# Hydrogen-Rich Exhaust Gas Handling Studies

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**Studies are currently under way to evaluate and develop test techniques for safely handling hydrogen exhaust products from hypersonic propulsion systems, as well as future liquid propellant rocket engines, during simulated altitude tests. Results of a study that considered several different handling concepts for air-breathing propulsion testing are presented. The study concluded that exhaust inerting to the oxidizer lean combustion limit is practical for direct-connect testing during normal engine operation, provided that bleed and plant surge-control airflows are negligible; however, diluent costs and exhaust pumping requirements can be high. The study also considered the so-called "hydrogen afterburning" concept. This technique uses injection of oxygen or air into the exhaust system to afterburn the exhaust products in real time in a controlled process. Hydrogen afterburning is attractive because of its potentially wide range of applicability and low secondary flow requirements. An analytical study was begun to investigate the technical details of hydrogen afterburning, which included the mixing of the exhaust products with the injected oxidizer and the potential for ignition and burning of the mixed gases. Conclusions from the analytical study were that adequate mixing can be obtained, and that autoignition and afterburning are feasible.**

## Introduction

SEVERAL proposed hypersonic propulsion systems use hydrogen fuel because hydrogen combustion has the ability to release a very large amount of energy very rapidly. Because of this large potential energy release, the presence of unburned hydrogen outside of the test article is a very significant hazard. In an altitude test facility, the use of rotating machinery to pump exhaust products to atmosphere increases the hazard, which can be catastrophic in terms of personnel safety and facility damage. Even without the tragedy of personal injury, the replacement value of the exhaust machinery is very high. The probability of ignition of a combustible mixture within the exhaust system is high because sparks, hot particles, or even hot exhaust can ignite a combustible mixture. Simplified engine cycle analyses indicate a high probability of unburned hydrogen being present in the test cell and facility exhaust systems during the altitude testing of both hypersonic propulsion systems and $LOX/LH_2$ rocket engines.

The presence of combustible species in test facility ducting is not a new problem. In fact, excess hydrogen has long been a problem in both liquid and solid rocket testing, in both altitude and ground level test facilities.[1-6] Combustible mixtures in altitude tests of rocket engines come from the combination of excess hydrogen in rocket exhausts and air inleakage into the subatmospheric facility exhaust systems. An acceptable procedure in altitude testing has been to inert the rocket exhaust flow to below the oxidizer lean combustion limit (OLCL), approximately 4% $O_2$ by volume at standard temperature and pressures (STP), by adding a flow of nitrogen to the exhaust stream. Because the amount of oxidizer present is low (inleakage is usually very low), and rocket tests are typically of short duration, large amounts of diluent may not be required. For this reason, the nitrogen inerting technique has been a safe, cost-effective solution to handling hazardous exhaust.

Carbon dioxide inerting has also been used. Carbon dioxide has an additional advantage that some of the $CO_2$ dissociates when it is mixed with the hot rocket exhaust gas and releases oxygen, which then recombines with some of the hydrogen in the exhaust to form water. Thus by adding $CO_2$, the flow is inerted, the hydrogen content is reduced, and the gas constant ("R-factor") of the exhaust is lowered simultaneously. A lowered R-factor means increased exhaust density, at given pressures, which reduces exhaust gas pumping power requirements.

In general, the inerting techniques rely on bringing the exhaust flow to either above the upper, or below the lower, flammability limit of the combustible exhaust gases. A reasonably complete literature exists on the flammability limits for hydrogen-air and hydrogen-oxygen-nitrogen mixtures, even at low pressures, and in both dry and wet conditions.[7-21] Thus, the technical basis for hazardous exhaust handling by inerting is well supported by data.

Another theoretical approach to hydrogen-rich exhaust gas handling is real-time afterburning. This process involves controlled combustion of the exhaust flow within the facility ducting during the engine operation. This approach has been studied at the Arnold Engineering Development Center (AEDC) and elsewhere.[1,2,5,6]

This paper discusses the overall combustible exhaust gas handling problem. The paper is divided into three parts. The first section discusses anticipated test requirements. The second section describes a study of the various handling techniques. The last section presents a computational study of some fluid-mechanical details of real-time afterburning.

## Test Requirements

There are anticipated requirements for altitude testing of air-breathing hydrogen-fueled hypersonic propulsion systems

Presented as Paper 88-3019 at the AIAA/ASME/SAE/ASEE 24th Joint Propulsion Conference and Exhibit, Boston, MA, July 11-13, 1988; received Jan. 27, 1989; revision received Sept. 28, 1989. 

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and rocket engines. Some of these tests may be conducted in facilities that were never intended for use with hydrogen fuel. In many facilities there will be a need to conduct the tests such that combustible mixtures are not present in the facility exhaust system. This section discusses several methods to meet this need. The results reported herein are oriented primarily toward the hypersonic engine test requirements; however, except where noted, the work is also applicable to the rocket test requirement.

There are several factors that must be considered before a particular exhaust gas handling technique is selected for engine tests. These factors include facility operational parameters and engine characteristics. The first operational parameter is the test mode. Two different test modes for air-breathing engine testing are the "direct-connect" mode, where virtually all of the supplied air flows through the engine, and the "freejet" mode. In a freejet mode, the engine is embedded within an airstream, and an amount (often large) of the supplied air flows around and over the engine. If the bypass air is captured and mixed with the engine exhaust, a large amount of $O_2$ will be flowing through the exhaust system making the accepted approach of $N_2$ inerting of the exhaust to the OLCL impractical, since very large amounts of diluent would be required.

Another operational parameter of critical importance is test duration. Rocket tests are typically of short duration, on the order of a minute. Exhaust gas handling techniques that are applicable to short duration or intermittent testing may not be practical or cost effective for long duration testing, which can occur in air-breathing propulsion tests.

Engine operational characteristics determine the sources of unburned hydrogen. There are two main hydrogen ($H_2$) sources in testing both air-breathing hypersonic and rocket propulsion systems during normal engine operation. The primary source of unburned hydrogen in the exhaust will result from fuel-rich operation of the propulsion systems. Both hypersonic air breathers and rockets are likely to operate fuel rich for performance requirements. For hypersonic engine ground tests in the Mach 0 to 4 range (the range examined by the authors), equivalence ratios (ER) between 1 and 1.6 were considered. The other potential exhaust gas $H_2$ source is ejected, nonregenerative cooling flow. In cases where the amount of cryogenic $H_2$ required for cooling is larger than needed in the combustor (regenerative amount), excess cooling fuel is dumped overboard. Hydrogen coolant dumps are not anticipated for the Mach 0 to 4 flight regime because airframe and engine cooling requirements are relatively low.

