PROPULSION FOR HYPERSONIC TRANSPORT AIRCRAFT

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ABSTRACT

The problem of achieving efficient, long-range hypersonic flight with turboramjet engines is examined. The performance of integrated engine-airframe configurations is studied in order to obtain meaningful results. After including appropriate off-design penalties and observing a 2-lb ft^2 limit on sonic-boom overpressure, the typical range of 2,800 nautical miles is calculated for a Mach 6 cruise airplane (125 passengers and 500,000 lb gross weight) with JP fuel. To obtain longer ranges, improvements must be made in the assumed engine performance or an unconventional fuel such as hydrogen must be employed.

INTRODUCTION

The aviation industry today is still in the process of regaining the stability of operation that was shaken by the introduction of subsonic jet transport planes. Nevertheless, we now apparently stand at the threshold of a new era characterized by supersonic commerical flight and even more enterprising concepts such as aerospaceplanes. With this background, it seems not inappropriate to examine the characteristics of still more advanced commercial transport airplanes that operate at hypersonic speeds. Since a number of excellent studies of this problem have already been made by others (e.g., Refs. 1–3), this paper might be described as a reappraisal of one possible approach to the topic. Although the primary interest herein is in the propulsion system, it has been necessary to consider the engine and airframe as an integrated unit in order to obtain meaningful results. The results are hence generally presented in terms of predicted airplane performance.

Perhaps the obvious question to be asked is why the aviation industry and the paying passenger should be interested in hypersonic aircraft in the first place. One answer is that higher speed has always been desirable for its own sake. For commercial purposes this translates into a desire by the passenger to reach his destination more quickly. The virtue of hypersonic flight in this regard is examined in Fig. 1. Similar figures have been shown many times before. Perhaps the most significant difference here is that the author pessimistically (?) assumed an average delay of 2 hours due to ground transport to and from the airport, check-in, baggage handling, holds due to weather or heavy traffic, etc. Furthermore, the climb and descent phases of the flight have been included with an average acceleration of 0.2 g.

We note a very significant reduction in total trip time as the cruise Mach number is raised from 0.9 to 3.0. However, the benefits of further speed increases are much smaller. For example, at a range of 4,000 nautical miles, doubling the cruise Mach number from 3 to 6 cuts the total time only from $4\frac{1}{2}$ to $3\frac{1}{2}$ hours. (On the other hand, of course, there is still a very appreciable reduction in flight time—about 40 percent.) Note too that at each

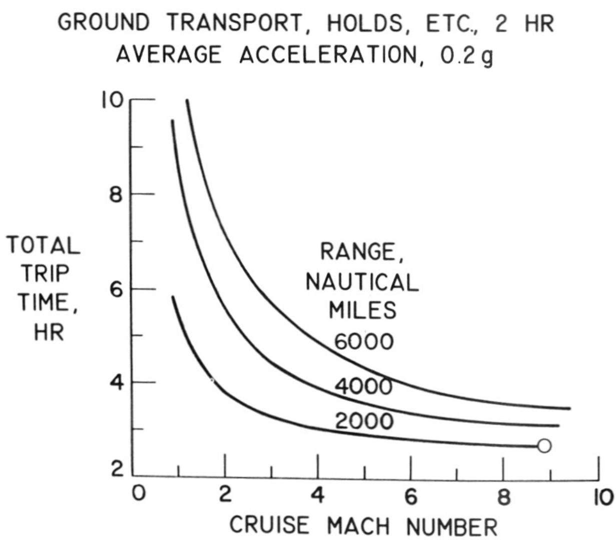

Figure 1. Aircraft transit times.

range there is a limit on useful speed capability reached when the full range is covered during the acceleration and deceleration phases of flight. In this region, time savings can be realized by climbing and descending faster but not by further increasing the airplane's maximum speed.

The value of time savings of the order indicated in Fig. 1 must be judged ultimately by the potential paying passengers. Without further discussion of this point, the remainder of this paper is devoted to the technical aspects of achieving effective hypersonic vehicles.

ENGINES

At the present time it appears that the proposed supersonic transport will employ either turbojets or turbofans. These engine types are limited both structurally and thermodynamically to speeds below approximately Mach 4. For hypersonic flight only a ramjet cycle is suitable. At lower speeds, of course, a ramjet is inefficient and must be supplemented by some auxiliary acceleration device.

Many types of propulsion system can provide the required low-speed acceleration for a hypersonic cruise vehicle. These types include turbojets, turbofans, air turborockets, pure rockets, etc. Furthermore, all these systems can be employed as entirely separate units from the high-speed ramjet, or they can be designed with the ability to convert internally to ramjet operation when required.

Reference 1 is a good example of the separated engine approach. This study assumed the use of turbojets at low speeds and external ramjets (ERJ) at high speeds. Some significant advantages of this approach are (1) elimination of the need to operate a single inlet and exhaust system over the complete speed range, (2) easier isolation of the relatively delicate turbojet from the high-temperature hypersonic environment, and (3) the ability to utilize the beneficial self-cooling characteristic of the external ramjet at high speeds.

The present paper, on the other hand, employs the combined propulsion concept of the turboramjet. This engine is essentially an afterburning turbojet during low-speed flight. A valve and ducting is provided, however, so that, at speeds above approximately Mach 3, the incoming air is by-passed around the rotating machinery directly into the afterburner, which now functions as a ramjet combustor. Some advantages of this system are (1) a weight saving due to the use of a single inlet and exhaust nozzle, (2) elimination of the need for two separate propulsion systems, with possible simplification of controls, fuel systems, etc., and (3) elimination of the drag penalty caused by the presence of a nonoperating engine.

It is not intended to imply by this choice that the turboramjet is necessarily the best engine for this application. More detailed, mutually consistent studies must be performed to compare engine types properly. Earlier work at NASA and elsewhere, however, yields the tentative conclusion that other engine cycles are unlikely to offer significantly improved performance.

ANALYSIS

The approach adopted in the study was to design an airplane of given gross weight carrying a specified payload and to determine the achievable range when flown over a realistic flight path. (Although it is recognized that multistaging or in-flight refueling can be very beneficial, they have not been considered herein. It is felt that such expedients will not be employed for commercial flight until all other reasonable alternatives have been exhausted.)

The sensitivity of the aircraft range to variations in major propulsion parameters was then studied. For each new set of engine design parameters a new optimum airplane design was generally determined. Once determined, however, the configuration was not varied during a flight, and appropriate off-design penalties were assessed.

