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Technical Notes

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Boundary-Layer Separation Due to Combustion-Induced Pressure Rise in a Supersonic Flow

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I. Introduction

Korkegi [1] correlates the pressure rises required to cause turbulent boundary layers to separate for two- and three-dimensional shock-wave/boundary-layer interactions. The correlations have been used in studies to predict or avoid boundary-layer separation due to two-dimensional [2–4] and swept [2,5] shock-wave/boundary-layer interactions.

For two-dimensional interactions, Korkegi [1] correlates the static pressure downstream of the interaction at which incipient separation occurs, p_k, as a function of the static pressure upstream of the interaction, p_c, and the Mach number upstream of the interaction, M_c, in the form

$$\frac{p_k}{p_c} = 1 + 0.3M_c^2 \quad \text{for } M_c \leq 4.5 \tag{1}$$

Another mechanism for producing adverse pressure gradients in supersonic flows is a combustion-induced pressure rise in a supersonic flow. The correlation has been cited by several researchers in relation to separation due to such pressure rises [6,7]. A series of supersonic combustion experiments was performed as part of the ground-testing phase for the HyShot I and II supersonic combustion flight experiments [8–10]. Tests were performed on the flight combustor configuration for a range of simulated flight conditions and fuel flow rates. The results from those tests provide useful data for checking the applicability of the Korkegi correlation for boundary-layer separation due to combustion-induced pressure rises in supersonic flows.

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II. Experiments

The HyShot supersonic combustion experiments were conducted in the T4 Stalker tube at the University of Queensland, Australia. Different driver and shock-tube filling conditions were used to generate conditions in the combustor that would be attained on the nominal trajectory at altitudes of 34.4 ± 1.2, 28.6 ± 0.9, 22.9 ± 0.6, and 21.6 ± 0.5 km. For the shock-tunnel experiments, the Mach 6 axisymmetric nozzle was used. The nominal stagnation enthalpy for the HyShot I and II experiments was 3 MJ/kg. At this enthalpy, the nozzle produced a Mach number of 6.5 ± 0.08 at its exit.

The same model design was used for the HyShot I and II flight experiments [8,9]. The model was designed to test supersonic combustion in a constant-area duct of rectangular cross section. It consisted of an intake compression wedge, combustion chamber, and thrust plate (Fig. 1). The intake was deigned to spill both the compression shock and the boundary layers formed on the intake surfaces and to generate new boundary layers on the four walls of the combustor.

The model was designed for a flight Mach number of 7.6 for altitudes ranging from 20 to 35 km. The inlet compression wedge angle δ for the flight model was 18 deg. To match the Mach number in the combustor between the Mach 6.5 tunnel tests and the Mach 7.6 flight tests, the inlet compression wedge angle for the shock-tunnel tests was decreased to $\delta = 17$ deg. The shock-tunnel conditions were set so that the pressures and temperatures in the combustor would be similar to those expected in the flights at the different altitudes. The combustor was a constant-area duct, 300 mm long, 75 mm wide, and 9.8 mm high. Piezoelectric pressure transducers (PCB Piezotronics, Inc., types 111A26, 112A21, and 112A22) were used to measure pressures on the inner surface of the combustor and on the thrust plate. The inner surface of the combustor had 16 pressure tappings along its centerline at 13 mm centers, and there were 11 pressure tappings along the centerline of the thrust plate, also at 13 mm centers. The first pressure tapping in the combustor was located 90 mm from the leading edge of the combustor, and the first in the thrust plate was located 13 mm from the start of the thrust plate. The sensing surfaces of the pressure transducers were thermally protected by covering them with 25 μm cellophane disc. Some of the cellophane discs failed during the test program and the data from those transducers have not been used in the present analysis.