Handling techniques must also be evaluated for abnormal operations or system failures of either the facility or the engine. For both hypersonics and rockets, one type of failure is a fuel line leak or rupture. Another failure, which can occur during the development and qualification testing of hypersonic engines, is a combustor no-flame situation. In a no-flame situation, fuel is added to the engine airstream, but combustion either does not take place or the flame blows out. These kinds of failures put very large amounts of unburned fuel in the facility exhaust flow, and obviously, there is also a large amount of $O_2$ flowing through the system, in both freejet and direct-connect test modes. Diluent requirements for engine combustor failures will be drastically different than for cases of normal engine operation.

Combinations of the factors outlined above show that there is wide variety in the types of problems that can be encountered in such testing. The utility and practicality of using inerting to handle the exhaust hazard in such tests needed to be examined and compared to other possible exhaust gas handling techniques. The study was conducted for this purpose.

## Study of Hazardous Exhaust Handling

The hazardous exhaust handling study was based on assumed hypersonic propulsion system test requirements discussed in the previous section, which were ER up to 1.6, Mach number from 0 to 4, and altitude from ground level to 70,000 ft. Furthermore, a generic flight trajectory was assumed for a vehicle, which permitted the relative flow rates of fuel and air to be defined. Several exhaust gas handling techniques were then evaluated in terms of the propulsion system test requirements. All combinations of the aforementioned factors were considered, except for the fuel line rupture. This failure situation was not included because of the difficulty in quantifying the amount of fuel spilled. The no-flame combustion condition was considered. The conclusions that were obtained from the study are presented below.

### Inerting Techniques

The study first considered the application of inerting techniques to various test scenarios. For the purpose of the study, the fuel lean combustion limit (FLCL) for hydrogen mixtures was considered to be 4% $H_2$, by volume, under standard conditions of temperature and pressure. The OLCL was taken as 4% $O_2$, by volume, the minimum required to support combustion, also at STP.

#### *Direct-Connect Testing*

First, for a direct-connect test of an engine operating normally, the study found that nitrogen inerting to the OLCL is feasible for facilities with small air inleakage. The exhaust flow from a direct-connect test during normal engine operation is inert (no oxidizer) until there is an air inflow produced by an inleak or inbleed. Examples of air inbleed include test cell cooling and turbomachinery surge control flows. Diluent added to the exhaust need only account for the inbleed or leakage air inflow, which is normally small. However, the total amount of diluent required (and thus the diluent costs) may become significant for a long duration test. Further, some facilities may use large quantities of air inbleed for surge control or test cell pressure control. This facility characteristic defeats the technique of using nitrogen inerting to the OLCL for hazardous exhaust handling.

To inert to the FLCL during normal engine operation requires a larger amount of diluent. The amount in this case depends upon the specific engine operating characteristics (equivalence ratio and coolant dump). The value is also a function of the amount of water vapor that is condensed out of the exhaust flow by any exhaust dehumidification/cooling processes. The total quantity of diluent required is large for a long duration test. If nitrogen is used as the diluent, the costs may become prohibitive. The diluent flow rate, along with the required test flow, may exceed the exhaust plant capacity, depending upon the test requirement (engine size, equivalence ratio, etc.). Inerting to the FLCL, rather than the OLCL, has the advantage that air can be used as the diluent. In this case caution must be used because the flow will pass through an oxidizer lean state at the engine exit to a fuel lean state (inert) somewhere downstream of the diluent addition point(s). The composition of the flow will pass through a combustible mixture. Combustion may occur in this transition zone, and thus uncontrolled afterburning might occur.

As previously stated, carbon dioxide was used as a diluent for handling hazardous exhaust in rocket testing. However, for hypersonics testing, an analysis has shown that the exhaust density may not be a critical issue for direct-connect testing, at least for the lower Mach number regimes. Further, because the hypersonic engines produce exhaust gases at lower total temperatures than rocket exhaust, resulting in less $CO_2$ dissociation, diluent mass flow rates for $CO_2$ inerting are higher than for $N_2$ inerting. Therefore, there is no advantage to using $CO_2$ inerting for hypersonic testing.

For a direct-connect test experiencing a failure mode, such as a combustor no-flame situation, the amount of diluent required to inert the flow becomes very large. To inert to the OLCL, diluent flows of 3–4 times the test cell inlet airflow are required based on hypothetical test requirements. To inert to the FLCL, diluent requirements may be an order of magnitude above the cell inlet airflow. For a full-scale test article, the diluent flows could easily exceed plant pumping capabilities. In the event of a no-flame condition, the test would have to be shut down very quickly (response time on the order of milliseconds) and the system purged with large air or $N_2$ flows dumped into the exhaust flow. Thus, rapidly responding hydrogen detectors and flame detectors will be necessary for test control and monitoring.

*Freejet Testing*

In a freejet test, during normal operation, the situation is different. The flow leaving the test cell will likely be fuel lean overall because of the large airflow spilled around the test article. Depending on engine equivalence ratio, freejet nozzle to engine mass flow ratio, and exhaust dehumidification, the overall exhaust flow could be below the FLCL. Mixing of an engine exhaust containing free hydrogen and the freejet bypass stream should cause localized combustible regions. If the overall flow is not below the FLCL, then the flow can be inerted to the FLCL with additional injected air. The amount of air required is a function of the engine and facility characteristics.

Theoretically, the flow could be inerted to the OLCL using nitrogen. For the hypothetical test requirement, the $N_2$ requirement could vary from 1 to 4 times the cell inlet airflow, depending on engine ER, freejet mass flow spill ratio, and dehumidification. The diluent costs will therefore be prohibitive. Thus, inerting to the OLCL for freejet testing is not as attractive as air inerting to the FLCL.

For a freejet test during a no-flame situation, the required diluent flow rates are again very large. Given the assumed test requirement, to inert to the OLCL using nitrogen, the diluent requirement could be well over 4 times the cell inlet airflow. To air inert to the FLCL, the diluent requirement is approximately 3 times the cell inlet airflow. The total facility mass flow (engine flow + bypass flow + diluent flow) would most likely exceed the facility capability, except in tests of very low flow rate engines. Again, in the event of a detected no-flame situation, the test would have to be shut down quickly and the exhaust system purged.

*Alternative Handling Technique*

Because of the limitations cited above, cost-effective alternative methods were sought to handle combustible exhaust flows safely. One concept that has shown particular promise is real-time afterburning. This approach has been used for some rocket tests.[1,2] Conceptually, the real-time afterburning technique is based on injection and mixing of an oxidizer with the hydrogen-rich exhaust gas produced by the propulsion system, so that the exhaust gas hydrogen and injected oxidizer react in a controlled fashion, reducing the hydrogen concentration to below the FLCL.

For a direct-connect test utilizing afterburning, the relative amount of injected oxidizer required to burn the excess hydrogen is small. Air would be an ideal oxidizer for hypersonics testing, because the cost savings over an oxygen injection system would be significant. On the other hand, required oxidizer flow rates may be much higher for tests involving large liquid rocket engines. For these tests, oxygen should be used as the oxidizer to reduce the overall flow rate being pumped by the exhaust plant. Afterburning has an advantage that the oxidizer flow rate required would not change in the event of a flameout since the required oxygen would already be present in the flow. For an air-breathing engine in a freejet test with a typical value of freejet-to-engine mass flow ratio, it is likely that all the oxygen required to afterburn would already be available in the bypass stream. The major issues for freejet testing would be how to mix the engine exhaust and bypass streams and to initiate and sustain the combustion process.

