11th Australasian Fluid Mechanics Conference
University of Tasmania, Hobart, Australia
14-18 December 1992

AXISYMMETRIC SCRAMJET THRUST PRODUCTION

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ABSTRACT

Experimental and computational results for thrust production in an axisymmetric scramjet with wall fuel injection are reported. Test conditions were established in a reflected shock tunnel which were energy equivalent to flight at Mach 17. Thrust was determined by integrating static pressure measurements along a conical nozzle. Computational modelling of the flowfield was done with a Parabolic-Navier-Stokes code and good agreement was found between measured and predicted combustor pressure distributions. The level of agreement between the nozzle gross thrust coefficients is also good. Measured nozzle thrust is found to increase with increasing fuel equivalence ratio and is primarily a result of the combustion induced pressure rise. The effect of two-dimensional wave processes in the nozzle are also examined.

INTRODUCTION

A primary objective of hypervelocity scramjet tests is to evaluate the thrust producing potential of a combustor configuration. For a given nozzle area ratio, a one-dimensional analysis ignoring finite rate chemistry in the nozzle shows the thrust force produced depends on the nozzle entrance Mach number and pressure alone. In a two- or three-dimensional expansion geometry, non-uniformity of the Mach number and pressure at the combustor exit plane and in the nozzle may modify the thrust achieved.

The degree of modification and whether the Mach number profile augments or reduces the thrust relative to a uniform profile depends on the transverse location and sign of the Mach number gradient, and the shape and length of the thrust nozzle (Stalker 1989). In a two-dimensional scramjet, operating at hypervelocity, a partially mixed and reacted fuel jet will form a low Mach number region in the combustor exit plane. The sign of the Mach number gradient created by the jet depends on whether the fuel is injected along the wall, or is centred in the duct, separated from the wall by unmixed air. The latter case has been studied by Stalker and Morgan (1984) for a plane two-dimensional scramjet. It was found that the interaction of the centred expansion which originates at the nozzle expansion corner, with the lower Mach number fuel jet, produces compression waves and a thrust increment on the nozzle surface relative to a fuel-off situation.

To achieve the nozzle geometric area ratio necessary for sufficient thrust production in a working scramjet, a three-dimensional expansion geometry is required. Presently, a wall-injected axisymmetric scramjet and conical nozzle is studied. This geometry achieves a larger geometric area ratio than a plane geometry of the same expansion angle and therefore includes in a readily analysable flowfield some of the advantage of a three-dimensional expansion.

Experiments with an axisymmetric scramjet with a conical thrust nozzle are described and measured pressures are compared to a computational solution. The nozzle thrust produced is calculated and the influence of fuel equivalence ratio on nozzle performance is evaluated by examining the nozzle thrust coefficient.

EXPERIMENTAL CONDITIONS AND APPARATUS

The tests were performed in a reflected shock tunnel with a contoured nozzle producing uniform air flow at Mach 5.3, pressure of 16.0 kPa, and temperature of 2000 K. The pressure level at the exit of the facility nozzle was confirmed by flat plate static pressure measurements. The Mach number and temperature values were determined from a finite rate chemistry calculation of the nozzle flow (Lordi et al. 1966) terminating when the static pressure achieved the measured value. The test gas was calculated to be out of equilibrium having 33% and 12.5% by mass of its oxygen in atomic oxygen and nitric oxide forms respectively.

The axisymmetric combustor and nozzle configuration are shown in Figure 1. Hydrogen fuel was injected at Mach 1.9 from an annular wall jet at an angle of 15° from the combustor wall. Fuel conditions were varied by adjusting the injector plenum pressure to achieve equivalence ratios of 0.93, 1.91, and 3.05. A thrust nozzle is formed by a conical expansion of 10° half angle. It begins from a sharp corner and terminates at an exit geometric area ratio of 9.17. Upstream of injection a short 0.851 area ratio conical inlet captured the facility flow producing a 25% increase in static pressure. This inlet was necessary to fine tune the test condition to achieve desired combustor inlet conditions. Combustor and thrust nozzle instrumentation included wall static pressure transducers at 25 mm intervals in the combustor and nozzle.

COMPUTATIONAL MODELLING

Simulations of the complete scramjet and nozzle flows were done using the SHARC computer code developed by Brescianini (1992a). SHARC is a two-dimensional, steady, parabolic Navier-Stokes solver using the finite volume method of method of Patankar and Spalding (1970) with the pressure field calculated by the SIMPLE algorithm, (Elghobashi and Spalding, 1977).