ASSUMPTIONS

Airframe. A conventional airplane configuration is considered (Fig. 2), featuring a thin delta wing and four underslung engines. The takeoff weight is 500,000 lb. The useful payload, consisting of 125 passengers plus baggage, is 26,125 lb. An uncooled airframe constructed when necessary of superalloy steels is assumed for purposes of weight estimation.

Engines. It is frequently stated that close integration of the airframe and engines becomes increasingly desirable as flight speed is raised. For computational simplicity, pod-installed engines were assumed in the present analysis. It is not unlikely, however, that a more sophisticated approach may be necessary in practice to achieve the performance obtained herein.

The engine inlets are located within the wind pressure field. Primarily, this design has the virtue of reducing the engine size (and weight) needed to produce a given amount of thrust at high speeds. In addition, there is a modest improvement in specific impulse plus a lessened sensitivity to angle of attack.

Regenerative cooling of the engines is employed during acceleration and cruise and is supplemented where necessary by insulation and water cooling. Since engine cooling is not critical during the acceleration, an afterburner equivalence ratio of 1 was used during this phase of flight.

Inlet performance is based on a constant capture area, with an axisymmetric translating spike. The nozzle throat area was allowed to vary within practical limits, but exit area was considered fixed.

Performance Levels. As a propulsion specialist, it is not the author's intent to defend any of the detailed assumptions regarding the airframe. Such esoteric factors as wing warp, directional stability, transition Reynolds numbers, panel flutter, etc., held no part in this study. Instead, the approach taken was to reproduce, by simple means, a certain desired level of achievement in such parameters as lift-drag ratio and structural weight. The desired levels were based on the very extensive work done by the U.S. Air Force, NASA, and industry, most recently in regard to supersonic transports and orbital booster systems.

Some insight into the performance levels assumed for both the engine and the airframe is offered by Fig. 3. Overall engine efficiency, cruise lift-drag ratio, and hardware weight fraction (structural plus engine) for typical vehicles are shown at various cruise Mach numbers. Engine efficiency (to which cruise range is directly proportional) increases with speed up to beyond Mach 5; this fact offers some initial hope that hypersonic vehicles may possess some intrinsic advantage apart from mere speed. (It is only the incorporation of some sizable component inefficiencies at high speeds that makes the engine efficiency peak at the point

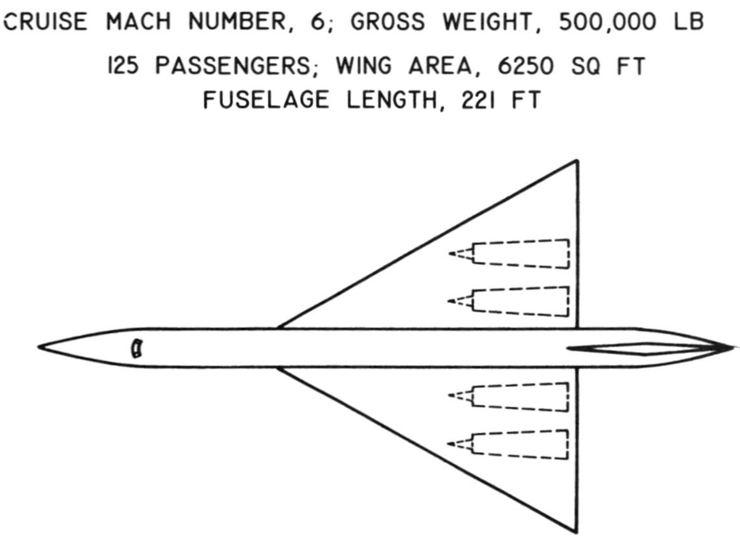

Figure 2. Typical hypersonic transport.

shown. More optimistic assumptions regarding nozzle performance, for example, would raise the optimum Mach number still further.)

Although hypersonic velocities may be desirable in terms of engine efficiency, this is not the case for the other two quantities represented in the figure. The cruise lift-drag ratio decreases somewhat with speed. A typical value is 5.9 at Mach 6. Values of this order are suggested by a number of experimental and theoretical studies (e.g., Refs. 4 and 5).

Another detrimental consequence of greater speed is the degradation of structural efficiency due to the increased aerodynamic heating. The structural fractions shown in Fig. 3 were based on a hot structure operating at the average equilibrium-radiation temperature corresponding to the cruise condition. The engine weights were obtained with empirical relations that represented a moderate advance beyond current technology. A typical value of installed engine thrust-to-weight ratio (including inlet, exhaust nozzle, and nacelle) is 4.5 at takeoff.

Although not explicitly shown in Fig. 3, still another penalty incurred by higher flight speeds is the greater amount of energy required to accelerate to the cruise condition, coupled with the need to compromise both the airframe and the engine to operate effectively over a wider range of off-design conditions.

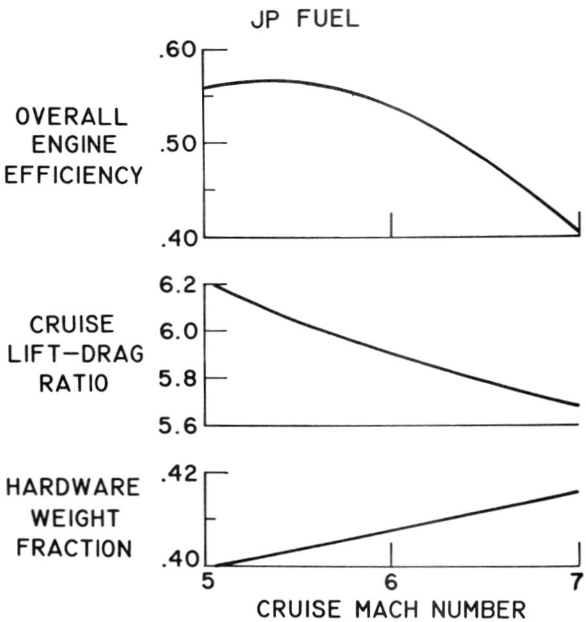

Figure 3. Typical aircraft performance levels.

Flight Path. The only way to evaluate these off-design losses is to compute airplane performance corresponding to a complete flight. A large number of such flights were studied with the aid of a large-scale electronic computer. The flight path employed during climb is indicated in Fig. 4.

