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Technical Notes

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Unstarted Inlet for Direct-Connect Combustor Experiments in a Shock Tunnel

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A COLLABORATIVE project[1] between Australia and Japan has been conducted in which supersonic combustion has been compared between a free-piston-driven shock tunnel[2] (T4) and a vitiation-heated blowdown tunnel[3] (VAG). Under normal operation T4 produces freestream Mach numbers considerably higher and total pressures at least an order of magnitude higher than the VAG. To match the combustor inlet conditions of the VAG, an unstarted inlet (converging-diverging nozzle) directly connected to the combustor has been used to further process the shock-tunnel freestream flow before combustion. Given the large subsonic region produced, the successful use of this unstarted inlet has important implications for testing in impulse facilities and is the subject of this Note.

The free-piston shock tunnel[2] operates by using a free piston to compress adiabatically the driver gas for the shock tube. Upon primary diaphragm rupture, a shock wave propagates along and reflects from the end wall of the shock tube, leaving a stagnant high-enthalpy reservoir of test gas. This reservoir feeds into a nozzle downstream of the shock tube. The nozzle then produces a hypersonic flow with useful duration of up to several milliseconds.

Through suitable choice of the driver operating conditions, primary diaphragm thickness, and shock-tube initial pressure, T4 is capable of nozzle reservoir enthalpies and pressures in the ranges 2–15 MJ/kg and 10–80 MPa, respectively. Several contoured axisymmetric hypersonic nozzles of different exit-to-throat area ratios (for nominal Mach numbers 4, 6, 8, and 10) are available. A Mach 4 nozzle was used in the present work, for which the uniform test core is approximately 100 mm in diameter.

The nominal combustor inlet conditions required for the present work were Mach 2.5 and a total pressure of 1 MPa. If this Mach number were produced by a Mach 2.5 nozzle fed from the shock tube, the necessary area ratio would result in severe restrictions on the size of models that can be tested. Furthermore, the combustor inlet total pressure would be the reservoir pressure of the hypersonic nozzle.

To match the VAG combustor inlet conditions, a different arrangement was necessary. It is possible to employ pairs of wedges in a supersonic diffuser arrangement to process the tunnel freestream flow through a system of oblique shocks and arrive at the required conditions. However, this method is also constrained by geometry: the size of the inlet of such a diffuser would be much larger than the exit dimensions of the relevant hypersonic nozzle used. Instead, an unstarted converging-diverging nozzle was employed, directly connected to the combustor (Fig. 1).

A pair of wedges with parallel sideplates capture the flow, which is then fed into a short parallel duct that forms the throat of a Mach 2.5, two-dimensional contoured nozzle (which was designed for a perfect gas flow using the method of characteristics). This nozzle in turn was directly connected to the start of the scramjet duct at the leading edge of the two-dimensional fuel injector strut. The length of the nozzle was 96 mm (throat to exit). The fuel injector leading edge is situated at the nozzle exit.

The angle of the wedges to the incoming flow was set to 25.8 deg so that the mutual reflection of the attached shocks on each wedge impinged on the wedge surfaces upstream of their trailing edges. Application of the conditions behind the incident and reflected shocks (computed with perfect gas relations) to the boundary-layer separation criteria of Korkegi[4] indicates that the shock impingement is strong enough to separate the boundary layer. This in turn would choke the flow between the wedges, which was indeed observed through pressure measurements (see the following). At lower wedge angles the reflected shocks pass beyond the trailing edge before interaction with the boundary layer, and their reduction in strength caused by the corner expansion leaves the boundary layer attached. This was observed through pressure measurements that indicated that the flow through the nozzle was not choked. Forced boundary-layer separation in turn caused inlet unstart: a normal shock moved upstream of the wedge pair, allowing excess incoming fluid to spill outside the wedges, whose inlet area (0.01 m^2) exceeded the value for which choked flow could be sustained through the nozzle throat. Thus, by choking the shock-tunnel freestream flow, a nozzle reservoir for the Mach 2.5 scramjet inlet nozzle was formed, enabling simulation of the VAG flow. To obtain a total pressure of

