Liquid-Fueled Supersonic Combustion Ramjets: A Research Perspective

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A review of past and current research applicable to liquid-fueled supersonic combustion ramjets is presented and discussed. An assessment of its strengths and shortcomings is made and, finally, a list of research opportunities that merit consideration in this rapidly expanding area of airbreathing propulsion is presented.

Nomenclature

A = area
$\bar{C}_f$ = average combustor wall skin friction coefficient
D = isolator duct diameter
d = injector diameter
ER = fuel-air equivalence ratio
f = fuel-air ratio
h = penetration height, enthalpy
Δh = driving enthalpy (Ref. 85)
M = Mach number
P = static pressure
$\bar{q}$ = fuel-to-freestream dynamic pressure ratio, average
q = combustor wall heat flux
$R_{e\theta}$ = Reynolds number based on momentum thickness
S_d = length of precombustion shock structure in supersonic combustion
S_s = total length of precombustion shock structure
T = temperature
w = fuel jet lateral spreading
$\dot{w}$ = mass flow rate
x = axial distance downstream of injection
θ = fuel injection angle
θ_m = boundary-layer momentum thickness
η_c = heat release combustion efficiency
$\bar{\tau}$ = average combustor skin friction

Subscripts

0 = freestream
1 = conditions behind inlet bow shock
4 = supersonic combustor inlet/isolator upstream of precombustion shock
5 = supersonic combustor exit
a = air
f = fuel, final
gg = gas generator
j = injector
t = total (zero velocity) condition
w = wall

Introduction

THE potential for the combustion of storable liquid fuels (see Ref. 1 for the gaseous fuel counterpart) in supersonic airstreams for efficient propulsion engines at hypersonic speeds has been recognized since the late 1950's.[2-4] This propulsion cycle, commonly referred to as the supersonic combustion ramjet (scramjet), has an additional feature that permits it to operate over a much wider flight Mach number M_0 range than most other airbreathing engine cycles: at low M_0 and high fuel-air equivalence ratios (ER), the combustion process is initially subsonic, similar to that of a conventional subsonic combustion ramjet, but it accelerates to a supersonic speed prior to exiting the combustor. On the other hand, regardless of ER, it is entirely supersonic at low M_0 and low ER as well as at high M_0. This is referred to as dual-mode combustion and permits efficient operation of the engine at $M_0 = 3$ to 8–10 for liquid fuels and up to orbital speeds for gaseous fuels. The upper limit for the liquid-fueled cycle is, of course, due to energy consumption by dissociating and ionizing species at elevated temperatures, which cannot be compensated for by additional fuel as in the case of, for example, a diatomic gas such as hydrogen.

Early experimentation on the scramjet cycle began in the late 1950's with tests of external burning on double wedges at Mach 5 to demonstrate the feasibility of combustion in a supersonic airstream using a liquid fuel.[5,6] These were followed by a series of inlet tests and internally ducted connected-pipe combustor tests during the 1960's and early 1970's using a variety of liquid fuels, ignition enhancement aids, fuel injector configurations, and combustor geometries at simulated flight conditions up to Mach 7, all culminating in free-jet engine demonstration tests.[7]

Since these tests, most of the effort in developing scramjet engines has been directed toward demonstrating the viability of a hybrid engine cycle, denoted the dual-combustor ramjet (DCR),[8] which incorporates the best features of the scramjet and subsonic dump combustor ramjet to overcome the necessity of using logistically unsuitable liquid fuels, fuel additives, or fuel pilots to achieve the required combustor performance in a pure scramjet.

Concomitant with these development tests have been a number of more fundamental, albeit much less extensive, efforts focused on understanding some of the more fundamental issues of the scramjet and DCR engines as well as the supersonic combustion process itself. The purpose of this paper is to identify the major areas of research, review past basic research efforts, discuss ongoing research, and present a series of research opportunities in need of exploration in this rapidly expanding field.

Engine Cycles/Operation

Prior to discussing the research efforts, a brief digression describing the engine cycles and components is necessary in order to put some focus or perspective into the discussions. Accordingly, Fig. 1 presents generic schematics of both the scramjet and DCR engine cycles. In the scramjet engine (Fig. 1a),[9,10] the freestream air is diffused and contracted in the inlet to a supersonic Mach number M_4, which is typically 0.4–0.5 M_0. Fuel is injected downstream of station 4. The combustion process generates a shock train in the vicinity of

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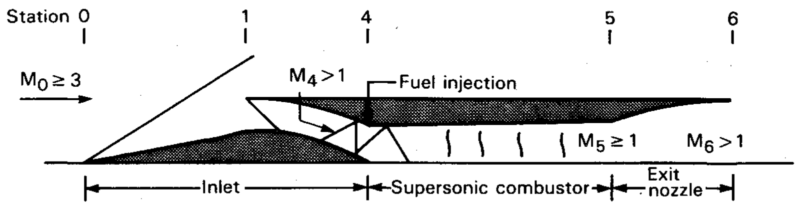
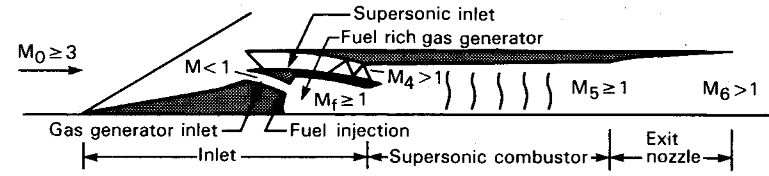
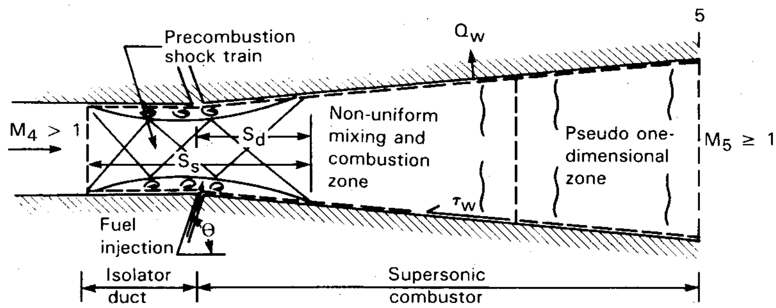


a) Supersonic combustion ramjet.