Optimized trajectories yielding the minimum possible fuel consumption can be derived by proper mathematical techniques. Unrestricted optimum paths generally result in high speeds at rather low altitudes. In real situations, various operational constraints or limitations are imposed. Those considered herein, as indicated on Fig. 4, are related to (a) maximum allowable sonic-boom overpressure and (b) allowable internal engine pressure. Various airframe constraints, such as aerodynamic heating, were not specifically considered. The illustrated flight path, however, is felt to be a realistic one.

CALCULATION PROCEDURE

The major independent parameters for each airplane calculation were cruise Mach number, takeoff wing loading, takeoff acceleration rate, and maximum engine-inlet area. These parameters effectively size the wing, the turbojet cycle, and the ramjet cycle since gross weight and payload are fixed. Weights of the various airplane components, in particular the fuel, can then be computed.

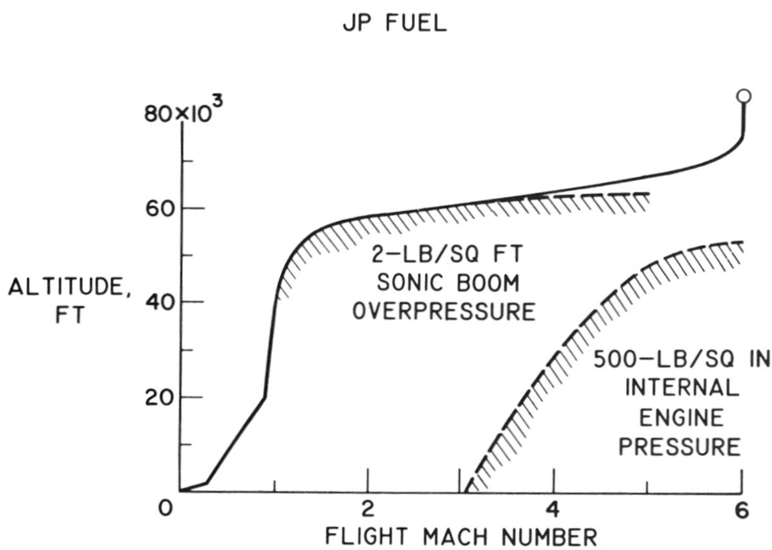

Figure 4. Airplane climb trajectory.

The resultant airplane design is then "flown" through a complete flight. The climb and acceleration phase is specified by a table of altitude against Mach number, which observes the appropriate limiting boundaries. Upon reaching the cruise Mach number, the airplane climbs until reaching the altitude corresponding to maximum cruise range. At this altitude the engine is throttled back, the airplane levels off, and a Breguet cruise path is followed until the usable fuel is exhausted. Primarily because of the engine cooling requirements, it is probably impractical to employ a gliding descent. It was hence conservatively assumed that the last part of the cruising fuel was really used in the terminal deceleration, and no additional descent range was included.

In lieu of more detailed studies of fuel reserve requirements, it was assumed that 10 percent of the total fuel load was held in reserve for holds, etc. (This assumption is not a minor one, as the reserve fuel weighs about as much as the payload.)

RESULTS AND DISCUSSION

In the first part of this section, the effect of the various major airplane and engine design parameters on range will be examined.

BASIC DESIGN PARAMETERS

Sonic Boom. One of the most critical factors affecting the performance of supersonic transports is the limitation placed on allowable sonic-boom overpressure. This factor is equally important for the hypersonic airplane. Maximum overpressures are generally produced during the climb phase, just after the airplane accelerates into the supersonic region. The sensitivity of the boom overpressure to the altitude at this point is illustrated in Fig. 5.

Because of the large size of the airplane being considered, the sonic boom is very intensive unless quite high altitudes are reached. As also shown in the figure, however, high altitudes significantly penalize airplane range. This penalty results from the need for larger, heavier engines and a less efficient climb path. The magnitude of boom that will be tolerated by the public is still a matter of great controversy. An acceleration overpressure of 2.0 lb ft^2 is being used in current proposals for the supersonic transport and will also be assumed for the remainder of this paper. This overpressure corresponds to transonic acceleration altitudes of the order of 56,000 ft (as calculated by techniques developed at the NASA Langley Research Center, e.g., Ref. 6).

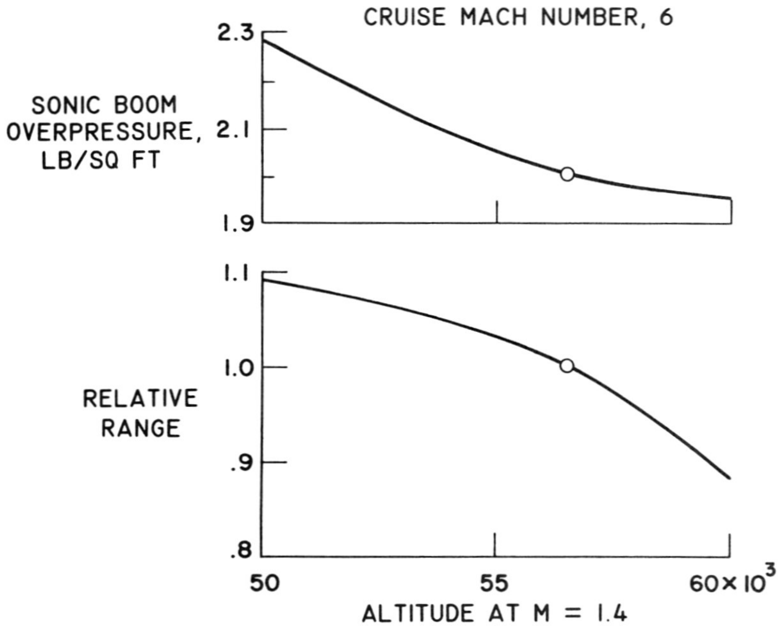

Figure 5. Effect of transonic altitude.

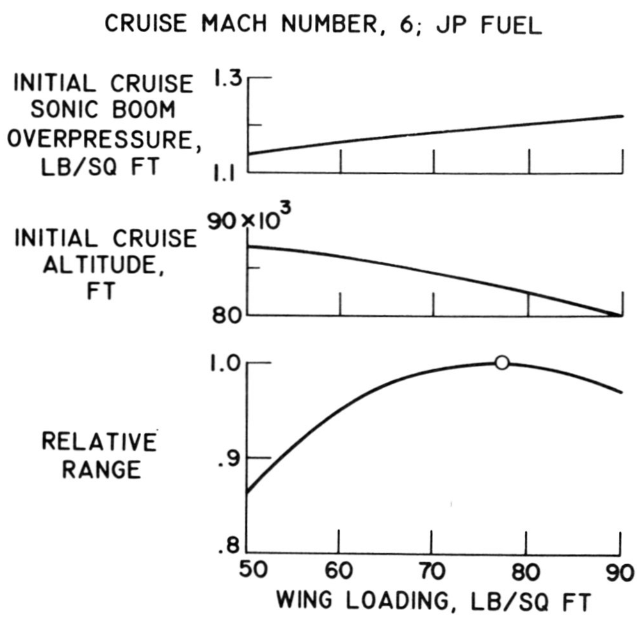

Figure 6. Effect of wing loading.

