# REALLY HIGH SPEED PROPULSION BY SCRAMJETS

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## Nomenclature

| | | |
|---|---|---|
| $A$ | area | |
| $D$ | drag | lb. |
| $h$ | static enthalpy (prime denotes isentropic value, see Fig.6) | $\frac{\text{Btu}}{\text{lb}}$ |
| $I_f$ | fuel specific impulse | $\frac{\text{lb sec}}{\text{lb}}$ |
| $J$ | mechanical equivalent of heat ($J = 778 \frac{\text{ft.lb}}{\text{Btu}}$) | |
| $K_D$ | process efficiency | |
| $m$ | weight | lb. |
| $M$ | Mach number | |
| $p_t$ | total pressure | $\frac{\text{lb}}{\text{ft}^2}$ |
| $p_{2i}$ | ideal static pressure at station 2 | $\frac{\text{lb}}{\text{ft}^2}$ |
| $p$ | static pressure | $\frac{\text{lb}}{\text{ft}^2}$ |
| $q$ | dynamic pressure | $\frac{\text{lb}}{\text{ft}^2}$ |
| $R_e$ | Reynolds number | |
| $T$ | thrust | lb. |
| $t$ | temperature | °K. |
| $V$ | velocity | $\frac{\text{ft.}}{\text{sec}}$ |
| $\eta_{KE}$ | kinetic energy efficiency | |
| $\rho$ | density | $\frac{\text{lb}}{\text{ft}^3}$ |

## Subscripts

| | |
|---|---|
| 0,2,3,4 | Reference stations (Fig. 2). |
| f | fuel |
| t | total conditions |

# Really High Speed Propulsion

E. T. Curran and J. Swithenbank

## INTRODUCTION

A quiet revolution is currently taking place in the field of propulsion, as engineers begin to appreciate the remarkable capability of the scramjet. The scramjet, or supersonic combustion ramjet, is a very simple device consisting only of an intake, combustion chamber and nozzle. It differs from the ramjet in that, at hypersonic speeds, the air velocity is only decreased by a few percent in the inlet, and the combustion heat release occurs at supersonic velocity. Although its efficiency is small at low supersonic speeds, Fig. 1 shows that it can achieve high efficiency over a very wide velocity spectrum. Furthermore, for much of this range, a fixed geometry (2)* can be used with little loss in efficiency (as indicated by the dashed line on Fig. 1), thus presenting unparallelled capability for a simple engine unit.

In the same way as the subsonic-combustion (conventional) ramjet has been successfully combined with a turbojet in the Nord Griffon Turboramjet power plant, the scramjet can probably be combined with a turbo-engine to produce a unit equally suitable for operation from pedestrian speeds to orbital speeds or above. Such a propulsion system could well power a near ultimate vehicle

* Numbers in brackets indicate references at the end of paper.

for atmospheric flight - a very real prospect which must excite all aircraft engineers. Furthermore, this propulsion system is no longer just a theoretical possibility, but its feasibility has been confirmed by extensive computer studies and exploratory experimental tests. The engine is complementary to a new aircraft concept, in the historical series of classical swept and slender aircraft, which is beginning to take shape largely as a result of U.K. effort (1). Their union would produce a new aeroplane - a fairly narrow delta, lifting-propulsive body, in which the engine and the airframe would be completely integrated. Interestingly enough the low speed performance of the vehicle is also likely to be acceptable (1).

Such vehicles may be suitable for various hypersonic missions, covering the range from high speed boost to long range cruise. In the case of the boost mission, it may be more advantageous to combine the scramjet with a rocket rather than a turbo-machine. The rocket may be either a simple first stage booster or a unit integrated with the main scramjet system. At hypersonic speeds, particularly in the case of boost missions, the engine inlet capture area must be almost equal to the projected frontal area of the vehicle. The need for integration of the airframe and engine, even from the earliest design stage, thus becomes apparent.

Despite the high promise of the vehicle, there are obviously many problems to be overcome in the following fields:-

1. Lightweight airframe structures
2. External thermal protection and fuel tank insulation
3. Aerodynamic problems, including boundary layers, stability and control.
4. Reliable, lightweight, vehicle systems.

However such problems are beyond the scope of this paper and attention will now be turned to the propulsion system. To avoid duplication of other articles contained in this special issue, the analysis of engine performance will be directed to the high hypersonic speed range in excess of Mach 10.

To a great extent, the incentive to achieve air breathing boosters accelerating from currently attainable speeds (say Mach 4) to Mach 7, depends on the potential performance above Mach 7, since the step from Mach 4 to Mach 7 alone would barely justify intensive effort. However, the prospect of a simple fixed geometry engine then operating from Mach 8 to Mach 18 is most attractive.

