HYPERSONIC RAMJETS

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Abstract—Hypersonic ramjets using supersonic combustion are discussed. Previous papers show that the supersonic combustion ramjet (SCRJ) offers better performance than the conventional ramjet (CRJ) between Mach 7 and 10.

This paper discusses the diffuser, combustor, and nozzle design problems and shows that unlike the CRJ the SCRJ imposes no insuperable problems as speed is increased even up to satellite levels. Much research and development will be required.

Performance estimates for the SCRJ, using the best current data suggest high propulsion efficiencies up to Mach 20, thus making it attractive for long range high speed aircraft, and for boosting large tonnages into orbit.

INTRODUCTION

It is well known that the ramjet engine does not have any useful performance until the flight speed of the vehicle to which it is attached begins to approach the speed of sound; however, it has clearly demonstrated its usefulness in the low supersonic regime (Mach 2–3) as a means of propelling anti-aircraft missiles such as Bloodhound and Bomarc. Studies have indicated, moreover, that in combination with the turbo-jet engine, it offers advantages over purely turbo driven jet engines for aircraft cruising at Mach 3[1], and that further it has possibilities as a means of propulsion for hypersonic aircraft at speeds around Mach 7[2].

It has usually been considered however, that the field of usefulness of the ramjet will end at a forward speed such that the stagnation temperature of the inhaled air reaches the conventional limiting flame temperature resulting from adiabatic combustion of the particular fuel to be used. Such a limitation, analogous in some ways to the temperature limitation on flight speed of turbo jet engines, would appear to restrict the use of ramjets to speeds of about Mach 7.

It is the purpose of this paper to show that there is good reason to believe that this limitation does not, in fact, apply. It is shown that, if certain assumptions are satisfied, the ramjet engine can operate, and operate efficiently, at extremely high speeds indeed. If the so-called flight corridor be, in fact, broad enough to permit consideration of an air supported vehicle achieving escape velocity, there appears to be no fundamental reason to exclude equally the possibility of propelling such a vehicle with an air breathing engine, thus achieving economies in fuel consumption that are impossible with present or future chemical rocket engines.

THE BASIC PREMISE

The basic argument can be illustrated as follows. Imagine a simple laboratory experiment in which is burned, under adiabatic conditions, a given mixture strength of a given fuel with air or oxygen. A certain final flame temperature will result which will not be as high as that calculated on the assumption of a complete reaction between the fuel and the oxygen because of the effects of dissociation. If the same experiment were performed in a large satellite travelling in orbit, an observer travelling in the satellite would find the same final flame temperature as did the observer on the ground, but of course, from the point of view of the observer on the ground, the total temperature of the flame in the satellite experiment has not only the value measured at the satellite, but also the equivalent of the 26,000 ft/sec of translational speed, and this would in fact, be a very high stagnation temperature. The essential point here, of course, is that the fact that although the molecules of air and fuel in the satellite experiment happen to have a very large and uniform translational velocity, this does not in any way affect their collision frequency and consequently the equilibrium of reaction.

It seems clear, therefore, that provided the stream of air which is travelling at high velocity with respect to an engine is not slowed to stagnation conditions but has its diffusion restricted to such a limit that the static temperature after diffusion is below the normal flame temperature, there should be no fundamental reason why fuel could not be burned and its energy usefully released in the combustion chamber.

It is easy to show that as the forward speed goes up, so must the speed after the diffusion if a low value of the static temperature is to be maintained, and this forces the use of supersonic combustion. After years of designing combustion systems in which a very low velocity was necessary for stable combustion, it is perhaps surprising to talk of supersonic combustion, but as will be shown later there is no reason to doubt that this is possible, either on theoretical or experimental grounds.

We find, moreover, that once we accept the premise of supersonic combustion, there appear to be many other incidental benefits. Firstly, the diffusion problem is much simplified because there is no necessity to have a normal shock; secondly, since we never get near a value of Mach 1, we avoid the high values of heat flux corresponding to this condition; and thirdly, the problems of the expansion nozzle are easier because the change in velocity required in the nozzle is less. There are also possible advantages arising from wholly supersonic flow in making it feasible to dispense with complete enclosure of the combustion system, thus giving an external engine with better cooling and possible augmentation of lift. On the debit side, is the fact that momentum losses corresponding to heat addition in a supersonic stream are obviously much greater than those for a similar temperature ratio in a subsonic stream, but it appears that at hypersonic speeds this disadvantage is more than compensated for by the other advantages.

Summarizing the argument, we believe that if means can be devised to mix and burn fuel with a fast flowing stream of air, without at any time bringing the bulk of that air stream down to a velocity at which the conversion of kinetic energy into static temperature brings values higher than the conventional stoicheiometric maximum temperatures, then it appears reasonable to suppose that the ramjet engine can be used to at least satellite velocities. It will be noted that this approach to the hypersonic ramjet does not call for large amounts of molecular re-association in the propelling nozzle. In the following sections of the report, the various factors are considered in more detail.

THE DIFFUSER

It is obvious that there is a progressive loss of total pressure in a supersonic diffuser as the exit Mach number is progressively reduced from the flight Mach number.

This is illustrated in Fig. 1 which shows fixed geometry inlet performance results obtained at the UAC Research Laboratories[(3)]. Here the oblique shock loss, the boundary layer loss, the normal shock loss and the subsonic loss are shown individually for diffusion to subsonic velocities with external compression. In the case of an SCRJ, only the oblique shock loss and boundary layer loss would be present, and the large improvements in performance are immediately apparent.

