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EFFECT OF FUEL STAGNATION TEMPERATURE ON SUPERSONIC COMBUSTION WITH TRANSVERSE INJECTION

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ABSTRACT

An investigation was conducted into the effect of varying fuel stagnation temperature on combustion in supersonic flow. Fuel was injected from a wall injector at an angle of 30° to the free stream in a constant area duct. Wall pressure was measured to determine the time for complete reaction to occur. Simple incompressible theory predicted that with the equivalence ratio raised to account for the increased stagnation temperature that no difference should occur between the hot or cold injection. The experimental results supported this prediction.

INTRODUCTION

Accelerating scramjet powered vehicles need to fly at high density altitudes in order to develop sufficient thrust. Heat transfer to the vehicle is therefore very high and necessitates the use of active cooling with cryogenic Hydrogen fuel. All past duct experiments in [1]hypervelocity flow have used fuel reservoirs at room temperature. An experimental study was performed to determine the effect of raising the stagnation temperature of the fuel reservoir and the subsequent effect of the increased fuel velocity.

The most likely type of fuel injection to be used for scramjet engines is transverse wall injection through a circular injector at some angle to the flow. This type was utilised for the experiments with the injector angle set at 30°. Some simple theory was applied in an attempt to quantify the result.

ANALYSIS

Past experiments with transverse injection have shown that the jet quickly takes the form of a pair of counter rotating vortices. Broadwell and Breidenthal (1984) performed a series of experiments by injecting an alkali with coloured pH indicator into a colourless acidic free stream. The experiments measured the distance at which all the alkali and acid had mixed to a molecular level and reacted to change the pH causing the indicator to become clear.

In their analysis they suggest that if viscosity is ignored, except in the dissipation of microscale vortices, then the only global length in the flow is given by

$$l = \left(\frac{m_j V_j}{r_{fs} V_{fs}^2} \right)^{0.5}$$

where m_j is the jet mass flow, V_j is jet/wake velocity, ρ_{fs} is the free stream density and V_{fs} is free stream velocity. This represents the ratio of the jet (or wake) momentum flux to the vorticity driving force. Broadwell and Breidenthal go further to suggest that if the diffusion time is short compared to time for entrainment, the velocity ratio V_{fs}/V_j is large, and for equal jet and free stream densities, then the length scale for complete reaction is given by

$$x_r = const. \cdot \left(\frac{V_{fs}}{V_j} \right)^{0.5} (\Phi + 1)^{1.5} A_o^{0.5}$$

where A_o is the injector exit area and Φ is the equivalence ratio.

An attempt was made to extend the validity of this theory to a compressible gas flame. The free stream static temperature and pressure were sufficiently high such that combustion should occur as soon as mixing had reached a molecular level. This is similar to the acid-alkali reaction in that the process is limited by the mixing rate rather than reaction rate. The pressure rise on the walls of the duct was measured to determine when combustion occurred.

APPARATUS

Experiments were performed using the T4 free-piston reflected shock tunnel at the University of Queensland. A constant area duct with a 50mm * 25mm cross-section was used to simulate the combustion chamber of a scramjet powered vehicle. A single circular injector of Ø7mm was placed on the lower 50mm surface at an angle of 30° to the flow. Pressure transducers were placed along the centreline of the upper and lower surfaces of the duct. The room temperature fuel was supplied via a Ludwig tube separated from the duct by a fast acting valve. Hot fuel was supplied via a Nitrogen driven gun tunnel. The fuel stagnation temperature was measured using a calorimeter heat transfer gauge with the theory outlined by Rose and Stark (1958). The temperature of the fuel at the gun tunnel reservoir was measured at 730K and at the exit from the 30° injector at 625K. Both the Ludwig tube and gun tunnel supplied steady pressure during the test time.

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1 High velocity and high temperature flow.

EXPERIMENTS

The experiments were performed at three different air velocities and two different fuel velocities. The shock tunnel shock speed and stagnation pressure were used to calculate the reservoir conditions for the nozzle. A non-equilibrium nozzle flow code was then utilised to compute the free stream conditions. The fuel conditions were calculated by isentropically expanding the measured fuel stagnation conditions until fuel and free stream static pressures were matched within the confines of the duct. These conditions are given in tables I and II.

