# Review of SCRAMJET Propulsion Technology

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## Nomenclature

$a$ = acceleration
$A_a$ = capture area
$A_b$ = thickness of body
$D$ = drag
$I_e$ = effective specific impulse
$L$ = lift
$o$ = ratio of fuel burned to stoichiometric value
$T$ = thrust
$V$ = velocity
$W$ = weight
$\dot{W}$ = fuel weight burned per second
$W_f$ = fuel weight

## Introduction

THE possibility of using supersonic combustion air-breathing engines as propulsion systems for hypersonic vehicles has been investigated during the last few years. Such an approach was proposed several years ago[1,2] and was considered skeptically at the beginning by many. Presently it is one of the promising engine schemes for recoverable space vehicles, for small-weight and small-volume nonrecoverable launchers, and for hypersonic cruising airplanes. Many of the initial results of the investigations are largely due to a few groups of researchers supported by the Aeronautical Propulsion Laboratory of Wright-Patterson Air Force Base, in the field of applied research and development, and by the Air Force Office of Scientific Research in the field of basic research in combustion. The results have generated a wide interest in the fields. As a consequence, many new groups recently have joined the development effort. This rapid increase in scope and effort of the program tends to generate some confusion on the best technical approach, on the objectives considered to be important, and on the goals of such effort, to justify a few lines of discussion attempting to define some of the important and less recognized characteristics of a practical SCRAMJET.

In discussing several characteristics of a SCRAMJET engine for a different field of application or propulsions systems, the parameter considered as predominant is the specific impulse when the variation of specific impulse as a function of flight Mach number is investigated for different cycles. A preliminary conclusion could be reached that the field of application for a turbojet engine is between 0 and 5–6, and subsonic burning ramjet, between 1.3–2 and 6–8, whereas the SCRAMJET is considered to be useful mainly above $M = 5$–$6$. The rocket engine is competitive for many applications in all Mach number ranges because of the small structural weight and cost and the ability to give thrust at zero velocity outside the atmosphere and possibly at very high Mach numbers.

Any conclusion based on a single parameter is too superficial to be correct; a specific correct conclusion can be reached only when a given application is considered and a detailed study is performed. However, some of the important parameters involved in this type of analysis can be determined on the basis of a more general discussion. The two main uses of any engine are to produce thrust for acceleration to the required speed, and also, in many cases, to produce thrust for cruise. When a rocket engine is considered, usually a large part of the acceleration occurs outside the atmosphere. Therefore, the specific impulse of the engine can be used directly for the determination of the acceleration. Then the only requirements are that the thrust be higher than the weight at takeoff and that, at the end of the mission, the acceleration be below a given maximum. The limitation of maximum acceleration can impose a severe condition to the rocket propulsion engines when the value of the maximum acceptable acceleration is small (1.5–2 $g$), because of the large variation of vehicle weight during flight; in this case, the use of multiple rocket engines or the use of throttlable engines is required. The rocket engine appears very attractive for nonrecoverable launchers when there are no limitations on mass, inert weight, and volume, and presently it is used extensively for such application. The structural weight is percentually small, even if the actual value of the structural weight and total weight is high, and the design is simple. The cost of a launching system is a function of many parameters. The important parameters are structual weight, total weight, and total volume, which are related to opera-

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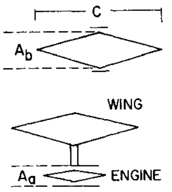


**a) Wing-engine. Equivalence ratio $\phi_a$, fuel flow $W_{fa}$.**

**b) Wing plus engine. Equivalence ratio $\phi_b$, fuel flow $W_{fb}$; $C_{Do} = 0.08$; $A_b/C = 0.10$; $C_t = 0.28$ at $\phi = 1.0$.**

**Fig. 1 Comparison of wing-engine with wing plus engine.**

tional cost and development cost, etc. Therefore, a factor against the rocket is that in any given application, the volume and total weight of a system using rocket engines is larger than for a system using air-breathing engines. The air-breathing propulsion system permits reducing volume, weight, and, possibly, structural weight of a system; however, it requires flight in the atmosphere and, therefore, introduces all the complications due to the aerodynamic forces and to aerodynamic heating. The most impressive advantages of such a propulsion system for a nonrecoverable launching application are related to a reduction of volume and total weight, which are important in some special applications.

In a recoverable system the requirements of aerodynamic flight are present in any case, and substantial additional structural weight related to the flight in the atmosphere decreases drastically the already small mass fraction that represents the payload of a rocket launching system. For recoverable launching systems the more efficient airbreathing engine appears to be very attractive, even though only used for acceleration. For a hypersonic airplane the engine is also used for acceleration and must also be efficient during cruise. The acceleration to high speed must be obtained in a short time in either case.

For hypersonic application the engine must be capable of performing in a large range of Mach numbers, must be simple and compact, and must be able to operate efficiently in a large range of thrust levels. The possibility of having different thrust level for a given flight condition requires that the engine operate efficiently at different values of equivalence ratio (ratio between fuel consumption and fuel consumption for stoichiometric mixture in the burner). This flexibility is a necessary characteristic if the engine must be of practical use; therefore, the design must be capable of guaranteeing such flexibility.

