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Techniques for detailed heat transfer measurements in cold supersonic blowdown tunnels using thermochromic liquid crystals

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Abstract

The paper presents methods for measurement of convective heat transfer distributions in a "cold flow" supersonic blowdown wind tunnel. The techniques involve use of the difference between model surface temperature and adiabatic wall temperature as the driving temperature difference for heat transfer and no active heating or cooling of the test gas or model is required. Thermochromic liquid crystals are used for surface temperature indication and results presented from experiments in a Mach 3 flow indicate that measurements of the surface heat transfer distribution under swept shock wave boundary layer interactions can be made.

1. Introduction

Vehicles travelling at supersonic speeds are subjected to high thermal loads that can be most severe in regions of shock wave boundary layer interactions. Measurement of heat transfer at such conditions in ground-based test facilities usually requires that the test gas (or the test model) be heated in order to provide a driving temperature difference between the model and the gas for heat transfer to occur. For example, pebble-bed heaters [1] are used in supersonic and hypersonic blowdown wind tunnels to raise the temperature of the test gas to stagnation temperatures up to 1700 K. Impulse facilities, such as light-piston tunnels [2], gun tunnels [3] and shock tunnels [4], can also be used to raise the temperature of the test gas for heat transfer testing. Lee et al. [5] and Martinez-Botas et al. [6] used different techniques to heat models prior to arrival of the test flow in blowdown tunnels.

In a supersonic blowdown tunnel, in which the test gas is unheated, a small temperature difference for driving heat transfer already exists. This is because the adiabatic wall temperature will usually be lower than the total temperature of the test gas (due to the fact that the recovery factor is not unity and because of the temperature drop across the main valve of the tunnel due to the Joule–Thompson effect). In a transient test in such a facility, in which both the test gas contained in the reservoir and the test model are initially at ambient temperature, the surface temperature of the model will decrease during a test. The surface temperature is driven towards the adiabatic wall temperature and the rate of approach will depend on the local heat transfer rate. At any position on the model, the heat transfer rate to the surface, q, can be written in terms of the local convective heat transfer coefficient, h, and the difference between the local adiabatic wall temperature, T_{aw}, and the surface temperature, T_s, according to

$$q = h(T_{aw} - T_s). \tag{1}$$

(This assumes that radiative heat transfer is small.) By monitoring the time history of surface temperature, it is possible to determine the local heat transfer rate using conventional analysis, assuming minimal lateral conduction and a semi-infinite substrate (e.g., Schultz and Jones [7]).

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Nomenclature

c	specific heat
f	see Eq. (5)
g	see Eq. (8)
G	see Eq. (14)
h	heat transfer coefficient
k	thermal conductivity
M	Mach number
p	pressure
q	heat transfer rate
r	recovery factor
St	Stanton number
t	time
T	temperature
u	flow speed
x, y	spatial coordinates
δt	time step
γ	ratio of specific heats
ρ	density
τ	integration variable
Ω	see Eq. (7)

Subscripts

0	stagnation
∞	freestream static
aw	adiabatic wall
P	constant pressure
ref	reference location
s	surface

While this "passive" technique for measuring heat transfer has been recognized in the past, it has usually been dismissed. This is because the driving temperature differences for heat transfer will be small and the accuracy of measurement of heat transfer rates will depend on the accuracy with which the small temperature differences can be measured. Preliminary tests at Mach 2 showed that this technique produced promising results [8,9] and experiments at Mach 3 using thin-film resistance thermometers indicate that it is possible to make accurate heat transfer measurements in a supersonic blowdown tunnel without heating or cooling the test gas or model [10].

In this paper we investigate the possibility of full-surface heat transfer measurement using only the small natural temperature difference between T_{aw} and T_s to drive the heat transfer and thermochromic liquid crystals (TLCs) for surface temperature indication. TLCs change colour with temperature so the distribution of temperature on a test surface coated with a layer of TLC can be inferred from the colour of the layer. The range of temperatures over which a liquid crystal goes from colourless, through the visible spectrum of colours and back to colourless is referred to as the temperature band of the TLC. Around room temperature TLCs are available with bands as small as 1 °C (narrow band) and up to around 20 °C (broad band). In this paper we use only narrow-band TLCs.

Results from a series of experiments in a Mach 3 flow in The University of Queensland (UQ) supersonic blowdown tunnel are presented. The motivation is to determine whether a conventional, transient, supersonic blowdown wind tunnel could be used to obtain qualitative and/or quantitative measurements of the complete heat-transfer distribution over a surface without actively heating or cooling the model or test gas. This could potentially increase the utility of such facilities to routine heat transfer as well as aerodynamic testing.