With internal compression inlets similar advantages are obtained since the bleed or area variation required to satisfy the starting condition will be comparatively small.

COMBUSTION

Numerous experiments have shown that if fuels are injected into a hot stream of air, combustion will occur spontaneously after a certain time delay even though there is no flameholder. Typical results for hydrogen

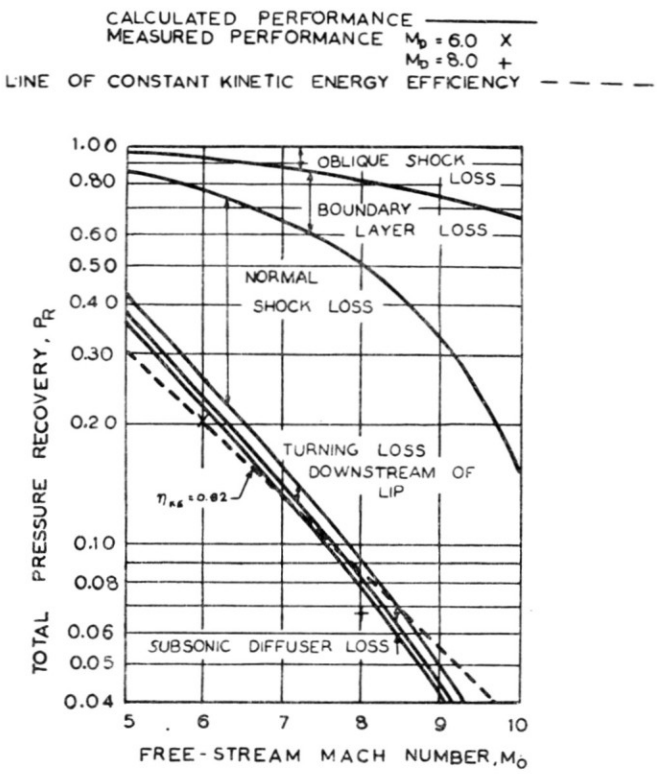

FIG. 1. Hypersonic inlet performance.

kerosene and n-butyl nitrite are shown in Fig. 2[7], and it will be seen that at temperatures of the order of 800–1000°C, the delay times are generally of the order of milliseconds. Over the lower temperature ranges there is an essentially linear relationship between the time and the logarithm of the temperature and extrapolation to temperatures of the order of 1500°C show delay times of the order of microseconds. The slight curvature of this relation is probably due to a physical mixing delay, superimposed on the inherent chemical delay. As would be expected for the reasons given above, the results also show that combustion delay is independent of velocity over a range up to 1500 ft/sec. and this can safely be assumed to hold at any velocity. Hence, no reason can be seen why, if fuel can be injected or mixed with the air stream without producing a strong shock system, it will not burn in times of the order of microseconds so that even at velocities of thousands of ft/sec the actual physical length of the combustion region is going to be very small indeed.

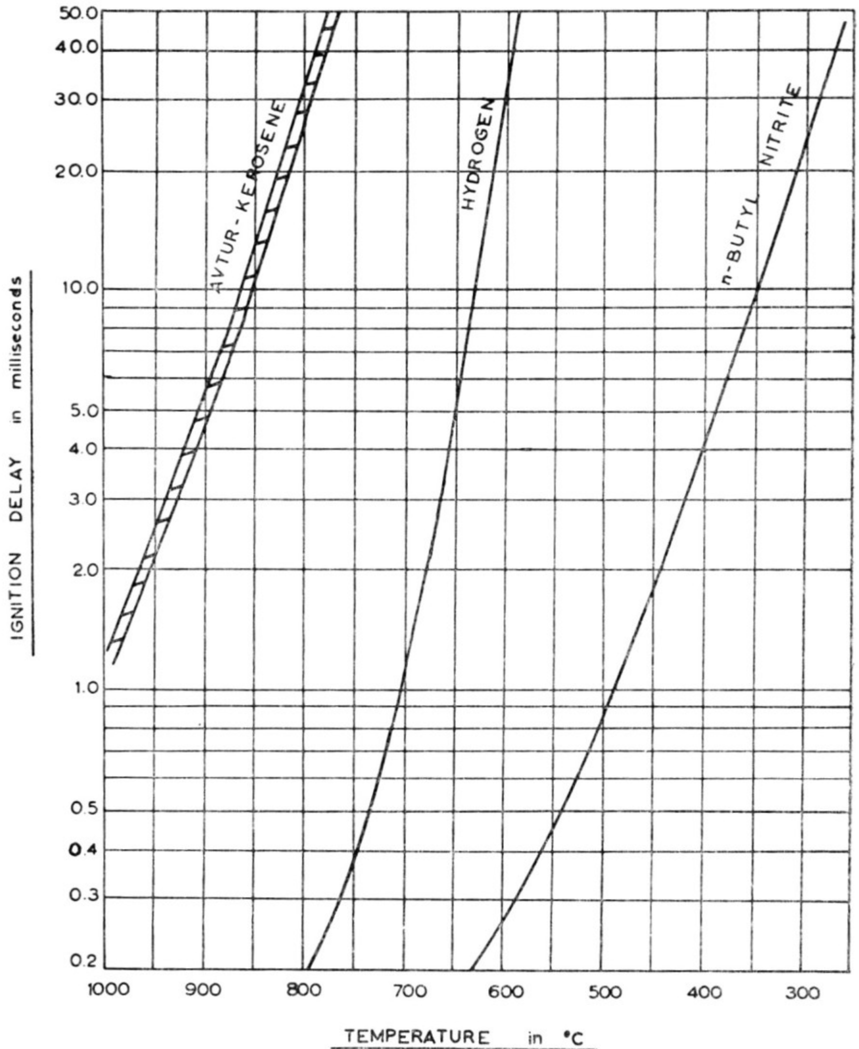

FIG. 2. Ignition delay vs temperature.