ENGINE PERFORMANCE

A schematic arrangement of a scramjet is shown in Fig. 2, and various stations are identified. The performance of such an engine in relation to other power plants can be presented as a graph of fuel specific impulse vs. Mach number as shown in Fig. 3. This figure also shows the relative performance of a scramjet operating on hydrogen and conventional hydrocarb fuel. The superiority of hydrogen fuel in the upper hyperson speed range is apparent, and for this reason hydrogen must be used at the highest flight speeds. The scramjet specific impulse band shown on Fig. 3 indicates the typical sensitivity of performance to assumed component efficiencies.

From Fig. 3, it can be seen that the scramjet performance overtakes that of the ramjet at about Mach 7, although if necessary the scramjet could be used from about Mach 4. This change-over arises because the intake diffuser losses become prohibitive in a conventional ramjet at high Mac numbers, whilst the combustor losses become prohibitive in a scramjet at low Mach numbers.

An important parameter in the design of a scramjet is the amount of diffusion carried out in the intake. This amount is usually specified either by the velocity ratio $\frac{V_2}{V_o}$ or enthalpy ratio $\frac{h_2}{h_o}$ across the intake. The effect of diffusion on performance is illustrated in Figs. 4 and 5 for a scramjet engine operating at 16,000 ft./sec. If one consider first Fig. 4, the variation of engine performance with diffusi is shown for constant values of $\eta_{KE}$. As the amount of diffusion is increased ($V_2/V_o$ decreased) the engine impulse steadily improves. This is because at a given value of $\eta_{KE}$ the entropy rise across the inlet is fixed, and as $V_2/V_o$ decreases, the associated combustion loss decreases. Hence an overall improvement in performance occurs.

However in the practical diffusion process the entropy rise across the inlet is a function of the amount of diffusion performed, and thus $\eta_{KE}$ will decrease as the diffusion increases. In early performance studies, before test results became available, various empirical relationships were postulated relating the inlet kinetic efficiency to the amount of diffusion. Thus Dugger in reference 3 utilized

$$\eta_{KE} = 0.94 + 0.06 \frac{M_2}{M_o}$$

In this expression, as $M_2$ approaches low subsonic values, $\eta_{KE}$ tended to the value 0.94, an acceptable value for subsonic combustion ramjets. However, with zero diffusion, $M_2$ was equal to $M_o$, and $\eta_{KE}$ became equal to unity. Another analysis (4) introduced the impirical relationship

$$\eta_{KE} = K_D + (1 - K_D) \left(\frac{V_2}{V_o}\right)^2$$

where $K_D$ was constant at a given flight condition. It was subsequently realized that $K_D$ possessed thermodynamic significance and was of interest as a parameter of inlet efficiency. This parameter is becoming increasingly popular both in cycle analyses (5) and in intake studies (6,7). It can readily be shown (8) that $K_D$ can be expressed as:-

$$K_D = \frac{h_2 - h_o'}{h_2 - h_o}$$

compared to

$$\eta_{KE} = \frac{h_t - h_o'}{h_t - h_o}$$

Fig. 6 illustrates the difference between these definitions. The important point about $K_D$ is, however, that at constant $K_D$, the entropy rise across the intake increases as the amount of diffusion is increased. Fortunately the assumption of constant $K_D$ does describe quite well the operation of a given class of inlets at a specified flight condition. Thus in Fig. 7 the variation of $\eta_{KE}$ with $V_2/V_o$ for a simple two shock inlet is seen to follow approximately a constant $K_D$ line. Improved $K_D$ values are to be expected from more sophisticated inlet designs. It is thus more appropriate to present the performance data shown in Fig. 4 in terms of $K_D$ as shown in Fig. 5. It will be noted that for each value of

$K_D$ an optimum amount of diffusion exists: also as $K_D$ increases, the optimum amount of diffusion increases.

The above performance illustration has been given for constant-area combustion: in an interesting approximate analysis of constant pressure combustion, Builder (5) shows that for this type of engine the optimum amount of diffusion is given by the expression

$$\frac{h_2}{h_o} = \sqrt{\frac{K_D}{1 - K_D} \cdot \frac{Q}{h_o}}$$

where Q = heat energy input to the cycle (per unit weight flow through the cycle)

and it is assumed that the nozzle efficiency is unity.

Thus for the constant pressure cycle the optimum amount of diffusion is approximately a simple function of $K_D$ and flight speed. The corresponding variation of velocity ratio as a function of flight velocity for various values of $h_2/h_o$ is shown in Fig. 8.

In the practical design of inlets to achieve high $K_D$ values, the factors which minimize losses through the intake must be considered.

1. Shock losses - strong shocks may be avoided by the use of a slender spike inlet (with a small apex angle), followed by an isentropic turning region to attain high

pressure ratios.

2. Boundary layers - at hypersonic speeds, viscous losses tend to be comparable with the shock losses, and the boundary layers become comparatively thick. The height of the narrow annular gap obtained with an axisymmetric design would be comparable with the bounda: layer thickness, and more complicated designs must be used. These must have a small wetted surface area to minimise drag, and reduce the heat loading. On high trajectories (i.e. low q) the intake boundary layer is laminar and the adverse pressure gradients must not be so large as to cause separation. The interaction o: cowl shocks wi+h the boundary layer can also cause separation, and the possibility of "bleeding" the bound: layer by lateral flow must be considered.