Wing Loading. The major airplane design parameter considered herein is the wing loading (takeoff gross weight divided by wing planform area). Excessively low values of this parameter yield large, heavy wings. High values yield airplane configurations with relatively small lifting surfaces compared with the fuselage frontal area and hence with a low lift-drag ratio. As shown in Fig. 6, there is a compromise value of wing loading that results in maximum range. Recall that each airplane design was left free to select its best cruise altitude. This altitude is primarily a function of wing loading and is also shown in the figure. The corresponding sonic boom overpressures during cruise are shown at the top of the figure. The indicated values are rather modest (the current supersonic transport design cruise overpressure is 1.5 lb/ft^2); still lower overpressures can be obtained, although with penalty in range, by employing lower than optimum wing loadings and higher than optimum cruise altitudes.

Engine Sizing. Figure 7 illustrates the procedure followed to determine the proper sizes for the turbojet and ramjet portions of the engine. Maximum range is generally obtained by selecting turbojets large enough to provide a takeoff thrust-weight ratio of about 0.5. For an airplane with four engines, this ratio corresponds to a design airflow of about 560 lb/sec.

As suggested by others, a convenient parameter for specifying ramjet size is the ratio of ramjet airflow to turbojet airflow.* Although range is not nearly so sensitive to this parameter, an optimum value also exists.

Turbojet Design. The two major design parameters for turbojet cycles are the compressor pressure ratio and the turbine-inlet temperature. The importance of these factors is shown in Fig. 8. (In performing these calculations, not only were changes in cycle performance included, but also the estimated associated changes in component weights.) With the postulated weight inputs, range increases significantly with turbine inlet temperature until values of the order of 3000°R are reached. There is less sensitivity to variations in design compressor pressure ratio although an optimum value does exist, increasing from about 7 at 2000° to about 10 at 3000°R.

One might expect range to be rather insensitive to the turbojet design, since cruising is purely in the ramjet mode. The turboaccelerator weight, however, is quite significant, and a considerable amount of fuel is consumed in accelerating with the turbojet. Higher turbine-inlet temperatures benefit both these areas. (Note that this high-temperature operation is needed for only a short period during each flight and so may be easier to achieve for this application than for a Mach 3 airplane.)

In the remainder of this paper, the rather conservative value of 2500°R has been employed for turbine-inlet temperature.

* More exactly, the ratio of corrected ramjet airflow at the conversion Mach number to corrected turbojet airflow at sea-level static conditions.

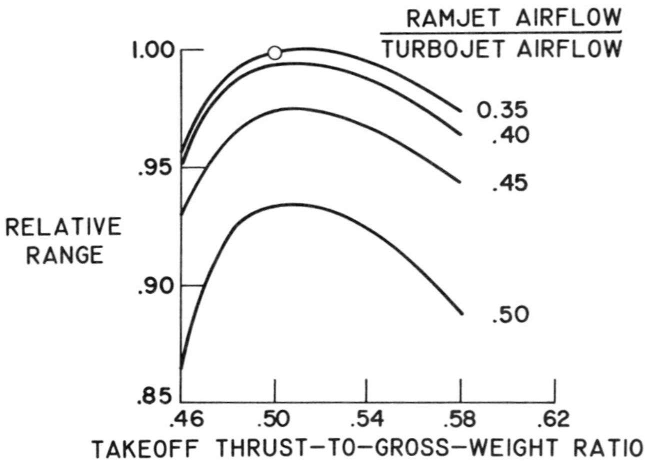

Figure 7. Effect of engine sizing.

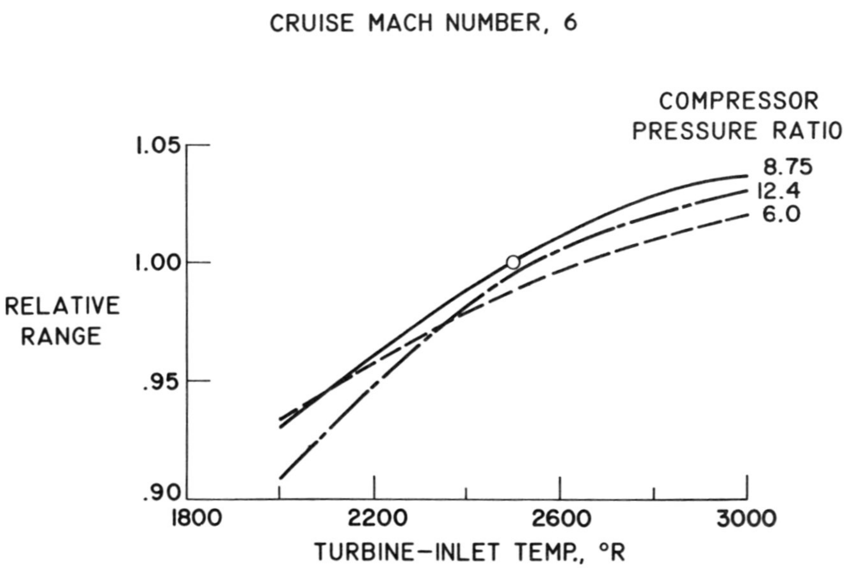

Figure 8. Effect of turbojet design parameters.

Exhaust-Gas Recombination. At hypersonic flight speeds the combustion temperatures are high enough to cause substantial dissociation of the combustion gases. Large amounts of energy are absorbed in this process. If the energy is not regained as the gases expand through the nozzle, serious losses in engine performance will result. At Mach 7, for example, if the expansion is so rapid that no recombination takes place, both the thrust and the specific impulse of the ramjet are reduced by about 25 percent from the potentially available values [7].

The seriousness of this problem was studied by use of a calculation procedure based on the Bray sudden-freezing criterion [8]. As indicated in Fig. 9, the predicted "kinetic" engine performance at a typical Mach 7 operating condition is nearer to the equilibrium—rather than to the frozen-expansion case. Since much of the airplane fuel is consumed at lower speeds where the dissociation losses are smaller, the resultant decrement in airplane range does not appear to be serious. Equilibrium performance is hence assumed throughout this paper.