The hydrogen afterburning technique was considered to have valid potential because of its wide range of applicability and low operational costs. However, there were some significant uncertainties in the afterburning technology that required examination. These include 1) determinating the best method to inject and/or mix the oxidant with the exhaust; 2) ensuring that autoignition and afterburning do occur for all conditions of the engine and facility operational modes; and 3) obtaining adequate system pressure control and pressure recovery. Of special concern is the fact that the ignition and afterburning would have to occur at low static pressures, less than half an atmosphere. Flammability limits[7-21] and ignition delay times[22-32] are both affected by low static pressures. Thus, the afterburning technology will have to overcome both reaction-limited and mixing-limited combustion processes.

## Analytical Model of Afterburning

### Definition of the Injection and Mixing Configuration

Analytical studies of real-time hydrogen afterburning were initiated because of the potential benefits of this technique for both air-breathing and rocket propulsion system testing. Previous feasibility studies conducted on rocket applications of afterburning[5,6] recognized that both analytical and subscale experimental studies were required to address the safety and practicality of the afterburning technique. The previous analytical studies were based on both equilibrium and finite rate chemistry, in both one- and two-dimensional flow models. The two-dimensional analytical studies were based on nearly constant-pressure, parallel mixing conditions. It was recog-

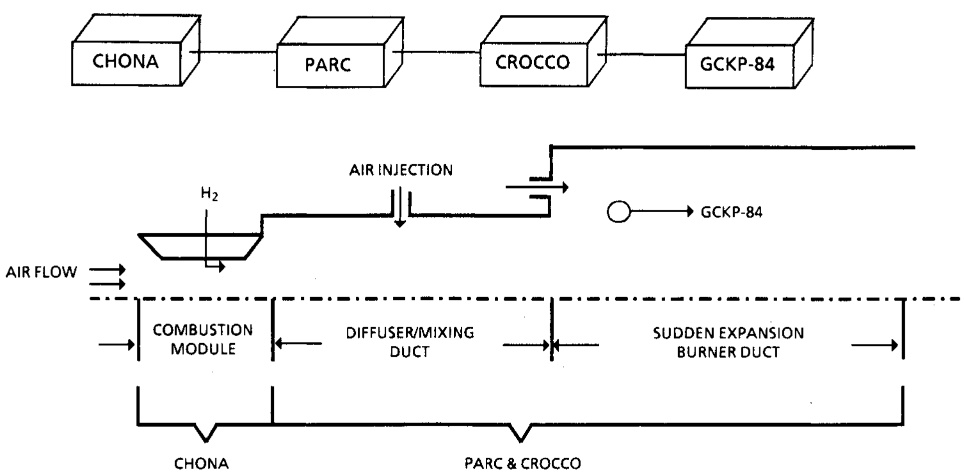



Fig. 1 Real-time $H_2$ afterburning analytical model.

nized that the parallel mixing case would not permit a practical afterburning system because lengths required for complete mixing were of the order of several hundred engine exhaust diameters. Thus, it was recognized that transverse jet injection of the oxidizer was required for adequate mixing with the exhaust stream.

Transverse injection, the injection of a fluid jet perpendicular to a moving stream, has been recognized for a long time as a method to increase the mixing rate of the jet and the surrounding flow, compared to coflowing or parallel injection.[33-41] However, in the present afterburning application, the situation is reversed, compared to previous studies, because the injected jets are air or $O_2$, and the main moving stream is a fuel-rich exhaust flow.

It is also important to note that supersonic combustors may require fuel injection from wall steps or into sudden-expansion geometries for flameholding and steady combustion processes.[42-45] Thus, successful supersonic combustors will probably require both transverse injection and parallel injection, together with wall steps and/or sudden-expansion geometries for adequate overall mixing, combustion, and flameholding. Moreover, of key importance is the distance between the transverse injection jets and the wall steps or sudden-expansion locations. Experiments have shown[42,45] that injection upstream of the steps or expansions resulted in best combustor performance. Therefore, the supersonic diffuser-burner design that was chosen for the present study incorporated both transverse and parallel jet injection, a wall step, and a significant increase in cross-sectional area ratio, or sudden expansion. Figure 1 shows a schematic of the supersonic diffuser-burner cross section with the afterburning oxidant jet injection locations indicated.

To study analytically the mixing and flow in the ducting system indicated schematically in Fig. 1, a Navier-Stokes-based computational fluid dynamic scheme was required. In particular, a fully elliptic, finite-rate chemistry, shock-capturing, fully viscous (including turbulence model) code was required, capable of predicting both transient and steady-state, three-dimensional flow in ducts and diffusers. At the time of the study, such a computer code was not available,[46,47] although, recently, codes with the necessary capabilities have been reported.[48,49] Therefore, the approach taken by the authors was to assemble a component-type, overall analytical approach based on three available computer codes.

### Overall Analytical Model

Figure 1 depicts the overall analytical approach that was adopted. Four independent computational steps were used to investigate afterburning flowfields. The four steps used three codes, CHONA, PARC, and GCKP-84. CHONA[50] is a one-dimensional, equilibrium chemistry code, which was used to compute engine exhaust gas flow conditions at a given flight condition, based upon specified engine operational fuel/air ratio (or ER) and an assumed inlet performance. The PARC code[51,52] is a two-dimensional or three-dimensional, perfect gas, fully elliptic, fully viscous, Navier-Stokes code with shock-capturing capability. PARC was used to predict mixing of the engine exhaust gas with the injected oxidizer within the specified duct geometry (see Fig. 1). The resulting flowfields obtained with PARC include temperature, but not gamma or molecular weight effects. A subprogram called CROCCO was written to convert the PARC code total temperature results into local species concentrations, using predicted local total temperature, referenced to the exhaust gas and injected air total temperatures; that is, CROCCO was used to apply the Crocco analogy. The Crocco analogy assumes that turbulent diffusion of total temperature is the same as turbulent diffusion of species, i.e., that unity turbulent Lewis number conditions apply. Thus, total temperature distributions imply conserved atomic mass fractions of the species from the exhaust (or, alternately, the oxidant). Thus, the total temperature values were used to specify local exhaust gas concentration. The exhaust gas concentrations and the oxidant concentrations were then input either to CHONA, to predict local equilibrium chemistry, or they were input to GCKP-84. The GCKP-84 code[53] is a one-dimensional chemical kinetics code, which was used to predict local ignition characteristics of selected samples of the mixed flow, based upon the PARC and CROCCO results.

In this overall analytical approach, mixing and combustion processes were partially decoupled, because of the perfect gas limitations of the (then current version of the) PARC Navier-Stokes computer code.