The use of TLCs for surface temperature indication leads to the possibility that detailed heat transfer distributions may be obtained in a single test. TLCs have been used in a range of aerothermal tests since the original work of Klein [11]. They have been found to be an excellent means of measuring the surface temperature in heat transfer experiments and have the advantage of giving surface temperature information over the complete test surface. Early disadvantages of the technique have been overcome with improved stability of materials and data processing methods. Micro-encapsulated crystals exhibit a reversible, repeatable response and, for the most part, are insensitive to effects of pressure and surface shear.

Recent examples of the use of TLCs include transient tests in which the gas temperature changes suddenly and the heating of the surface is monitored [12] and where the model is pre-heated prior to a test [6]. The surface can also be actively heated with the flow on until a steady state is achieved and the TLCs indicate contours of constant surface temperature [13]. In compressible flows, Klein [11] allowed the model to reach its adiabatic wall temperature and used TLCs to indicate the distribution of T_{aw}. The test gas has also been heated in a blowdown facility [14] and in compression-heated, short-duration facilities [2,15]. In some cases, accurate distributions of heat transfer are obtained while in others only qualitative information about heat-transfer distributions is claimed.

The potential of the technique presented in this paper is that it could be used in many conventional supersonic, transient blowdown or suckdown wind tunnels to obtain qualitative indication of hot and cold spots on a given model. The liquid-crystal technique has the advantage over discrete instrumentation, such as thin-film gauges, that it gives information about the heat transfer distribution over an area rather than at discrete points. This enables higher resolution in the variations in heat transfer to be obtained. There are other techniques for full-field thermal imaging such as thermal paints [16], infrared thermography [17] and thermographic phosphors [18]. The liquid-crystal technique has advantages over these techniques in that only illumination in the visible spectrum is required, images can be recorded on video tape without need for special optics or filters and the colour changes in the liquid-crystal layer are reversible and repeatable.

Further, the technique may be suitable for making detailed quantitative heat transfer measurements in intermittent supersonic wind tunnels.

The aim of this paper is to investigate the suitability of using TLCs to obtain both qualitative and quantitative information about heat transfer in cold supersonic blowdown tunnels. The background theory is presented in Section 2. For conditions typical of those in a supersonic blowdown wind tunnel, simulations of the surface temperature time histories that would be obtained on a model in the test section are reported in Section 3. It is shown how such information could be used to infer both Stanton number and recovery factor on a model but that, under some circumstances, it may be difficult to determine both. Experiments made in a supersonic blowdown wind tunnel using narrow-band TLCs are detailed in Section 4. In Section 5 results from the tests are used to show the type of information that can be obtained using one of the proposed techniques.

2. Theory

Techniques for analysis of transient surface temperature measurements to infer heat transfer rates [7,19] are well established and are used routinely in impulse facilities such as shock tunnels, gun tunnels and light-piston tunnels [4,15,2]. They have also been applied in transient transonic and supersonic blowdown tunnels [6,10].

Using a one-dimensional semi-infinite model for heat transfer, the heat transfer rate can be written in terms of the surface temperature time history as [7],

$$q(t) = \frac{1}{2}\sqrt{\frac{\rho c k}{\pi}} \int_0^\infty \frac{T_s(\tau)}{(t-\tau)^{3/2}}\,d\tau, \quad (2)$$

where ρ, c and k are the density, specific heat and thermal conductivity of the substrate. Assuming that the heat transfer is due to convection only, the heat transfer rate can be expressed in terms of the heat transfer coefficient and the driving temperature difference as in Eq. (1).

In order to generalize a wind tunnel measurement of heat transfer rate to flight conditions (at the same Mach and Reynolds numbers) the heat transfer is expressed in terms of a Stanton number,

$$St = \frac{h}{\rho_\infty u_\infty c_p}, \quad (3)$$

where ρ and u are the freestream density and speed and c_p is the specific heat of the test gas. Thus, in order to obtain an instantaneous value of the Stanton number during a wind tunnel test, it is necessary to measure q, T_{aw}, T_s, ρ_∞ and u_∞. The adiabatic wall temperature is the parameter that is most difficult to determine. From a measurement of the total temperature, T_0, and by assuming a value for the recovery factor, r, it is possible to obtain a value for T_{aw}. An alternative method for inferring T_{aw} from the time history of heat transfer rate, surface temperature and total temperature by extrapolation on measured data is described below.