3. Geometry - an external intake geometry is dictated by the requirement for radiation cooling. Fortunately such intakes can be started more readily than internal compression designs, nevertheless starting can present some problems. An important factor is the leading edge design, which should be sharp to minimise drag, and must also be swept back if possible to reduce drag and heat transfer effects.

So far only the effect of inlet operation on overall performance has been discussed. In general, however, the cycle performance of scramjets at a given flight condition depends on many variables e.g.:-

Mode of heat addition (e.g. constant p, A, M, $\rho$)

Fuel type, injection conditions, equivalence ratio

Combustion efficiency

Nozzle efficiency including non-equilibrium effects

Overall engine area ratio ($A_4/A_o$)

If the intake is designed to give the optimum velocity ratio, the static temperature at the entrance to the combustion chamber is approximately 5 times the ambient temperature (for $M_o > 10$). This results in temperatures in excess of 2000°R at all altitudes, and the possibility of spontaneous ignition of hot hydrogen when injected into such hot air is immediately apparent. The time required to complete the hydrogen/air reaction has been extensively studied (e.g. Refs. 7 and 24) and in common with other chemical reactions, the time is a function of pressure, temperature and fuel concentration. For this particular reaction, the effect of fuel concentration (or mixture strength) is small, and for combustion chamber pressures in excess of 12 psia, the time is always less than $10^{-4}$ seconds (Fig. 9). For the values of $V_2$ given above, the maximum distance necessary to complete combustion is only a few inches, and for most practical cases in this speed range, it is a tiny fraction of an inch.

This reaction length is based on instantaneous mixing of the fuel and air. In practice, mixing and reaction occur together, and realistically we must therefore consider the length required by the combined process. The mixing of fuel and air is broadly due to three mechanisms which act at success levels of gas motion:

1. Molecular diffusion
2. Turbulent eddies
3. Macro-mixing by vortex effects

Molecular diffusion - even for such a mobile species as hydrogen - is much too slow to give significant mixing between hydrogen and air in the time available in passing through a scramjet combustor.

Turbulent mixing of concentric jets (or two dimensional jets) has been extensively studied for this application (e.g. Refs. 9, 10 and 11), and it has been shown that a length of abo 100 jet diameters are required to obtain significant mixing. Depending on the choice of injection geometry, this mechanism may be adequate to achieve a short combustion chamber.

Macro-mixing by vortex effects is encountered, for instance, in a cross-stream injection system. In this case, th jet develops a double vortex structure, and entrains air about three times more rapidly than does a simple downstream injectio system. Cross-stream injection into a supersonic airstream introduces shock losses, and the vortices may be produced more efficiently by a swirl system. Plate 1b shows the effect of swirling a supersonic jet before injection into ambient air, (Plate 1a is without swirl), and the increased penetration is apparent. A promising technique is the injection of fuel into the vortex sheets produced for example by a highly swept delta aerofoil injector mounted at incidence in the combustion chamber. The fuel would then be injected from the leading and/or trailing edges of the injector. In such a system, the vortex serves to distribute the fuel through the air, whilst the turbulent eddy viscosity effects would complete the mixing. The powerful effect of vortex mixing is illustrated by the mixing chambers on by-pass engines in which mixing between two almost equal gas streams is accomplished in a length almost equal to the duct diameter. The problems of wall cooling and wall friction must also be considered, and these can both be alleviated by injecting hydrogen along the wall. Furthermore, the additional fuel specific impulse obtained by downstream injection becomes very significant at the higher hypersonic speeds. However, expansion of the hydrogen to very high velocities would reduce its temperature and could result in prohibitively slow combustion. It should also be noted, that a fixed geometry injector, which gives an exit static pressure equal to the combustion chamber pressure at one flight Mach number will be mismatched at other Mach numbers. Thus although very rapid mixing can be obtained, the optimisation of an injector configuration is a careful compromise between mixing length and impulse loss effects.

Turning now to the exit nozzle; its purpose is to expand the gas towards the ambient pressure as efficiently as possible. Reference to Figure 2 shows that there is no 'nozzle in the conventional sense, as the diverging combustion chamber blends directly into the 'nozzle' to form an integral unit. The absence of a throat means that the uniform velocity profile usually produced by the throat is not generally present. Thus the non-uniform profile likely to be produced by the combus system will persist through most of the nozzle flow field. At the higher altitudes the flow through the nozzle will be frozen. The impulse loss due to frozen flow depends on the amount of dissociation energy frozen into the exhaust gas. Thus in addition to the altitude (pressure) effect, the static temperatu at the end of combustion also effects the non-equilibrium loss. The static temperature is very sensitive to the intake velocity ratio, so it is to be expected that the non-equilibrium loss wi increase with decreasing intake velocity ratio.

