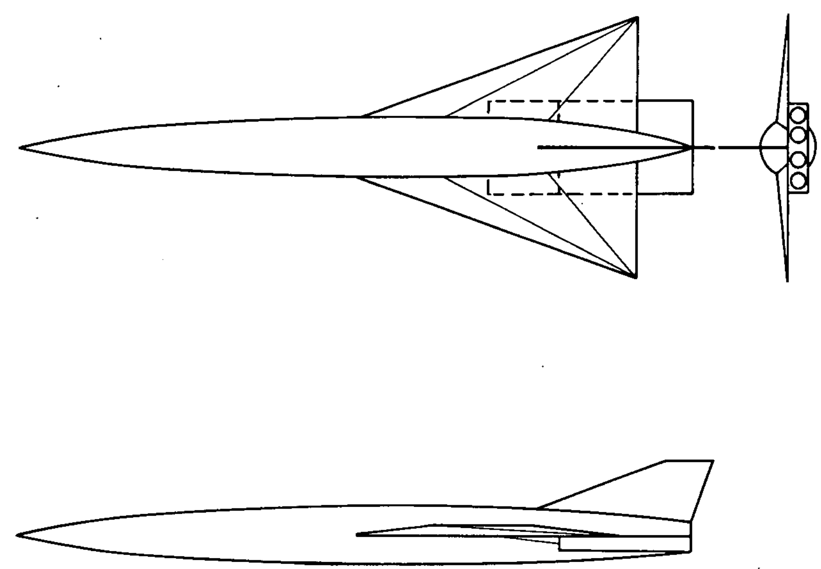
Figure 1.

FUEL CHARACTERISTICS

	CONVENTIONAL 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/FT^3	51	4.5
BOILING TEMPERATURE, °R (1 ATMOS)	820-915	36

Figure 2.

TYPICAL CONFIGURATION



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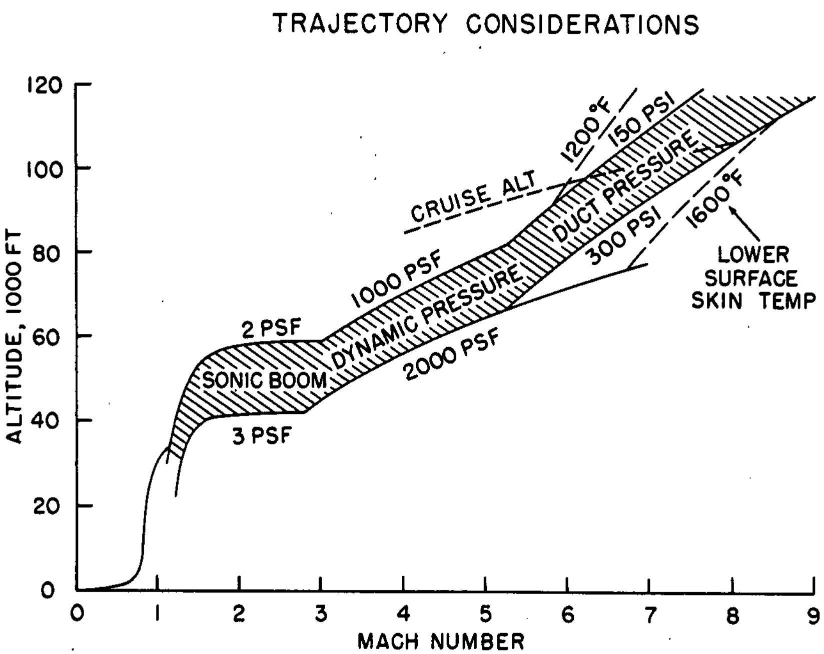