In an accelerating air-breathing engine that flies horizontally in the atmosphere, an equivalent specific impulse can be defined (i.e., one that is equivalent to the rocket specific impulse) which must be used in the acceleration equation

$$dV = gI_e d\ln W \tag{1}$$

Such an equivalent specific impulse is defined by the expression

$$I_e = T - D/\dot{W} \tag{2}$$

where $T$ is the thrust of the engines, $D$ the drag of the vehicle, and $\dot{W}$ the weight of fuel burned per second in the engine. The acceleration is given by

$$a = T - D/W \tag{3}$$

where $W$ is the weight of the aircraft. This implies that, in order to have good fuel-utilization during acceleration, the thrust of the engine must be as high as possible, the drag as low as possible, and that $T - D$ must be very high. Now $T - D$ depends on the design and size of the engine, on the fuel air ratio in the engine, and on the altitude of flight. An engine good for acceleration must be designed to be efficient at $\Phi$ slightly larger than 1, whereas for cruise the engine must be efficient at $\Phi$ lower than 1. The engine must be designed with negative external drag, zero or small spillage in a wide range of Mach numbers, and favorable interference with the vehicle. A large part of the airplane volume must be utilized as inlet and nozzle of the engine. Then the variations of impulse of the flow entering the engine already include the skin-friction drag and the pressure drag of a large part of the airplane, and therefore, the drag of the vehicle is reduced by the presence of the engine. In addition, the vehicle must be able to fly at the lowest possible altitude because $T - D$ is proportional to the air density. The problem of flying at low altitude is essentially a structural design problem; the aerodynamic forces increase with the density and, therefore, the stresses increase. The problem of aerodynamic heating of the structure is not strongly affected by the increase in density if the system is regeneratively cooled; however, it is affected if radiation cooling is used. From a structural point of view, the use of supersonic combustion burning simplifies the problem because the maximum pressure rise is small and, therefore, the stresses are reduced. Classical turbojets using subsonic combustion and subsonic combustion ramjets reach a static pressure in the burner of the order of 6–10 times the pressure required for supersonic combustion for similar requirements. This is a very important characteristic of the SCRAMJET engine, which tends to push the range of the SCRAMJET to Mach numbers as low as possible.

Airplane engine integration is a very important requirement in order to obtain good system performance for both acceleration and cruise. For cruise the fuel consumption is directly related to the drag of the airplane and the engine combination. Large reduction in volume drag can be obtained by a satisfactory integration when a low air specific impulse engine is used (pounds of thrust per pound of air per second, entering the engine). This consideration tends to favor the ramjet with respect to the turbojet, even when the ramjet has lower specific fuel impulse as isolated engine. In order to emphasize the importance of engine airplane interference, let's consider the following example, which is of an academic nature, but which serves to illustrate the importance of this point. Let's assume that the vehicle is represented schematically by a two-dimensional wing having 10% maximum thickness. Two limiting cases can be considered. In one case the engine is integrated completely in the airplane, (Fig. 1a), whereas in the second case the engine is a separate power plant as shown in Fig. 1b. The capture area of the two engines is not the same, the capture area of the engine in Fig. 1a is considered as a parameter $A_a/A_b$. Each value of $A_a/A_b$ corresponds to a different value of fuel-to-air ratio in the engine. In both cases the thrust is equal to the drag of the system; then an equivalent specific impulse $I_e$ of the engine can be defined in each case, given by $I_e = D/\dot{W}$. In Fig. 2, the ratio $W_{fb}/W_{fa}$ is given as a function of the ratio $A_a/A_b$. The quantity $W_{fa}$ is the fuel required by the system shown in Fig. 1a, and $W_{fb}$ is the fuel required by the system shown in Fig. 1b. In the case of Fig. 1a, the only external drag is the skin-friction drag outside the engine. The value $\Phi$ for the engine of Fig. 1a is indicated on the curve. In some military applications analyzed it has been found that an average value of $\Phi = 0.5$ for cruise is practical. Then the interference effect can duplicate the value of the equivalent specific impulse of the engine. This possibility

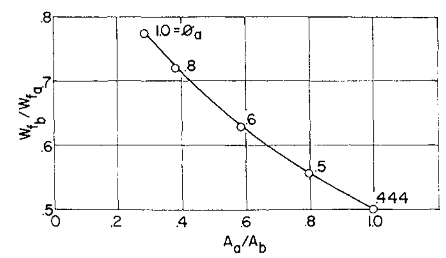


**Fig. 2 Fuel consumption of wing-engine with respect to wing plus engine, $M_\infty = 10$.**

does not exist for a rocket engine, and is very much reduced for a turbojet engine. However, it requires that the SCRAMJET design be tailored to the airplane. If life is required, a lifting engine must be designed where the top and bottom are not equivalent. In this case flat plate $L/D$ with only small skin-friction drag can be obtained theoretically.

The importance of good aerodynamic matching between engine and airplane is indicated by the preceding example. The achievement of such matching depends strongly on the type of air-breathing engine selected, on the design criteria selected for a given type of engine, and on ability to develop a simple engine design that can be tailored to many different vehicle configurations. Engines that require circular cross-section types of ducts are less suitable, from the point of view of good interference, because they require inlets that are difficult to tailor to the airframe configurations. The same is true for engines using large variations of geometry, because the introduction of variable geometry makes it difficult to have inlets with entrance cross sections having one dimension substantially different from the other and therefore very wide.

## Cycles and Performances

The subsonic combustion ramjet engine usually has a constant-area burner. For subsonic combustion the constant-area burner does not change strongly the static pressure, and produces favorable gradients in the boundary layer; therefore, the use of constant-area burning does not create any aerodynamic problem and is practical. The situation is different for supersonic combustion, because the supersonic combustion can produce large pressure rises and, in some cases, can choke the channel. The combustion mode in supersonic combustion must be selected on the basis of cycle performances, combustion stability, boundary-layer separation, and ability to obtain a satisfactory design, capable of operating in a wide range of flight conditions. In addition, the combustion mode is related to interaction between different components of the engine and, therefore, affects the development procedure of a successful engine. Large differences can be found in cycle performance between different modes of combustion.