FUEL INJECTION

It seems to the writers that the more serious problem is that of injecting the fuel without causing undue disturbance to the main stream, and in particular, injection must be achieved without strong shocks. Mixing prior to combustion must also be extremely rapid if the length of the mixing region is to be acceptable. The influence of various fuel injection techniques on these factors is now discussed.

(i) Direct injection of gaseous or liquid fuel either up-, cross-, or down-stream is the conventional method for combustion systems, however the formation of shocks on the structural or fuel jet appears inevitable in this case. The shock losses are minimized for oblique downstream injection, but the fuel is quickly accelerated to the stream velocity, and unless it is very highly reactive (such that mixing is accelerated by the combustion process), satisfactory operation appears unlikely.

(ii) The reaction rate is increased for a hot fuel hence one possible approach is to preheat the fuel, and two methods of preheating are proposed. The first is to use the fuel as a convection coolant for those hot parts of the engine and airframe where some form of cooling has to be provided. The second is to use a mono-propellant fuel (such as methyl acetylene) in a precombustion chamber, and inject the hot reactive products into the air stream as a subsonic or supersonic fuel jet.

(iii) The boundary layer in the engine diffuser will be at a high temperature and moving relatively slowly. If the fuel is introduced into this layer through a porous wall, cooling of the wall is achieved, while the hot over-rich film is gradually accelerated up to the stream velocity. After an ignition delay period the combustion and mixing take place in the combustion chamber. Preliminary experiments have shown that this technique is complicated by interaction of the combustion with the normal viscous effects.

EXPANSION

Propulsive nozzles for use at high gas temperatures pose many problems, especially as any small inefficiency becomes a large loss in overall engine performance. Optimation of the nozzle is therefore very important and it will be seen that the supersonic combustion cycle offers many advantages to the designer. The most significant factors are expansion ratio, internal and external drag, heat transfer, weight and dissociation.

Since an all supersonic nozzle requires no throat, the expansion ratio, internal drag, heat transfer and weight problems are all reduced.

At hypersonic flight speeds, the high temperatures can lead to considerable dissociation in the engine. If recombination is not achieved in the nozzle, a serious loss of thrust occurs. For a supersonic cycle, the temperature and dissociation can be held at a low level at entry to the nozzle, thus permitting equilibrium exit flow to be obtained over a much wider range of conditions than for a subsonic combustion cycle.

At this point the effect of scale should be mentioned. Since recombination is time dependent, it will be easier to achieve recombination in a large nozzle than a small model. However recent data has shown that the transition from equilibrium to frozen flow is not very sensitive to scale effects[8].

HEAT TRANSFER CONSIDERATION

The heat transfer to the engine components is the sum of the convection and radiation effects. For any gas stream, the convective heat transfer to a wall is approximately proportional to ϱ_V, and thus is a maximum at a Mach number of 1. An engine with supersonic Mach numbers throughout will not attain this maximum convective heat transfer at any point.

Wall heating by radiation energy from the gas stream is a more important factor at high gas temperatures, however, the much lower gas temperatures obtained when the flow is supersonic reduce this problem.

The above generalizations indicate that the heat transfer considerations are much easier for a supersonic combustion cycle, although detailed analysis depends on the engine geometry and stream temperature profiles.

It may be of interest to discuss the geometry briefly: in supersonic flow, the stream along a wall is bounded by the Mach lines rather than by any enclosing wall immediately opposite. This is increasingly significant as the Mach number rises and the Mach lines are more nearly parallel to the wall. We feel that at high Mach numbers it will be possible to remove the second wall without affecting the engine flow significantly, and the engine becomes completely external. The advantages of such a power unit whether fitted to the wings or body are immediately obvious, since

(i) intake starting is simplified,
(ii) radiation cooling can be employed,
(iii) the power plant weight is almost nil.

THEORETICAL CYCLE CALCULATIONS

Symbols and Abbreviations:

A	Area
CRJ	Conventional ramjet
C_V	Nozzle velocity coefficient
H	Height (ft)
L/D	Lift/drag ratio
M	Mach number
M	Molecular weight
M_0	Undissociated molecular weight
p	Static pressure (lb/in.^{2})
P	Total pressure (lb/in.^{2})
q	Dynamic pressure (lb/ft^2)
f	Range
SCRJ	Supersonic combustion ramjet

S/R	Entropy (non-dimensional)
sfc	Specific fuel consumption lb/(hr lb)
t	Static temperature (°C or °R)
T	Total temperature (°C or °R)
V	Velocity (ft/sec)
w_1, w_2	Initial and final weights
W	Air mass flow (lb/sec)
X_n	Net thrust (lb)
Z	Molecular weight ratio
α	Fuel/air ratio
η_e	Efficiency (%).