The adiabatic wall temperature can be written in terms of the total temperature, Mach number, M, and recovery factor as

$$T_{aw} = T_0 \frac{1 + r\frac{\gamma-1}{2}M^2}{1 + \frac{\gamma-1}{2}M^2}, \quad (4)$$

where γ is the ratio of specific heats. If the recovery factor is constant during the test, this can be written as

$$T_{aw} = fT_0, \quad (5)$$

where f is a constant for any position on the test surface. From (1), (3) and (5),

$$\Omega = fSt - St\frac{T_s}{T_0}, \quad (6)$$

where

$$\Omega = \frac{q}{\rho_\infty u_\infty c_p T_0}. \quad (7)$$

If the heat transfer rate varies during a test (due to changing surface temperature and/or changing total temperature), by plotting Ω vs T_s/T_0, it is possible, in principle, to infer both the Stanton number (from the slope of the curve) and the factor, f, (by extrapolating to zero heat transfer rate). Similar techniques to determine adiabatic wall temperatures have been used in transient tests [6] and by varying the driving temperature for heat transfer in a series of tunnel runs [5,20].

Thus, from records of the surface temperature time history at a number of points on the surface of the model, the Stanton number and adiabatic wall temperature can be determined. The accuracy of this technique in a run of a transient facility will be a function of the range of values of T_s/T_0 covered during the test. If the surface temperature does not reach the adiabatic wall temperature during the test, extrapolation of a curve fitted through the data will be required to find f (see Sections 3 and 4).

A broad-band TLC could be used to measure the time-history of surface temperature on the test surface by suitable selection of the temperature range for colour play of the TLC. The technique described above could then be used to determine the adiabatic wall temperature and Stanton number distributions. It is also of interest to determine if a narrow-band TLC could be used to obtain contours of surface temperature at different points in time and if the time taken for the surface to reach the temperature of the TLC could be used to infer St. The analysis will not be the same as the curve-fitting technique described above because a narrow-band TLC will give the surface temperature at a point on the surface at only one point in time (as the test model cools). It is possible to mix two narrow-band TLCs to obtain the surface temperature at two points in time and this is considered below.

The proposed analysis involves measurement of the complete time-history of surface temperature at one point on the test surface (using a thermocouple, RTD or thin-film gauge). From this time history and other tunnel measurements, the time history of Stanton number at this gauge location, St_{ref}, can be determined as described above. If the flow establishes quickly, this should be close to a step from zero before the test to a constant level dictated by flow conditions during the test. In a conventional blowdown tunnel, in which the test gas is supplied from a reservoir, the temperature of the gas in the reservoir may decrease as the pressure in the reservoir decreases. Therefore, consider the possibility that the tunnel stagnation temperature is not constant during the test. This will lead to variations in the freestream conditions and, consequently, also in the Stanton number during the period of nominally steady flow. For the analysis, it is assumed that the time-history, but not the level, of Stanton number at all points on the test surface is similar. This assumption implies that the heat transfer coefficient at any point on the surface is constant in time so that the Stanton number varies only with variations in the freestream flow conditions. This is reasonable for cases with a rapid flow start and only relatively small changes in stagnation temperature. An exception is in regions of transitional flow where intermittency of turbulence can lead to the heat transfer coefficient switching between laminar and turbulent levels. We write,

$$St(x,y,t) = g(x,y)St_{\mathrm{ref}}(t), \tag{8}$$

where g is a constant for any location on the test surface.

For a given stagnation temperature time-history, Stanton number time-history and initial model surface temperature, the complete surface temperature time history can be predicted for other locations on the test surface, assuming, as above, 1D conduction. Taking samples at constant time steps, Schultz and Jones [7] show that the local heat transfer rate at step n can be related to the surface temperature by

$$q_n = 2\sqrt{\frac{\rho ck}{\pi}}\sum_{i=2}^{n}\frac{T_{s_i} - T_{s_{i-1}}}{\sqrt{t-t_i}+\sqrt{t-t_{i-1}}} \tag{9}$$

and, from (1), (3), (5) and (8),

$$q_n = gSt_{\mathrm{ref}_n}\rho_{\infty_n}u_{\infty_n}c_{\mathrm{p}}(fT_{0_n} - T_{s_n}) \tag{10}$$

For constant time steps, δt, Eqs. (9) and (10) can be equated and the result rearranged to give the surface temperature at time step n as

$$T_{s_n} = \frac{T_{s_{n-1}} + fB_nT_{0_n} - \sum_{i=2}^{n-1}\frac{T_{s_i}-T_{s_{i-1}}}{\sqrt{n-i}+\sqrt{n-i+1}}}{1+B_n}, \tag{11}$$

where

$$B_n = \frac{1}{2}\sqrt{\frac{\pi\delta t}{\rho ck}}gSt_{\mathrm{ref}_n}\rho_{\infty_n}u_{\infty_n}c_{\mathrm{p}}. \tag{12}$$

Using Eqs. (11) and (12) the surface temperature time history can be predicted for any combination of f and g. The mixture of two TLCs will give the surface temperature at each point on the surface at two points in time and the times at which the TLCs change colour will depend on the f and g combination at each point. Thus Eq. (11) could be used to solve for two unknowns, f and g, given the surface temperature at two points in time. The suitability of this procedure is addressed in Section 3.