For zero yaw angle, the angle of attack of the model was varied from −4 to +4 deg in 2 deg increments. For a zero angle of attack, the angle of yaw was varied from 0 to 4 deg in 2 deg increments. In analysis of data in the present paper, the combustor entrance conditions were calculated from the oncoming mainstream conditions using oblique shock theory. The conditions at the exit of the nozzle were determined from the shock-tube filling pressure and temperature, the measured speed of the primary shock wave in the shock tube, the measured nozzle-supply pressure, and the geometry of the nozzle using equilibrium chemistry calculations [11,12]. Because of the variations in angle of attack, the Mach number at the entrance to the combustor duct, M_c, ranged from 2.2 to 3.6. For the full range of simulated altitudes and angles of attack, the static pressures at entrance to the combustor duct, p_c, varied from 17 to 300 kPa.

Fuel was injected perpendicularly to the oncoming flow from four 2 mm porthole fuel injectors located 40 mm from the leading edge of the combustor (see Fig. 1). Hydrogen was used as the fuel, and

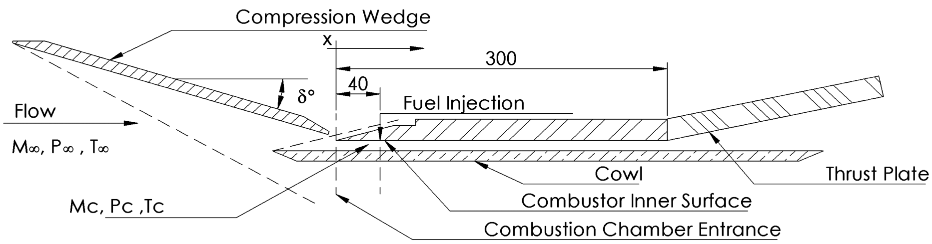


Fig. 1 T4 model of the HyShot scramjet.

combustion experiments were conducted for equivalence ratios from 0.20 to 1.20. Fuel was supplied from a Ludwieg tube, and the flow from the injectors was initiated approximately 10 ms before the test flow impulsively started through the nozzle of the tunnel. For the duration of the test, flow the fuel mass flow rates varied by less than 1.0%.

Further details of the HyShot I and II models and their orientation during the tests and the freestream conditions used can be found in [8].

The data acquisition system for the tunnel tests enabled simultaneous data capture on 21 channels at a sampling time of 1 μs with a memory buffer capable of storing 8192 sample points. Signals from sets of up to four transducers were multiplexed onto a single channel of the data acquisition system. The 95% confidence-interval uncertainty in the measured static pressure is estimated to be $\pm 4\%$.

The shock tunnel produces an impulsively starting flow. For the present tests, the duration of steady nozzle-supply pressures was several milliseconds. The nozzle flow takes approximately 1.0 ms to become established at the conditions of the present tests. This was determined by normalizing the pitot pressure in the test section by the nozzle-supply pressure with an appropriate time delay to account for the time it takes for the flow to pass from the nozzle-supply region to the nozzle exit. The ratio became constant when the nozzle flow was established. It takes some time for the flow to establish over and within the model and for the combustion to establish. For scramjet models in the T4 shock tunnel, this has been identified as being the time it takes for flow to traverse approximately 3 times the length of the model [13].

For the present conditions, this means that the flow will take approximately 1.0 ms to become established at the most downstream pressure measurement location on the thrust plate. Therefore, the pressure distributions were steady from approximately 2.0 ms after flow start. The end of the test flow in T4 occurs either when the nozzle-supply conditions are no longer steady or when the test gas becomes contaminated by the shock-tube driver gas. For the conditions of the present tests, the latter was often the relevant criterion, and previous tests to measure driver-gas contamination [14,15] have shown that contamination occurs approximately 5 ms after the start of the flow in the test section. Therefore, for the present experiments, the established test flow period was taken as being from 2.0 to 5.0 ms after flow arrival in the test section. Note that in the cases in which the combustor duct choked, the combustor flow establishment time was longer, and in some cases, the choking was not complete by the end of the test period.