Jet-Deflection. A well-known technique for improving airplane range consists of deflecting the exhaust gases downward to some extent. Although

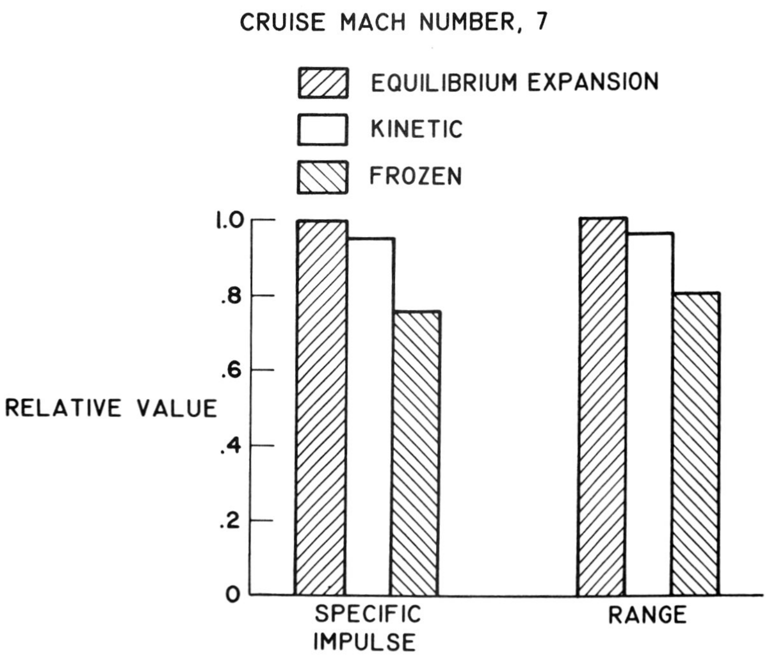

Figure 9. Effect of nozzle dissociation.

a small loss in useful horizontal thrust is suffered thereby, a considerable amount of upward force is generated. The reduced lift required of the wing then results in either a smaller, lighter wing or a decrease in induced drag. Figure 10 shows that about a 4 percent increase in range can be realized in this manner. As predicted by simple theory (e.g., Refs. 9 and 10), the maximum improvement occurs when the exit jet is deflected by an angle equal to twice the wing angle of attack. About three-quarters of the maximum benefit occurs if the engine is simply alined with the wing.

The rather modest gains illustrated are partly due to the fact that the inlet was placed within the pressure field of the wing. As pointed out in Ref. 9, the benefits of inlet location and jet deflection are not additive.

Cruise Mach Number. Figure 11 shows the effect of perhaps the most important airplane design parameter—the cruise Mach number. As a result of the complicated interactions between aerodynamics, structures, engine performance, and flight paths, a characteristic of decreasing range with increasing speed is obtained in the region considered. Of primary importance is what range is required for a practical airplane. Reference 11 suggests a minimum range of 3,500 nautical miles in order to meet the major needs of the potential long-range passenger market. This value was achieved at Mach 5. The corresponding reduction in flight time over a

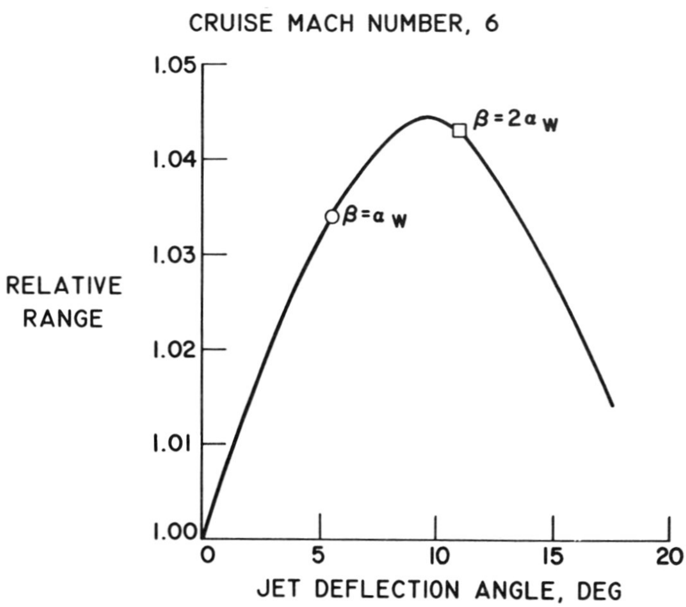

Figure 10. Effect of exhaust jet deflection.

Mach 3 airplane is about three-quarters of an hour (Fig. 1). Greater time savings would be expected at higher speeds. The range decreases so rapidly, however, that this saving is not obtained. At Mach 6, for example, the range is 2,800 nautical miles; the time difference is still about three-quarters of an hour (compared with a Mach 3 airplane of the same range).

RANGE IMPROVEMENTS

Since the vehicle performance presented up to this point is not entirely satisfactory, some techniques that may better the situation will be considered in this section.

Engine Cooling. One reason for the poor performance obtained thus far stems from the engine-cooling requirements. Although the JP fuel is assumed to cool the engine regeneratively, it is an inadequate heat sink at hypersonic speeds. In this study water was considered to supplement the fuel as a cooling medium when required. Also the engine surfaces were coated with a ceramic insulation in order to reduce the heat flux. Despite some fairly optimistic assumptions, the estimated cooling requirements for a Mach 6 vehicle still resulted in about 16,000 lb of cooling water plus a 5 percent increase in engine weight for insulation.