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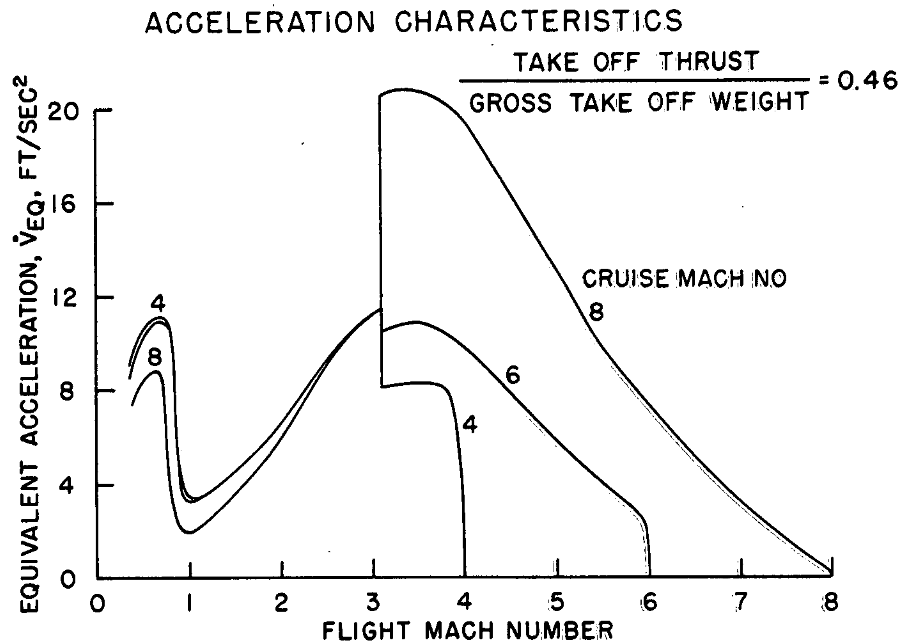
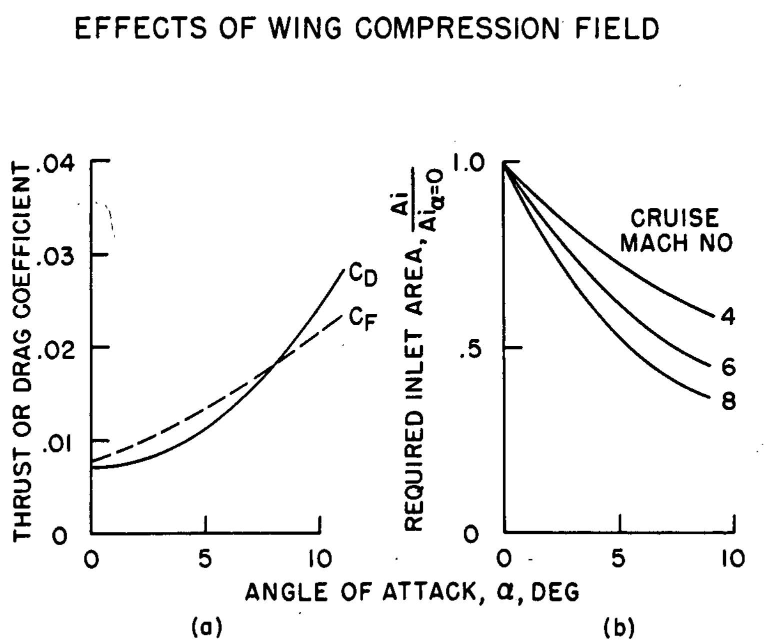


Figure 5.



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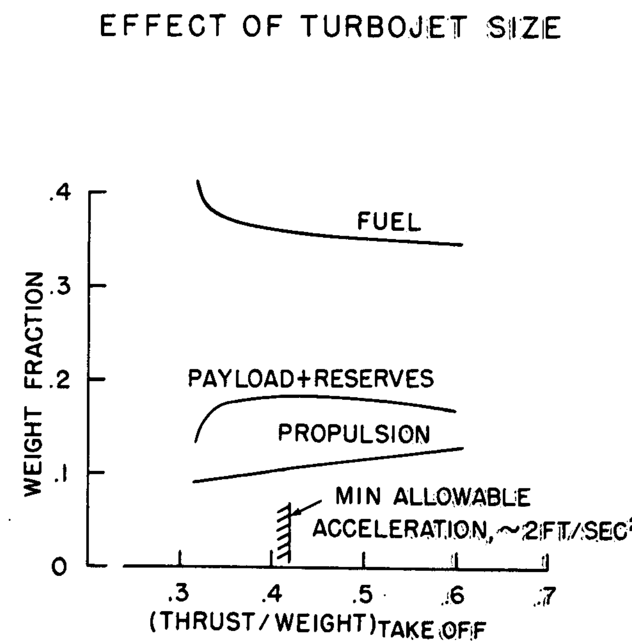

Figure 7.