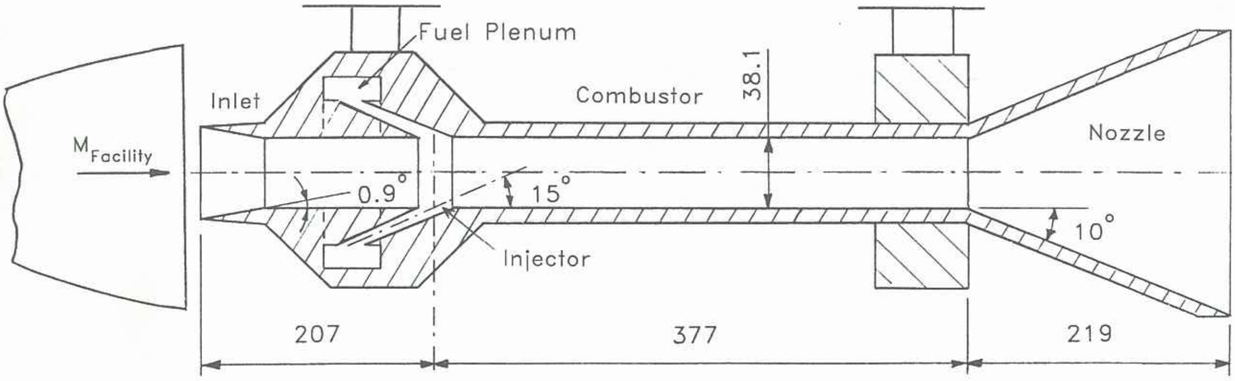

Figure 1. Configuration of axisymmetric scramjet and conical thrust nozzle (dimensions in mm).

Turbulence was modelled using a compressibility corrected k-ε1 turbulence model described in Brescianini (1992b). It was implemented in the manner described in that reference. For hydrogen-air reaction chemistry the 8-reaction mechanism as given by Evans and Schexnayder (1980) was chosen to allow rapid turn around of computer solutions. A spot check comparing to a more complete 25 reaction set from the same reference showed negligible difference in predicted scramjet pressure distribution.

The portion of the model from the inlet leading edge to the injection station was modelled using the facility exit conditions quoted above as initial conditions. The computational grid was slightly compressed toward the wall and 80 grid points were used over the duct radius. For fuel-on simulations the calculation was restarted at the injector station using a modified grid which included the inclined fuel jet. Again an 80 point grid was used, slightly compressed toward the outer wall. The air initial conditions were calculated to give the same boundary layer thickness on the injector wall as the inlet solution without retaining the details of the weak wave system formed in the inlet.

MEASURED AND COMPUTED PRESSURES

Measured and SHARC predicted distributions of wall surface pressure are shown in Figure 2a. Scatter bars represent the spread of data points over several (up to seven) tests and are seen to be typically within +/- 10% of the mean. No normalisation of the data was done to remove the scatter associated with repeatability of facility pressures, which accounts for approximately +/- 5% of the scatter. Where no bars are shown insufficient data points (less than three) were available to evaluate data scatter.

For the fuel-off case, the mean pressure level is well predicted while the measured wave pattern created by the inlet appears to damp more quickly than is predicted. The thrust surface pressure is of the correct magnitude although the shape of the distribution is not picked up by the calculation.

For the fuel-on case, a large amplitude standing wave pattern has been created by the slightly transverse fuel injector which is underexpanded by approximately a factor of three at the injector lip. Agreement in the combustor is excellent with the wave pattern and combustion and mixing induced pressure rise well predicted. The calculated injection shock persists into the thrust nozzle as seen by the bump in the computed solution there, while no clearly defined perturbation appears in the data at this location.

Measured and computed pressures for the other equivalence ratios tested are shown in subsequent Figures 2b and 2c. The injection induced shock system and the mixing/combustion pressure rise decrease in intensity for the lower equivalence ratios and the computed location of shock reflection in the nozzle moves downstream. The level of agreement between the computation and the data in the combustor is again seen to be good.