III. Results and Discussion

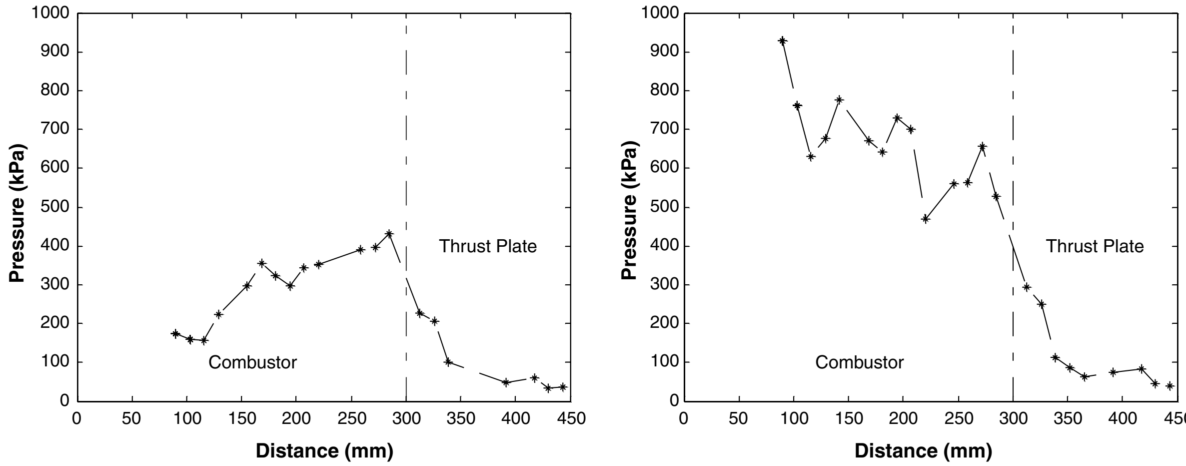
The preflight ground testing for the HyShot I and II supersonic combustion experiments involved 79 shock-tunnel shots in which fuel was injected. These were composed of tests at varying fuel equivalence ratios for each combination of angles of attack and yaw and simulated altitude.

For some test conditions, the pressures in the combustor indicated that the combustor duct was choked. A simple thermal-choking calculation was done for each condition using Rayleigh-flow analysis. The fuel flow rate required to reduce the Mach number in the combustor to unity from the value at entry to the combustor was calculated. A heating value for hydrogen of 120 MJ/kg, a combustion efficiency of 80%, and a specific heat of the flow of $1500\ \mathrm{J/kg \cdot K}$ were assumed. The fuel flow rates at which the initiation of choking was observed were not high enough for this to be caused by thermal choking. Therefore, choking of the duct is attributed to separation of the boundary layers on the walls of the combustor restricting the flow that is able to pass through the combustor. Boundary-layer separation is attributed to the adverse pressure gradients caused by combustion-induced pressure increase in the duct.

For fuel-off tests, the combustor duct started for all conditions and supersonic flow was observed in the combustor. Both unchoked and choked combustor flows were observed in experiments in which fuel was injected. The characteristics observed for unchoked flows were relatively low pressures in the combustor (corresponding to supersonic flow) and static pressures increasing with increased distance along the duct. The latter is characteristic of heat addition in a supersonic flow in a constant-area duct. For cases in which the duct choked, the pressures in the duct were higher (corresponding to subsonic flow) and the static pressures decreased with distance along the duct. The latter is characteristic of heat addition in a subsonic flow in a constant-area duct. Examples of pressure distributions measured at the end of the test period for unchoked and choked combustor flows are presented in Figs. 2a and 2b, respectively. Both results are from tests at a simulated altitude of 22.7 km with the model at zero angles of attack and yaw but for different fuel equivalence ratios. The result in Fig. 2a is for fuel injection at an equivalence ratio ϕ of 0.36 and the duct is unchoked. The result in Fig. 2b is for $\phi = 0.54$ and the duct is choked.