Note that if the recovery factor is assumed to be constant over the surface (and thus a value for f is assumed), a single measurement of the time at which the surface temperature reaches a given level could be used to obtain an indication of the level of Stanton number. That is, a single TLC could be used to obtain the distribution of St or a mixture of TLCs could be used to obtain more than one indication of the distribution. It may also be possible to obtain further indications of the Stanton number and adiabatic wall temperature by monitoring the warming of the model after the test flow has stopped. Initial analysis indicates that the latter is possible but it is not discussed further in this paper.

3. Simulation

Simulations of the techniques for measuring heat transfer described in Section 2 have been made to determine the suitability of the present methods. A turbulent boundary layer formed on a flat plate in a Mach 3 flow is considered. The Stanton number is set to be 1×10^{-3}. We consider a perspex substrate ($\sqrt{\rho ck} = 570$ J m^{-2} K^{-1} s$^{-1/2}$) at an initial temperature of 295 K in a flow of air with a total temperature also at 295 K and a total pressure of 700 kPa. These conditions were chosen

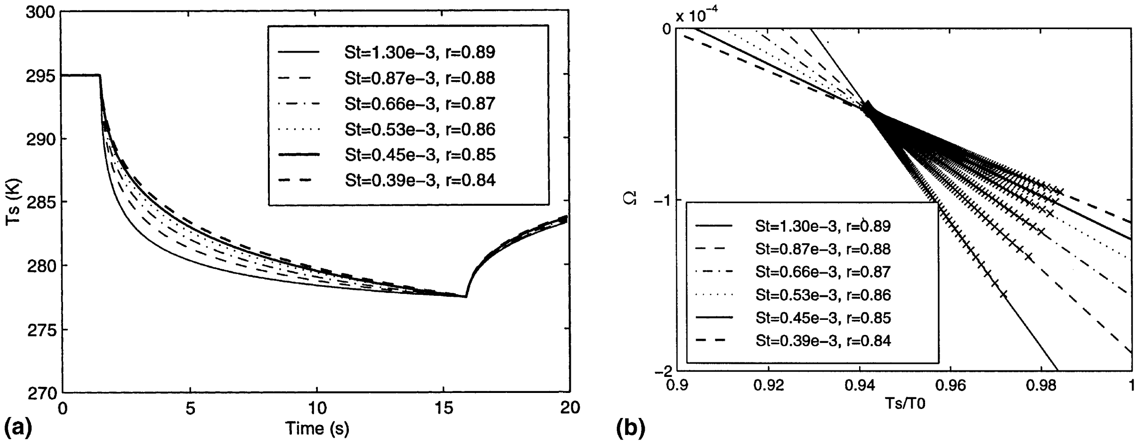

Fig. 1. Simulation results for $p_0 = 700$ kPa, $M = 3.0$, $T_0 = 300$ K (constant) for different St and r combinations: (a) surface temperatures; (b) curve-fitting results to obtain St (slope) and r (intercept).

because they are typical of those obtained in tests in the UQ blowdown wind tunnel (see Section 4). The total temperature of the flow is assumed to remain constant during each simulated test so that the adiabatic wall temperature does not change. The heat transfer coefficient is also kept constant during each simulation and the time history of surface temperature is predicted.

It is possible to obtain a given overall change in surface temperature during a test with different combinations of St and r. This can be seen by combining Eqs. (1), (3) and (5),

$$q = \rho_\infty u_\infty c_p St(fT_0 - T_s).$$

For constant flow conditions, ρ_∞, u_∞ and c_p remain constant and the same heat transfer rate can be achieved with either a low Stanton number and a low recovery factor [1] or a high Stanton number and a high recovery factor. This raises the question of whether it would be possible to infer both St and r from a measured surface temperature time history.

Results of simulations for a series of cases with similar overall drops in surface temperature during the test but for differing Stanton numbers and recovery factor combinations are presented in Fig. 1. For the combination of high Stanton number and high recovery factor, the surface temperature decreases more rapidly in the initial period, with the rate decreasing as the surface temperature approaches the adiabatic wall temperature. For the case of low Stanton number and low recovery factor, the rate at which the surface temperature decreases is more uniform. Consider the proposal in Section 2 that two narrow-band TLCs be used to determine the surface temperature at a point on the test surface at two points in time. For the present simulation, from the times at which the surface temperature was at two levels, say, 10 °C and 15 °C below the initial temperature, it would be possible to determine the unique St and r combination which would cause the temperature vs time curve to pass through the two points.