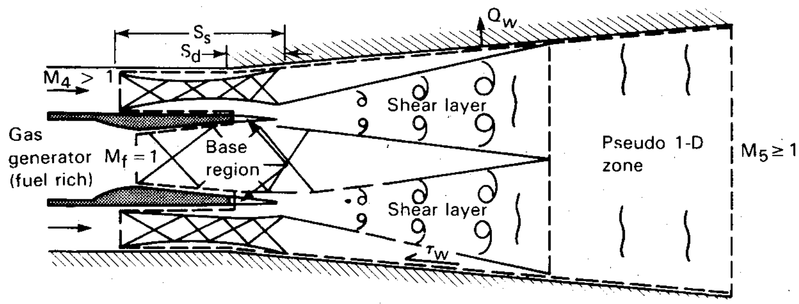


b) Dual-combustor ramjet.

Fig. 1 Schematic of generic supersonic combustion engines.



a) Scramjet combustor.



b) Dual-combustor ramjet.

Fig. 2 Schematics of scramjet and DCR combustion processes (at flow station 4).

the fuel injection station that can vary in strength between no shock and a normal shock, depending on M_4, ER, and the combustor geometry. Injection, vaporization, mixing, ignition, and combustion of the liquid fuel occurs in the supersonic combustor between stations 4 and 5 with $M_5 \geq 1$. The flow is then expanded in a diverging supersonic nozzle.

In the DCR cycle (Fig. 1b),[8] the major change is that a portion of the air captured by the inlet (typically one-fourth) is diffused to a subsonic Mach number to supply air to a small subsonic dump combustor. All of the fuel is injected into the dump combustor (or gas generator) so that it operates very fuel rich (typically, $ER_{max} \geq 4$). The fuel, however, is distributed such that a near-stoichimometric mixture is maintained in the central dome of the gas generator to maintain a stable pilot zone, with the remainder of the fuel distributed elsewhere. This permits the unreacted fuel to be vaporized, cracked, and preheated prior to entering the supersonic combustor, enabling the use of pure heavy hydrocarbons and, perhaps, metallized slurry fuels not heretofore possible in a scramjet. The remainder of the cycle is the same as the scramjet and, because the gas generator operates fuel rich, there are none of the combustor/inlet instabilities usually associated with subsonic combustion ramjets operating at or below stoichiometry.

Figure 2 presents more detailed schematics of the two generic, liquid-fueled supersonic combustion processes to illustrate the fundamental flow structures and regions of interest. Figure 2a represents a scramjet combustor with fuel injection from the walls through either discrete holes or flush slots with angle θ to the airstream. Staged injection is not precluded, but, for simplicity, only one fuel injection station is shown. Instream fuel injector struts are not shown because they require active cooling, especially at the higher M_0, and liquid hydrocarbon fuels do not, in general, have the cooling capacity required. A simple diverging cone frustrum is shown, but the qualitative features for other shapes, such as those containing constant-area sections or step increases in area, are the same. Here, the blockage due to the combined effects of the fuel penetration, vaporization, and heat release generates a "shock train" disturbance that, in general, originates upstream of the fuel injector ports and extends into the combustor. At moderate combustor inlet Mach numbers M_4, i.e., 1.5–3, and typical fuel-air equivalence ratios, i.e., 0.5–1.0, the pressure rise associated with the shock train is of sufficient strength to separate the incoming boundary layer, which then reattaches prior to exiting the combustor. The length of the shock train is defined as the length of the corresponding s-shaped pressure rise and is given as s_s. The distance that the shock train extends into the combustor is s_d. As flight speed increases, M_4 correspondingly increases and the strength of the pressure rise in the shock train decreases.

Within and downstream of the shock train, vaporization, ignition, mixing, and combustion are intensive, with large radial, axial, and, perhaps, circumferential gradients in the flow properties and chemical composition. This region is labeled "nonuniform mixing and combustion zone." Further downstream, the mixing and combustion is less intense and gradients are considerably weaker. This region is labeled the "1D zone" (cone-dimensional). The control boundary that forms the basis for mathematical models is comprised of the plane at station 4 upstream of the shock train, the combustor walls, and the combustor exit plane at station 5.

As previously discussed, the supersonic combustor operates as a dual-mode combustion system.[8-10] At low M_0 (or M_4) and ER, combustion is preceded by a compression field whose strength is equivalent to that of an oblique shock wave and the combustion process is entirely supersonic. As the ER (or heat release) is increased, the strength of the precombustion compression field increases until it reaches a maximum equivalent to a normal shock pressure rise. At this point, the combustion process begins in a subsonic flow, but reaccelerates through a thermal throat in the combustor such that $M_5 > 1$. Additional heat can be added without a change in the strength of the precombustion compression field until the thermal throat and station 5 coincide, i.e., until $M_5 = 1$. At low M_0, this limit generally corresponds to an $\mathrm{ER}_{\max} < 1$ and is a function of the properties at station 4 and the combustor geometry. Adding heat beyond this limit is possible, but M_5 will be subsonic, requiring a geometric throat in the exit nozzle. As M_0 increases, $\mathrm{ER}_{\max}$ increases until stoichiometry is reached with $M_5 = 1$. As M_0 continues to increase with $\mathrm{ER} = 1$, the strength of the precombustion compression field decreases to the equivalent of an oblique shock wave and the combustion process is entirely supersonic, which generally occurs when $M_0 = 5$–6. Another important feature of the flow is the wall skin friction $\bar{\tau}_w$. Whereas $\bar{\tau}_w$ and the resultant thrust losses are small in a subsonic combustion system, they can be quite significant in a scramjet combustor and vary by a factor of two to three, depending on the conditions at station 4 and the amount of heat release.