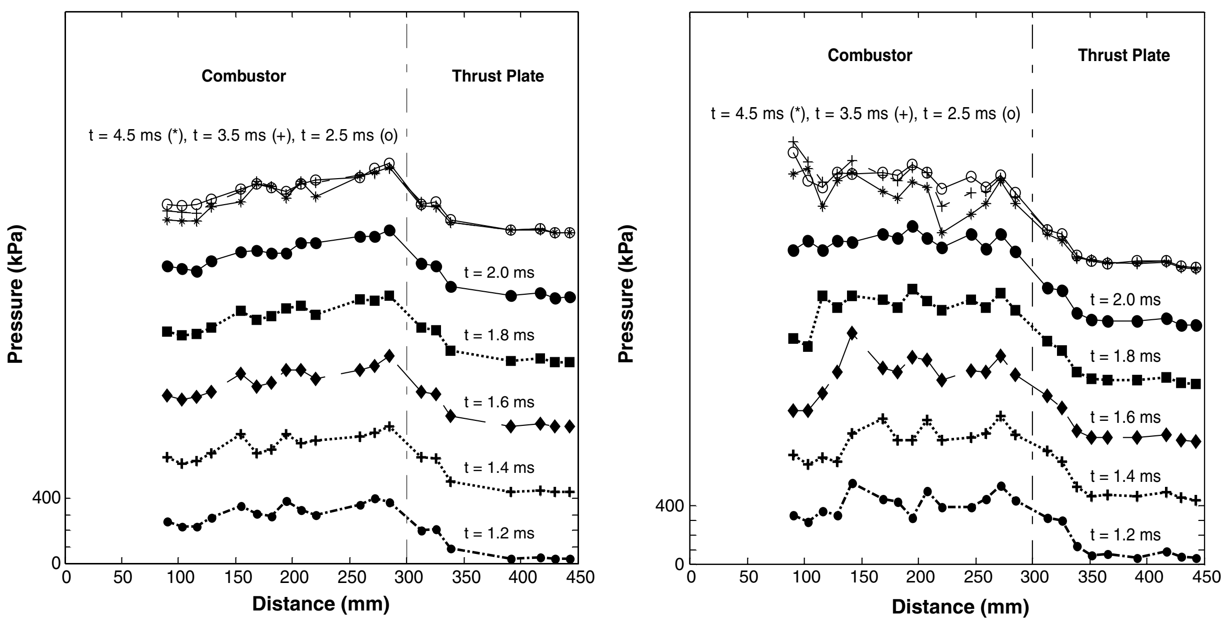
For all shots in which the combustor duct was identified as choking, it was observed that the duct initially started, but the choking process could be observed to occur as the run progressed. This choking process was observed to occur during the nozzle starting and flow establishment phases or, sometimes, during the test period. Similar processes have been observed in other shock-tunnel tests of scramjet combustors [16]. Sequences of pressure distributions measured for the sample shots in which there was no choking (6873) and in which choking occurred (6797) are shown in Fig. 3. These are the same shots used for the example traces in Fig. 2. The pressure distributions in Fig. 3 are shown for times during the flow establishment period and during the steady flow period. At each time, the pressures are averaged over a period of 200 μs. The plots for different times are offset in the vertical direction to enable the flow establishment processes to be seen.