Table I - Free Stream Conditions

cond	test gas	Static Temp K	Static Press. kPa	Dens. kg/m^3	Vel. m/s	Ma. #
a	Air	2163	69	.109	3633	4.22
b	N_2	2249	68	.102	3989	4.28
c	Air	1672	74	.155	3243	4.11
d	N_2	1428	59	.138	3350	4.48
e	Air	1307	68	.182	2990	4.25
f	N_2	1184	60	.171	3082	4.51

Table II - Fuel Conditions in Duct

Stag. Temp. K	Static Temp. K	Velocity m/s	Density Kg/m^3	Mach #
297	124	2150	.121	2.38
625	167	3689	.1	3.68

The reflected shock tunnel driver gas contained 22.3%Argon, 77.7%He to allow the shock tunnel to be run in tailored mode at the chosen test conditions. Typical wall pressure records in the duct for each of these conditions are given in figure 1 and show the near constant pressure during test period.

The cold fuel injection data was taken from earlier experiments by Morgan and Casey(1990).

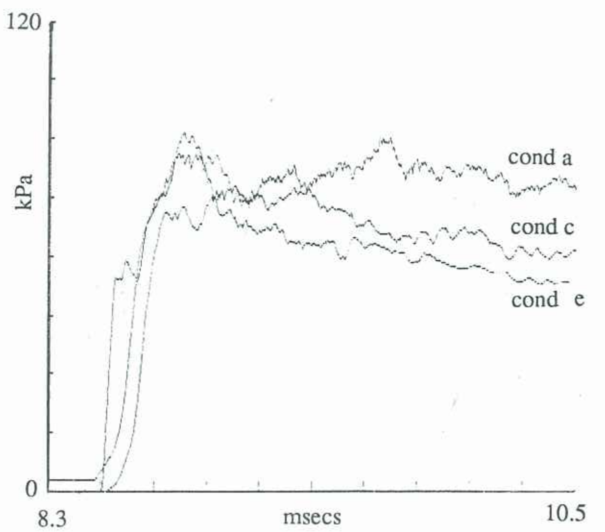
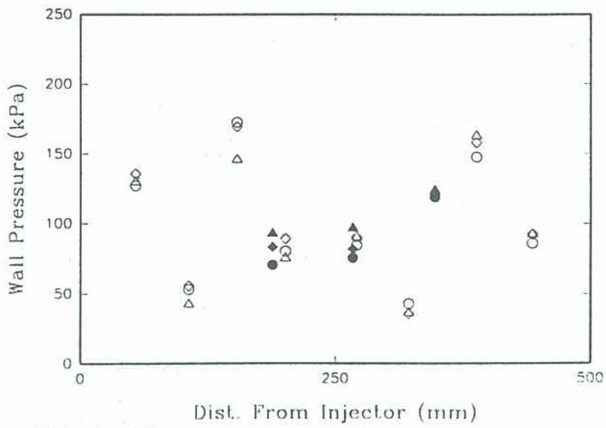


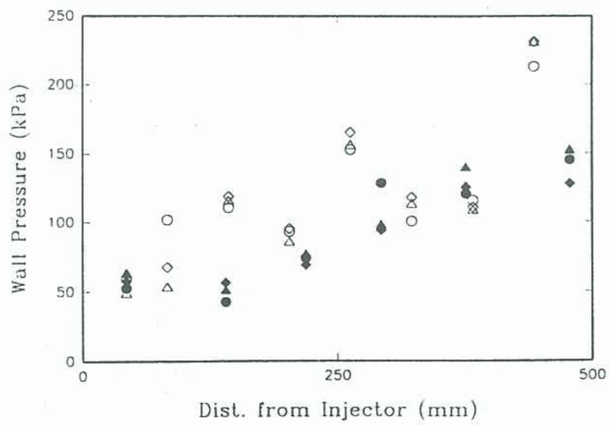
Figure 1- Typical duct pressures versus time