The curve-fitting and extrapolation technique for determining the Stanton number and recovery factor (described above in relation to Eq. (6)) can also be demonstrated using these simulation data. Data during the test period of the simulation are plotted in Fig. 1(b) for each case. It can be seen that different Stanton numbers give different slopes and that different recovery factors lead to different intercepts – lower values of T_s/T_0 occurring for zero heat transfer ($\Omega = 0$) as the recovery factor drops from 0.89 to 0.84. These data lead to accurate values of St and r by using the slopes to find St and extrapolation to zero Ω to find r.

In many blowdown tunnels, the total temperature of the test gas decreases as the air is drained from the reservoir. In the UQ blowdown tunnel, to a first approximation, the temperature drops linearly with time at a rate of about 0.8 °C/s (see Fig. 4). Another simulation has been performed for conditions similar to those in Fig. 1 but with the stagnation temperature dropping at 0.8 °C/s. Results are shown in Fig. 2. Again, results are shown for different St and r combinations that produce similar overall surface temperature drops. In contrast to the results for constant T_0 (Fig. 1(a)), it can be seen that there are smaller differences between the shapes of the surface temperature time histories for linearly decreasing T_0. Thus, very similar T_s time histories can be obtained for different St and r combinations (which produce very similar heat transfer rates).

Curve fitting this simulation data to determine St and r is shown in Fig. 2(b). It can be seen that the curves are still distinct but there is a smaller range of Ω covered

[1] Note from Eqs. (4) and (5) that f varies monotonically with recovery factor.

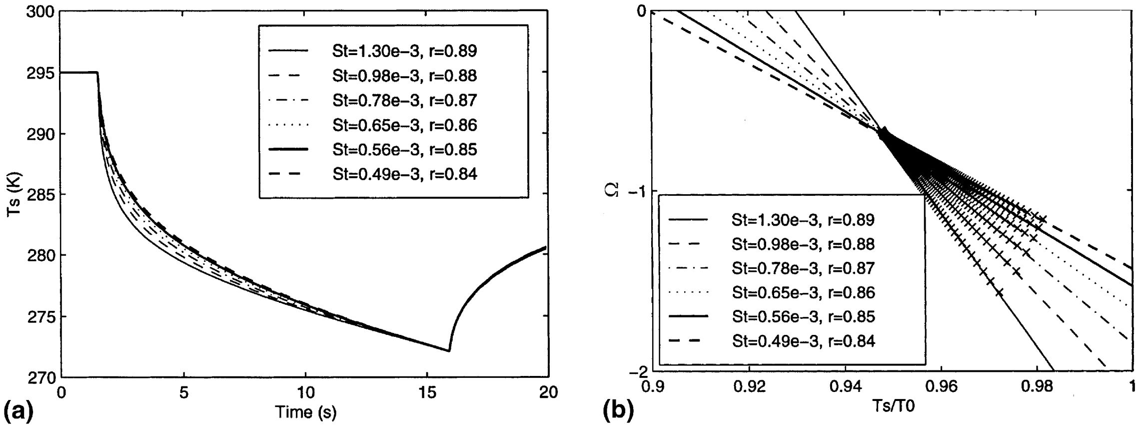

Fig. 2. Simulation results for $p_0 = 700$ kPa, $M = 3.0$, T_0 starting at 300 K and dropping at 0.8 K/s for different St and r combinations: (a) surface temperatures; (b) curve-fitting results to obtain St (slope) and r (intercept).

than when the stagnation temperature remains constant. Also Ω remains further from zero compared with results when T_0 is constant. This will lead to greater uncertainty in r because more extrapolation will be required to find it. Also the largest difference between the curves occurs at the start of the test when the magnitude of Ω is largest. This makes the determination of both Stanton number and recovery factor quite sensitive to errors in measurements, particularly since a large extrapolation is required to find r.

In summary, these results indicate that it is possible in principle to determine both Stanton number and recovery factor in a single test from two measurements of surface temperature. However, for the present tests, the combination of dropping total temperature and the expected level of Stanton number would lead to large uncertainties in heat transfer quantities. Consequently, a recovery factor will be assumed in the experiments reported below in order to determine the Stanton number from the tunnel results.