From the theoretical standpoint, the first impression of the supersonic combustion cycle is gloomy since energy and momentum considerations show that if heat is added to a supersonic air stream to choke the flow the same final temperature and pressure will be obtained whether supersonic heating or a normal shock followed by subsonic heating is used. It follows that at Mach numbers below 6 where the heat addition before choking a constant area combustion chamber corresponds to stoicheiometric hydrogen fuel, the supersonic cycle is inferior to the conventional ramjet, since the conventional ramjet can easily obtain and utilize extra diffusion.

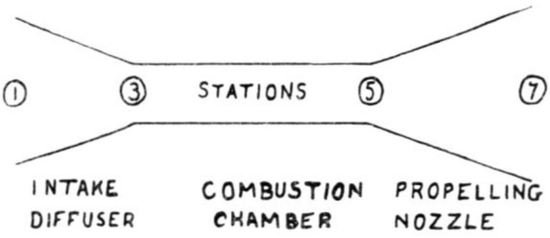

FIG. 3. Reference stations diagram.

Both references 4 and 5 compare the performance of hypersonic ramjet engines with subsonic and supersonic combustion, and show that the SCRJ overtakes the CRJ at about Mach 7. By Mach 10 the CRJ thrust is falling rapidly whilst the SCRJ efficiency is becoming very high.

To carry out theoretical cycle calculations at hypersonic speeds, imperfect gas effects must be taken into account by using Mollier diagram data for gas properties. To date we have generally used equilibrium flow throughout the engine, and inserted intake and nozzle efficiencies of any assumed value. Figure 3 illustrates the reference stations used in the following discussion. The results presented here are all based on a combustion chamber of constant area, and hydrogen fuel. The former restriction reduces the number of variables and precludes the evaluation of an optimum configuration, however practical considerations such as the rate of heat release override this limitation.

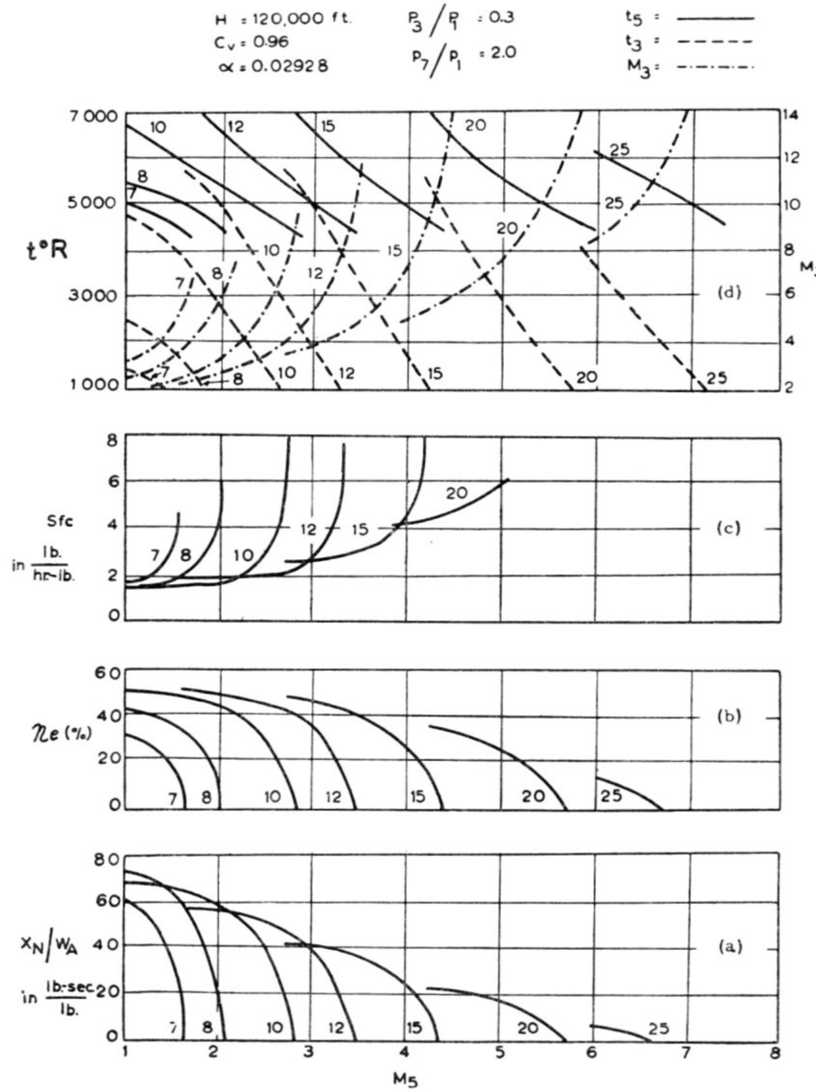

FIG. 4. Performance vs. Mach number after combustion.

Figure 4 shows the variation of impulse, overall efficiency, sfc, temperature and diffuser exit Mach number (M_3) for several flight Mach numbers plotted against the Mach number after combustion (M_5). The interesting characteristic of these curves is the knee which divides the region of almost constant performance from the region of low performance. In order to obtain the best intake and nozzle performance, the highest internal Mach numbers are required which do not prejudice the thermodynamic efficiency. Using M_5 as the parameter, a close approximation to the optimum internal Mach numbers can be obtained from the knee of the performance curve. This is plotted in Fig. 5 (*d*) as M_5 vs. M_1, and a linear relation appears acceptable.