In the limiting case in which the combustion occurs in a constant-area streamtube, the combustion produces large static pressure rise. It must be noted here that the one-dimensional concept of a constant-area streamtube is not equivalent to the concept of constant-area combustor. The air and fuel are not premixed when the air enters the combustor, but the fuel is injected locally in the combustor. The combustion is controlled by mixing; therefore, the combustion starts near the injection region and transmits waves in the air outside of the combustion region, as shown schematically in Fig. 3. As a result, the flow in the combustor is highly nonuniform, and peak pressures are reached locally in some part of the combustor. Such peaks can be several times higher than the average pressure at any given cross section of the combustor.

In a supersonic combustion engine design, three main problems exist related to 1) boundary-laryer separation, 2) aerodynamic heating, and 3) stability of combustion. The boundary layer in the inlet tends to separate from the walls and tends to produce reverse flow because of the very large

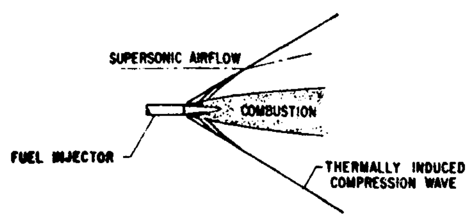


**Fig. 3 Sketch of supersonic combustion flowfield.**

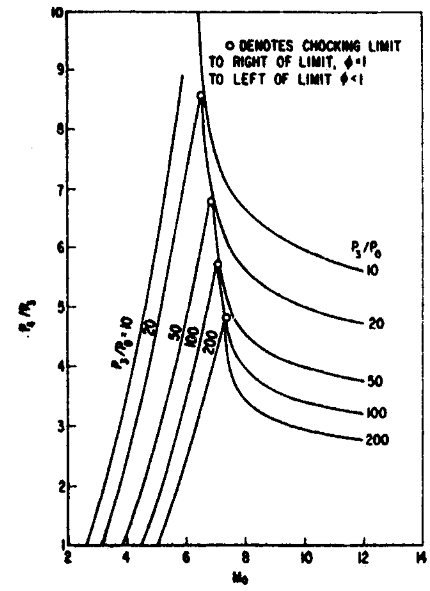


**Fig. 4 Pressure ratio across burner with constant-area burning.**

pressure rise produced in the inlet and in the combustor. The ability to avoid separation depends strongly on the local pressure gradients. Boundary-layer scoops can reduce the problem but cannot eliminate it. Several experimental investigations have been performed on separation of boundary layers in the presence of adverse pressure gradients. Some of the results are summarized in Refs. 3–6.

The maximum pressure rise obtainable in the absence of separation depends strongly on the wall temperature in the region of separation. Larger pressure rises could be obtained either by using smaller initial pressure gradients or by using porous walls. In the first case the inlet becomes extremely long, very heavy, and difficult to cool. Similarly, the use of continuous porous surfaces could solve theoretically the boundary-layer separation problem; however, it would introduce very serious heating and drag problems. For an engine capable of wide Mach number operation, the porous surface must be extended to a large region of the engine, and must be introduced in a zone where the static pressure is much higher than freestream pressure (region of combustor). Therefore, the amount of air discharged in order to avoid separation would constitute a large percentage of the capture mass flow.

It must be noted that at high static pressure, temperature, and velocity, the combustion occurs very rapidly; in the burner the pressure rise is controlled by supersonic combustion and is localized. Such pressure rise can produce separation unless the pressure rise is eliminated by expansion produced at the wall.

Figure 4 gives the average pressure rise across a constant-area burner calculated on the basis of one-dimensional flow, as a function of freestream Mach number for different values of pressure rise in the inlet. The discontinuites in the curves denote choking at the end of the burner for $\Phi = 1$. The curve for $M_o$ below choking corresponds to variable value of $\Phi$ and choking at the end of the burner.

Pressure rise produced in a short length, therefore, can separate the boundary. For this reason it appears difficult to avoid separation in a burner where combustion produces large pressure rises, especially when operating in a wide range of Reynolds numbers and Mach numbers. The pressure gradient depends on the combustion process (for example, reaction rates), which is affected by the Reynolds number. Because of the presence of the boundary layer, it can be concluded that comparing the performance of cycles having constant pressure combustion with the performance of cycles in which the combustion occurs in a rising pressure field