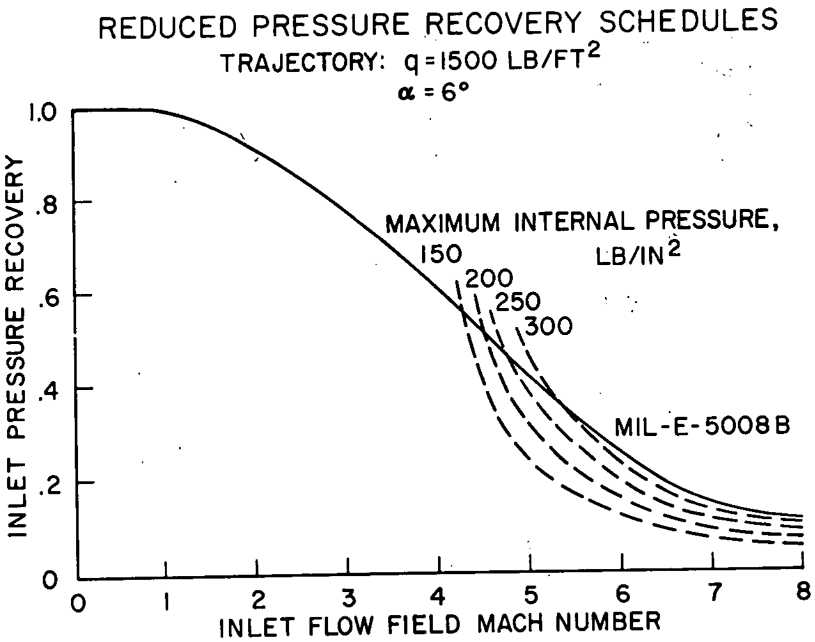

Figure 8.

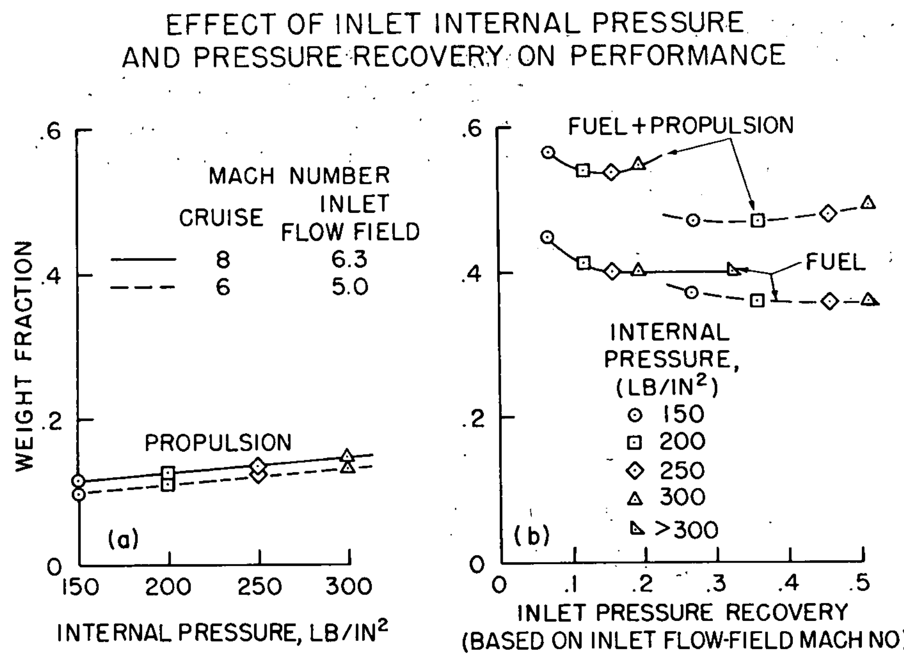

Figure 9.