MEASURED AND PREDICTED NOZZLE PERFORMANCE

The nozzle thrust, F_T, is obtained from the measured nozzle surface pressure distribution and geometry by numerical integration using Simpson's rule. The accuracy of this integrated value is limited by the number of points in the pressure distribution and the degree of data scatter noted previously. However, the relative smoothness of the nozzle pressure distributions should make the numerical integration procedure reliable. Therefore, it is expected that the thrust values are of accuracy comparable to the combustor data, +/- 10%. For comparison, values of F_T were calculated from the SHARC nozzle pressure distributions. These are summarised for the three fuel equivalence ratios and the fuel-off run, along with the measured values, in Table 1. The values computed agree with the data to well within the data reliability limit of 10%.

Table 1. Nozzle thrust as measured and predicted by SHARC

Equiv. Ratio	Measured Thrust F_T(N)	SHARC Thrust F_T(N)
0	51.7	48.0
0.93	67.3	66.5
1.91	80.3	77.0
3.05	89.0	89.0

Both measurements and predictions show increasing thrust is produced by the nozzle with increasing equivalence ratio. This increase reflects changes in both the mean pressure at the combustor exit, which alters the

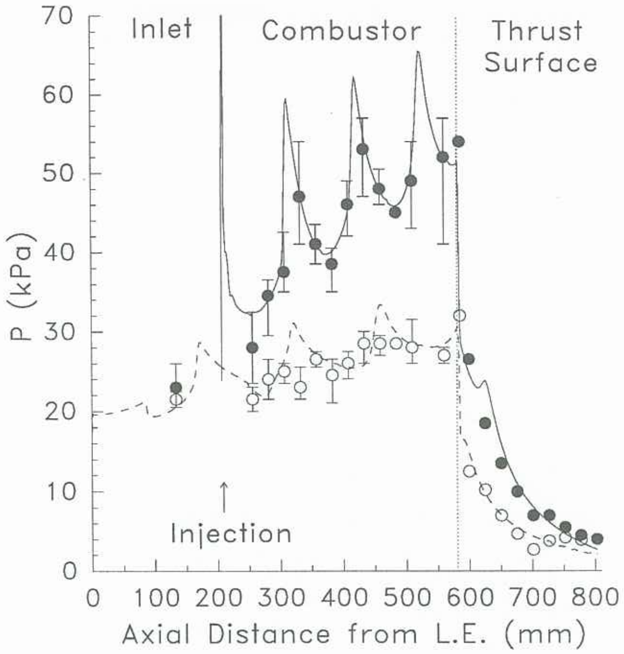

Figure 2a. Comparison of measured and SHARC predicted duct wall pressures for fuel-off (○ - - -) and equivalence ratio of 3.05 (● ——) tests.

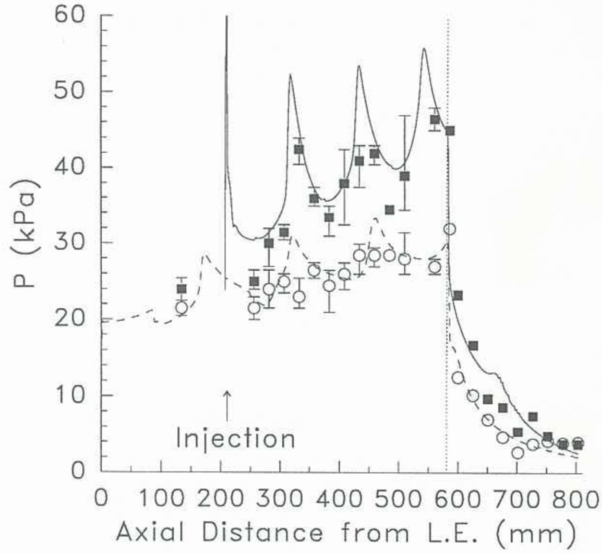

Figure 2b. Comparison for fuel-off and equivalence ratio of 1.91 (■ ——) tests.

pressure level in the nozzle, and changes to the Mach number distribution at the combustor exit, which results in a modification of the wave pattern in the nozzle. The relative importance of the wave pattern changes alone can be examined by forming a thrust coefficient, C_F, defined as the nozzle thrust normalised by the combustor exit area and mean pressure. A mean exit pressure was determined for each equivalence ratio by area averaging the SHARC calculated pressure distribution. Considering the good agreement found between the SHARC solution and the combustor pressure data, this mean pressure was used to form both the measured and SHARC C_F values.