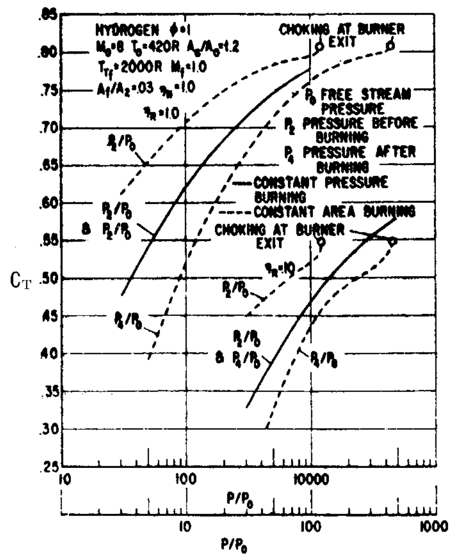


**Fig. 5 Ramjet performance comparison of constant-pressure and constant-area burning.**

is of interest only if the value of the maximum static pressure is kept constant. In both cases losses due to boundary layer and shocks can be considered to be similar. It must be noted here that the same conclusions are reached when the heat-transfer problems are considered. The most severe cooling and stress requirements occur in the burner. The problem of cooling of the burner surfaces is twofold. The total amount of heat to be removed must be minimized; in addition, the value of maximum heat-transfer coefficient must be minimized in order to keep the temperature of the wall within acceptable values, and, at the same time, to keep the wall thickness within practical limits. Both requirements tend to limit the maximum static pressure acceptable in the burner. Both effects indicate that the quantity that must be kept constant in any preliminary comparison is the value of the maximum static pressure reached at the surface of the engine, not the static pressure at the entrance of the burner (same inlet design), which is not a significant parameter in the boundary-layer or heat-transfer phenomena. If a comparison of performances is made for constant maximum pressure but not for constant static pressure at the beginning of combustion, any cycle analysis shows that the constant-pressure burner is better than the constant-area burner (Fig. 5). In Fig. 5 the thrust coefficient $C_T$ is given as a function of the maximum pressure rise for the stagnation pressure values of recovery of the inlet $\eta_r$ equal to 1 for flight Mach number of 8, and 15 for the case of constant-pressure and constant-area burning and the same maximum pressure. In Fig. 6 the results of the same analysis are plotted as a function of the pressure recovery of the inlet for $M_\infty = 10$, and for a maximum static

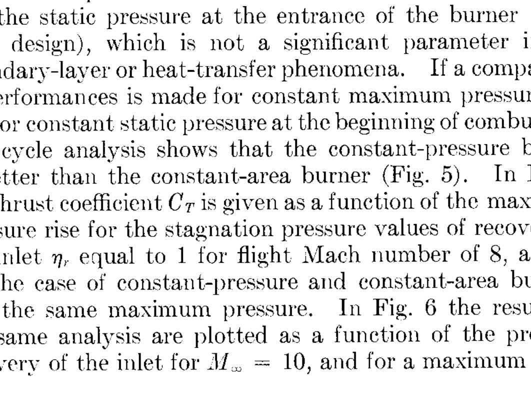


**Fig. 6 Specific impulse comparison constant-area vs constant-pressure combustion.**

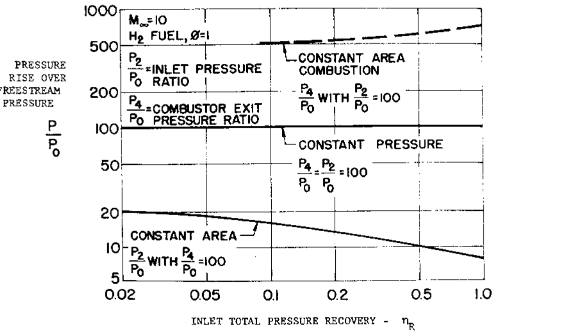


**Fig. 7 Comparison of internal pressure rise constant-area vs constant-pressure combustion.**

pressure rise of 100. In this figure the value of $C_T$ for the case of constant-area burner and same inlet design (same static pressures at the entrance of the burner) is also shown by the dotted line. The design represented by the dotted line has the same inlet design as the constant-pressure burner but much larger wall pressures at the end of the burner, as shown in Fig. 7.

The third problem, again the burning process with large pressure rise during combustion, is related to the stability of the combustion in a combustion process controlled by mixing. This problem is very serious and has not been considered until now. Consider, for example, the mixing process of air and hydrogen shown in Fig. 8. The analysis assumes chemical equilibrium. The initial temperature of the gases is 2500°R. The calculation is performed by a mixing-type analysis with chemical reaction. The pressure gradient along the axis is specified a priori. Both streams are initially supersonic but the analysis shows that the combustion process increases the static temperature and, therefore, the speed of sound. As a consequence the stream tends to become subsonic in the high-temperature region, even in absence of any pressure gradient, despite the small change in velocity due to mixing. Any small longitudinal pressure gradient produces very large effects in the subsonic region, which substantially decrease the velocity in the region of combustion.

In Fig. 8 it is assumed that pressure along the axis rises to 1.8 times the value at the beginning of mixing. Figure 9 shows two Mach number profiles at two stations; the Mach number at the axis of station $X/D = 4.75$ is of the order of 0.05. If the assumed pressure continues to rise only slightly, local reverse flow is obtained which can produce high instability in the flow. Figure 10 is a schematic of this flow configuration; Fig. 11 is a photograph of an experimental flame for these external and initial conditions.

It is also important to mention here that the large pressure rise produced during burning by the combustion at supersonic speed causes the separate inlet tests to be unidicative of the inlet performances in the actual engine. This is because the pressure rise produced by the combustion influences the boundary layer in the inlet, and can change

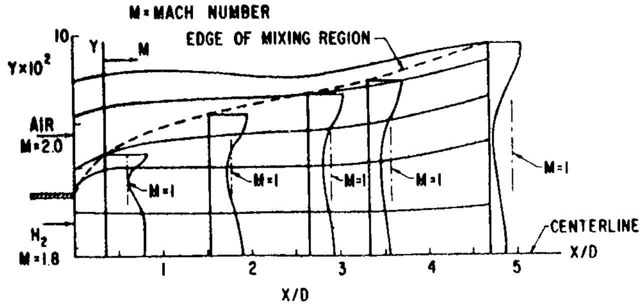


**Fig. 8 Flowfield calculation showing development of recirculation region in axisymmetric diffusion controlled combustion process with adverse pressure gradients.**