The cooling requirements of actual hypersonic engines may vary considerably from those estimated herein. This variation could occur simply from inability to calculate this factor properly, or it could arise from real

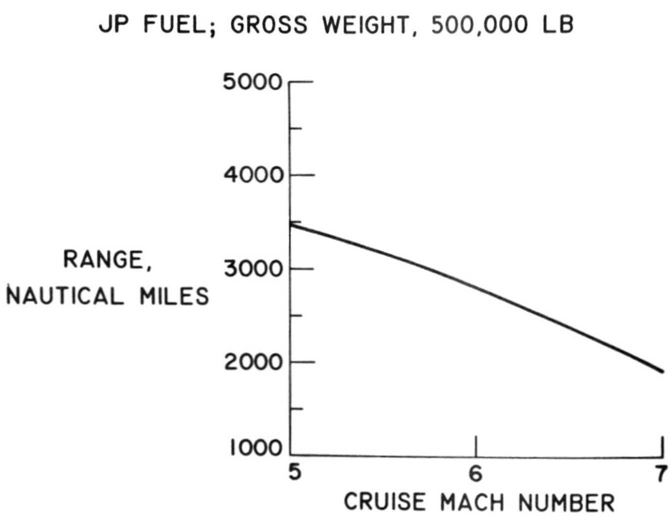

Figure 11. Effect of cruise Mach number.

differences in engine design. (For example, as already mentioned, the variety of ramjets employed in Ref. 1 has a greatly reduced liquid-cooling requirement as a result of enhanced radiation cooling.) Figure 12, which shows the range obtained at Mach 6 with various arbitrary variations in water-coolant flow, demonstrates the benefits that might result from efforts in this area. Because the design changes might also affect engine weight, the effect of this parameter is also shown. If an ingenious designer can in some fashion significantly reduce the coolant flow without excessive engine weight increase, an appreciable improvement in range will result. Even if the coolant flow could be reduced to zero, however, the indicated range is still unsatisfactory.

The reason that the improvement is not greater than it is arises from the somewhat optimistic estimate of the cooling problem that was used as a point of departure. The proper cooling needs may well be much higher than predicted herein, so that even the nominal 2,800-mile range could not be achieved.

Parametric Variations. In order to determine which other aspects of airplane-engine performance might most profitably be improved, arbitrary performance improvements of 10 percent were assumed in the major parameters. The resulting increases in range are shown in Fig. 13.

Higher airplane lift-drag ratios are very desirable as is lower structural weight. Since these quantities were selected rather arbitrarily in the first place, however, they will not be further discussed. An increase in gross

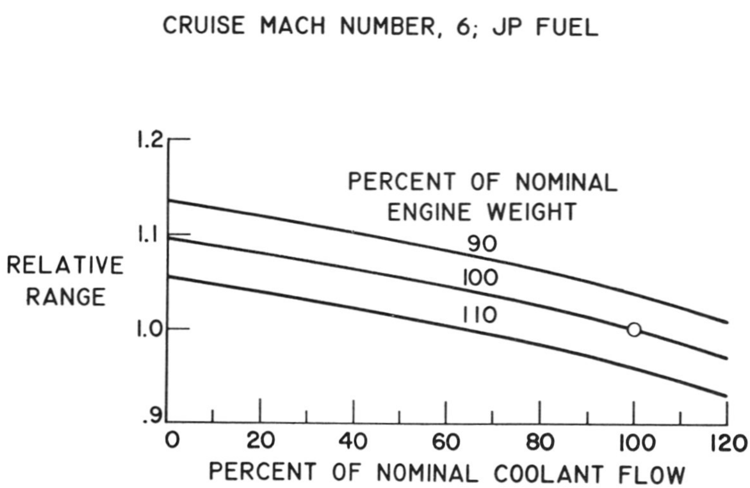

Figure 12. Effect of engine cooling requirements.

weight is helpful, but the payload fraction for a 500,000-lb airplane is only 5.2 percent, and further reductions are probably economically unsound.

In the propulsion area, the sensitivity to engine weight and specific impulse is shown. Engine weight is one of the critical items for the current supersonic transport designs. It is also important for hypersonic airplanes, where it accounts for about 13 percent of the gross weight. Numerically even more important is the engine specific impulse, both during acceleration and cruise. Increases in the turbine-inlet temperature, as already discussed, can benefit the acceleration phase. Other improvements require better component performance, especially during off-design operation. In particular, this refers to inlet pressure recovery and additive drag and exhaust-nozzle thrust coefficient. (The values used herein for a typical flight are shown in Table 1.)

If the 10 percent improvements shown in Fig. 13 could all be achieved simultaneously and the results were additive, the airplane range could be increased by 55 percent. The new range would be 4,350 nautical miles.

FUEL TYPE

An entirely different approach to the problem of securing longer ranges is possible; that is, to employ another fuel rather than the conventional hydrocarbon type assumed so far. This step is not one that can be lightly

TABLE 1

TYPICAL INLET AND NOZZLE PERFORMANCE

Mach number	Pressure recovery[a]	Additive drag coefficient[b]	Nozzle thrust coefficient[c]
0.3	0.950	0	0.862
1.0	0.950	0	0.922
1.3	0.950	0.154	0.943
2.0	0.936	0.052	0.947
3.1	0.812	0	0.979
3.1[d]	0.812	0.282	0.979
4.0	0.693	0.075	0.941
5.0	0.501	0	0.952
6.0	0.280	0	0.954

[a] Not including wing-pressure-field effects.
[b] Based on inlet capture area.
[c] Ratio of actual nozzle thrust to ideal thrust for full expansion to ambient pressure.
[d] Start of ramjet operation.

undertaken. All current commercial jet aircraft use kerosene or JP fuel. Introduction of a new fuel variety with possibly greatly different storage, handling, and safety qualities would cause tremendous operational difficulties. If the need were great enough, however, it could certainly be accomplished.

Approximately 10 years ago intensive studies were conducted at the Lewis Research Center and other organizations on alternative fuels suitable for military requirements (e.g., Ref. 12). On the basis of these studies, two fuels out of the many available have been selected for discussion herein. They are ethyldecaborane (EDB) and liquid hydrogen.

Fuel Characteristics. Figure 14 compares these possible fuels with ordinary JP fuel on the basis of heating value, density, and cooling capacity. EDB is one of a large family of boron-containing fuels which offer substantially higher heating values than JP fuel. EDB, with its 40 percent improvement, is actually one of the poorer members of the family in this respect. It is of interest, however, because of its high density, low toxicity, and general similarity in handling qualities to JP. On the other hand, in common with the rest of the boron-containing family, it has the serious disadvantage of forming boric oxide as a combustion product. At the