To illustrate the application of the analytical models, consider a case that represents the highest fuel flow rate for a hypothetical hypersonic vehicle. For this case, the inputs to the CHONA code were the combustor inlet conditions corresponding to the highest mass flow ingested along a given flight trajectory. The CHONA code computed the ideal, one-dimensional exhaust gas conditions. The exhaust conditions were then used to specify inlet conditions to the PARC Navier-Stokes model of the supersonic diffuser-burner exhaust ducting. The PARC code computed the mixing of the exhaust gas and injected air for afterburning within the supersonic diffuser and burner sections. The air was injected at two locations, either from the transverse injection ports in the supersonic diffuser, or as a parallel jet issuing from the wall-step location at the sudden expansion. Except for total pressure and total temperature, thermodynamic properties of the exhaust flow and the injected air were identical in the PARC-based mixing analysis. This approach should result in an acceptably realistic

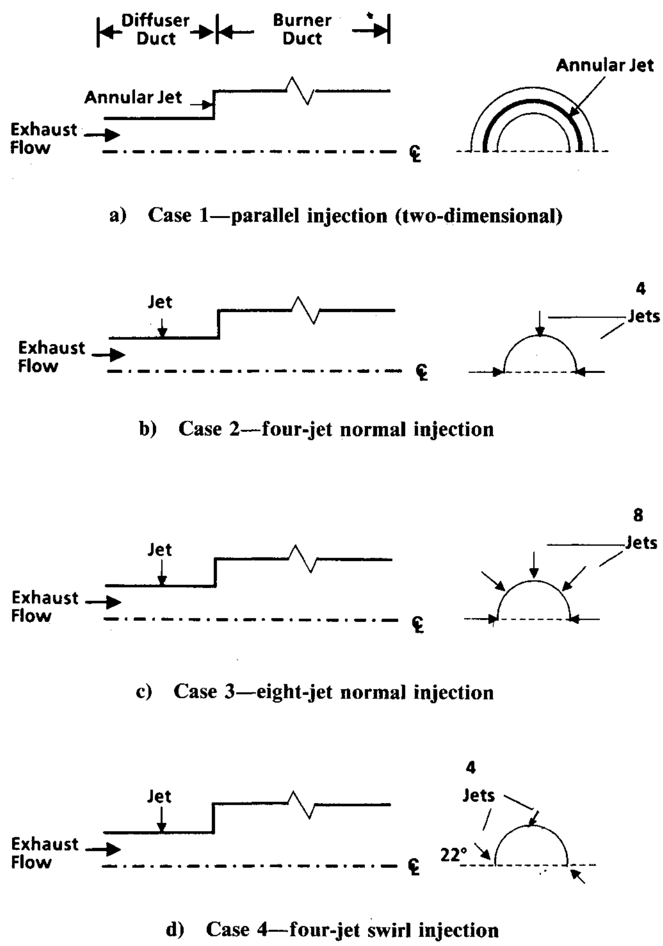



Fig. 2 Injection configurations used in analytical study.

computed mixing process, compared to actual behavior; however, gamma-dependent flow features, such as shock or expansion structures might be slightly less accurate.

Once the mixing flowfield was obtained by PARC, the species concentrations at each grid point were obtained as follows. Local mixture composition was computed using the CROCCO code that was based on the CROCCO relation

$$C_{\text{exhaust}} = (T_t - T_{t_{\text{jet}}}) / (T_{t_{\text{exhaust}}} - T_{t_{\text{jet}}})$$

where $C_{\text{exhaust}}$ is the local exhaust gas mass fraction, $T_t$ is the local mixture total temperature from PARC, and $T_{t_{\text{exhaust}}}$ and $T_{t_{\text{jet}}}$ are the specified engine exhaust and air-jet total temperatures, respectively. Then, $C_{\text{jet}} = 1 - C_{\text{exhaust}}$, where $C_{\text{jet}}$ is the total air mass fraction. Local flow thermodynamic properties could then be determined from $C_{\text{jet}}$ and $C_{\text{exhaust}}$ mass fractions at each location in the mixing flowfield. Basically, these "local flow" conditions (velocity, pressure, temperature, and species composition) were used only by serving as input to the GCKP-84 one-dimensional chemical kinetics model to investigate ignition and flame processes.

### Injection Configurations

For the real-time afterburning analysis, four basic air injection configurations were examined. All four configurations modeled the flow through the same supersonic diffuser-burner geometry. The modeled geometry and injection configurations are shown in Fig. 2. The duct geometry begins at the exit plane of a hypersonic engine with an axisymmetric supersonic diffuser section. The modeled diffuser length is 5 diam. At the diffuser exit, there is a sudden expansion to a constant area burner section. The burner duct has a length of 10 diam. The ratio of the diameters of the burner duct to diffuser duct is 2:1. This diameter ratio was chosen to avoid thermal choking under conditions of maximum heat release where a no-flame condition in the combustor leads to a requirement that all fuel be burned in the diffuser-burner, and also, the 2:1 diam ratio (4:1 area ratio) should provide for a satisfactory pressure recovery.[54,55]

#### *Parallel Injection from Stepped Wall*

The first case simulated parallel injection of an annular jet from the base of the sudden-expansion step, or stepped wall. Thus this case is a two-dimensional axisymmetric flow with injection through an annular ring. An axisymmetric version of the PARC code was used for the computation.

#### *Transverse Injection*

The other three cases simulated transverse injection at the midpoint of the diffuser duct. The first two of these cases used perpendicular injection with either four or eight discrete jets equally spaced around the duct circumference. The fourth case was a four-jet swirl-injection case. Swirl injection added a circumferential velocity component to the air jets. The injection angle was set at approximately 22 deg from the normal in a cross-sectional plane. The three three-dimensional cases used the same total mass flow for the injected air. The mass flow was 110% of the airflow required to make an overall stoichiometric fuel-air ratio, i.e., when fully mixed with the exhaust gas. The parallel injection case used somewhat higher mass flow than other cases for computational convenience, because of grid limitations in modeling the axisymmetric ring.

The normal injection cases used numerical mesh models that were wedge-shaped, three-dimensional geometries. "Periodic" and symmetry boundaries were defined and used to reduce the extent of the computational grid required in the circumferential direction. The four-jet normal injection model required a 45-deg symmetry segment, while the eight-jet normal injection model required a 22.5-deg symmetry segment. The four-jet swirl model was based on a 90-deg segment with periodic boundary conditions. About 22,000 grid nodes were used for the parallel injection computational model, while the three three-dimensional models used approximately 240,000 grid nodes each.

### Results and Discussion of Mixing Analysis

Results from the four cases are given in Figs. 3–6. These figures show computed concentration contours. Concentration contours are shown in the longitudinal plane and, for the three-dimensional cases, in cross-sectional planes. A contour level of 1 represents species concentrations identical to the engine exhaust flow, while a contour level of 0.838, indicated by a dashed line in the figures, represents a stoichiometric hydrogen-air mixture based upon the specified exhaust and jet total temperatures and mass flows. This contour represents the desired mixture (the overall ER was slightly less than one based on exhaust gas-to-injected-air mass flow rates).