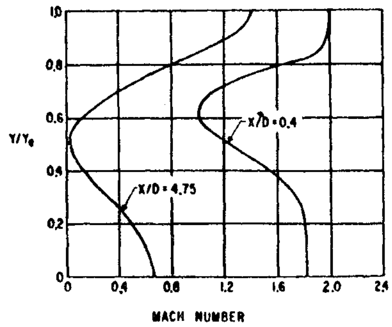


**Fig. 9 Mach number profiles near the injection point and near the stagnation point.**

completely the inlet performances. In this case, the inlet-burner combination must be tested together. The limitation is very severe for the development of SCRAMJETS at high Mach numbers. The inlet of a practical engine, for a particular constant-pressure type of burner, can be tested and developed at a reduced scale. It can be tested without simulating full stagnation enthalpy, either by using different gases or by introducing simple corrections. (This is especially important if the inlet is incorporated into the airframe.)

Because supersonic combustion permits variable burning Mach numbers, SCRAMJETS capable of operating in a wide range of Mach numbers are possible. On the basis of conceivable performances of components, SCRAMJETS more efficient than any other power plant can be predicted for Mach numbers between 5 and 25.

It must be noted that fuel specific impulse of the order of 1200 sec can be predicted for hydrogen fuel at orbital speed.[7] As shown in Fig. 12,[7] even some deterioration of component performance does not appear to have strong effects on the value of specific impulse obtainable at high Mach number. The problem developing a SCRAMJET engine for $M = 24$ is related mainly to the problems of testing and structural integrity, not to the development of extremely efficient components.

## Engine Design Requirements of Engine Components

### Inlet

The use of supersonic combustion is of great advantage for the design of an engine that can perform in a large range of flight Mach numbers without large change of geometry.[7] In Fig. 13a the average Mach number produced by an inlet at the entrance of the burner as a function of the freestream Mach number is indicated for stagnation pressure recovery of the inlet, and the capture areas schedule is shown in Fig. 13b. In practical inlet design, variations of streamtube contraction ratio as implied in Fig. 13 can be obtained.

### Combustion

The aerodynamic design of a combustor in SCRAMJET speed is related to two basic problems, mixing and combustion.

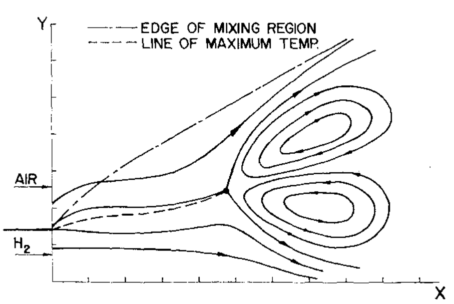


**Fig. 10 Schematic of off-axis stagnation point and recirculation region in an axisymmetric diffusion controlled combustion process with adverse pressure gradient.**

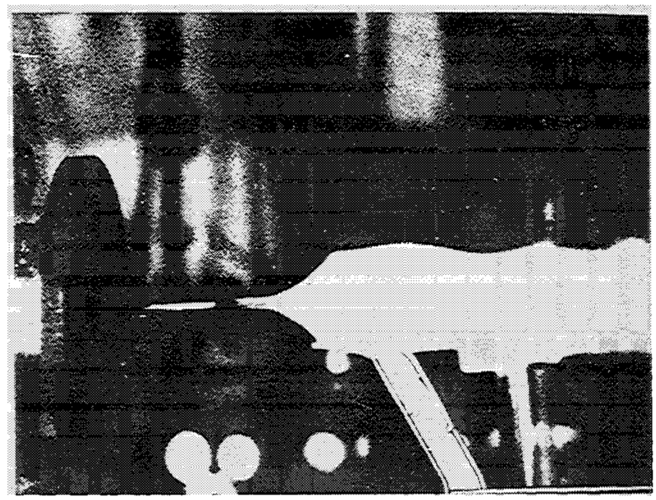

**Fig. 11 Photograph of supersonic burning with increasing pressure.**

Substantial progress has been made on the analysis of turbulent mixing of gases in the case of tangential injection. The problem of inclined injection is more difficult to analyze. Presently it is difficult to predict as functions of flight conditions with satisfactory approximation the distribution of fuel in airstream when the fuel is injected in the flow by means of inclined injection. Some comments must be made on the relative advantages of the two approaches. Inclined injection is interesting because, when the fuel is injected with a velocity component normal to the air flow, it reaches a region at some distance from the point where it enters the channel, because of its kinetic energy. Therefore, the fuel penetrates the air without needing a physical tube to bring the fuel totally into the stream. In addition, because of its inclination the fuel entering the stream produces a local compression and increases locally the pressure of the air in front of the jet and, therefore, facilitates combustion. However, such advantages are coupled with several disadvantages that make this approach rather unattractive for a practical engine.