Figure 1b corresponds to the DCR configuration wherein the prevaporized, heated, and cracked liquid fuel is injected axially, surrounded by a supersonic airstream. In this case, the "fuel" is, in fact, fuel-rich products of combustion that are injected at sonic or supersonic speeds (M_f), similar to wall or instream slot injection in a scramjet using a gaseous fuel. In this case, the major features consist of the combustion-induced shock train and attendant separated zone along the wall and at the exit of the gas generator, the wall skin friction, the nonuniform mixing and combustion zone, and the nearly uniform one-dimensional zone. However, unlike in the scramjet combustor, where fuel penetration and vaporization from the wall is a primary concern, here the mixing and combustion process is governed by a free shear layer emanating from the juncture of the air and "fuel" streams and complicated to some extent by the existence of a base flow region where the two streams meet. The control boundary for mathematical modeling is, again, represented by the dashed line.

Areas of Research

With the foregoing introduction, identification of the major areas of interest for research is accomplished by dividing the engine into components (viz., the inlet, isolator duct, fuel/fuel injector, combustor, and exit nozzle) and then listing the fundamental issues associated with each. The ultimate objectives, of course, are to understand the governing physical and chemical processes and alteration mechanisms in each area and to predict accurately these processes using mathematical models.

Engine Components

Inlet

For inlets (Fig. 3), the objective is to determine how much air is captured, what the total pressure recovery or kinetic energy efficiency is, and what the flow profiles are at its exit, e.g., pressure, velocity, temperature, species, and turbulence, regardless of geometry. Accomplishing this requires detailed knowledge of both the external and internally ducted compressible fluid dynamics with and without the presence of shock waves with imbedded regions of mixed subsonic and supersonic flows. In addition, a complete understanding of the multidimensional growth of laminar, transitional, and turbulent surface viscous layers, their interaction with strong and weak compression and expansion fields, separation and reattachment, alteration mechanisms (e.g., suction), and dissipation and transport properties such as Reynolds stresses and difussion coefficients is required.

Fuel/Fuel Injector

While it is difficult, at best, to separate the fuel and fuel injector from the supersonic combustor, such a distinction is made here in an attempt for clarity. The objective here, then, is to use a logistically suitable, high-energy-density, storeable liquid fuel, fuel blend, or slurry (with or without a pilot or ignition aid) and a fuel injection system that permit 85–90% or more of the energy release available to occur prior to the flow exiting the supersonic combustor. Fuels, fuel additives, and fuel pilots that are pyrophoric, toxic, caustic, excessively expensive, or any combination thereof are not considered suitable, e.g., boranes, silanes, aluminum alkyls, chlorine trifloride, etc. To meet this objective requires a thorough understanding of the rheological and thermochemical properties of the fuels and products species, including vaporization and ignition energies as well as chemical kinetics.

In addition, for wall injection, one must know what the penetration, atomization, vaporization, and ignition characteristics of a given fuel in various fuel injector configurations are for a given set of initial conditions in the incoming airstream and fixed supersonic combustor constraints (see Fig. 4). For axial injection, it is necessary to understand how the free shear layer grows, what controls this growth, what the transport properties are, and how varying initial conditions and the presence of a base region alter the mixing and growth processes. Integrated into this is the necessity to understand how pilot fuels or oxidizers and external ignition aids alter the basic ignition and chemical kinetics processes. This understanding must ultimately include the presence of the precombustion and fuel injection compression fields, as well as the attendant regions of the boundary-layer separation. Consequently, it is necessary to accumulate and integrate knowledge in rheology, hydrodynamics, stratified flows, compressible fluid dynamics with the shock waves present, two- or three-phase thermochemistry and kinetics, viscous flows and the control thereof, and turbulent transport properties and mechanisms if a viable, cohesive picture of these processes is to be achieved.

Supersonic Combustor

The objective within the supersonic combustor (Fig. 2) is to maximize the amount of heat release permissible and minimize total pressure or stream thrust losses within a given length constraint. To realize this requires not only an understanding of the fuel preparation and injection processes, but also a thorough knowledge of the processes governing supersonic

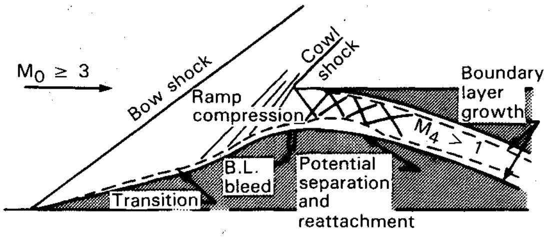


Fig. 3 Generic hypersonic inlet flow structure.

mixing and combustion as well as those precipitating other losses within the combustor. For fuel injection from the walls, this requires a knowledge of the governing transport (mixing) mechanisms and properties in the freestream, as well as along the walls, together with compressible fluid dynamics, viscous flows, and chemical kinetics and how they can be altered. For axial injection (e.g., in the DCR), additional knowledge is required, i.e., what mechanisms control the base flow region as well as the free shear layer mixing and combustion processes.

Exit Nozzle

The objective in the supersonic exit nozzle is to maximize the exit-to-inlet stream thrust ratio given certain geometrical constraints. In general, the exit and inlet areas and overall nozzle length are given and one must then choose a wall contour to maximize the stream thrust. This is not an easy task, since the initial (combustor exit) conditions are nonuniform, the average entrance Mach number is not constant (generally between 1 and 2), and the initial state properties and chemical species vary with the flight Mach number and fuel-air equivalence ratio. In addition, the expansion process is generally very rapid, so that the characteristic flow time may be faster than some of the chemical reaction rates (Damköhler number ≤ 1). Consequently, loss mechanisms include nonequilibrium kinetics, wall shear and heat transfer, divergence, and nonisentropic expansion of the inviscid core flow. Thus, a thorough knowledge of chemical kinetics, viscous flows with severe axial and normal pressure gradients, and compressible inviscid flows with chemical reactions and compression/expansion fields is required to understand, perhaps control, and, ultimately, predict the flow in this expansion process.