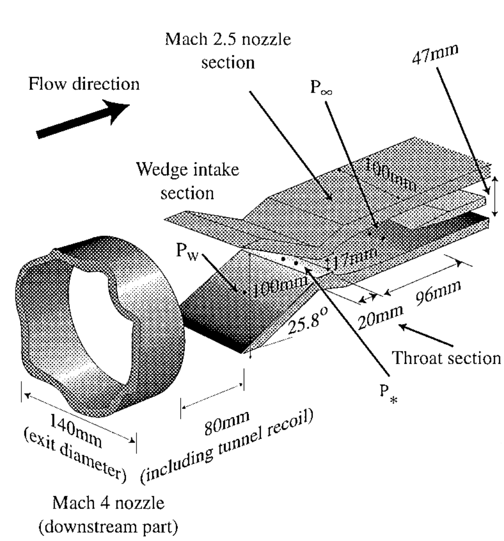

Fig. 1 Direct-connect inlet configuration downstream of the shock-tunnel Mach 4 nozzle.

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1 MPa, T4 conditions were chosen such that the Mach 4 nozzle pitot pressure equalled 1 MPa, with choking of the freestream flow providing the necessary reduction in total pressure to the level in the VAG.

Pressure transducers were used to monitor the surface pressure P_w on one of the wedges (at an area ratio of 3.25 relative to the throat), as well as the throat surface pressure P_* (one on the model centerline and one 15 mm off center, both 3.8 mm upstream of the trailing edge of the parallel throat region) and the exit surface pressure P_∞ of the Mach 2.5 nozzle (two transducers, matching the throat transducer locations). In addition, prior to fuel-on experiments, a pitot probe was installed in the center of the scramjet duct at the exit of the Mach 2.5 nozzle, replacing the injector. These pressure measurements were used to confirm that a choked nozzle flow had been established and to determine the actual flow conditions at the Mach 2.5 nozzle exit.

Figure 2a shows sample time histories for P_*, P_∞, and the exit pitot pressure P_{pitot}, normalized by P_w. P_w is theoretically 97.8% of the Mach 2.5 nozzle stagnation pressure P_0. Assuming this, the mean experimentally determined value of P_∞ / P_0 for all runs was $0.057 \pm 9\%$. Shown in Fig. 2b is the shock-tunnel nozzle reservoir pressure time history which, because of the extremely low enthalpy of the condition, was not easily tailored, and therefore slowly decays with time. It was desirable[1] to match the VAG static pressure upstream of the injector (59 kPa). The mean measured P_∞ / P_0 value was therefore used to determine the required nozzle reservoir pressure ($P_0 = 1035$ kPa). The time at which that value occurred, minus the shock-tunnel nozzle transit time (approximately 300 μs), determined the measured shock-tunnel nozzle reservoir pressure (typically 18.5 MPa). By using the equilibrium shock-tube code ESTC[5] with this measured pressure and primary shock speed (typically 1.60 km/s), the tunnel (and hence scramjet) nozzle reservoir temperature and enthalpy could be calculated.

Calculations of the Mach 2.5 nozzle flow using the quasi-one-dimensional nonequilibrium chemistry, and the equilibrium/frozen vibration nozzle expansion code NENZF[6] indicated that chemical relaxation is unimportant at these conditions. However, the choice of a vibrational excitation state (equilibrium or frozen) had a significant effect on nozzle exit conditions such as the static pressure and temperature (but not on the pitot pressure). The mean measured pitot-pressure ratio P_{pitot}/P_0 for four runs with the pitot probe in place was used in combination with NENZF to determine the mean effective area ratio $A/A^* = 2.71 \pm 0.37$ at the Mach 2.5 nozzle exit. The values of P_∞ / P_0 calculated for this area ratio for the two extremes of vibration were $P_\infty / P_0 = 0.056 \pm 0.012$ (frozen vibration) and $P_\infty / P_0 = 0.067 \pm 0.015$ (equilibrium vibration). The measured value agrees best with the frozen calculation, but also agrees with the equilibrium calculation to within experimental uncertainty. Thus, via theoretical calculations good agreement was observed between the measured values of P_∞ / P_0 and P_{pitot}/P_0, which provides confidence that the Mach 2.5 nozzle flow is behaving reasonably well. The measured throat pressures revealed transient instabilities. The off-center throat pressure was generally no more than 10% lower than the expected value of $P_* = 0.52 P_0$, but during the test time the on-center measurement typically fluctuated between the off-center value and a level up to 30% lower than expected. An unstable separation bubble on the centerline at the sharp expansion corner downstream of the throat region is possibly occurring, giving rise to the observed behavior.