A schematic description of the flowfield is shown in Fig. 14. The fuel is injected from a wall where boundary layer exists. In the case of injection of an inert gas, the shock produced by the injection separates locally the boundary-layer flow, and the compression starts somewhat upstream of the injection. The jet is deflected and the gas mixes with the air. The static pressure in the base region (region 2) of Fig. 14 of the flow is below freestream, whereas in region 1 the static pressure is above freestream. The pressure difference tends to bend the jet, which, at some distance from the wall, becomes parallel to freestream. Two sets of shocks are produced, one in front of the jet, and the other when the flow behind the jet reattaches to the wall. Such shocks produce losses in the flow outside, which must be considered in the cycle analysis and are difficult to determine. Combus-

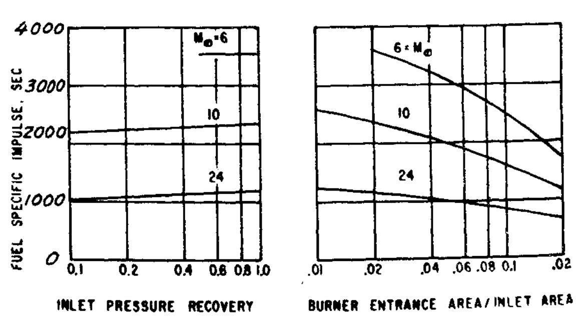


**Fig. 12 Variation of performance with design parameters.**

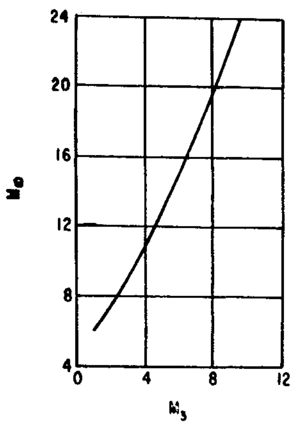

Fig. 13a Conditions at burner entrance for fixed-geometry engine.

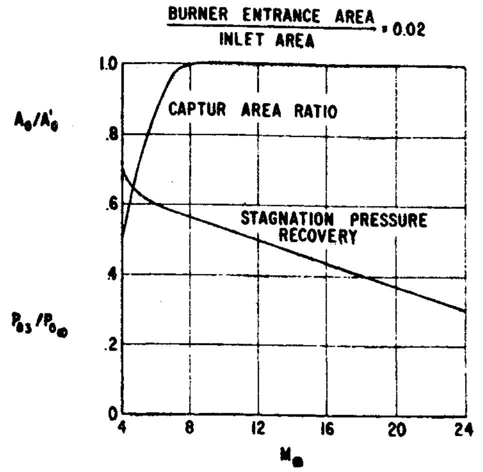


Fig. 13b Inlet performance for fixed-geometry engine.

tion starts in region 1 and propagates upstream into the separated region. In this region the heat-transfer rate is locally extremely high because the combustion occurs locally, very close to the wall. Heat transfer as high as 10 to 20 times the local heat transfer before injection has been measured experimentally (unpublished data) in this region. The combustion and the pressure rise tend to move the separation upstream and can separate the flow in the inlet. The flow in the region 2 behind the wall is a low velocity and the pressure is low. Since the temperature is high, however, combustion can take place here for some conditions. This tends to increase the local pressure, and, therefore, changes the trajectory of the jet. The amount of combustion taking place in region 2 is affected strongly by static pressure variation and by the amount of combustion taking place at 1; therefore it is very sensitive to variation of Mach number, altitude of flight, and type of separation produced. Thus the penetration is strongly affected by all such parameters. When the engine must be designed for a large range of flight Mach numbers, there exists the problem of matching the fuel penetration with the value of stagnation pressure of the fuel, which is determined by the amount of fuel that must be injected at a given flight condition station. Such matching requirements impose additional constraints on the selection of cycle design. At low Mach numbers the injectors located near the minimum areas of the flow must inject very small mass flow in order to avoid choking the stream, and the largest part of the fuel must be injected downstream. At high Mach numbers the opposite requrement exists; note that much of the fuel must burn near the wall and thus will increase the local heat transfer.

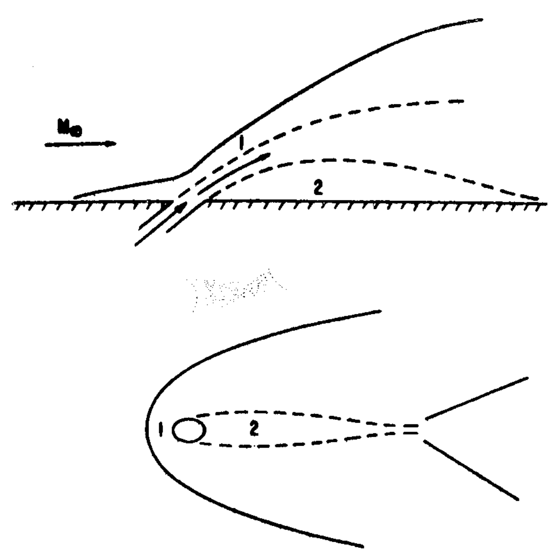

Fig. 14 Schematic wave or pattern vectored injection.

In addition, for any given flight condition and local flow conditions, the penetration of the jet depends on the stagnation pressure of the fuel. A change in fuel mass flow (variation of $\Phi$) changes the dynamic pressure of the jet and therefore the penetration and the region reached by the fuel; a small value of $\Phi$ corresponds to a large amount of burning near the wall and a small amount of burning in the mainstream as opposed to the requirements for good cooling characteristics. It must be noted that the cooling requirements are most severe at low values of $\Phi$.