#### *Parallel Injection*

Results from the parallel injection case are given in Fig. 3. The concentration contours show that there is little mixing taking place within the first 2.5 diam of the sudden-expansion burner duct inlet. From 2.5 to 10 diam (not shown), there is negligible change in the concentration profiles. After 10 diam there is still almost no oxygen present within the inner half radius. Conversely, almost no hydrogen has reached the duct wall. There is only a narrow ring where sufficient mixing has occurred to support combustion. Thus, for the case investigated, the parallel injection and mixing was inadequate.

#### *Four-Jet Normal Injection*

Representative results from the four-jet normal injection mixing model are given in Fig. 4. Concentration contours are shown in four different planes. Figures 4a and 4b show the contours at different duct cross sections. Figure 4a shows the contours in the plane of injection in the supersonic diffuser, while Fig. 4b shows the contours near the plane of the sudden expansion (burner inlet). Figures 4c and 4d show the contours in longitudinal planes of the supersonic diffuser. The contours of Fig. 4c are in the jet symmetry plane, the midplane of the injection ports, while Fig. 4d shows the contours in a plane located halfway between the jets, i.e., between injection ports.

The mixing performance of the four-jet normal injection was much superior to the parallel injection scheme. The in-

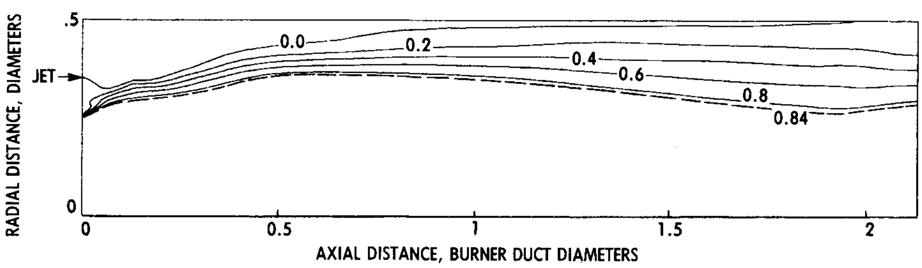



**Fig. 3 Concentration contours for parallel injection case.**

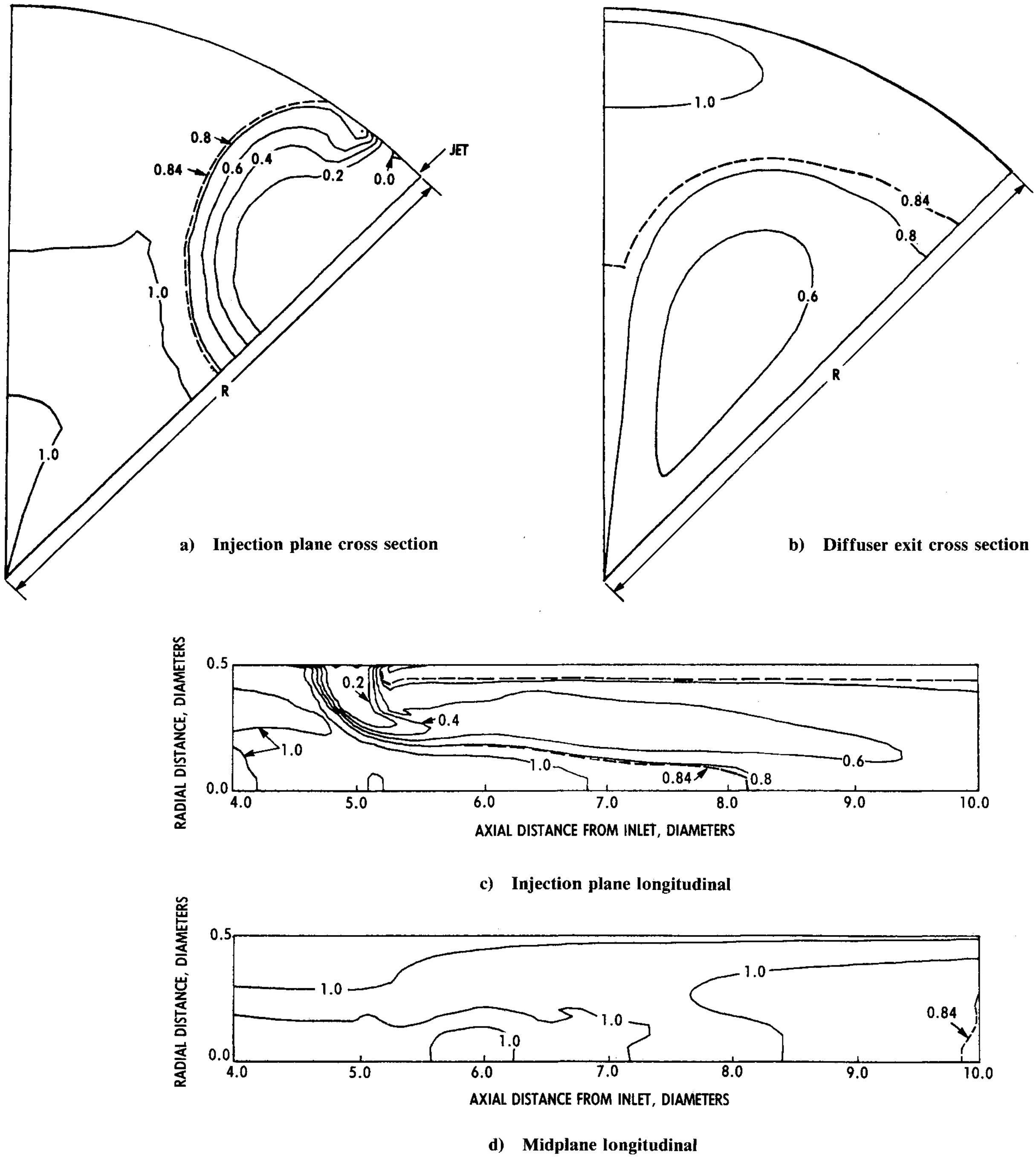



**Fig. 4 Concentration contours for four-jet normal injection case.**

fluence of the normal jet is clearly evident in Fig. 4a. From the cross-sectional contours, it can be seen that the engine exhaust (contour levels near 1.0) has moved toward the duct wall with the injection flow (contour near zero) nearer the centerline. In the longitudinal plane of injection, the flow is fairly well mixed. However, as Fig. 4d shows in the plane halfway between the jets, little mixing has occurred. Thus, it was assumed that better overall mixing could be obtained from using more jets around the circumference. An eight-jet case was therefore defined, modeled with a new mesh, and computed.