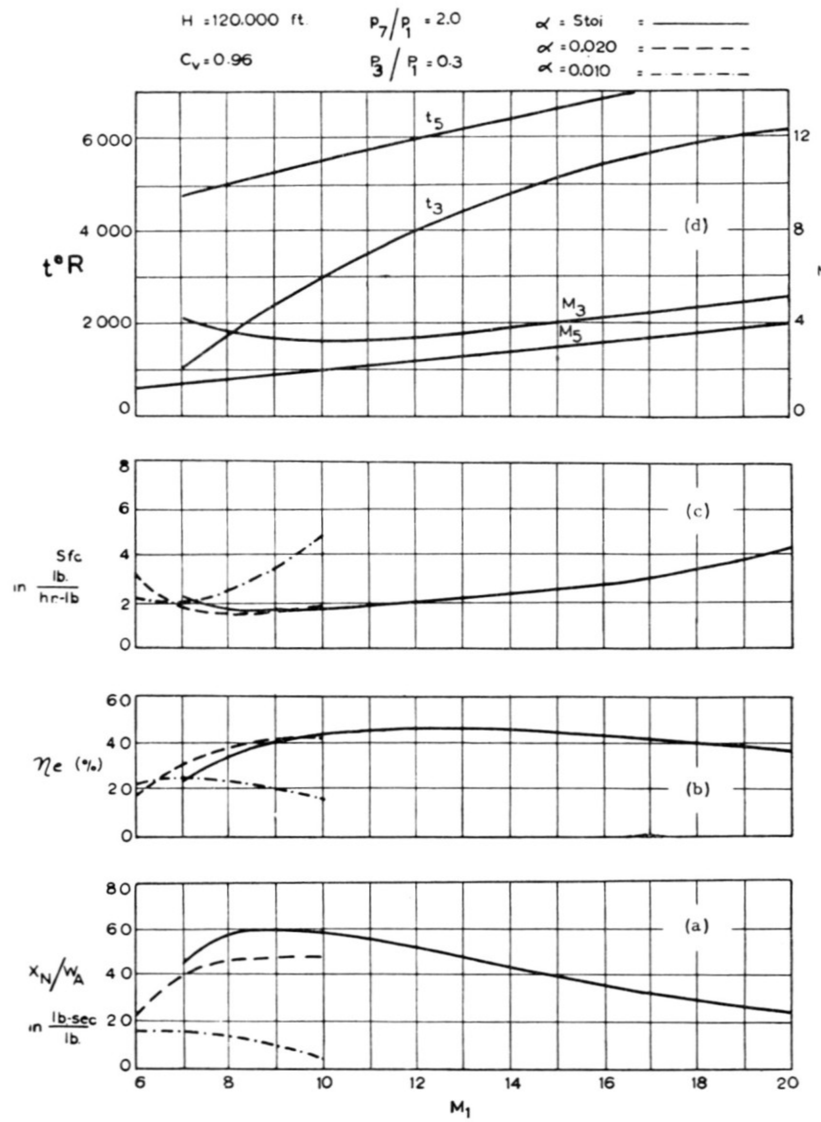

FIG. 5a. Performance vs. flight Mach number.

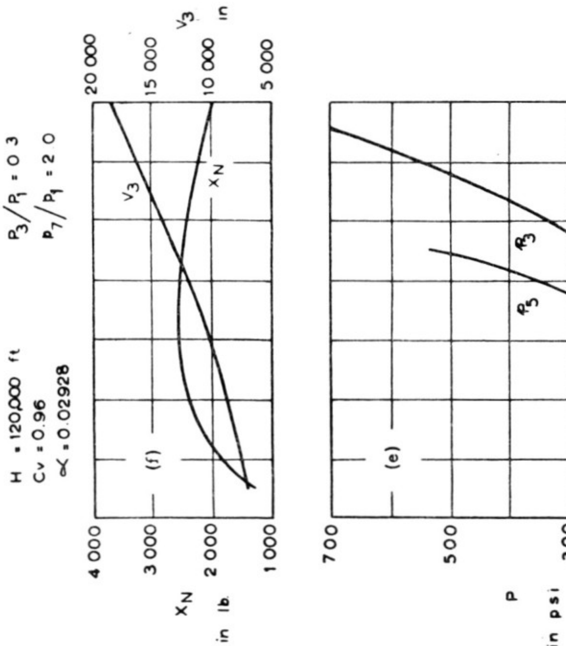

FIG. 5b. Performance vs. flight Mach number (cont.)

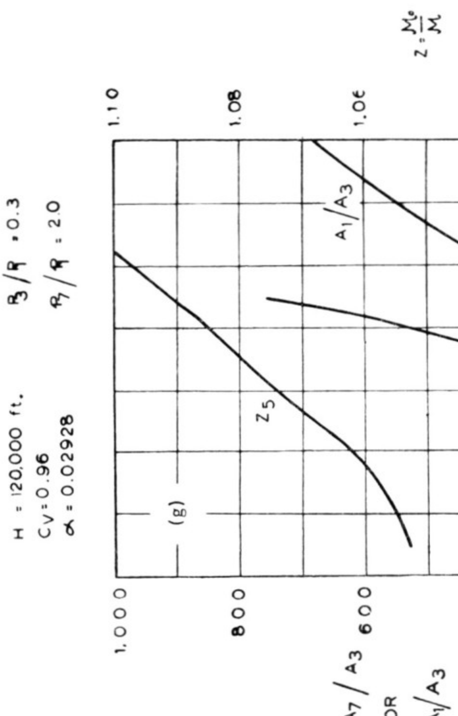

FIG. 5c. Performance vs. flight Mach number (cont.)