Furthermore, there is a loss in impulse due to nozzle wall friction and divergence of the exit stream, which will depend on the nozzle geometry. For example a short conical nozzle will give smaller friction loss, but large divergence lo whilst the converse holds for a long conical nozzle. The compromise between these two losses has been treated in Refs. 2

14 for a Mach 15 engine. Both references show that the minimum loss occurs for a nozzle wall angle between 10° and 15°. The losses also depend strongly on the nozzle area ratio, and as this is usually almost equal to the intake area ratio, it follows that the losses increase with decreasing intake velocity ratio.

For the example being considered here, the trend in overall loss attributable to these three effects is plotted against $V_2/V_0$ in Fig. 10. Two important conclusions can be drawn from this figure.

1. Since the losses increase with decreasing velocity ratio, the optimum $V_2/V_0$ will be increased above the value previously indicated.
2. The nozzle losses are generally large and must be carefully minimised. For example, the friction loss can be reduced by choosing a nozzle geometry with a small wetted surface area. For certain missions however, divergence of the exhaust downwards (thrust deflection) gives a lift component, which can increase the vehicle range (Ref. 13).

A further requirement is that most of the nozzle surface should be able to radiate heat away, and therefore must not be enclosed. Such radiation cooled, semi-open nozzles can be designed readily once the external flow field is known. However, the interaction between the internal and external flow fields leads to considerable difficulty in testing, since the engine and airframe cannot be evaluated independently as with conventional test techniques.

Despite the problems in defining the exact component performance, which can be achieved, reasonable estimates of component efficiency give a band of overall engine capability which is encouraging (Ref. 15). Furthermore, judicious integration of the components may avoid some of the problems discussed above. For example:

1. Fuel injection along the walls, near the diffuser exit, will prevent excessive heat transfer, and may alleviate shock boundary layer interaction problems.
2. Careful design of the fuel injector system may avoid undesirable flow profiles.
3. The exact geometry of shock generating components, such as the intake spike and cowl lip, can be arranged to give the minimum boundary layer disturbance. In particular, lateral motion of the boundary layer can provide an effective boundary layer bleed system.

A major problem of the system is cooling. Although large sections of the intake and nozzle can be radiation cooled, shielded sections and the combustion chamber will require regenerative cooling. Certain regions, such as leadin edges, which should be sharp to minimise drag, will require special ablative or transpiration cooling. However, it is encouraging to note that Ferri (Ref. 16) has shown that up to Mach 22, stoichiometric hydrogen fuel flow is sufficient to cool the vehicle. Furthermore, little additional fuel is required to cool at higher speeds, and significant increases in air specific impulse can be obtained by the mass addition effect of this extra fuel.

Considering now the overall performance of the engine, the features which determine the design at a given Mach number can be illustrated as shown in Fig. 11.

1. The hatched lines indicate the limits imposed by the static temperature for spontaneous fuel ignition.

2. The maximum area ratio which gives reasonable combustion chamber size (compared to the boundary layer), is indicated by the dashed lines.

3. The intake pressure ratio - which determines the structural loading is also plotted.

Fortunately the optimum intake velocity ratio required for the maximum fuel specific impulse gives convenient values of diffuser exit pressure, area and temperature. A similar result is obtained throughout the speed range from Mach 8 to Mach 20, and furthermore, the optimum angle for a two shock inlet only varies slightly over this speed range. Thus, as illustrated in Ref. 2 and Fig. 1, the off-design performance is so close to the ideal over this speed range, that a fixed geometry engine gives less than 10% loss in specific impulse.

The prospect for obtaining good performance above orbital velocities, depends on reducing viscous friction losses, to below the levels currently estimated.

Although the impulse values depicted in this article are comparatively high it must always be remembered that for boos missions it is most important that a very high thrust:drag ratio is maintained. Thus for a simple horizontal acceleration the velocity increment is related to the mass decrement as follows:-

$$dV = I_f \left(1 - \frac{D}{T}\right). \frac{dm}{m}$$

and thus the available impulse of the air breathing engine is effectively reduced by the factor $\left(1 - \frac{D}{T}\right)$.

Now as the thrust coefficient of the air breathing engine decreases as the flight speed increases, high thrusts can only be obtained by flying either at high q trajectories or by using engines of large frontal area: in either case a weight peanlty may be incurred with a consequent degradation in system performance. Thus the comparative performance of air-breathing engines for boost missions must only be based on an over all analysis of system performance, and never on comparative impulse values.

More precise studies of performance require definition of the engine/vehicle configuration so that thrust vector, boundary layer, heat transfer, and frictional effects can be evaluated. It will therefore be appreciated that for extensive cycle performance studies a digital computor is required, and that for refined performance estimates an actual project configuration must be established. It follows that in an article of this size the performance levels quoted must be regarded as approximate values only, to be treated with many reservations.