### Tangential Injection with Chemical Reaction

The turbulent mixing of a heterogeneous coaxial stream has been investigated in detail analytically and experimentally in the last few years, and many of the available results have been presented in Refs. 7–12. On the basis of present knowledge, the mixing in the absence of large pressure gradient can be represented accurately enough for engineering purposes by the analytical expressions suggested in these references. Good agreement between experimental and analytical results is found when diffusion and heat conduction processes are investigated. (These are the physical properties of interest in a mixing process used in combustion.) However, discrepances are found in many instances when the same types of analyses are applied to determine velocity decays and Mach number decays produced by mixing processes. Such discrepancies are not too important for the problems under investigation, but they are significant here because they often have been attributed to major shortcomings of the representation used. It is the writer's opinion that a careful analysis of the experiments where the velocity distribution has been measured for a mixing process could prove that the discrepancies found are not due to incorrect transport properties characteristic assumed in the analysis, but to incorrect interpretation of the experiments. Velocity variations in the mixing process can be generated by two different effects, a small pressure gradient in tangential or normal direction of the flow at the initial conditions, or viscosity. These pressure gradients usually exist in the free-flight experiments, whereas in controlled experiments any small longitudinal pressure gradient existing in the flow outside the mixing regions has equivalent effects. These small pressure gradients in the flow outside of the mixing region are present in any experiment, but they are usually

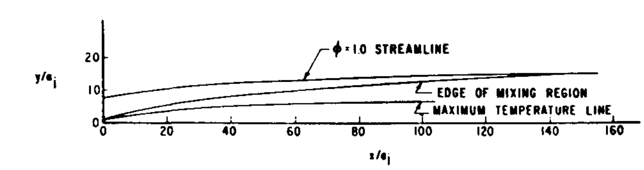


Fig. 15 Mixing and combustion region-two-dimensional constant pressure.

neglected and often not even measured. The experiments on mixing are usually on axially symmetric jets. In such a flow, small pressure gradients can produce large effects because of the focusing effect of the compression waves at the axis of the jet. The focusing produces large Mach number variations near the axis, changes the mixing profile, and can produce local shocks, which in turn affect the decay at the axis. Experiments have been performed at New York University in order to investigate this point.[13] A small disturbance has been placed at the wall of the tunnel. This disturbance, which produces very small variations far from the axis, induces large variations at the center of the tunnel, and affects strongly the Mach number decay at the axis of the mixing.

The mixing analysis has been extended with success to problems with chemical reaction. Several comparisons of experiments with analyses already have been presented for hydrogen air flame in the literature.[7,8,11,12] Recent unpublished experiments show that the agreement between experiments and analysis is very good, even in the case of hydrocarbon air combustion process, in spite of the additional uncertainties due to the more complex chemistry of the hydrocarbon.

### Analysis of Flowfield in the Combustor

The success encountered in analyzing combustion processes has encouraged several investigators to develop procedures that permit taking into account the effect of combustion outside the combustion region in the flowfield. The combustion process produces changes in the streamline shape which affect the pressure. Such deviations of direction propagate in the outside flow through waves. It is well-known that the effect of heat sources introduced in the flow is equivalent aerodynamically to the introduction of equivalent volume sources. In order to obtain the wave pattern produced by combustion in the flow, a detailed study of this equivalence has been performed for a mixing process with combustion. In the analysis, the pressure distribution has been assumed to be known along the mixing process, and the deviation of the streamline outside of the combustion region has been determined analytically.[12] A typical result of such a calculation is shown in Fig. 15.

The static pressure along the mixing in this example has been assumed to be constant. The shape of the streamline outside the constant region and the pressure distribution along such streamline define the outside flow constant with such process. This type of analysis has been performed first by using mixing types of approximations. Longitudinal pressure gradients are accounted for in this type of analysis, but normal pressure gradients are neglected. This simplification implies that pressure variations generated by combustion are assumed to travel normally to the axis of the

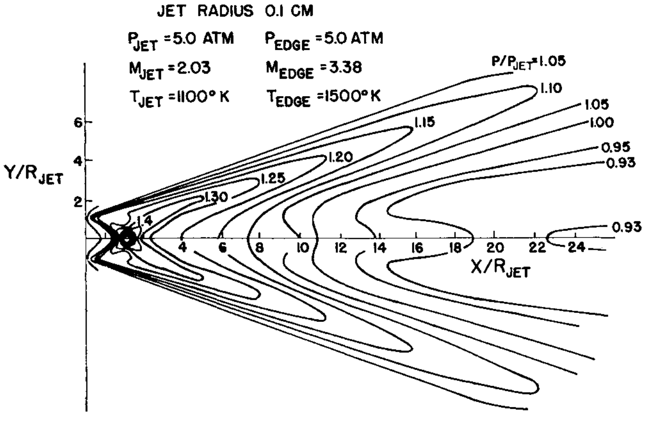


**Fig. 16 Pressure field for a freejet of hydrogen in air computed by a method of characteristics with viscosity and finite rate chemistry.**

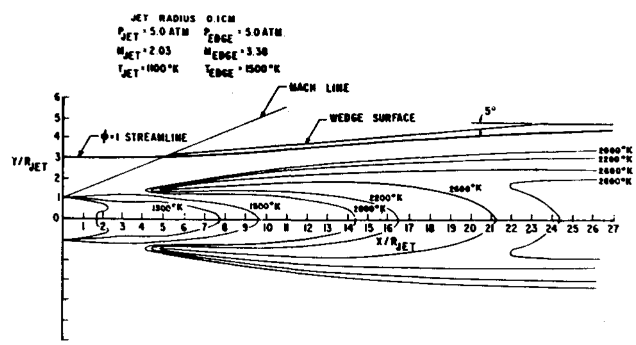


**Fig. 17 Temperature field and $\phi = 1$ streamline for a free-jet of hydrogen in air computed by a method of characteristics with viscosity and finite rate chemistry.**

mixing, not along Mach waves. In order to introduce corrections to the mixing analyses, a more complex type of analysis has been developed,[7,14,15] in which the effects of viscosity, conductivity, and diffusivity are included together with pressure gradients in both directions. The analysis is performed as in the case of inviscid flow. However, the terms of the equations which include the transport properties are kept and determined locally in the step-by-step numerical calculation at each point, and are assumed to be equal to the average between the value at initial and final for each step. A result of such analysis is shown in Figs. 16 and 17. The deviation of the streamline outside the mixing region given by such a calculation is in good agreement with the deviation obtained from a mixing analysis for similar initial conditions, but is displaced longitudinally. In addition, the flow inside the flame has large variation of static pressure. These differences are important when the reaction rates are slow, and reaction time is comparable with the time required for mixing. This comparison confirms that the model used in combustion calculations on the basis of mixing analysis is sufficiently approximate.