*Eight-Jet Normal Injection*

The eight-jet configuration is similar to that of the four-jet case with the major differences being that the mass flow per injection port was halved, as was the circumferential distance between injection port centerlines. The total mass flow rate was the same as the four-jet case. Representative results from the eight-jet normal injection mixing model are shown in Fig. 5. The results show that mixing in the regions between the jets was improved over the four-jet case. This improvement came at a sacrifice of radial penetration, however, since the same total pressure was used in each case, whereas the jet diameters were reduced by a factor of $1/\sqrt{2}$.

The normal injection results can be understood in terms of aerodynamic blockage. In the four-jet case, the blockage of the injection jets caused the exhaust flow to move radially outward in the midjet plane to the duct wall. In the eight-jet case, the blockage has increased circumferentially while weakening in strength (penetration to the centerline). As a result, the flow is well mixed out near the walls, but relatively unmixed near the centerline of the supersonic diffuser.

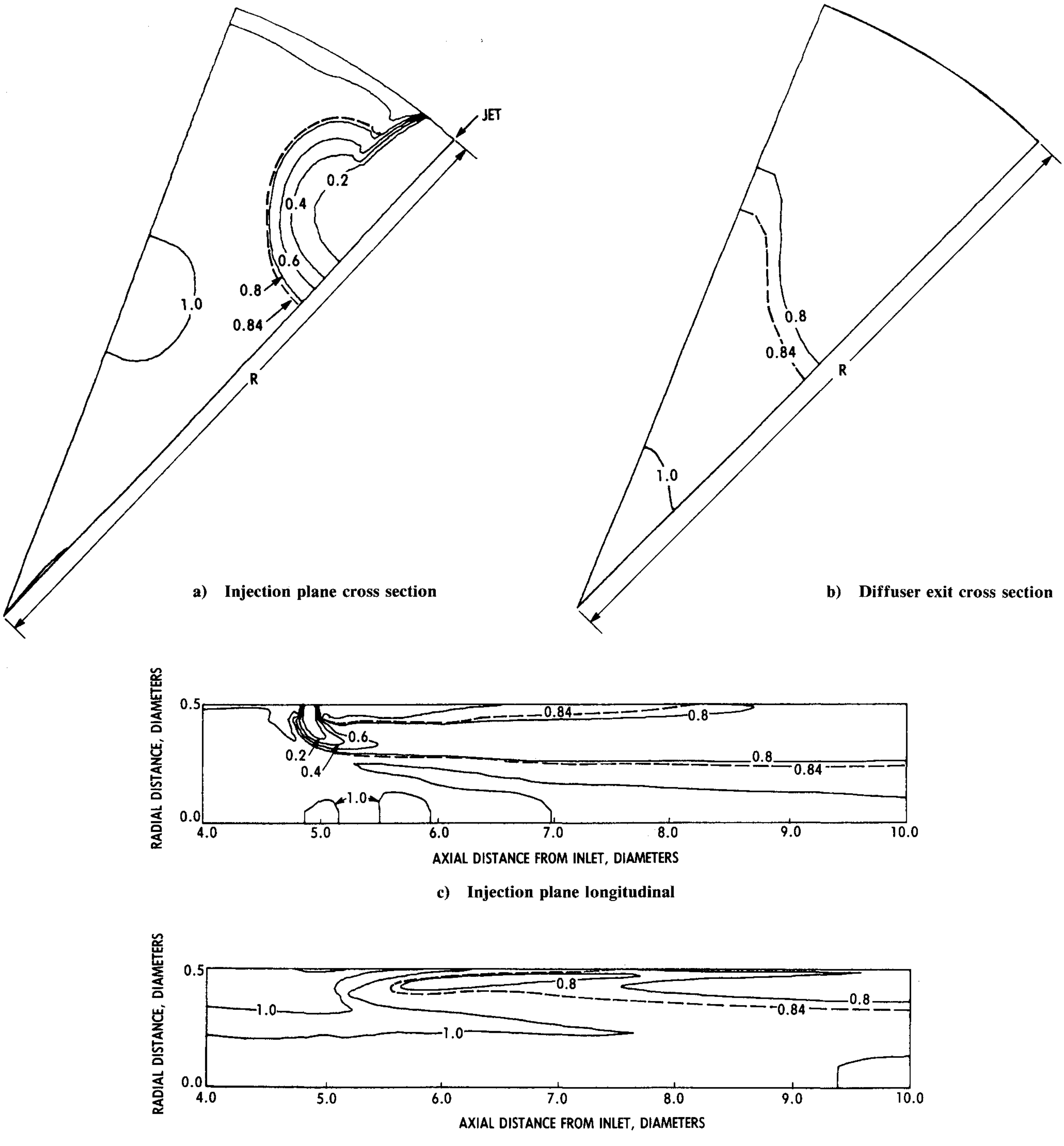
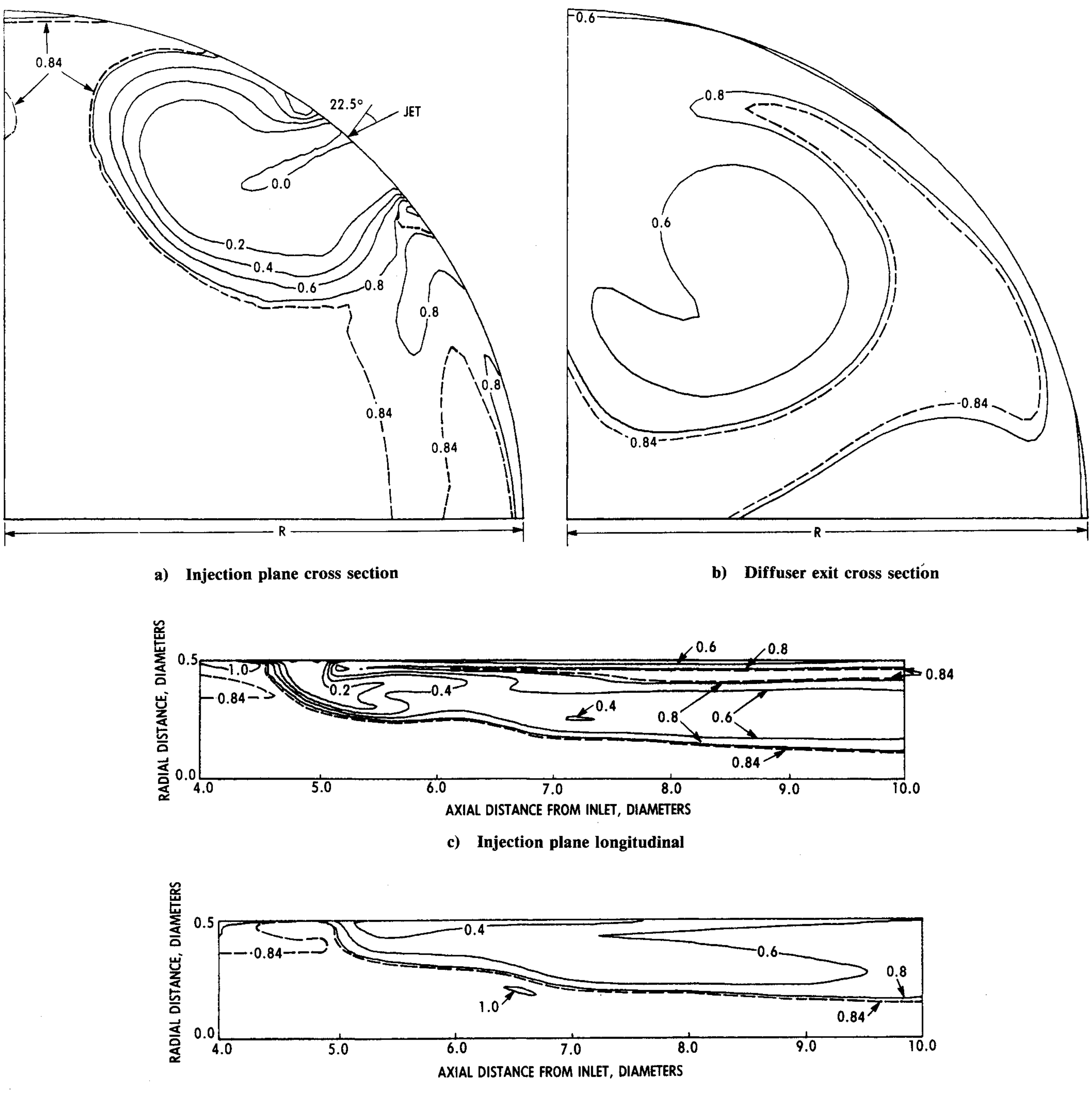