4. Experiments

The experiments were performed in the supersonic blowdown tunnel at The University of Queensland. This is a small facility with a 115 mm × 100 mm test section. A Mach 3 nozzle was installed for the present tests. Typical conditions for the tests were 650 kPa stagnation pressure and 300 K starting total temperature, giving a unit Reynolds number in the test section of 5.2×10^7 m^{-1}. The total pressure and total temperature of the flow were measured in a plenum chamber ahead of the nozzle. The flow Mach number is based on the measured total pressure and a measurement of the static pressure taken on the side wall of the tunnel just ahead of the model. Surface flow visualizations were made using an oil film composed of kerosene, titanium dioxide and oleic acid. In order to calibrate the TLCs and to obtain the complete surface temperature time history, a fast-response, foil, type-T thermocouple was attached to the measurement surface 65 mm from the leading edge. Data capture was performed using an A/D card in a PC at a sample rate of approximately 10 samples per second.

A flat plate aligned with the flow direction and with a sharp leading edge formed the test surface on which heat transfer measurements were made. The plate spanned the test section and was immersed 15 mm into the flow to generate a new boundary layer and to bleed the boundary layer formed on the upper nozzle block above the test surface (see Fig. 3).

Small, sharp fins were placed normal to the test surface and at angles of 8° to the oncoming flow to generate shocks that sweep across the surface and intersect. These shocks interact with the boundary layer formed on the test surface. This geometry is representative of an inlet for a scramjet and has been the subject of many investigations (e.g., [21,22]). Heat transfer characteristics in such interactions have proven difficult to predict [23,24]. In the present tests the thermocouple was located upstream of any influence of the shocks and thus gave a measurement of the undisturbed heat transfer to the plate.

The test plate was made of 15 mm thick clear perspex. Transient thermal conduction analysis indicates that the semi-infinite substrate assumption is suitable with this thickness of perspex for the present conditions. The top wall of the test section was modified to include a window of 50 mm thick perspex (see Fig. 3). The liquid-crystal colour play was viewed through the window and recorded using a video camcorder located outside the tunnel, above the test section. Fluorescent lights located above the window provided illumination.

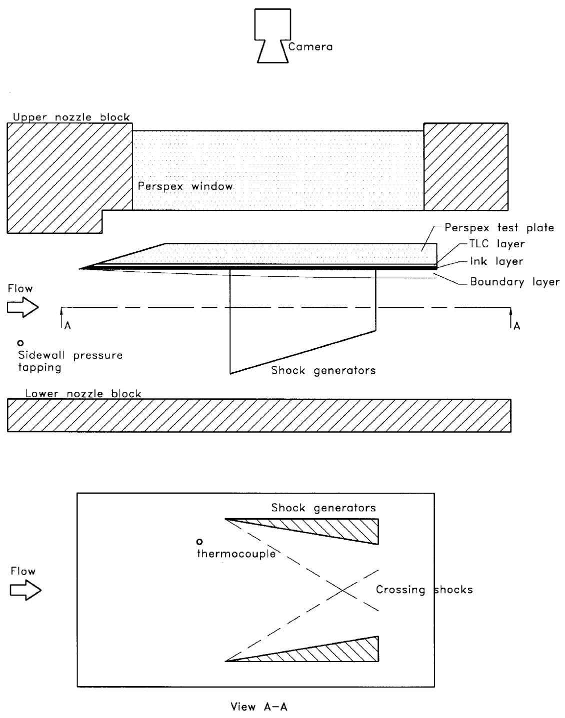

Fig. 3. Schematic of the test section of the tunnel showing the test plate.

Two, narrow-band, encapsulated liquid crystals (supplied by Halcrest, Poole, UK) with colour plays occurring over approximately 1 °C at nominal temperatures of 10 °C (TLC_1) and 5 °C (TLC_2) and were mixed with water and an aqueous binder. The surface of the test plate was sprayed with about 10 thin coats. The surface was then coated with a black, water-based ink to provide a mat-black background against which the TLC colour play could be viewed.

An LED was used to provide a timing signal to synchronise the video and computer-recorded data. A voltage was applied to the LED about one second before the tunnel was run using a switch activated by the tunnel operator. This signal was used to trigger the data acquisition program and the activation of the LED was visible on the video recording. The two frames between which the LED was activated could be clearly identified. An LED timer was used to record the elapsed time on the video record.

5. Results

Data from a test are presented in Fig. 4. It can be seen that the tunnel is started and stopped rapidly, that the stagnation temperature decreases as the run proceeds and that the surface temperature measured at the gauge location continuously decreases during the test. After the test the surface temperature recovers towards ambient and the temperature of the gas in the tunnel plenum chamber remains below ambient.