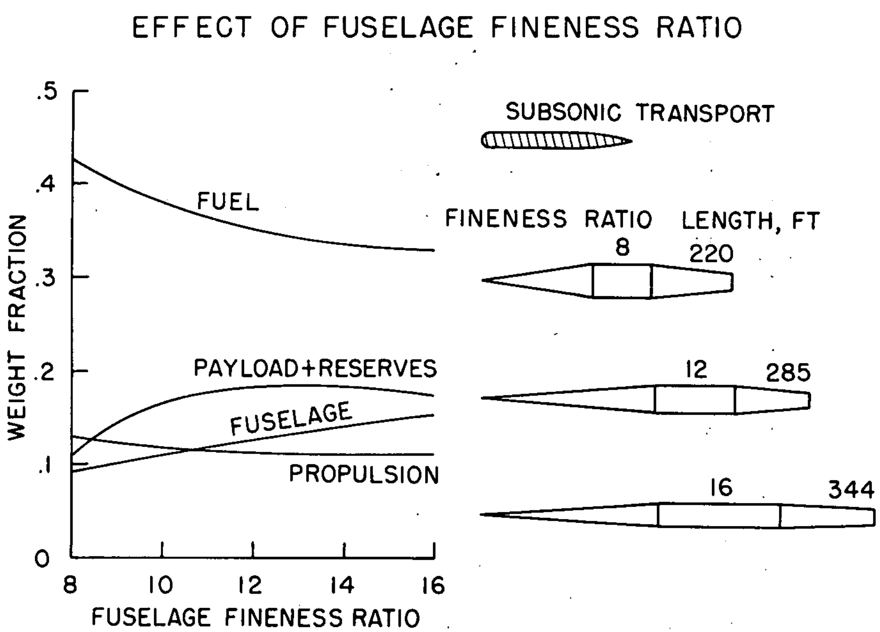

Figure 10.

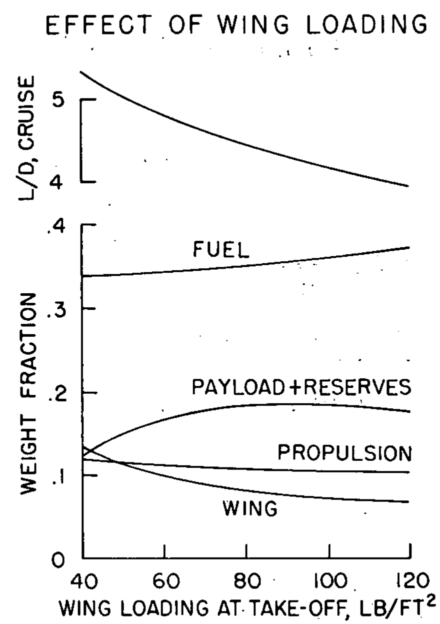

Figure 11.

EFFECT OF ASPECT RATIO

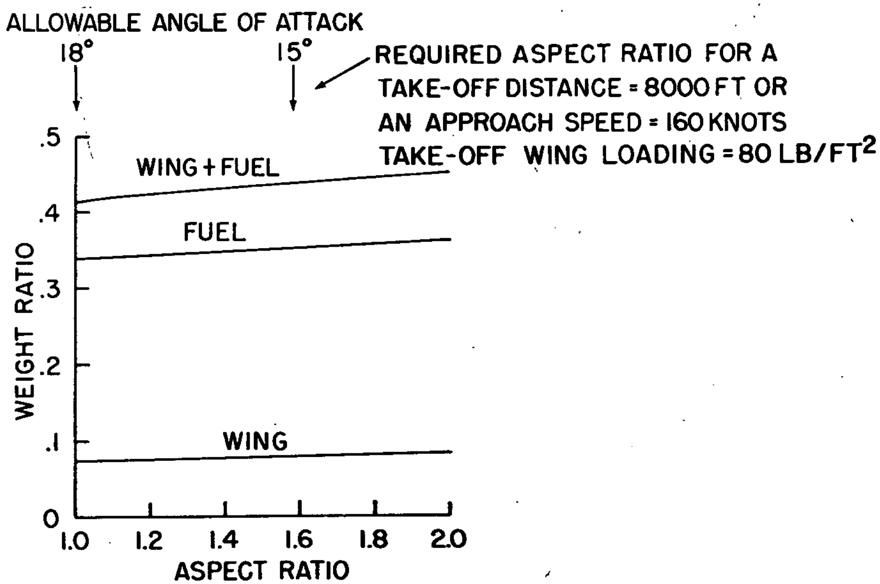

Figure 12.