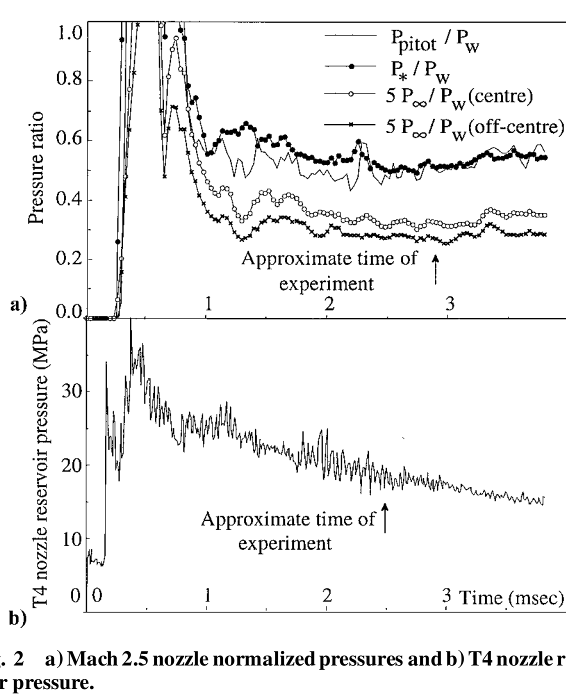

Fig. 2 a) Mach 2.5 nozzle normalized pressures and b) T4 nozzle reservoir pressure.

Table 1 T4 and VAG combustor flow conditions

Parameter	T4	VAG
P_0, kPa	$1035 \pm 5\%$ (meas.)	$1012 \pm 1\%$ (meas.)
M_∞	$2.47 \pm 6\%$	$2.44 \pm 0.2\%$
T_0, K	$2105 \pm 6\%$	$2214 \pm 2\%$
h_0, MJ/kg	$2.40 \pm 6\%$	$2.79 \pm 1\%$
P_∞, kPa	$59 \pm 9\%$ (meas.)	$59 \pm 1\%$ (meas.)
T_∞, K	$1025 \pm 13\%$	$1258 \pm 2\%$
v_∞, m/s	$1560 \pm 4\%$	$1753 \pm 1\%$

The flow conditions at the inlet to the scramjet, just upstream of the injector, are thus taken to be the mean of the calculated values for the two extremes of vibration, except for P_∞ and P_0, which are taken as the measured values. These conditions, together with the measured and calculated Mach 2.5 nozzle reservoir conditions, are summarized in Table 1. The corresponding freestream conditions in the VAG are also provided for reference. The uncertainties quoted in the table account for uncertainties in the shock tunnel and Mach 2.5 nozzle reservoir conditions, the measured pitot pressure in the nozzle, and the state of vibration. The T4 total enthalpy and temperature are somewhat lower than the VAG conditions (14 and 5%, respectively). This is a result of the chosen shock-tunnel operating conditions. At the higher primary shock speeds the behavior of the Mach 2.5 nozzle inlet was unstable, and so a slightly lower shock speed was selected.

Conclusions

Despite the very short flow duration, it is possible to conduct direct-connect combustor experiments in impulse facilities by using an unstarted converging-diverging nozzle located in the freestream flow. Such an unstarted inlet contains a very large subsonic region, which would be expected to require considerable time to establish and reach steady flow. Pressure measurements indicate that steady flow was achieved after approximately 1.5 ms of freestream flow, which at low tunnel enthalpies leaves sufficient time for meaningful combustor measurements to be made before driver gas arrival.[7] This result, which is evidence that shock tunnels are capable of establishing flowfields where large subsonic regions are influential, is important to the ground testing community.

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Reference Spray Combustion Facility for Computational Fluid Dynamics Model Validation

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Introduction

THE operation of industrial thermal oxidation systems, for example, power generation and chemical waste incineration, is generally based on *a priori* knowledge of the input stream physical and chemical properties, desired stoichiometric conditions, and monitoring of a few major species in the exhaust. The optimization of these systems is relying increasingly on computational models and simulations to provide relevant process information in a cost-effective manner. However, for computational models to be used confidently, one needs reliable data for specifying model initial/boundary conditions and experimental/numerical comparative analysis of conditions within the reactor to validate the models. Discussion of these issues is the focus of the work carried out in the AIAA Computational Fluid Dynamics (CFD) Committee on Standards.[1]

The objective of this Technical Note is to introduce the reader to a new reference spray combustion facility being used to provide benchmark experimental data with associated quantitative uncertainties for the validation of multiphase combustion models and submodels. Reference 1 distinguishes between four levels of complexity when collecting data for CFD validation. These four levels are the unit problems, benchmark cases, subsystem cases, and the complete system. The four levels increase in complexity in the order stated. The current facility corresponds to a benchmark case and, therefore, is considerably less complicated than a complete thermal oxidation system. The benchmark case does, however, contain important elements of the system, such as multiphase flow, complex transport phenomena, and chemical reaction. Also, benchmark facilities have the flexibility to provide greater diagnostic analysis of flowfield phenomena than allowable in full-scale complex systems.