For shot 6873, the flow with combustion establishes by time 2.0 ms and remains quite steady during the test period. Toward the end of the test period, the tunnel nozzle-supply pressure begins to drop and the pressures in the duct also start to drop. For the higher equivalence ratio shot (6797), the pressures toward the front of the combustor (90–130 mm) at time 1.4 ms are relatively low and near the levels that would be expected for supersonic duct flow. Note that this distribution was measured before the steady test flow period. The pressures toward the rear of the duct are approximately twice those at the front. A disturbance is observed to propagate toward the front of the duct as time progresses. This disturbance is interpreted as being due to separation of the boundary layers in the duct due to the adverse pressure gradient to which the boundary layers are subjected. The separation reduces the effective flow area, thereby restricting the flow and pushing a shock (or series of shocks) upstream to eventually unstart the duct. By time 2.5 ms, the duct is fully choked, and for the next 2 ms, the pressure distribution in the duct remains relatively



a) Shot 6873, $\varphi = 0.36$ b) Shot 6797, $\varphi = 0.54$

Fig. 2 Measured pressure distributions for unchoked and choked combustor flows (simulated altitude 22.7 km; zero angles of attack and yaw).



a) Shot 6873 (unchoked) b) Shot 6797 (choked)

Fig. 3 HyShot scramjet pressure distributions (simulated altitude 22.7 km; zero angles of attack and yaw).

steady. During this established flow period, the pressure distribution down the combustor is indicative of subsonic combustion. The flow establishment periods for shots in which the combustor choked were generally longer than for cases in which the combustor flow remained supersonic and, in some cases, the duct-choking process was not complete until after the nominal test period.

Choking of the combustor, attributed to boundary-layer separation due to combustion-induced pressure rises, was identified to occur within the test period in 12 fuel-on shots, and in 10 fuel-on shots, complete choking did not occur until after the test period. The maximum static pressure measured in the combustor during the test time, $p_{\max}$, has been normalized by the pressure calculated at entrance to the combustor, p_c, for all fuel-on shots. Taking into account the uncertainty in flow conditions and pressure measurements, the 95% confidence-interval uncertainty in the ratio $p_{\max}/p_c$ is estimated to be $\pm 11\%$. The results are plotted as a function of the Mach number at the entrance to the combustor, M_c, in Fig. 4. Shots for which choking of the combustor was identified are indicated by closed symbols and, when choking was not observed, open symbols are shown. An asterisk is used to show the results from cases in which the choking process started during the test period, but was not completed until after the test period. The Korkegi correlation from

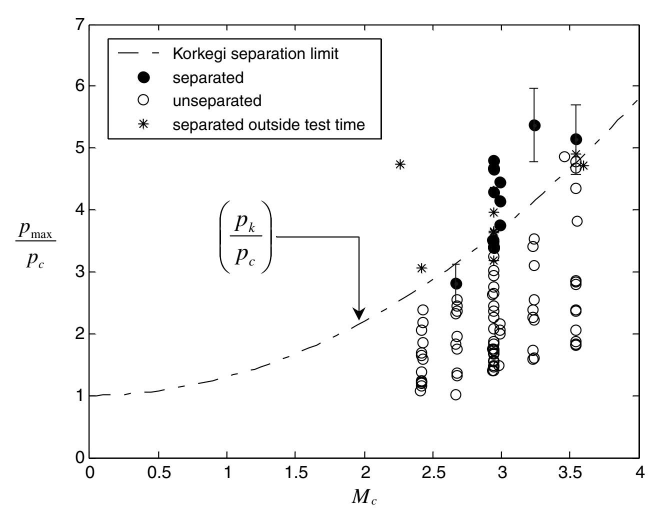


Fig. 4 Combustion-induced pressure rises in a constant-area scramjet duct required to induce boundary-layer separation and comparison with the 2-D correlation by Korkegi [1].

Eq. (1) is also shown. It can be seen that the correlation gives a good indication of shots in which the duct chokes for the range of duct entry Mach numbers covered by these experiments.

IV. Conclusions

Korkegi [1] presents a correlation for the pressure rise induced by a shock-wave/boundary-layer interaction required to separate turbulent boundary layers in supersonic flows. Results from the present tests on a constant-area supersonic combustor indicate that the correlation gives a good indication of the pressure rise induced by combustion heat release required to cause boundary-layer separation for combustor entry Mach numbers between 2.4 and 3.6.

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