The method used here to infer the heat transfer rates from the surface temperature data is a deconvolution technique [25]. Noting that the expression for the variation in surface temperature as a function of the heat transfer rate can be written as a convolution integral of the form

$$T_s = \int_0^t G(\tau)q(t-\tau)\,d\tau, \tag{13}$$

where

$$G(t) = \sqrt{t-\tau}, \tag{14}$$

standard deconvolution techniques employing FFTs (e.g., [26]) can be used to obtain the heat transfer rates. The heat transfer rate (Fig. 4(c)), negative because heat is transferred from the model to the flow, reflects the difference between the adiabatic wall temperature and the surface temperature (Fig. 4(b)) – the heat transfer rate is initially high and slowly increases again towards the end of the test as the stagnation temperature begins to drop more quickly. After the test, the heat transfer rate drops to zero. During some tests large fluctuations in the measured heat transfer time history were measured. This was attributed to boundary layer transition effects since the thermocouple was located not far downstream of the expected transition location. Tests in which such fluctuations were observed were not used in further analysis.

Fig. 4(d) shows the data during the test period plotted as Ω vs T_s/T_0. The curve-fit gives a Stanton number of $0.00133 \pm 5\%$ and a recovery factor of $0.894 \pm 0.2\%$. The Stanton number is close to the predicted value of 0.00127 for these conditions using a reference temperature method and the recovery factor is in agreement with that expected for a turbulent boundary layer (0.89) at the location of the thermocouple. In some tests the curve-fit was not as good and over 15 tests the Stanton number was 0.00127 ± 0.00050 and the recovery factor was 0.89 ± 0.016 (the errors representing 95% confidence intervals on the scatter). If a value for recovery factor for all tests was assumed to be 0.89, the Stanton number could also be averaged during the steady flow period for a test (see Fig. 4(e)). This gave an average Stanton number over the 15 tests of 0.00131 ± 0.00017.

We turn now to the results obtained using the narrow-band TLCs. Since two TLCs were mixed together before being applied to the test surface, assuming a constant value for recovery factor enables two estimates of the Stanton number distribution to be obtained. For the flat plate test surface the Stanton number is expected to vary due to boundary layer growth but by less than