Modeling

The objective in modeling supersonic combustion engines is ultimately to accurately predict the thrust of the engine for all flight Mach numbers, altitudes, angles of attack, and fuel-air equivalence ratios. Accomplishing this generally requires models for each of the components that can be molded into an overall engine cycle model, which, in turn, can be incorporated into either a component or engine cycle optimization procedure. The models should vary in complexity from useful engineering tools, such as integral analyses, to solutions of the full Navier-Stokes equations with viscosity and chemistry included. Each must also be experimentally verified where possible, since each model requires assumptions about some of the physical processes taking place. Reference 11 gives an excellent review of the modeling approaches currently in use, under development, or planned; the reader is referred to it for the details of those efforts.

Prior to discussing the research efforts, two revolutionary tools that have significantly enhanced our ability to understand these flows should be discussed. The first is the dramatic reduction in computational time and increased storage available in class VI computers over the past five years that permit at least some solutions of these complex physical processes not heretofore possible. While great strides are being made in solving the mathematically complex models of these flows, a number of simplifying assumptions of the governing processes are still necessary for solutions to be achieved (e.g., turbulent transport mechanisms and chemical reactions, to name two) and, as stated above, experimental data with sufficiently detailed measurements do not exist to verify these assumptions or to validate the computational results.

The second significant change is the development of nonintrusive measurement techniques over the last 10 years or so (see, e.g., Ref. 12). These techniques, while still evolving, permit or will soon permit measurements of particulate size and mean and fluctuating velocity, temperature, density, species (selectively), and pressure using Doppler shifts, light scattering, florescence, cineradiography, or a combination thereof. Consequently, while still somewhat expansive, it is now possible (or will be in the near future) to spatially and temporally resolve most of the flow and transport properties within these complex flows and greatly enhance our understanding of the governing physical and chemical processes.

Past and Current Research

With this introduction and background, the remainder of this paper will focus on past and current research efforts in each of these areas (other than modeling), identify deficiencies in our understanding of the governing physical or chemical processes and concomitant data base; and, in the next section, discuss research opportunities and attempt to prioritize them in a logical fashion. For convenience, this section is subdivided into the five engine component areas previously discussed.

Inlets

Past research on hypersonic inlets (Fig. 3) has generally been limited to testing scale models and measuring air capture using a throttle, stream thrusts using force balances, exit flow properties using pitot and cone static pressure probes, wall static pressure distributions (see, e.g., Refs. 7, 13, and 14), and empirical correlations of the resultant inlet efficiency.[15,16] Boundary-layer control devices using injection or suction have also been tested, especially in the Soviet Union.

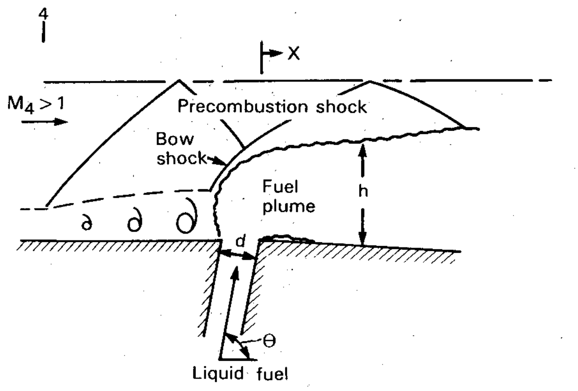
Research of a more fundamental nature, such as on the transition to turbulence and boundary-layer separation, is well-documented empirically on sharp-edged cones and wedges as well as flat plates (see, e.g., Refs. 17-21), including the effects of tunnel noise and surface roughness. There is also a sound data base of mean flow properties in turbulent boundary layers with and without pressure gradients, some of which include turbulence intensity measurements (see, e.g., Ref. 22).

None of these data, however, measure the flow near the wall in sufficient detail (for the flow speeds of interest) to spatially and temporally resolve the fundamental dissipation and transport mechanisms, i.e., the Reynolds stresses and wall shear. It is these that are of interest in hypersonic inlets, especially in internally ducted supersonic flows with compression and expansion waves, in order that the viscous total pressure losses as well as regions of separation can be modeled and predicted with confidence.

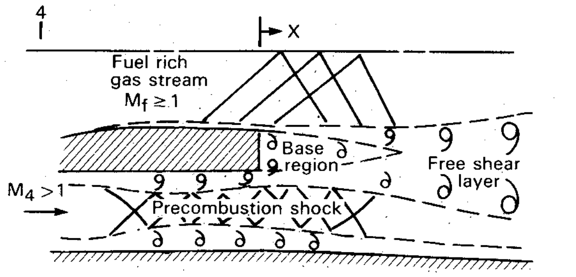
The only current relevant work in this area is being done at NSWC/White Oak by Yanta[23] where two- and three-dimensional laser Doppler velocimetry (LDV) measurements are being made at various axial stations in a two-dimensional inlet model, using suction to control the initial boundary-layer thickness at Mach 4. Measurements include profiles of mean and fluctuating velocities, Reynolds stresses, wall static pressure, and skin friction. However, there is still a strong need for additional spatial and temporal measurements of temperature, pressure, or density, as well as velocity, under differing initial conditions using state-of-the-art diagnostic techniques.

Isolator

Previous efforts to understand the structure of the precombustion shock system for scramjets[24-27] have centered on simulating the shock structure by throttling a supersonic



a) Scramjet.



b) Dual-combustor ramjet.

Fig. 4 Fuel injection flow structure.