Using this relation the performance is plotted against M_1 in Fig. 5 (*a*, *b*, *c*, *d*, *e*, *f* and *g*.) Here a broad performance plateau between Mach 10 and 17 can be seen. Figure 5b also shows that weaker fuel/air ratios than stoicheiometric are required at Mach numbers below 10.

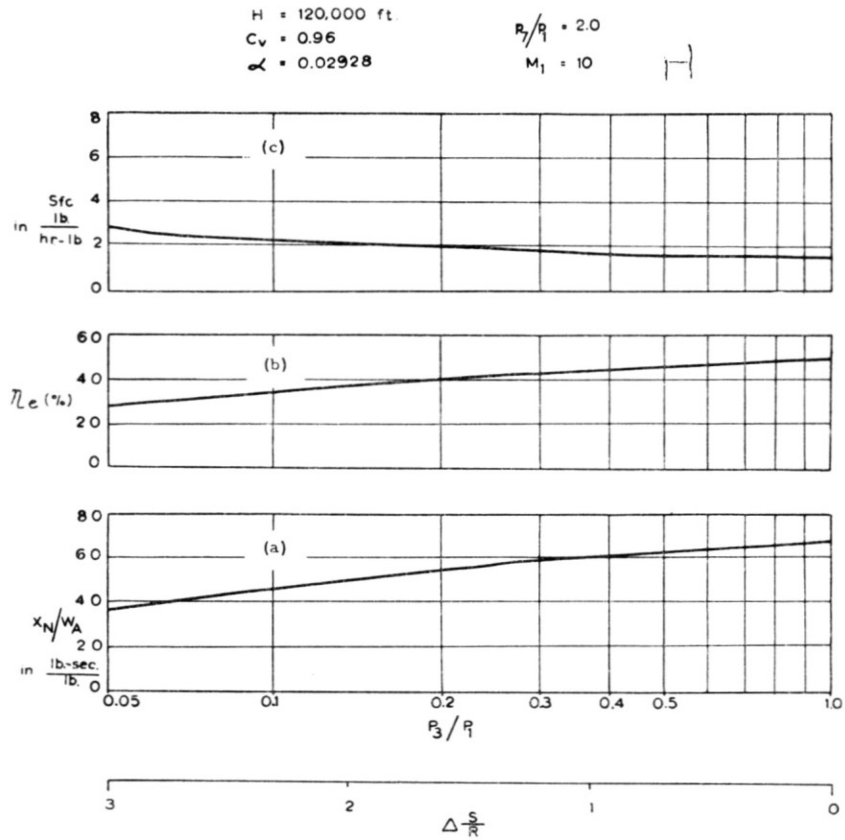

FIG. 6. Effect of intake recovery on performance.

The trends illustrated in Fig. 5 will be distorted since constant intake efficiency, constant nozzle velocity coefficient and constant nozzle exit conditions were assumed. The effects of these three variables are illustrated in Figs. 6, 7 and 8 at a fixed condition of Mach 10 at 120,000 ft. From these graphs it can be seen that performance is very sensitive to nozzle velocity coefficient, and relatively insensitive to the other two variables.

It is well known that heat transfer and lift considerations indicate that hypersonic aircraft must fly at higher altitudes at high velocities; that is in the so called "flight corridor". SCRJ engines are similarly restricted to this corridor, as the thrust is too low above the corridor and the

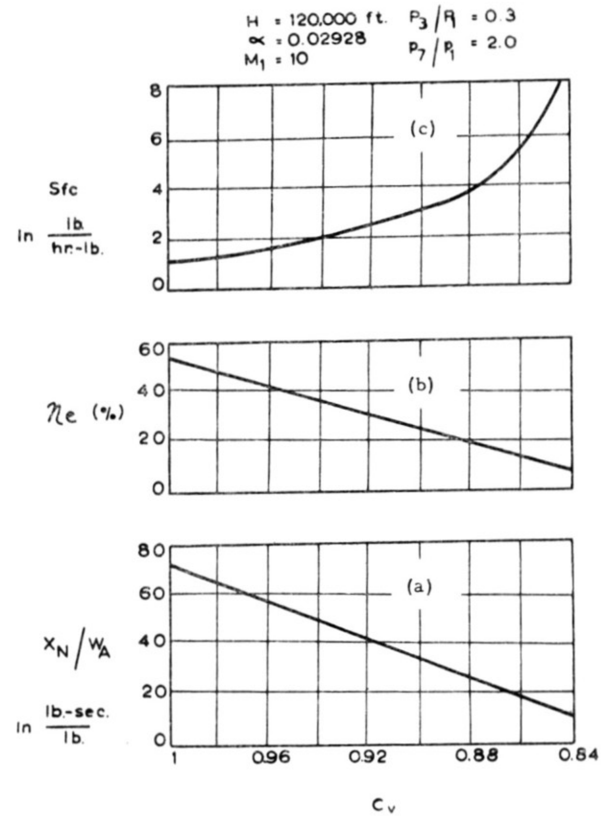

FIG. 7. Effect of nozzle velocity coefficient on performance.