Thrust coefficients (Table 2.) from both the measurements and the SHARC solutions show only slight variation (up to 6%) with fuel equivalence ratio, while the thrust itself varies considerably. Thus, the major factor influencing the thrust as the equivalence ratio changes is the mean static pressure at the combustor exit, while alteration of the Mach number distribution and subsequent nozzle wave pattern changes do not strongly influence the thrust produced. This strong pressure, weak wave pattern dependence is consistent with the fuel being well mixed across the duct, resulting in a relatively uniform Mach number distribution which only weakly interacts with wave propagation.

Although only a slight dependence of thrust on wave pattern changes due to fuel addition has been demonstrated for this flow, it is of interest to determine how close the conical nozzle is to an optimum thrust contour for the given area ratio and combustor exit conditions. An optimum contour acts to cancel all waves intersecting its surface, producing a uniform pressure exit plane.

To see this, SHARC was run for the four fuel conditions with a uniform lateral pressure condition imposed in the combustor and nozzle. The nozzle thrust found is thus nearly equivalent to a perfect, wave cancelling expansion in a contoured nozzle of the same

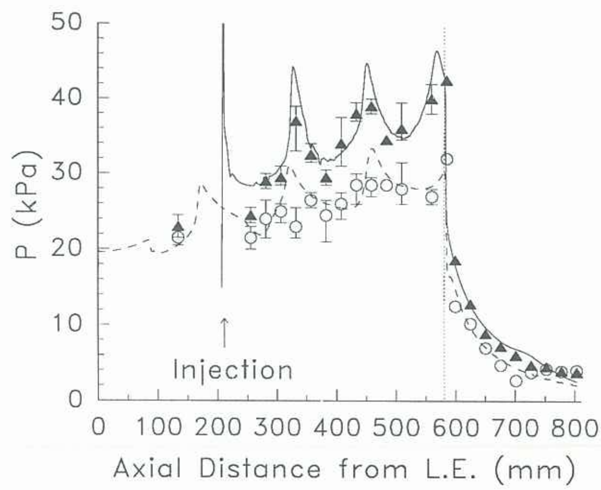

Figure 2c. Comparison for fuel-off and equivalence ratio of 0.93 (▲ ——) tests.

Table 2. Thrust coefficients determined from measured nozzle pressure and predicted by SHARC for conical and contoured nozzles.

Equiv. Ratio	Measured C_F Conical	SHARC C_F Conical	SHARC C_F Contoured
0	1.49	1.39	1.55
0.93	1.47	1.46	1.55
1.91	1.50	1.43	1.50
3.05	1.41	1.41	1.47

geometric area ratio. (Slight differences arise in the combustor exit conditions from the absence of waves in the combustor, however, combustor exit stream thrust is found to agree to within 1% of the full wave capturing solution). The "contoured" thrust coefficients calculated are given in Table 2. It is seen that the conical nozzle performance, as simulated by SHARC, approaches the ideal contour for the highest equivalence ratio, and is within 89% of the optimum for the worst, fuel-off case. Therefore, it is expected that the thrust measured in the conical nozzle is within approximately 10% of the thrust that would have been achieved had a perfectly contoured nozzle been tested.

CONCLUSIONS

Surface pressure distributions have been measured and thrust values calculated for a scramjet with a conical expansion nozzle when tested in a shock tunnel at conditions equivalent to flight Mach 17. Flowfield simulations with a parabolic Navier-Stokes code were found to predict the combustor pressure distribution and the thrust produced by the nozzle to within the data scatter of +/-10%. The details of the pressure distribution in the nozzle, including the location of injection generated wave reflections, is not well predicted, however, considering the agreement in integrated thrust, this discrepancy appears less important.

The nozzle thrust coefficient was examined to determine the effect of fuel addition on the combustor exit plane Mach number distribution and on the wave pattern formed in the nozzle. The addition of fuel was found to alter the thrust coefficient by a maximum of 6% for both the measured and predicted nozzle flows. However, changes in absolute thrust were much greater indicating that combustion induced pressure rise was the major contributor to thrust.

An ideal wave cancelling prediction of nozzle performance for the calculated nozzle exit Mach number distribution was found to give up to 11% greater thrust coefficient than the conical nozzle geometry which was tested. It is expected therefore that the measured thrust values are also approximately 90% of the maximum achievable for the nozzle geometric area ratio.

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