**Fig. 5 Concentration contours for eight-jet normal injection case.**

*Four-Jet Swirl Injection*

To again try to improve the overall mixing of the exhaust gas with the injected air, a four-jet transverse injection scheme was defined using angled jets that would induce a swirl or circumferential flow component.

Representative results from the four-jet swirl injection mixing model are shown in Fig. 6. The cross-sectional contours of Figs. 6a and 6b show that the radial penetration of the swirl case is superior to the eight-jet normal injection case, but that the injected air from the four-jet swirl case does not penetrate as deeply into the exhaust stream as the normal injection case. However, the longitudinal plots of Figs. 6c and 6d illustrate the mixing in the between-jet planes is greater for the four-jet swirl case compared to the four-jet normal injection case. When compared to the eight-jet normal injection case, the swirl case is more nonuniform out near the walls, implying poorer circumferential mixing, than the eight-jet case. Additional study is required in order to establish the number of jets, their mass flows, injection velocities, jet diameters, and angles, for the optimum air injection technique in the supersonic burner. However, a result of this part of the study was the observation that, if satisfactory mixing occurred in the supersonic diffuser, it was enhanced in the burner section of the ducting. Thus, the quality of satisfactory mixing was established by the performance of the jet mixing process in the supersonic diffuser.




a) **Injection plane cross section**

b) **Diffuser exit cross section**

c) **Injection plane longitudinal**

d) **Midplane longitudinal**

**Fig. 6 Concentration contours for four-jet swirl injection case.**

**Results and Discussion of Ignition and Burning Study**

A limited theoretical study was conducted to investigate whether exhaust-air mixtures would autoignite at the pressures and temperatures expected in the mixing zones. Ignition and flammability of the mixtures at low pressures is a major concern with the afterburning test technique. The literature on the ignition and combustion of hydrogen fuel injected into an airstream, the situation in scramjet combustors, has been studied in depth.[22-32,36,56-58] These kinds of studies were typically of the autoignition of hydrogen, at various temperatures, being injected into relatively low static temperature, supersonic airstreams. As pointed out earlier, the situation in the afterburning exhaust case is reversed. Here, cold air or $O_2$ is injected into a hot, supersonic exhaust stream containing excess (unburnt) hydrogen. The significant advantage for autoignition in this case is the relatively hot, approximately 3700°R, exhaust gas static temperature.[59] The disadvantage is that the static pressures in the ducting will be of the order of 1-1.5 psia ($\sim$0.1 atm), whereas in a scramjet combustor, the static pressures are expected to be of the order of 5-8 psia ($\sim$0.5 atm). Thus, a numerical study of the autoignition properties of selected "samples" of the mixed exhaust-airflows, which were computed using the PARC analysis, was conducted to see if any unusual or unexpected behavior (failure to autoignite) would be predicted for mixtures expected to autoignite. We clearly recognize, and have stated earlier, that the only acceptable verification of the afterburning technique would be by experiment. Further, for fail-safe operation of full-scale facilities, for under both normal and no-flame engine operating conditions, pilot torches and other active ignition sources would be required.

To conduct the analytical study of autoignition, the GCKP-84[53] was used to examine autoignition, ignition delay, and the combustion efficiency (in terms of free hydrogen consumption

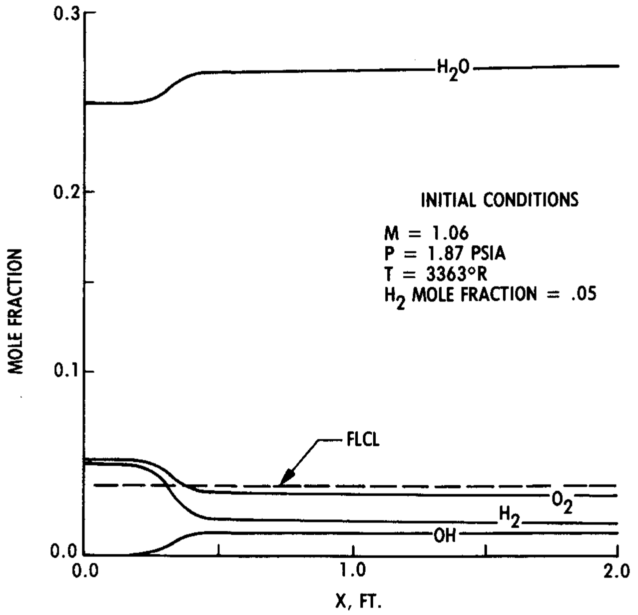



**Fig. 7 GCKP-84 ignition investigation results.**

by afterburning). GCKP-84 was used instead of the various formulas for ignition reported in the literature (e.g., Refs. 22, 24, 25, and 36) because this program allowed the study to be based on detailed chemical kinetic reaction mechanisms, and, further, it would permit various computational options to be exercised such as including heat loss from the mixture, or treating the reaction process as either a plug-flow or a stirred-reactor model of the localized combustion. And, as stated, the program permitted the degree of completeness of the $H_2$ combustion to be obtained, which the ignition formulas would not provide.

To perform the study, three sets of reaction steps were used, having 17, 49, and 60 individual reactions, respectively. The reason for conducting the investigation with three sets is that it has been found[60,61] that inclusion of some of the trace radical species, such as $H_2O_2$, and $HO_2$ provide for better predictions of ignition delay times and flame propagation for both fuel-lean and fuel-rich combustion zones. Further, species such as CO, $CO_2$, and $NO_x$ also play a role at higher temperatures. Therefore, the three reaction sets were constructed to test the sensitivity of the predicted autoignition to the presence or absence of the given species.