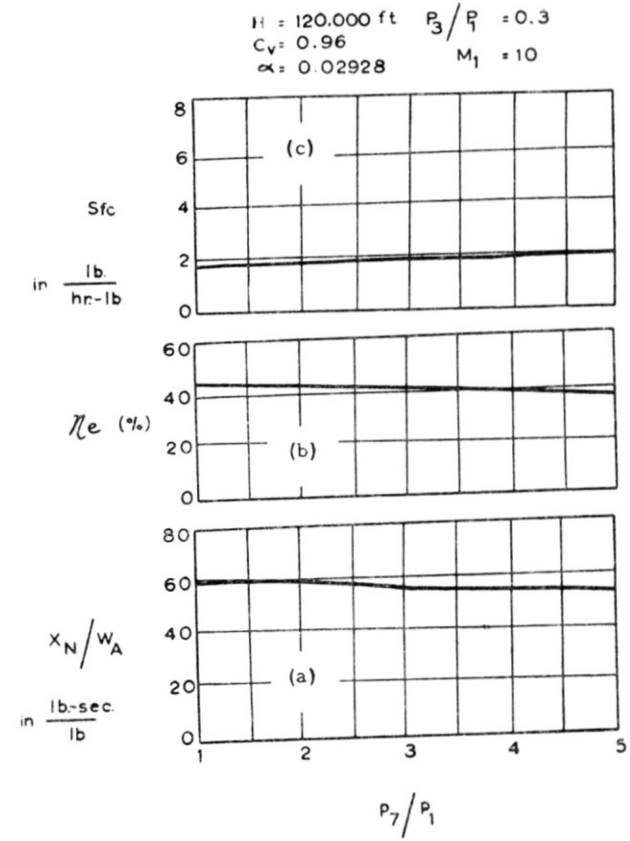

FIG. 8. Effect of nozzle exit pressure on performance.

pressures excessive below the corridor. The engine pressures shown on Fig. 5 (*e*) illustrate this as they were calculated at a constant altitude of 120,000 ft and become excessive above Mach 15.

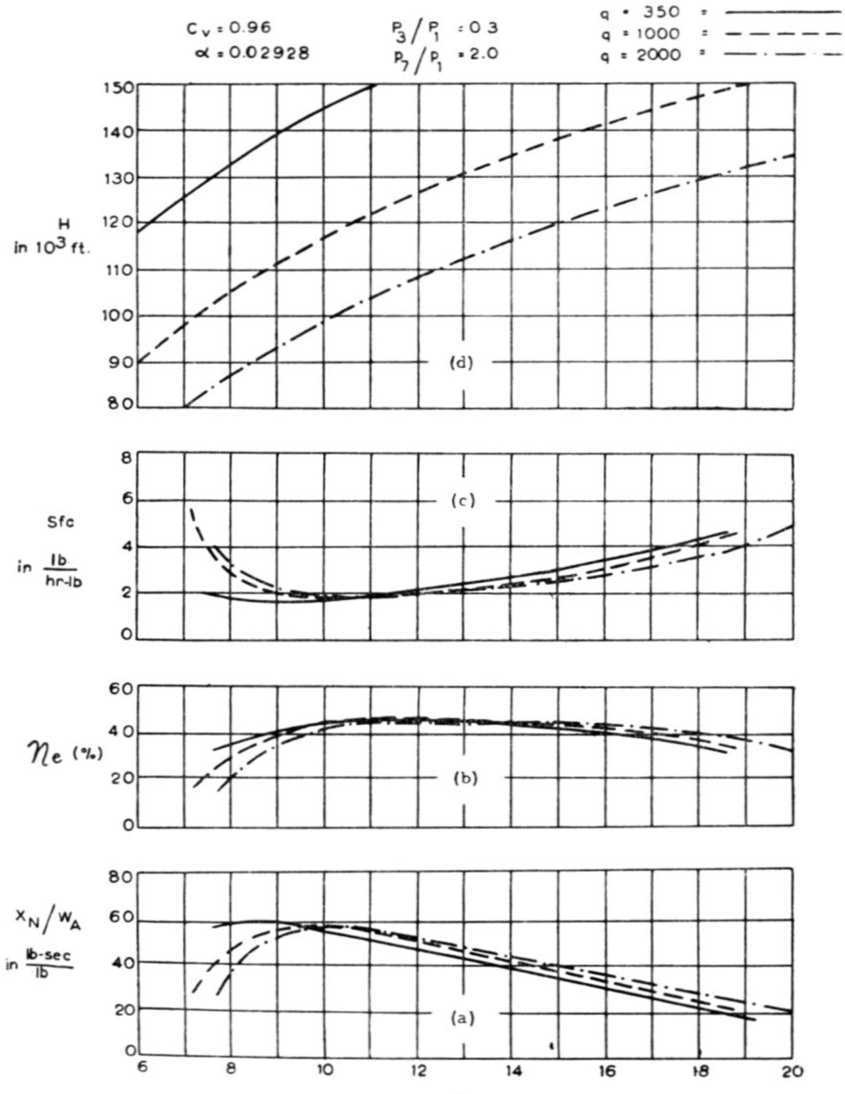

FIG. 9. Effect of dynamic pressure on performance.

Figure 9 has been plotted to take the altitude effect into account by using the altitude corresponding to dynamic pressures of 350, 1000 and 2000 lb/ft^2. In general, trajectories at the lower dynamic pressures are more efficient for cruise applications, whilst the higher dynamic pressures are suitable for accelerating flight.

APPLICATIONS

Possible applications of these engines can be derived by considering the theoretical cycle calculations. The significant parameters are the high propulsive efficiency and high specific impulse. These indicate an attractive performance as a long range cruise engine, and as a booster stage for space flight.