The current facility configuration and baseline operating conditions were defined in direct response to collaborations with modelers and their needs. A coupled experimental/computational approach is used to involve modelers directly in the program. Sample data are presented from the baseline case, and a more extensive compilation of the experimental data is available from the authors.[2]

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Facility Design

The reference spray combustion facility, shown in Fig. 1, has evolved into a well-characterized and controlled system that can handle different 1) process liquid fuels and wastes, 2) atomizer designs, and 3) combustor configurations. A variety of state-of-the-art diagnostics are employed to characterize the input liquid stream, spray (droplet size, velocity, and number density), and exhaust emissions (particulates and chemical species). The facility permits examination of the effects of air swirl, atomizer design, fuel type, and air preheat on spray structure, combustion and emission characteristics. Experiments can be carried out under a variety of conditions, where the flow parameters (airflow rate, swirl number, and inlet air and fuel temperature) and fuel injector characteristics (fuel flow rate, spray angle, and atomizer configuration) are variable over a wide range. The unique aspect of this reference facility is the availability of advanced in-situ diagnostics that can be used to provide data for a wide range of operating conditions.

The experimental facility includes a stainless steel chamber with a variety of windows and ports for introducing both nonintrusive optical diagnostics and intrusive probes. The chamber allows for better-controlled evaluation of spray and emission data, as well as the study of different combustor configurations and heat transfer. The chamber height is 1.2 m, and the inner diameter is 0.8 m. All chamber inlet and boundary conditions are monitored to provide accurate data for modelers.

The facility includes a swirl burner with a movable 12-vane swirl cascade. The cascade is adjusted to impart the desired degree of swirl to the combustion airstream that passes through a 0.10-m-diam passage and flows around the fuel nozzle. The flow rate of the combustion air is monitored using a sonic nozzle, and a series of thermocouples are used to measure the temperatures of the reactor wall and exhaust gas. The fuel flow rate is measured with a turbine meter, and the pressurized liquid fuel is forced through an interchangeable spray nozzle. The fuel flow rate, combustion airflow rate, wall temperatures, and exiting gas temperatures are continuously monitored and stored on a personal computer.

The burner is fired upward along the vertical axis of the chamber. A stepper-motor-driven traversing system translates the entire burner/chamber assembly permitting measurements of the spray at selected locations downstream of the nozzle. Note that the reactor exit is off-axis, which permits insertion from the ceiling of thermocouples and gas-sampling probes facing directly into the flame, but also makes the problem nonaxisymmetric. Flexibility of the facility will allow for modification to the axisymmetric configuration. The relevant dimensions necessary for modeling the multiphase combustion within the facility are presented in Fig. 2. Additional details on the design of the burner are available in Ref. 2.

The reacting fuel spray is characterized with a two-component phase Doppler interferometer (PDI). Information provided by the system includes statistics on the droplet size, axial and radial components of velocity, number density, and mass flux. Air velocity measurements, that is, mean and rms values, are also carried out using the PDI by adding seed particles to the combustion air or by using hot-wire probes. Gas-phase species concentrations are measured using Fourier transform infrared (FTIR) spectroscopy. An FTIR spectrometer equipped with a deuterated triglycene sulfate detector is used for extractive sampling of chemical species in the combustor emissions. A gas-sampling system, consisting of an air-cooled sampling probe, a heated gas line, and a vacuum pump, facilitates the transport of the sample gas extracted from the spray combustor into the single-pass gas cell in a continuous manner. The sampling probe was designed to aerodynamically quench chemical reactions occurring within the gases being sampled. The sampling gas line was also provided with a means for purging. Typically, the gas-sampling probe is inserted into the exhaust gas stream in such a way to probe the conditions at the selected exit plane (see Fig. 1). The extracted gas samples are analyzed with the FTIR spectrometer, and