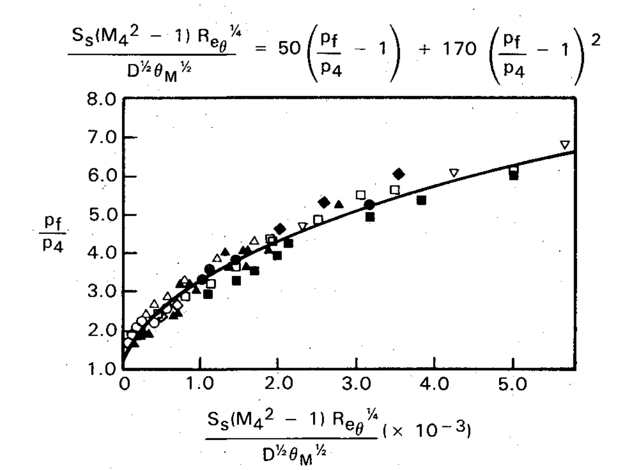


Fig. 5 Correlation of precombustion shock experimental data for scramjet (from Ref. 28).

airstream in a constant-area circular or rectangular duct and measuring wall static pressure distributions, boundary layer and in-stream, normal and axial pitot pressure distribution, and wall skin friction over a Mach number range M_4 of 1.5–2.7. The resulting data base has been empirically correlated[28] as a function of the overall static pressure rise, M_4, Reynolds number, boundary-layer momentum thickness, and duct geometry; the results are shown in Fig. 5. This correlation has been compared with experimental data (see, e.g., Ref. 28) and the agreement is quite good. Some attempts to correlate measured results in free-jet engine tests[7] with incipient separation predictions[21] have been made, but the agreement is not very good in those tests in which an isolator duct was used between the inlet and combustor. Current efforts in this area (see, e.g., Ref. 29) are focused on lower values of M_4 (below 1.5) using essentially the same measurement techniques.

For the DCR, a similar set of experiments is currently underway[30] to determine a correlation as a function of θ_m, M_4, M_f, mass flow split between the supersonic and "fuel" jets, and geometry. Figure 6 shows typical wall static pressure distributions for two different cases with $M_4 = 2.4$ to illustrate the large effect that different mass flow splits have on the shock structure.

While these results are, as in the case of the inlet data, very useful in development programs, they do not address the fundamental mechanisms (viz., turbulent transport and dissipation), other than in a cursory sense, that control the ability of the boundary layer to withstand a certain pressure gradient prior to separating nor the resultant shock structure and possible reattachment point. Such studies are required to fully understand and, ultimately, predict these flows.

Liquid Fuels and Pilots

Quite a large number of liquid fuels, fuel blends, and fuel pilots for scramjets have been tested in connected-pipe combustor and free-jet engine tests over the past 20 years[7,31–35] in an attempt to find a fuel that was energy-density efficient, would burn to near completion in the residence times available (typically <1.5 ms), and also be logistically suitable. Unfortunately, those fuels, fuel blends, and pilots that did perform well were not logistically suitable, i.e., they contained toxic, pyrophoric, or carcenogenic components that were unacceptable. Monopropellant pilots, which are logistically suitable, were also tested, but could not sustain the desired degree of heat release.[36] Thus, there still exists a need to develop a high-energy-density, storable liquid fuel or fuel pilot that is both highly reactive and safe.

The DCR hybrid combustion concept evolved as a direct result of this limitation. It does permit pure heavy hydrocarbon fuels[37] to be used and achieve the desired energy release in the residence times available at hypersonic speeds.

Slurry Fuels

Several slurry fuels, most notably those containing magnesium and THMCPD have been tested in a scramjet environment, but combustion was not sustained.[32] Other types, such as carbon or boron suspended in a heavy hydrocarbon have not been tested. Of the two, the boron slurry is the most desirable if the long ignition delay associated with the B_2O_3 particle coating can be overcome. There is current research addressing these issues.[38] In addition, the DCR concept offers a propulsion concept that may aid in overcoming this problem, since it permits the boron particles to be extensively preheated prior to entering the supersonic combustor.

Wall Fuel Injection

The injection, penetration, vaporization, mixing, and ignition process of liquid or slurry fuels is complicated at best, especially with the pressure of the precombustion shock (Fig. 4a). In order to make understanding the structure and mechanisms controlling it more tractable for research, it is generally accepted that the first step is to investigate the nonreacting, precombustion shock-free case. A number of investigators, most notably Schetz and his co-workers at Virginia Polytechnic Institute and State University have generated an extensive data base on the penetration h, spreading w, and unsteady nature of single- and multiple-port fuel injectors,[39–48] as well as some data on droplet size and distribution,[48–51] rheology,[51,52] mixing,[53] and ignition/flameholding.[54] These investigations also include variations in freestream Mach number M_4, injectant-to-free stream dynamic pressure ratio $\bar{q}$, injector hole size d, shape and angle θ, and injectant type (different liquids or slurries). The primary measurement technique used was very-high-speed photography with some recent measurements using species sampling probes[53] and Fraunhofer diffraction.[49]

Figure 7 is a photograph of the liquid injection process, showing some of its more salient features as well as a species sample probe. Here, $M_4 = 4$, $\bar{q} = 6$, $d = 2.0$ mm (0.08 in.),