EFFECT OF CRUISE MACH NUMBER

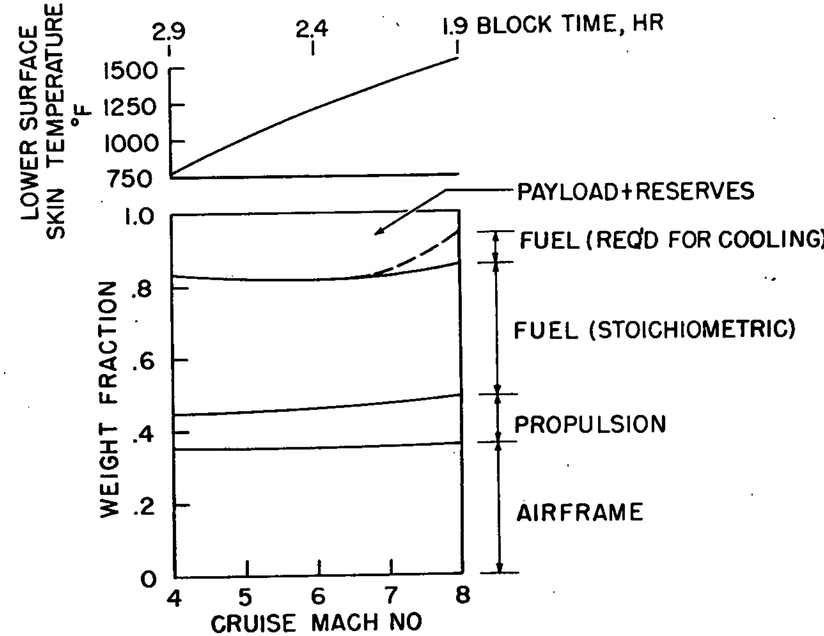

Figure 13.

TYPICAL FUSELAGE STRUCTURE

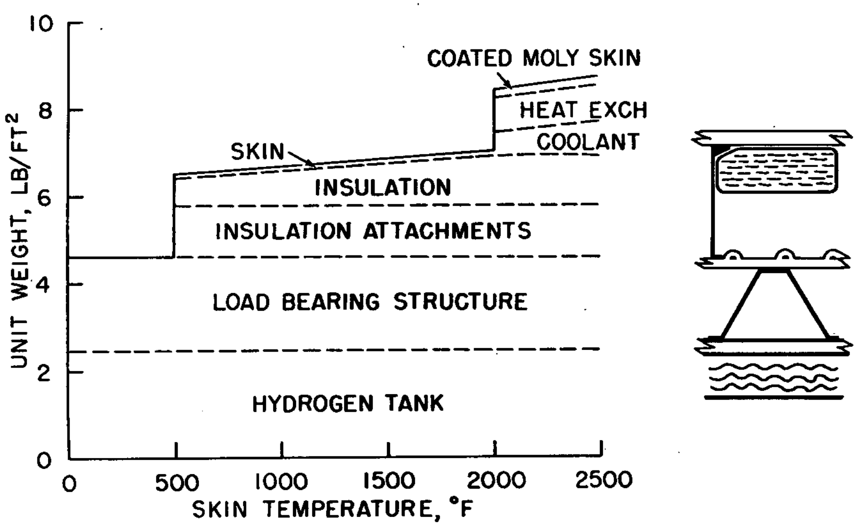

Figure 14.

HYPERSONIC TRANSPORT

RANGE-PAYLOAD CAPABILITY

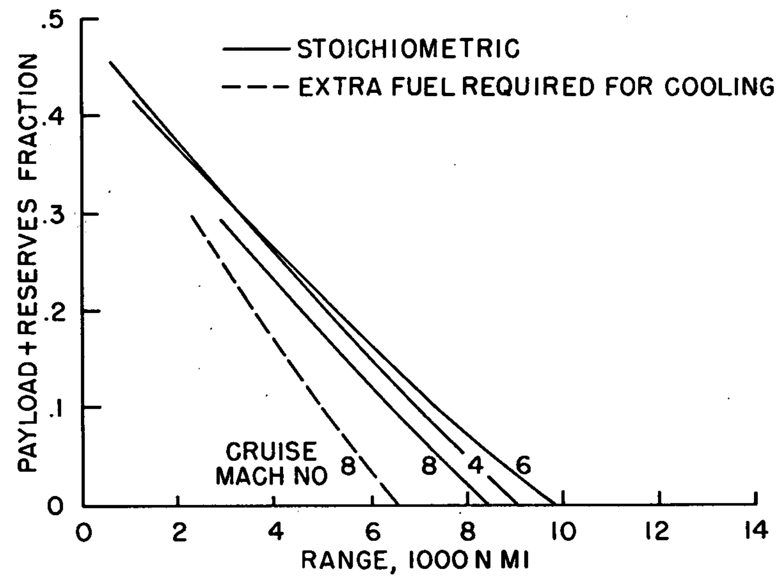

Figure 15.