The foregoing discussion shows that there are many detail problems to be solved before the hypersonic aircraft is fully developed. Most of the solutions will require experimental verification and we must consider how such data is to be obtained. In general, certain non-dimensional parameters such as Mach number and Reynolds number define the phenomena being studied. It is often more convenient to carry out experiments at appropriate values of these parameters, than to duplicate the full scale operating environment.

In the case of an integrated, scramjet powered, hypersonic vehicle, this approach poses many problems. This can be illustrated by considering some of the problem areas and noting some of the parameters to be simulated.

| Problem Area | Parameters |
|---|---|
| Inviscid hypersonic flow field | $M$ |
| 3 Dimensional boundary layers | $R_e$, $V$, heat flow |
| Mixing | $V_f/V_2$, $\rho_f V_f/\rho_2 V_2$ |
| Combustion kinetics | $h_2$, $V_2$, $p_2$, $t_2$ |
| Nozzle flow | $h_3$, $V_3$, $V_{external}$, $p_3$ |

Although the required Mach numbers, and to a lesser exten Reynolds numbers, can be obtained in current hypersonic test facilities, the enthalpy and pressure levels which are necessary to study the reacting flow regions are very difficult to attain. The stagnation pressures and temperatures at a flight Mach number of 20 at 170,000 ft. are 2,000,000 psi and 13,000°K respectively. These conditions are beyond the reach of any existing ground test facility, and careful consideration must be given to the means for obtaining valid data. Assuming scale models can be used, continuous test facilities using pebble or arc heaters are limited to about Mach 8 at representative flight-altitudes. Pulse type facilities such as shock tunnels can achieve about Mach 17 but the testing time is limited to a few milliseconds (see also Ref. 6). A shock tunnel which is in use at Sheffield University for the study of supersonic combustion is illustrated in Plate 2, and a concentric injector configuration for supersonic combustion research is shown in Plate 3. The limitations of these test techniques suggest that it would be foolhardy to proceed to a full scale aircraft on the basis of such results alone. The range of variables which can be investigated is so wide that it is easy to be distracted at the research phase and accumulate data which is outside the realm of engineering application. For example the chemical kinetics need only be studied for the particular fuel of interest at the pressures and temperatures to be encountered. Thus integration of the research and development aspects is particularly important in this field, if significant progress is to be made for reasonable investment. For this reason it is essential that both research and development work, and project studies, proceed in close harmony. Thus a research programme of this nature must be oriented towards a given vehicle system. In view of the indeterminate nature of future cruise and boost missions, a most suitable project would be the development of a hypersonic flight test vehicle to explore the hypersonic flight range.

Such a vehicle, apart from providing data essential for the design of flying propulsive bodies, would have a wide technological 'fall-out' in a variety of related activities, such as the tracking of high-speed vehicles, radar/radio phenomena, and re-entry problems. Two approaches can be contemplated, the first is typified in recent reports (Ref. 17) from the U.S.A., that a Mach 12 hypersonic research aircraft is being considered as a follow on to the X-15. It should also be noted that already a stretched version of the X-15, the X-15-A2, will subsequently be flying as a hypersonic ramjet test vehicle. A hypersonic research aeroplane and its support facilities are probably beyond European funding or objectives - nevertheless an alternative approach, that of using small free flight test vehicles is well within U.K. experience and funding. Such a vehicle, possibly of approximately 8"-12" diameter, could be boosted to speeds in the Mach 7-10 region, and subsequently accelerate over a limited speed range to provide relevant test data. The speed range would be limited as the quantity of low density hydrogen fuel carried would be severely restricted. The size and weight of the vehicles would have to be kept to a minimum as even small vehicles would require quite substantial first stage rocket boosters - for this reason it is well worth exploring alternative launch techniques. Certainly the cheapest alternative is the gun launching technique and it is interesting to note that Ref. 18 quotes launch and range costs at approximately £1500 per shot. It is considered that an experimental and analytical programme constructed around a scramjet flight test vehicle could act as a focus for European studies in the scramjet area.

It is noteworthy that scramjet research is already proceeding apace at DVL in Germany (Ref. 19), and ONERA in France (Ref. 20), and indeed Italy (Ref. 21), Spain (Ref. 22), and Russia (Ref. 23) have contributed to this field. It would be a great pity, if the original ideas of British Aerodynamicists, for a new class of aircraft, were to lack the support of an advanced propulsion technology - especially since such a technology could be founded on a relatively inexpensive experimental programme.

CONCLUSIONS

1. The scramjet has high performance potential over a remarkably wide speed range, extending possibly to about orbital velocity.

2. The scramjet may be readily integrated with the waverider hypersonic vehicle which has been concieved largely in the U.K.

3. The scramjet/waverider can also be integrated with a turbojet or rocket to yield a vehicle having acceptable low speed performance, and thus becomes a near ultimate aircraft.