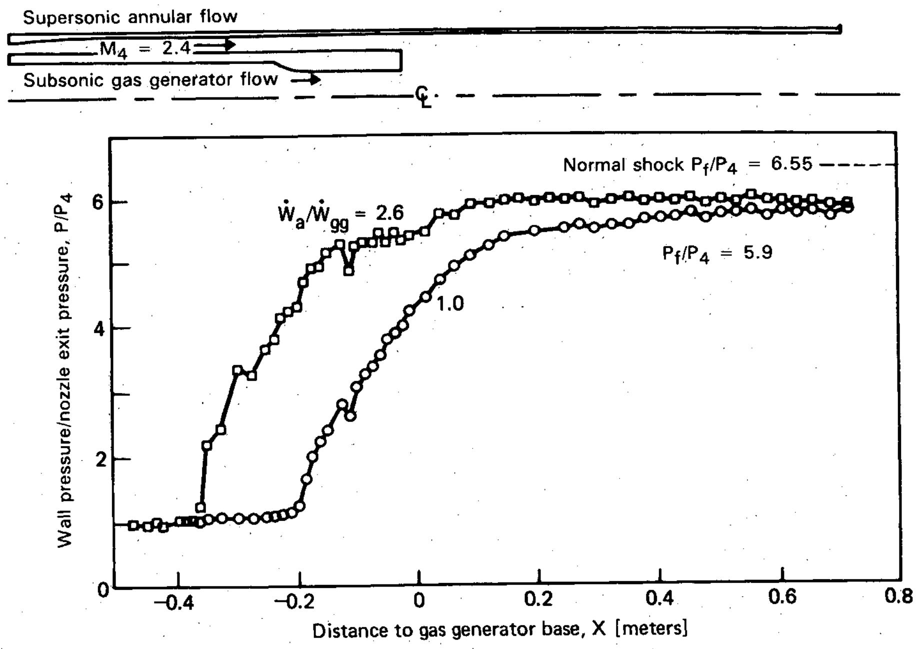


Fig. 6 Effect of mass flow ratio variation on DCR precombustion compression field wall pressure distribution (from Ref. 30).

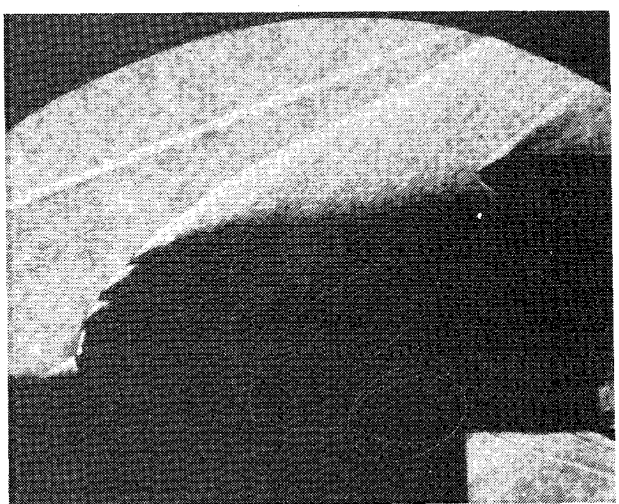


Fig. 7 Spark schlieren photograph of $\bar{q}=6.0$ liquid jet in Mach 4 airstream (from Ref. 53).

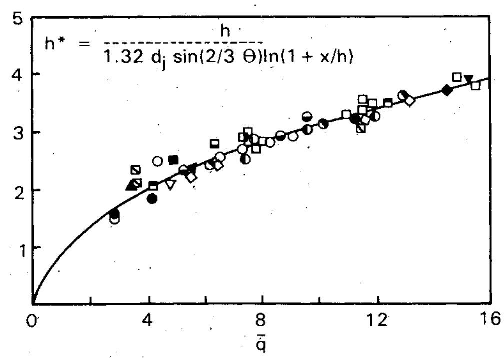


Fig. 8 "Reduced" penetration correlation of circular injectors (from Ref. 47).

and the injectant is water. The bow shock and wavy, unsteady interface between the liquid and air streams are clearly evident, as is the upper bound on penetration. This limit on penetration has been empirically determined for a variety of initial and boundary conditions, injector geometries, and fuel types; the resulting correlation is given in Ref. 47.

Figure 8 presents this correlation for circular injectors and illustrates how well all of the data (using photographic techniques) collapse to a single curve. Recently, however, results using the species probe,[53] as well as some Russian literature,[55,56] indicate that the correlation given in Ref. 47 and, by inference, photographic techniques may underestimate the actual penetration by ~25%.

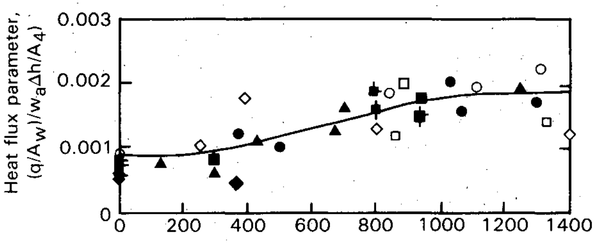
In either case, there is a physical limit on the actual penetration, which, in turn, limits the size of the supersonic combustor entrance without instream injection. For example, if $d=2.54$ mm (0.1 in.) and $\bar{q}=10$, both of which are practical upper bounds in a scramjet, then the penetration at $x=51$ mm (2 in.) is $h=50$ and 74 mm (2 and 2.9 in.) at $x=254$ mm (10 in.). For a more typical $\bar{q}$ of 5, $h=37.8$ mm (1.49 in.) at $x=51$ mm (2 in). Consequently, a practical limit on the entrance diameter or height of this combustor is 76–100 mm (3–4 in.).

While there is a large empirical data base on nonreacting liquid injection that is quite useful in development programs, a number of more fundamental scientific issues, some of which have been touched upon in previous studies, need to be addressed in future programs. These issues include using recent nonintrusive diagnostic techniques to determine the mean and time-dependent mechanisms controlling fuel penetration and spreading, droplet breakup and vaporization, mixing, and ignition of liquid and slurry fuels. On a more fundamental basis, these include a combination of shock/boundary-layer interactions, unsteady mixing, ignition, and combustion in the presence of compression/expansion waves, three-phase flows with and without thermochemistry and kinetics, and rheological effects of the fuels (see, e.g., Ref. 57). There are no ongoing efforts in this area.

Ignition aids, other than fuel pilots or additives, are not discussed here because they either do not alter or enhance ignition and combustion and/or require stored energy in an amount approaching that stored in the fuel tanks. These include photochemical, ionization, radical producing, and other ignition sources external to the fuel-air stream. This, however, does not preclude the discovering of one that is effective and uses a total energy that is at least one or two orders of magnitude less than that carried in the fuel.