First assuming the engine is to be used for cruise propulsion; the Breuget range equation may be applied:

$$r \propto \frac{V}{\mathrm{s\,f\,c}} \cdot \frac{L}{D} \cdot \log \frac{w_1}{w_2}$$

For supersonic flight, lift/drag ratio and weight ratio are almost constant, thus the range increases with V/s.f.c. This in turn is directly proportional to the engine overall efficiency. Thus hypersonic ramjets are particularly attractive for long range aircraft, and various detailed calculations have shown that operating costs would be lower than for current aircraft(2).

For boosting applications, the low fuel consumption and low weight of the hypersonic ramjet together with the possibiblity of supporting the vehicle by wings, suggest that a ramjet powered recoverable booster would be an economical way to accelerate equipment into space. In a typical 3 stage booster calculation where a ramjet stage is used to accelerate from Mach 1 to Mach 13 and rocket engines are used for the first and third stages. The payload in a 200 mile orbit is 3.25%, which compares very favourably with a conventional rocket booster.

The payload per pound of fuel is even more attractive since a lifting booster would be recoverable, and fuel consumption rather than weight is the important parameter. The increased economy is particularly important when large tonnages must be carried into orbit(6).

DEVELOPMENT

The problems of developing hypersonic ramjets hinge largely on the design of suitable ground facilities to test complete engines. The main problem here is to obtain representative total temperatures and pressures Pebble bed heat exchangers can be used up to 4000°F (2200°C) which corresponds to a flight Mach number of 7.5. Beyond this, none of the current ground techniques satisfy all the requirements, and free flight testing is the only solution at present.

Due to the large cost of testing complete full scale engines, extensive component testing appears advisable, and very detailed analysis of the results must be used to predict the effects of combining the components. This aspect of development thus becomes research and is discussed below.

RESEARCH

Experimental research work can be divided into the principal fields: Intake, Combustion, and Nozzle, plus heat transfer problems which are common to all three.

Preliminary intake testing can be carried out with conventional hypersonic wind tunnels.

Present high temperature air facilities are not capable of attaining representative hypersonic combustion conditions for satisfactory periods. However if we remember that the static temperature largely determines the combustion process, it follows that by reducing the test velocity from the representative value to some lower value, the required temperature can be obtained. The ignition delay distance will then be shorter than the true value, however suitable corrections can be applied in the analysis. Alternatively a more reactive fuel could be used for the tests than intended for complete engines. This latter approach is particularly useful for studying the fuel injection and mixing aspects of the combustion process.

Nozzle research is important since the overall performance of the engine is particularly sensitive to nozzle efficiency. The problem of testing the recombination of the dissociated combustion products in the nozzle is especially difficult and many laboratories are engaged on this problem.

The solution of heat transfer problems is necessary to operate the test rig on most of the previous investigations. However many papers have been written on this subject and it will not be discussed here.

A brief description of our test facility is perhaps of interest. This consists of a zirconia pebble bed heat exchanger operating at air pressures up to 100 psia, and temperatures up to 4000°F (2200°C). The temperature loss at the outlet is about 100°C in 7 minutes of running. The air mass flow is 0.1 lb/sec and this flow can be extracted from the test section at a pressure of 0.5 psia by a vacuum pumping system. The vacuum system contains two stages of steam ejectors in series followed by a mechanical pumping stage. The pebble bed is heated with propane burning in air preheated by an auxiliary natural gas fired heat exchanger to 900°C

CONCLUSIONS

1. The formerly accepted speed limitations of air breathing engines are not in fact valid and useful performance can be obtained up to satellite velocities.
2. At hypersonic velocities (above Mach 7) the supersonic combustion ramjet cycle becomes more attractive than the CRJ, and remains highly efficient up to satellite velocities.

3. Very high efficiency intake diffusers appear possible if supersonic exit velocities are required.
4. Supersonic combustion of conventional fuels is a theoretical possibility at the high temperatures attained in hypersonic flight.
5. Nozzle expansion problems are simplified if the entry Mach number is high, and dissociation level controlled, hence high efficiency nozzles are anticipated.
6. Heat transfer problems are reduced in the supersonic combustion engine compared to the conventional engine.
7. Although supersonic combustion is subject to large pressure losses, theoretical calculations indicate that high propulsive efficiencies and acceptable specific fuel consumption may be obtained at hypersonic speeds.
8. Hypersonic ramjets are attractive both for long range cruise aircraft, and for boosting large tonnages into orbit.
9. Engine development techniques are such that expensive test equipment would be required. Research tests can however be carried out with relatively small scale facilities.
10. Research at McGill is presently in the early stages, and if satisfactory, development will be purely a matter of money which can be justified only by the high performance of the engine.

ACKNOWLEDGEMENT

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DISCUSSION

W.F. HILTON: Is the lecturer familiar with the proposal made in my paper at the Commonwealth Spaceflight Symposium, London, August 1959, to inject the fuel by means of a tripod, the legs of which were swept back behind the Mach angle? This should avoid difficulty due to shock waves from fuel injection.

Furthermore, by using an octagonal duct, it is possible to achieve 8 shock compression through crossed shocks without shock-boundary layer interaction, and without flow deviation from an axial direction. The shock waves would reach the opposite wall after combustion had taken place.

J. SWITHENBANK: In reply to Dr. Hilton's suggestion of fuel jets swept behind the Mach angle we agree that this would reduce the shock interaction in this region; however, the fuel injection problem lies more in the rapid mixing which must be achieved. We are currently conducting tests and analysis of this problem.

Concerning the octagonal intake, this is an interest ing concept and we would be pleased to hear the results of the experimental tests which he has under way.