4. Hydrogen fuel is the obvious choice in the upper hypersonic speed range.

5. One of the main factors influencing engine optimisation is inlet velocity ratio. However, even a fixed geometry engine can be expected to give good performance from Mach 8 to 18.

6. Drag forces can become more significant with increasing flight speed, and care must be taken to minimise the vehicle surface area. Laminar, transition and turbulent boundary layers are all expected to cover significant portions of the

surface.

7. Combustion kinetics are unlikely to present a problem on most flight trajectories.

8. Optimisation of fuel/air mixing techniques may depend on further studies of vortex mixing. Nevertheless very short combustion chambers are possible.

9. The nozzle is generally integral with the combustion chamber, and integral with the body. Testing of engine and airframe cannot be readily separated.

10. It appears that overall cooling of the vehicle can be accomplished with the engine fuel flow.

11. Ground testing alone will not give adequate information for the confident design of a hypersonic aircraft.

12. A scramjet flight test programme, based on an 8"-12" diameter vehicle, would be an inexpensive method of advancing British technology in this field.


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Fig. 1. Comparison of Scramjet Efficiency with the Efficiency of Conventional Power Plants.

Key. A Variable Geometry Scramjet Hydrogen Fuel Inlet Process Efficiency $K_D = 0.9$

B. Fixed Geometry Scramjet $M_{DESIGN} = 15$ (Ref. 2)

C. Conventional Ramjet.

D. Turbojet with Reheat

Fig. 2. Schematic Arrangement of a Scramjet.

Fig. 3. Comparison of Scramjet Performance ($I_f$) with that of other power plants and ideal impulse.

Key. A. Maximum possible specific impulse - hydrogen fuel.

B. Hydrogen fuel Scramjet specific impulse band.

C. Turbojet with reheat - kerosene fuel.

D. Ramjet - kerosene fuel.

E. Scramjet - kerosene fuel (Ref. 18).

F. Rocket

Fig. 4. Variation of Engine Performance ($I_f$) with intake Diffusion ($V_2/V_o$).

$V_o = 16000$ ft./sec.

Constant area combustion exit area = inlet area.
Stoichiometric hydrogen air
Normal fuel injection
Equilibrium nozzle flow

Fig. 5. Variation of Engine Performance ($I_f$) with intake Diffusion for constant values of $K_D$.

$V_o = 16000$ ft./sec.

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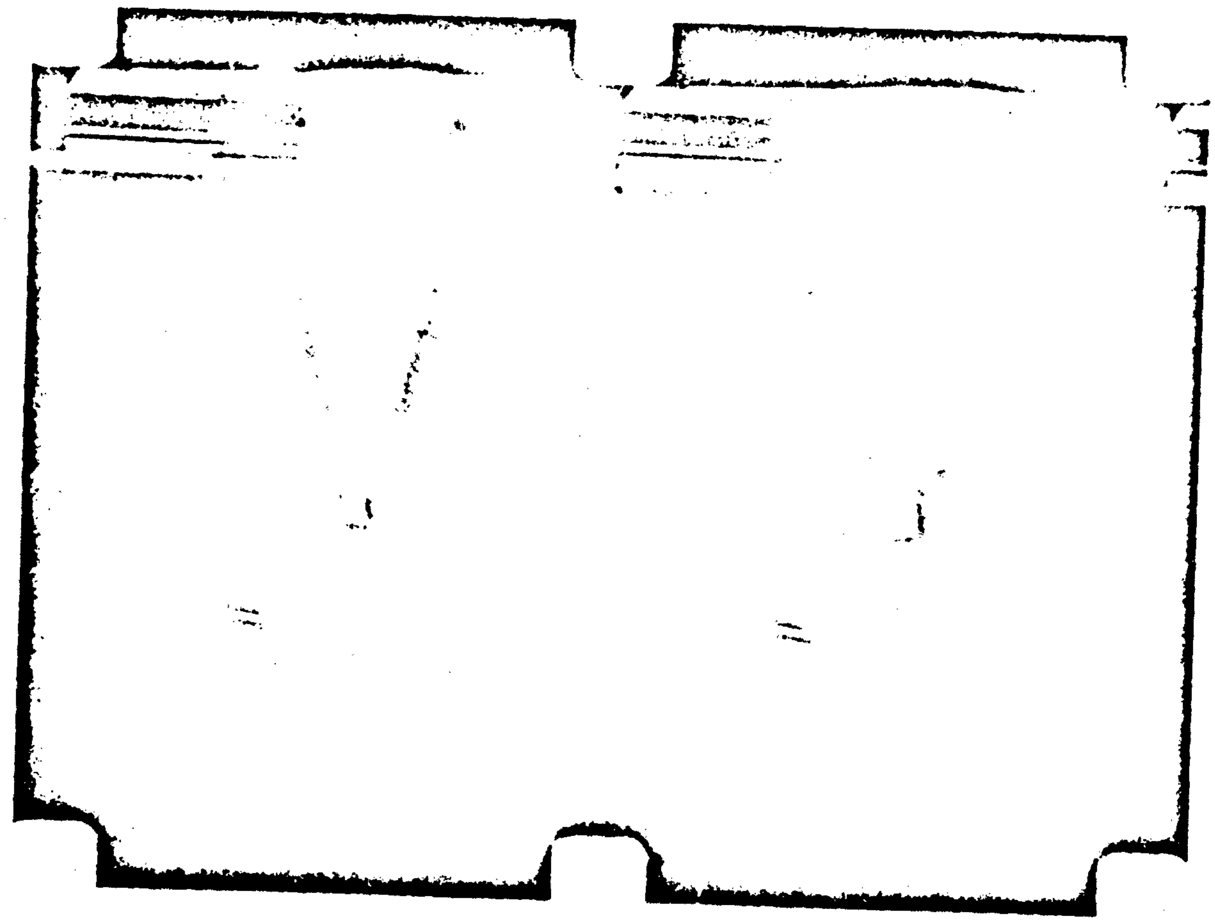