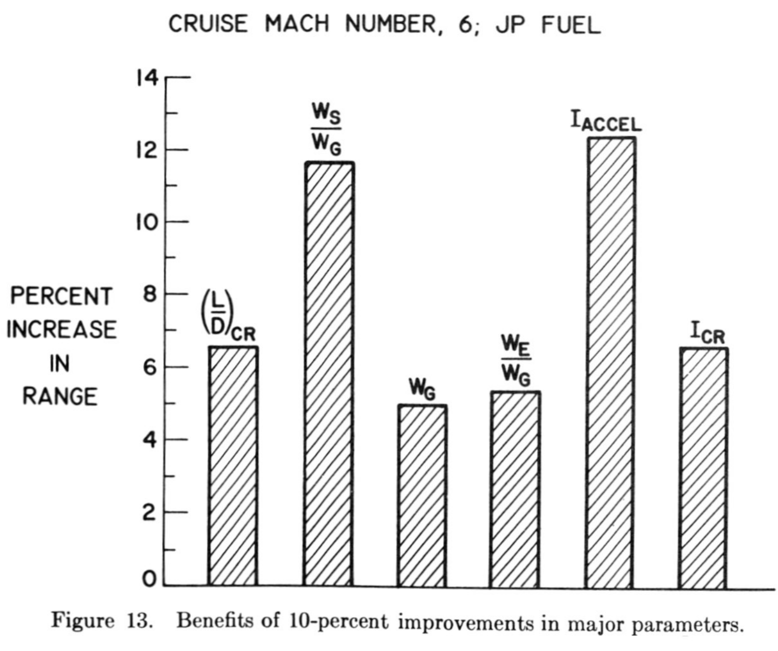

Figure 13. Benefits of 10-percent improvements in major parameters.

temperatures found in a turbojet combustor, the oxide exists as a rather viscous liquid that rapidly fouls the interior engine surfaces including the turbine. At the higher temperatures found in a hypersonic-ramjet combustor, the oxide is formed as a gas. It tends to condense, however, during the nozzle expansion process. If equilibrium condensation does not take place, severe losses in thrust and specific impulse occur, analogous to the nozzle recombination problem previously discussed. (Equilibrium expansion and no deleterious fouling are assumed in the following discussion.) Another disadvantage of EDB, as shown in Fig. 14, is that it is an even poorer coolant than JP fuel. The section entitled Engine Cooling indicates how serious a problem this can be.

Various liquefied gases, such as liquid methane, have been suggested as a solution to the cooling problem. Only the ultimate cryogenic fuel, liquid hydrogen, will be considered herein. Its heating value is nearly three times that of JP fuel, and its cooling capacity about eight times as great. Its density is very low, however, only one-tenth that of JP fuel, which thus requires very bulky fuel tanks. Furthermore, even this density can be realized only if the hydrogen is cooled to $-423°F$. Some obvious storage and handling problems are thus posed.

Airplane Performance. Figure 15 shows the airplane ranges that were calculated for the various fuels. In this figure it was assumed that *no*

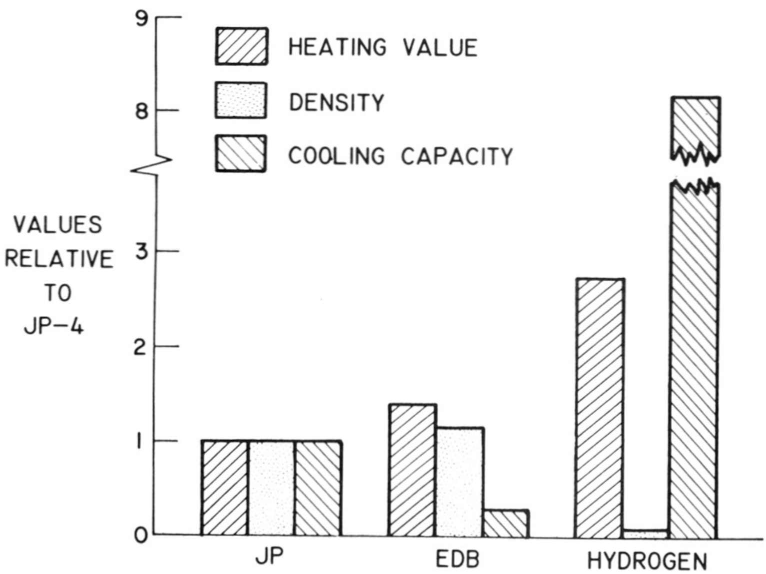

Figure 14. Fuel characteristics.

auxiliary engine cooling was required for any fuel. Our first expectation might be that the range is directly proportional to the heating value of the fuel. This, however, is modified by several factors: Differences in fuel density affect airframe structural fraction and lift-drag ratio, the proportions of fuel consumed during the climb and the cruise phases are altered, and the different thermodynamic properties of the combustion products affect both the thrust and specific impulse. The result of these factors is that both hydrogen and EDB yield a substantially longer airplane range than does JP fuel. The increase in range for hydrogen, however, is only 95 rather than the 170 percent predicted by the heating value; similarly, for EDB, the increase is 27 instead of the expected 40 percent.

The picture is somewhat different if the engine-cooling requirements are incorporated. Both JP fuel and EDB require substantial amounts of water for cooling. Hydrogen, however, is a good enough coolant that no water is necessary in the speed range considered. The result, as shown in Fig. 16, is that hydrogen is now clearly superior to EDB, which in turn is only slightly better than JP fuel. All three fuels are severely penalized by increasing flight speed; however, hydrogen still yields a very adequate range at Mach 7.

Operating Cost. The results of the previous section are given in terms of achievable range for a given gross weight. A more meaningful criterion for

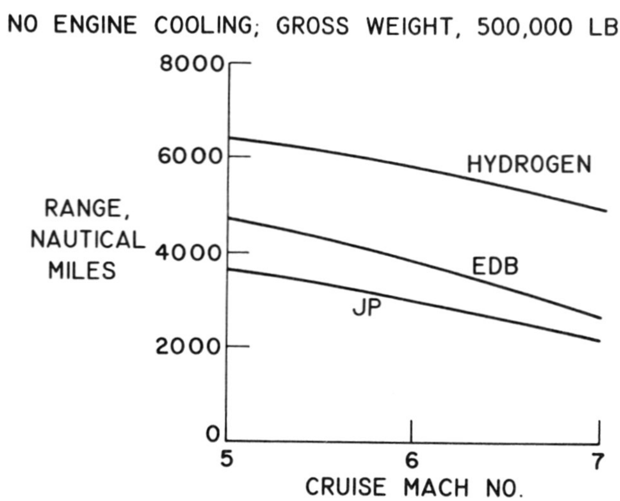

Figure 15. Comparison of different fuels.