### Nozzle

The design of a practical nozzle in a supersonic combustion engine is of importance, especially at high Mach numbers where the variation of total impulse due to the engine is a small fraction of the total impulse of the flow entering the engine. However, very little research has been performed in this field. The following problems must be investigated: 1) the effect of nonuniformity at the entrance of the nozzle, 2) the effect of viscous effects and under expansion of the nozzle, and 3) the effect of nonequilibrium flow under various flight conditions and fuel-air ratio.

The nonuniform flow can be due to nonuniformity of species concentration and temperature, as a consequence of imperfect mixing, and to presence of pressure gradients and shocks. Because the flow is often nonaxially symmetric, a three-dimensional type of analysis is required. The viscous effects and underexpansion are sources of important losses, and optimization procedures that include effect of nonuni-

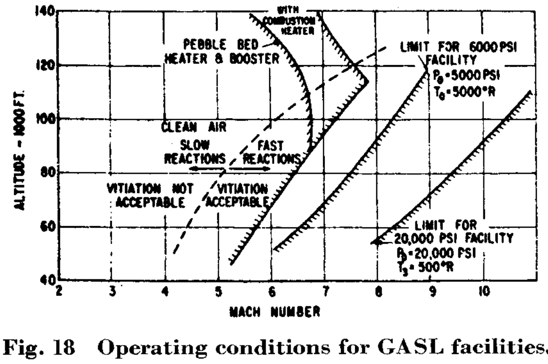


**Fig. 18 Operating conditions for GASL facilities.**

formity and viscous effects are required for good nozzle design but are not available at present.

The effects on nonequilibrium flow are important at high Mach number and high altitude, and can be reduced by selecting the correct burner pressure and temperature and the optimum fuel concentration. The amount of work performed until now in the investigation of this component is too small with respect to requirements. Good progress has been made toward sufficient understanding of chemical nonequilibrium reaction processes in the nozzle and development of method of analysis for axially symmetric nozzles.

Numerical programs are available which can calculate with sufficient accuracy the chemical reaction along a given streamline, if transport properties are neglected and the pressure is given. In addition, axially symmetric or two-dimensional nozzles with flow chemically active can be analyzed by the method of characteristics when transport properties are neglected. The method described before and applied to mixing and combustion processes could be applied to the nozzle design to take into account transport properties, if required. The two-dimensional, boundary-layer problem also can be treated analytically with sufficient approximation.

## Engine Development and Testing

In order to test air-breathing engines on the ground, stagnation conditions, pressure, and enthalpy must be simulated. Presently, because of good knowledge of chemical reaction rates for the hydrogen air reaction, the flight regime, from SCRAMJET engine test point of view, can be divided into two fields. One field considers the low flight Mach number and high-altitude high Mach number where the reaction time (which corresponds to a reaction length) is small but significant with respect to residence time and physical dimensions in the combustor. The second region includes all the flight conditions where the static conditions in the burner (static pressure and temperature) are sufficiently high so that the time required for complete reaction is very small with respect to the time required for mixing. The definition of the two fields depends on the fuel used, on the engine design (burner static conditions), and/or the temperature of the fuel at injection.

When the combustion is fast enough, the condition of simulation of the air properties and of stagnation enthalpy required in order to determine engine performances can be somewhat relaxed. Then vitiated air can be used for the tests where the gas used contains the correct amount of oxygen but has products of combustion in place of nitrogen. The results obtained from such tests do not correspond directly to test in air. However, small corrections can be introduced, obtained on the basis of analysis, in order to extrapolate measured values to flight conditions. The stagnation enthalpy and pressure of the test can also be reduced, because the turbulent mixing is only slightly affected by variation of Reynolds number, while the reaction rates above a given temperature can be considered to be infinitely fast. Figure 18 indicates projected capabilities of total engine tests of present facilities and indicates the division between the two regions of requirements.

## Concluding Remarks

The SCRAMJET technology has made great progress since the meeting of AGARD in Milan, Italy in 1960 where such schemes were discussed in detail for the first time.[1,2] Presently we are in the second phase of acceptance of this new technological approach.

Two main difficulties exist for the development of practical engines; the engines size for practical application becomes so large as to forbid test of a complete engine, and the tests related to structural integrity require full stagnation conditions. Both requirements tend to indicate that the engine proper must be divided into the following three regions:

1) The entrance region of the inlet, which usually has small thermal stresses and can be tested separately.

2) The final region of the inlet, the burner region, and the entrance region of the nozzle. This part has relatively small dimensions and is the most critical part for aerodynamic heating and stresses.

3) The last part of the nozzle, which also must be tested separately.

Because of the present industrial organizations, the first and last parts probably will be handled by the airplane manufacturers, whereas the central part will be designed and built by the engine manufacturers. Then the engine manufacturer must solve the problem of designing an engine that can accept several entrance and exit shapes and conditions, and can be tested on the ground, with good simulation of the entrance conditions.

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