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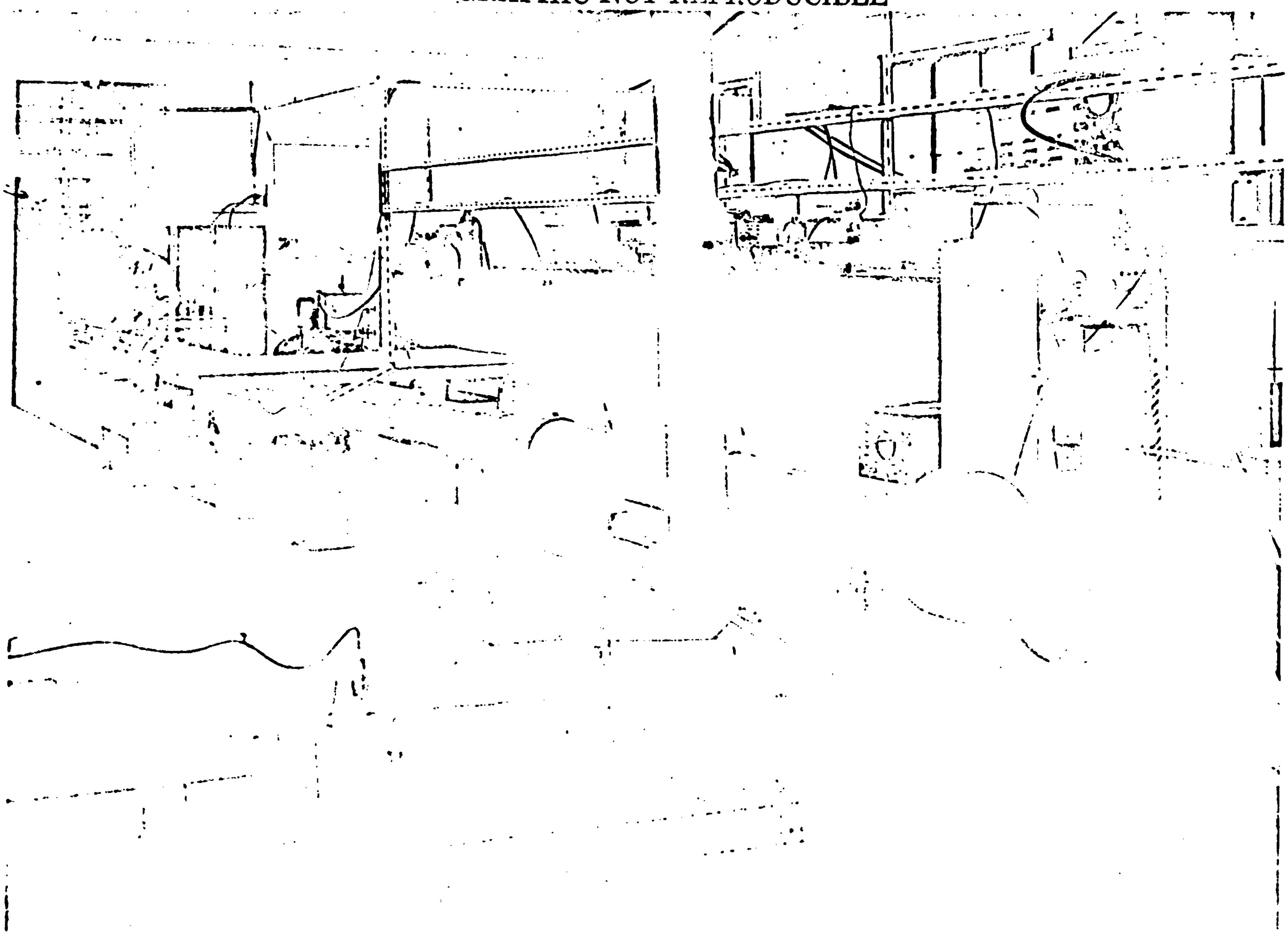