The 17-step reaction mechanism included only reactions between $H_2$ and $O_2$ and is shown in Table 1. Species such as $CO_2$ and $N_2$ are present only as chemically inert third-body species, denoted by the M in the reaction mechanism. The 49- and 60-step reaction mechanisms are not shown. The 49-step reaction mechanism includes the nitrogen-oxygen-hydrogen reactions with CO and $CO_2$ present only as chemically inert third-body species. The 60-step reaction C-H-O-N system included the C, CO, and $CO_2$ species in the ignition sensitivity study.

Figure 7 shows a typical result of the GCKP-84 code ignition calculations. Results from the 17-step reaction mechanism are shown. The 49- and 60-step reaction mechanism results are very similar, with slightly reduced ignition delay times. A plug flow analysis based on finite rate chemistry was applied to an initial mixture of exhaust gas and air having a Mach number (1.06), temperature (3363°R), pressure (1.87 psia), and hydrogen mole fraction (0.05) obtained from the PARC and CROCCO codes at a point in the mixing zone flowfield near a jet injector, a region where afterburning should initiate,[56-58] at least in the scramjet combustor process. For the afterburning flow, these regions were hot and contained unburned hydrogen at the inner edges of the mixing zones. The one-dimensional finite rate chemical process was computed under constant pressure, and the results are presented in terms of spatial distance from the start of the process.

All three reaction mechanisms (17, 49, and 60 reactions) indicated that afterburning was initiated and that about 65% of the free hydrogen in the initial mixture burned off within a few feet downstream of ignition. This leaves the resulting afterburned mixture well below the fuel-lean combustion limit. Equilibrium calculations indicate that the maximum expected temperature rise of the mixtures resulting from the afterburning process would be on the order of 500°R. Thus, the additional burden placed on the facility spray cooling system would be small for implementation of the afterburning method in existing test cells. The temperature rise in the diffuser burner will be much higher when the total hydrogen propellant flow must be afterburned there in the event of an engine no-flame condition, and the facilities must be designed for this thermal loading. For the no-flame case, the static temperature of the exhaust flow will be low, implying that the facility cannot depend on autoignition, as stated previously. Thus, as a recommendation, the operational facility should have torches or other pilot flame ignition sources in operation at all times.

## Conclusions

### Exhaust Handling Study

The major conclusions obtained from the hydrogen hazard handling study are as follows.

1) For direct-connect testing of a hypersonic engine during normal operation, nitrogen inerting to the oxidizer lean combustion limit is feasible in facilities with low air inleakage or inbleed. Long duration test costs may be high, however.

2) For freejet testing of a hypersonic engine during normal operation, air inerting to the fuel-lean combustion limit is feasible for facilities with adequate flow capacity. Mixing of the exhaust flow and the bypass air and/or diluent air is an unresolved issue, and some afterburning should be expected in the exhaust duct.

3) For either direct-connect or freejet testing of a hypersonic engine during a no-flame condition, no inerting technique will allow continued operation of the system.

4) The real-time afterburning technique has the potential to handle both normal and no-flame test conditions and is applicable to a wide range of test requirements and facilities, because of the low injected airflow requirement. Mixing and low pressure ignition and combustion are major issues. A supersonic diffuser-burner design, such as shown in Fig. 1, based on transverse jet injection is recommended for the direct-connect test mode. This same design may also apply to the freejet test mode, but adequate mixing must be assured.

**Table 1 GCKP-84 17-step reaction mechanism ignition model**

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| Reaction no. | 1 | O | + $H_2$ | | = OH | + H | |
| | 2 | O | + $H_2O$ | | = OH | + OH | |
| | 3 | H | + $O_2$ | | = OH | + O | |
| | 4 | $H_2$ | + OH | | = $H_2O$ | + H | |
| | 5 | OH | + OH | | = O | + $H_2O$ | |
| | 6 | H | + $O_2$ | + M | = $HO_2$ | + M | |
| | 7 | O | + O | + M | = $O_2$ | + M | |
| | 8 | H | + H | + M | = $H_2$ | + M | |
| | 9 | H | + OH | + M | = $H_2O$ | + M | |
| | 10 | $H_2$ | + $HO_2$ | | = $H_2O$ | + OH | |
| | 11 | M | + $H_2O_2$ | | = OH | + OH | |
| | 12 | $H_2$ | + $O_2$ | | = OH | + OH | |
| | 13 | O | + $HO_2$ | | = OH | + $O_2$ | |
| | 14 | OH | + $HO_2$ | | = $H_2O$ | + $O_2$ | |
| | 15 | $2HO_2$ | + M | | = $H_2O_2$ | + $O_2$ | + M |
| | 16 | H | + $HO_2$ | | = OH | + OH | |
| | 17 | $H_2O_2$ | + H | | = OH | + $H_2O$ | |

**Analytical Study**

Major conclusions of the present analytical study of the hydrogen afterburning exhaust gas handling concept are as follows.

1) Process validation is needed; specifically, this means the analytical model must be validated by experiment, for both the direct-connect test mode (transverse injection scheme) and for the freejet test mode.

2) Parallel oxidizer injection in the burner section resulted in inadequate mixing.

3) Four-jet normal injection in the supersonic diffuser provided improved mixing over the parallel injection configuration. Mixing performance in the longitudinal midplane, between jets, is low.

4) The eight-jet injection in the supersonic diffuser produces improved mixing circumferentially, but inferior radial penetration as compared to the four-jet case due to the increased blockage. Increasing air-jet injection pressure may help radial penetration, but this should be studied carefully to avoid too much blockage in the supersonic diffuser.

5) Mixing can be enhanced by adding a swirl velocity component to the injection in the four-jet case. This configuration achieves good circumferential mixing with moderate sacrifice in radial penetration.

6) Comparison of the injection configurations suggests that a configuration can be found that will provide adequate mixing, radially and circumferentially, within the supersonic diffuser. Further, if acceptable mixedness can be achieved in the supersonic diffuser, it will be enhanced by the sudden-expansion burner section downstream.

7) Ignition and afterburning are predicted to occur at the expected conditions if the flows are sufficiently mixed. The distances required for combustion to less than the fuel-lean combustion limit of the mixtures are short, of the order of a few feet at worst.

8) The exhaust gas temperature rise due to afterburning will be relatively low (+500°R maximum) under normal engine operating conditions. For afterburning in a no-flame condition, the temperature rise in the burner will be high but will not exceed the maximum anticipated for configurations that must handle the possible modes of operation.

9) As a recommendation, provisions should be made to ensure ignition and afterburning for all conditions since an engine no-flame condition will produce an exhaust flow that may not autoignite. Therefore, the operational facility will require pilot flames, torches, and perhaps physical flameholders in related locations of the burner, even though the burner itself is a sudden-expansion-type design.

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