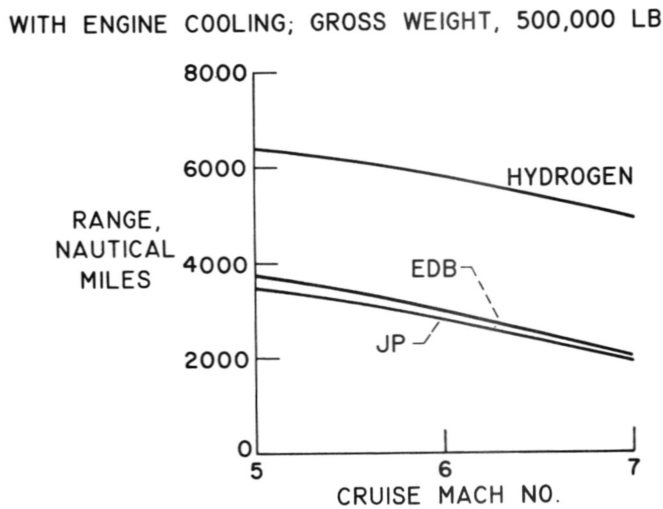

Figure 16. Comparison of different fuels.

commercial applications is cost. As suggested by Ref. 2, the following simplified expression for direct operating cost (DOC) has been used:

$$DOC = \frac{W_g}{W_l}\left[\frac{W_f}{W_g}\frac{\left(\frac{C}{W}\right)_f}{R} + \frac{W_{\text{emp}}}{W_g}\frac{\left(\frac{C}{W}\right)_{\text{emp}}}{V_b}\right]\left[\frac{\text{cost}}{(\text{lb})\ (\text{mile})}\right]$$

where W_g = gross weight
W_l = useful payload
$(C/W)_f$ = cost per pound of fuel
R = range
W_{emp} = empty weight
$(C/W)_{\text{emp}}$ = cost per pound of hardware, \$0.45/(lb)(hr)
V_b = block speed

Typical operating costs for a cruise Mach number of 6 are shown in Fig. 17 as a function of fuel cost. (EDB is not shown since it offers little benefit in range over JP fuel.) Because the gross weights of the two airplanes are the same, they achieve different ranges as indicated. The direct operating cost is primarily influenced by fuel unit cost. If hydrogen could be procured for 2 cents per pound (as can JP fuel), then it would not only double the range but would also halve the direct operating cost. At the present time, however, hydrogen is considerably more expensive than JP fuel. If its cost were 20 cents per pound (a value somewhat lower than at present), the direct operating cost would be 5 cents per seat-mile compared with 1.8 cents for the JP vehicle. These costs compare with a similarly computed figure of 1.5 cents for a hypothetical Mach 3 transport.

Not too much significance should be attached to Fig. 17. The cost calculation was overly simplified, and the ranges were not equivalent. Also, all cost numbers are probably misleading for such an advanced and poorly defined system as this one. It does serve as a reminder, however, that merely looking at range and gross weight is not adequate for predicting commercial success.

CONCLUDING REMARKS

The presented analysis has examined the problem of achieving satisfactory commercial hypersonic flight, with emphasis on the propulsion aspects. With conventional hydrocarbon fuel, a useful range of 3,500 nautical miles is estimated for a Mach 5 aircraft. In order to attain an appreciable time advantage over Mach 3 vehicles, however, achieving greater ranges and/or higher speeds is desirable.

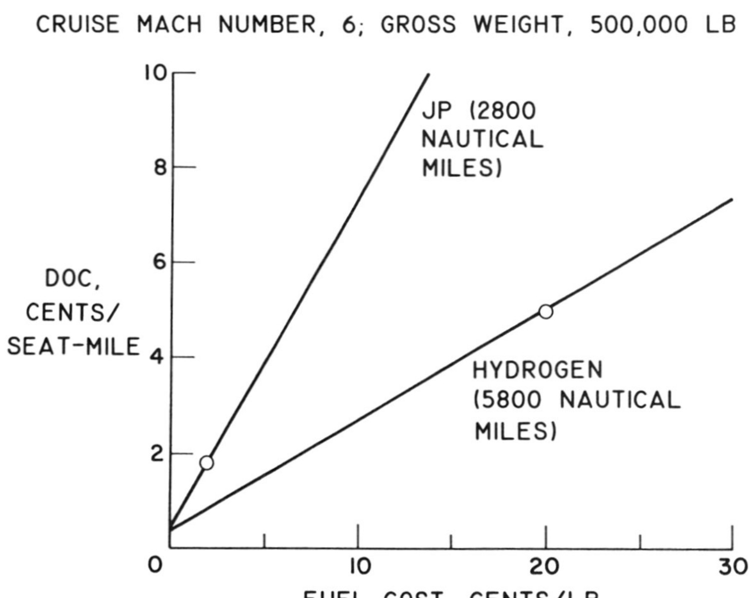

Figure 17. Direct operating cost.

A number of improvements in the propulsion system can be visualized that would aid this situation. Lighter-weight engines with better-matched variable-geometry inlets and nozzles are desirable. Reduced cooling requirements are helpful. High turbine-inlet temperatures are useful; as a result of short operating times they would be easier to achieve for this application than for the supersonic transport. A basic change that cannot be ruled out is the possible use of other engine systems than the turboramjets of the present study.

Such propulsion improvements coupled with better airframe characteristics (beyond the present, already optimistic assumptions) could make hypersonic flight with hydrocarbon fuel quite feasible. Alternative approaches are also possible. More complex operational techniques such as staging or refueling were arbitrarily rejected in this study. The paper has, however, considered a technique of scarcely less formidable proportions: the use of unconventional fuels. While ethyldecaborane offered little benefit, liquid hydrogen did yield very superior performance. A possible further advantage of hydrogen not yet fully explored lies in the use of special engines that employ the cryogenic properties of liquid hydrogen.

A major problem worth reemphasis is that of sonic booms. If an overpressure of much less than 2 lb ft^2 in the vicinity of airports is required, then the difficulty of designing useful hypersonic aircraft is greatly aggravated. A reduction of only 0.1 lb ft^2 causes over 10 percent loss in range. On the other hand, the overpressures during cruise are moderate; in this respect the hypersonic vehicle seems better than a supersonic transport.

In view of the many assumptions that went into this study, any conclusions that are drawn must be viewed with some caution. It does appear that achieving a useful hypersonic aircraft will present quite a challenge. If the conventional JP fuel is retained, substantial improvements in performance are necessary. On the other hand, hydrogen offers adequate flight performance but has a cost and logistics problem. Since the presented study was so limited, neither a pessimistic nor an optimistic conclusion is justified. While it is thus too early to proclaim the advent of hypersonic transports, it is believed that the results warrant making more refined analyses and conducting supporting research to improve our ability to realistically evaluate this mode of transportation.

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