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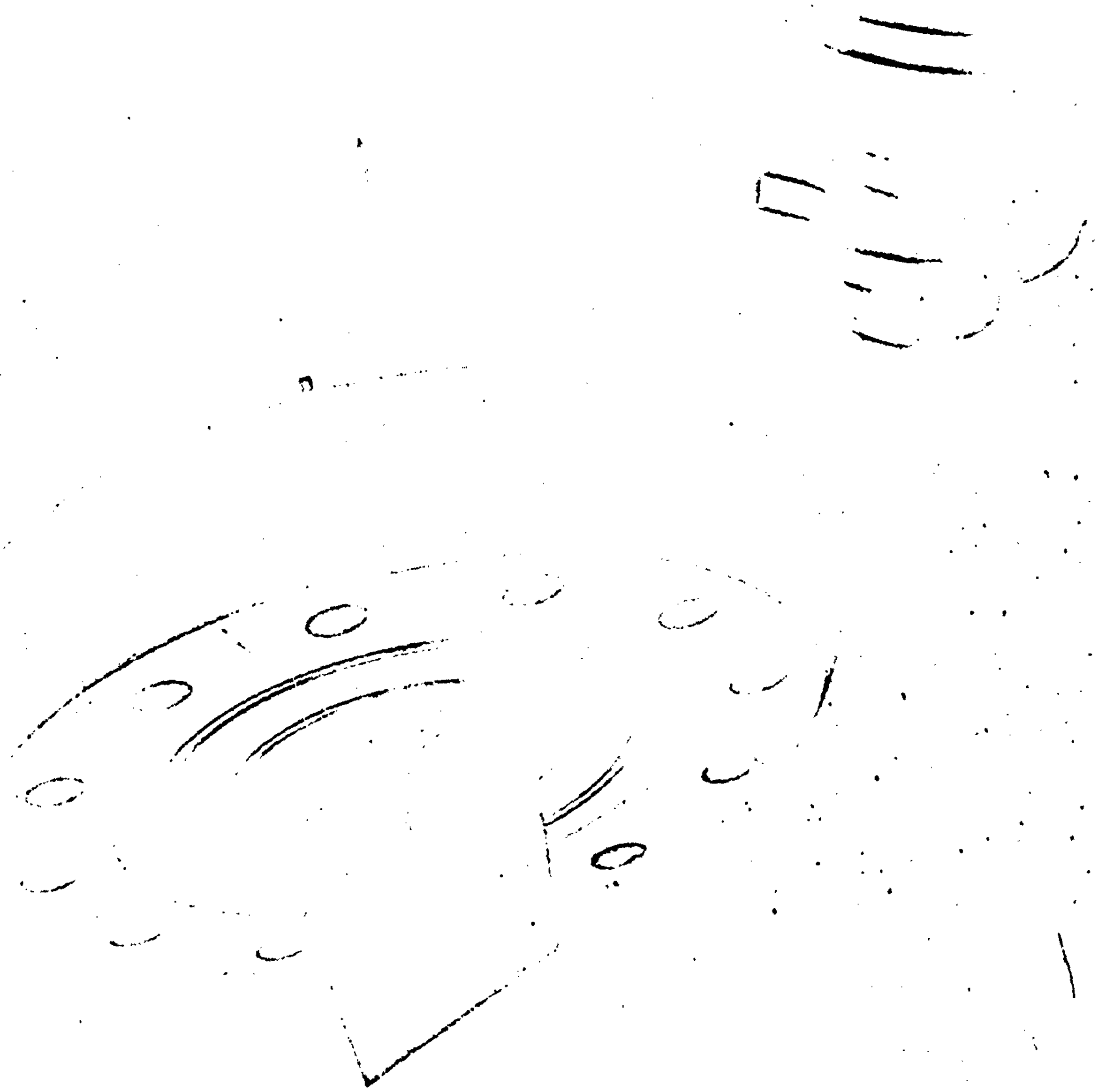

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4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

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13. ABSTRACT

The high overall efficiency of the scramjet engine is potentially maintained over the speed range from Mach 5 to 20. At higher speeds, Hydrogen fuel is essential and a fixed geometry appears feasible. Optimization of the intake diffusion can be carried out most easily in terms of the intake process efficiency and velocity ratio. The optimum amount of diffusion is reduced when realistic nozzle losses are included. The development of a hypersonic flight test vehicle is recommended.

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| 14. KEY WORDS | LINK A | | LINK B | | LINK C | |
|---|---|---|---|---|---|---|
| | ROLE | WT | ROLE | WT | ROLE | WT |
| Supersonic Combustion Ramjet<br>Scramjet; Air-breathing Combustion<br>Supersonic Combustion<br>Hypersonic Flight<br>Diffusional-type Supersonic Combustion | | | | | | |

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