RESULTS

Measured wall pressure versus distance from the injector are presented in figures 2 and 3. Figures 2a and 2b show the pressure for the wall surface with and without the injection orifice respectively. It shows that the pressure in the duct is approximately the same for the cold and hot injection for all conditions. The large variation in pressure is due to the bow shock off the transverse injection jet causing waves not only vertically but also horizontally. In an attempt to minimise the problem the combustion pressure(fuel+air) was divided by the non-combustion (fuel+N_2) pressure. This effectively sets the wave disturbance pressure to one, leaving only the combustion pressure. It also removes the calibration error between transducers and between the hot and cold injection data which were recorded with different equipment. The results (figures 3a,b) show that when the fuel bow shock disturbances are normalised out, the pressure rise due to combustion is no different for the case of hot or cold injection in the shape or magnitude of the pressure rise . The delay before the pressure rise is also the same.

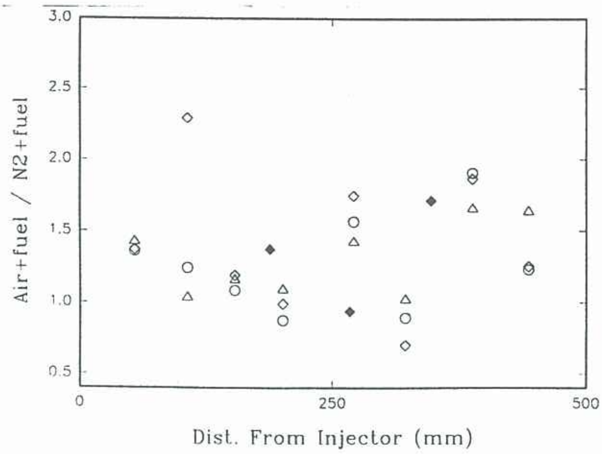


a) Injector side

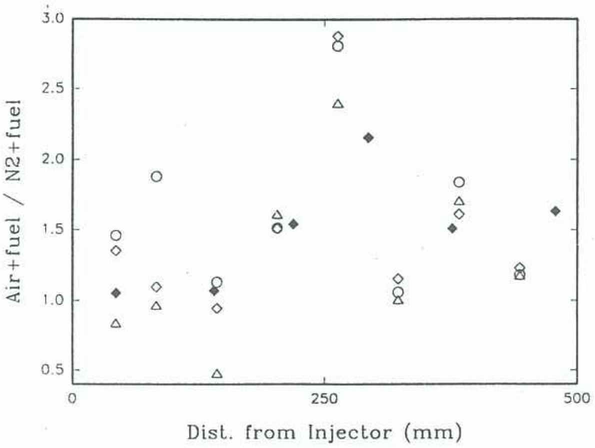


b) Other side

Figure 2- Wall pressure versus distance for Air+Fuel
○-a(hot) △-c(h) ◇-e(h) ●-a(cold) ▲-c(c) ◆-e(c)



a) Injector side



b) Other side

Figure 3- Normalised wall pressure versus distance
○-a/b(hot) △-c/d(hot) ◇-e/f(hot) ◆-e/f(cold)

Table III gives the values[2] of the equivalence ratio and reaction length scale. The theory suggests that with the equivalence ratio raised to account for the raised velocity of the free stream, that there should be no difference in the length scale between the hot and cold injection. This prediction was borne out by the experiments.

It is important to note that Broadwell and Breidenthals assumptions of equal density of the air and fuel and a large velocity ratio V_{fs}/V_j were not true for the experiments.

Table III - Comparison of Theoretical and Experimental Reaction Scale ratios

Test Cond	Equiv. Ratio Φ		Theoretical length scale			Expt. length scale
	hot	cold	hot	cold	hot/cold	hot/cold
a/b	1.78	.98	.029	.023	1.26	-
c/d	1.48	1.0	.023	.022	1.05	-
e/f	1.36	.95	.021	.02	1.02	1

[2]Note that velocities and densities are averaged between the Air and Nitrogen flows

CONCLUSION

Preliminary results suggest that for mixing limited flames such as found in supersonically combusting ramjets, the reaction length scale as outlined by Broadwell and Breidenthal appears to be applicable as a scaling parameter for compressible gas flames.

References

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