Axial Fuel Injection

For convenience of discussion, the axial injection problem (Figs. 2b and 4b) can be split into two categories: base flows



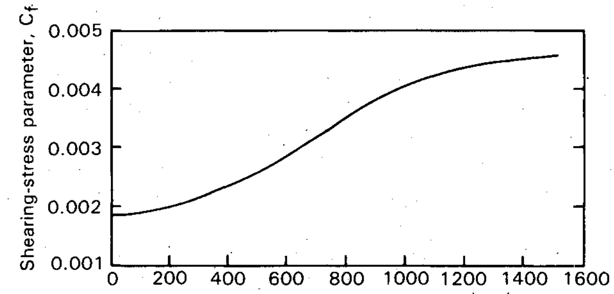


Fig. 9 Measured supersonic combustor of a) wall heat flux and b) deduced wall skin-friction coefficient (from Ref. 85).

and free shear layers. While base flows do have free shear layers at their boundaries, e.g., in the flow over a rearward-facing step without base injection, the shear layers of interest here are those generated by the axial intersection of two streams, each with an axial Mach number greater or equal to unity.

While there exists a substantial body of data on base flows in a supersonic stream (see, e.g., Refs. 58–62), most of the data consist of high-speed photographs, wall static pressure distributions, and some limited instream pitot pressure surveys. Recently, however, there have been several studies in which hot wires[63] and LDV's[64] were used to measure Reynolds stresses and turbulence intensities, which are essential if the fundamental physics controlling these base flows is to be well understood and predictable. Additional studies such as these and those in which the base region is generated by a bluff body between an air and fuel jet are needed if a true understanding of base flows and their influence on mixing, flameholding, and combustion is to be realized. Furthermore, measurements of turbulent transport properties as well as thermochemistry will be required to achieve these goals.

Our current understanding of the mechanisms governing and/or controlling the mixing and combustion processes when two sonic or supersonic streams merge to form a free shear layer, such as that shown in Fig. 4b, is very limited. The experimental data in the literature are generally 10–20 years old (see, e.g., Refs. 65–69) and provide only limited axial mean flow measurements of the shear layer thickness and/or pitot pressure and gaseous species profiles. While these data are useful in specific development programs, they do not document or describe the flow in sufficient detail, nor are they systematic enough to permit an adequate description of the flow structure or the mechanisms controlling it to be made. However, the data do show, for example, a distinct difference in the nonreacting growth rate of the shear layer when compared with an "equivalent" incompressible free shear layer's growth (one with the same density and velocity ratio between streams). The supersonic shear layer grows less rapidly.[67,70]

Although our understanding of the turbulent mixing and combustion processes in supersonic free shear layers is in its infancy, there is a firm body of information and basic understanding of the physical processes governing the free shear layer mixing process in incompressible[69,71-73] and compressible subsonic flows[74,75] This, in turn, provides a sound basis and point of departure upon which to build an understanding of the supersonic mixing processes in the free shear layer. Specifically, these incompressible free shear layers consist of large-scale, coherent vortical structures whose growth (or thickness) is dependent upon vortex pairings and the ratio of velocity and, to a lesser extent, upon the density between streams. Furthermore, the growth of these free shear layers can be altered by using harmonics or subharmonics of the natural frequency of the vortical flow to either suppress or excite vortex pairing. Whether the structure, growth, and alteration mechanisms in supersonic free shear layers are the same needs to be addressed.

Ultimately, chemical reactions must be included if the combustion, as well as mixing, processes are to be understood and, if possible, enhanced. Currently, our fundamental knowledge of the coupling of the combustion and mixing process in free shear layers is, essentially, nonexistent for supersonic flows and very limited in the incompressible case. Two recent studies[76,77] in incompressible shear layers indicate that, for small amounts of heat release, there is little, if any, alteration of the shear layer structure. However, when moderate amounts of heat addition are permitted, the large coherent vortical structure persists, but entrainment into the mixing layer decreases with no observed change in its thickness. The work reported in Ref. 75 shows that both large-scale coherent structures and small-scale eddies are needed to promote good mixing (large-scale structures) and maintain a high chemical heat release efficiency (small-scale eddies). These latter data, however, were obtained in a compressible, rather than incompressible, shear layer. As these results illustrate, there are insufficient data in any of the flow regimes having chemical reactions to permit a clear picture of the flow structure and coupling between the fluid dynamics and combustion to be made.

Consequently, there is a need to apply new and/or existing experimental techniques to supersonic free shear layers with and without heat release to temporally and spatially resolve the mean and fluctuating properties if a thorough understanding of these critical processes is to evolve. Currently, there are some limited initial efforts in this area.[1,78,79]

Combustors

Flame stabilization and combustion, along with wall friction and heat-transfer losses, are the major processes of concern in the supersonic combustor (in addition to the fuel injection and mixing processes previously discussed). Flame stabilization and combustion are intimately tied to the fuel injection, vaporization, mixing, and ignition processes as well as combustor geometry and, as such, just about all of the experimental combustion data available are configuration dependent (see, e.g., Refs. 1, 7, 28, 31, 32, 37, and 80–82). Of the configurations tested, the one that has consistently produced heat release combustion efficiencies η_c in excess of 85–90% in a scramjet is one having a 76–100 mm (3–4 in.) entrance height or diameter, followed by a 25–35% step increase in area, followed by a constant-area section, followed by a diverging area section with a divergence angle of 0.5–1.5 deg. Thus, flame stabilization is achieved by using a rearward-facing step and efficient combustion by initially eliminating and finally restricting the acceleration (or expansion) rate of the flow to prevent quenching of the reactions over a wide range of initial conditions: $M_4 = 1.5$–3.5, $P_4 = 0.5$–5 atm, $T_{t4} = 700$–2200 K (1260–3960°R). There are no fundamental data available in the literature on flame stabilization and combustion for the DCR.

It is important to note that, whereas total pressure losses attributable to wall skin friction are generally small in subsonic flows, they can be quite significant in supersonic flows, especially in the presence of an exothermic reaction. Based on an extensive data base of global heat-transfer data,[7,85] Billing

Table 1 Research Opportunities in Supersonic Combustion

	Engine component					
Research area	Inlet diffuser	Isolator duct	Wall fuel injector	Axial fuel injector	Supersonic combustor	Exit nozzle
Boundary layers						
Transition	X					
Turbulent structure	X	X	X		X	X
Turbulent transport and dissipation mechanisms	X	X	X		X	X
Separation and reattachment	X	X	X		X	X
Wall shear and heat transfer	X	X	X		X	X
Thermochemistry			X		X	X
Alteration mechanisms	X	X	X		X	X
Base flows						
Influence of initial and boundary conditions				X	X	
Shear layer/separated flow structure				X	X	
Rear stagnation point				X	X	
Turbulent transport and dissipative mechanisms				X	X	
Wall shear and heat transfer				X	X	
Ignition				X	X	
Flameholding/thermochemistry				X	X	
Geometrical influences				X	X	
Alteration mechanisms				X	X	
Free shear layers						
Influence of initial and boundary conditions				X	X	
Structure and growth				X	X	
Turbulent transport and dissipative mechanisms				X	X	
Ignition				X	X	
Thermochemistry				X	X	
Alteration mechanisms				X	X	
Wall injection						
Influence of initial and boundary conditions			X		X	
Separation and reattachment			X		X	
Turbulent structure			X		X	
Turbulent transport and dissipative mechanisms			X		X	
Geometric influences			X		X	
Rheological influences			X		X	
Ignition			X		X	
Thermochemistry			X		X	
Alteration mechanisms			X		X	
Inviscid flows						
Influence of initial and boundary conditions	X	X	X	X	X	X
Shock/expansion structure	X	X	X	X	X	X
Turbulence levels and influence	X	X	X	X	X	X
Diffusive mixing				X	X	X
Thermochemistry			X	X	X	X
Alteration mechanisms	X	X	X	X	X	X

et al. have developed a modified Reynolds analogy for an average value of combustor wall shear $\bar{\tau}_w$ that agrees reasonably well with some limited measurements. A plot of both the heat-transfer data and deduced average shear is shown in Fig. 9 vs the heat release for liquid fuels (the correlation for gaseous fuels is slightly higher.[85] From this, it is apparent that there is a fundamental change in the physics governing wall shear in the presence of chemical heat release compared to the accepted mechanisms governing it without heat release. Without heat release, one would expect a slight increase in $\bar{\tau}_w$ (typically 5–10%) as the driving enthalpy increases, not an increase of a factor of two or three.

Although a reasonable body of data exists for combustors, the measurements made are generally limited to mean combustor exit pitot pressure and species sample profiles, wall static pressure distributions, wall heat transfer, and steam calorimetry to measure overall heat release (see Refs. 83 and 84 for a discussion of these state-of-the-art techniques). Thus, one can determine what the global combustion efficiency, total pressure loss, exit species, and, with an appropriate Reynolds analogy, the average value of the skin friction are.

While all of these results are directly applicable to development programs, they provide little, if any, insight into the basic physical and chemical processes governing the combustor. Thus, measurements of the mean and fluctuating flow properties, turbulence levels, chemical reactions, local wall shear, etc., are extremely important. While it is difficult to apply laser diagnostic techniques to these flows, the data gathered would be extremely beneficial in enhancing our understanding of, and ultimately modeling, the turbulent aerothermochemical mechanisms controlling these processes in both the scramjet and DCR.

Exit Nozzle

Other than wall static pressure distributions and some nozzle exit pitot pressure and species sample surveys in free-jet engine tests (see, e.g., Refs. 7 and 33), there are no data on exit nozzles in scramjets or DCR's, primarily because of test facility limitations. While one could, perhaps, apply some of the limited chemical laser data to understanding the fundamental processes governing this expansion process, it should be remembered that the objective in chemical lasers is to maintain chemical nonequilibrium, whereas the opposite is sought in scramjets. Consequently, there is, again, a need to apply nonintrusive and other instrumentation techniques to rapidly expanding supersonic flows with and without chemical reactions, with particular emphasis on the turbulent viscous layer along the wall with large axial and normal pressure gradients, as well as the inviscid core where the Damköhler number may be less than unity for certain species.

Research Opportunities

Based upon the preceding discussions, one can surmise that many of the fundamental issues associated with supersonic combustion ramjet engines are not well documented or thoroughly understood. While there is a reasonable, albeit limited, data base applicable to specific engine development programs in most areas, there is a real need and desire to extend this knowledge down to a more fundamental level so that the basic physics and chemistry governing and/or controlling the supersonic combustion processes can be understood and predicted with a high level of confidence.

Table 1 is an initial attempt to relate the basic scientific areas that are not well documented or understood to the various components of the liquid-fueled scramjet and/or DCR. These comprise the research opportunities and include boundary layers, base flows, free shear layers, wall mass addition (or injection), and inviscid core flows with and without chemical reactions. Under each of these headings are the more detailed issues of interest, with the major common thread throughout being the desire to understand and predict the structure, transport, and dissipative processes in internally ducted supersonic turbulent flows, as well as to alter or control them (see, e.g., Ref. 86).

While it would be desirable to pursue the underlying science governing all of these areas concurrently, practical considerations dictate that they be prioritized to some extent. As such, and from the author's perspective, it is suggested that the order of priority for the major headings be as follows: 1) free shear layers, 2) boundary layers, 3) base flows, 4) wall injection, and 5) "inviscid" flows. Finally, typical initial conditions at each of the engine stations (Fig. 1) are recommended to be $M_0 = 3$–8, $M_4 = 1.5$–4, $M_f = 1$–2, and $M_5 = 1$–2.5 for flight within the tropopause.

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