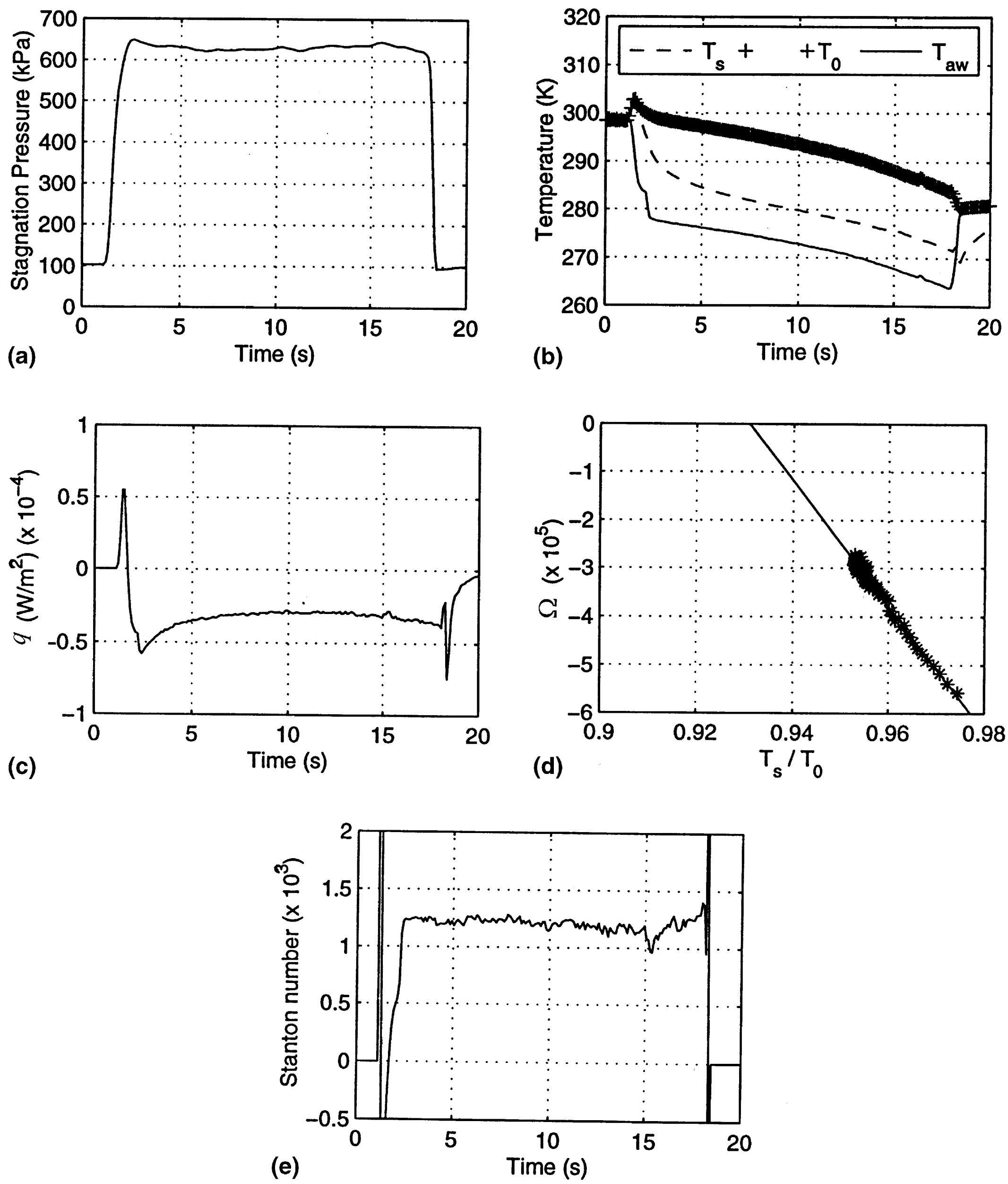

Fig. 4. Experimental data from a test in the UQ blowdown tunnel: (a) p_0; (b) T_s, T_0, and T_{aw}; (c) q; (d) curve fitting; (e) St.

±10% over the measurement region. The colour change times for each of the two TLCs for the flat test plate with no shock generating fins indicate a variation of Stanton number within this range.

In order to determine the Stanton number from the TLC images, the following procedure was used. The TLCs were calibrated for each test by noting the times at which the TLC contour crossed the location of the thermocouple and noting the temperature indication from the thermocouple at those times. For a range of Stanton number levels spanning the expected range for the experiment, surface temperature time histories were predicted using Eq. (11). For each Stanton number level and using the calibrated TLC temperatures, the times at which the surface temperature should cross the TLC levels were identified. An example of the result of this procedure is shown in Fig. 5. A polynomial was fitted through the data and this was used to interpolate to find

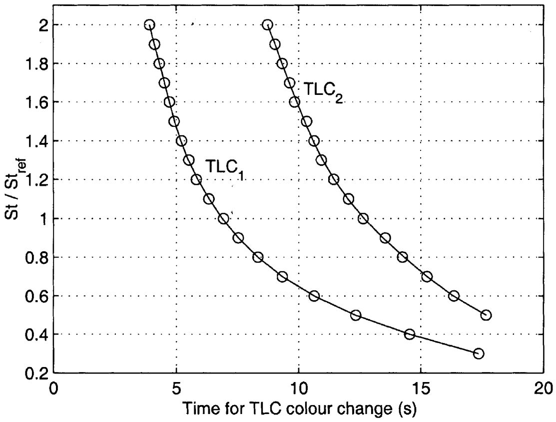

Fig. 5. Experimental calibration of the enhanced Stanton number from the time taken for the surface to reach the colour change temperature of the higher (TLC_1) and lower (TLC_2) temperature liquid crystals. Circles indicate data points and lines are polynomial fits.

the Stanton number, given the time at which the TLC contour was recorded.

The plot of Stanton number vs time for TLC contours (e.g., Fig. 5) gives an indication of the sensitivity of inferred heat transfer rate to uncertainties in timing. For example, in Fig. 5 the high slope of the *St* vs time curve for TLC_1 at high levels of *St* indicates that any error in the measured time for colour change will lead to a large error in *St*. Thus, synchronisation of the image record and data record is very important. At lower heat transfer levels, the sensitivity is reduced. The lower temperature coating, TLC_2, should be more accurate for higher heat transfer levels. The response time for TLCs should be suitable for the present tests based on measurements of response times of several milliseconds made by Ireland and Jones [27] for TLC coatings typical of the present tests.

The measured Stanton number levels for the colliding shock interaction are shown in Fig. 6. The corresponding limiting streamline pattern is shown in Fig. 7. These heat transfer results are averaged over five tests and the contours shown are Stanton number levels normalised with the Stanton number measured at the thermocouple location. The shock generators are shown in black and dashed lines show the locations of the shocks. Reduced heat transfer levels can be seen in the vicinity of shock intersection and high levels were measured close to the shock generators. There is evidence that there may have been leakage between the wedges and the test plate for the heat transfer tests not present in the visualization test. Near the tips of the wedges the extent of spanwise influence is larger for the heat transfer results than in the flow visualization (note the locations of the 0.85 and 0.95 contours in Fig. 6).

There are limited experimental data available on the heat transfer distribution in colliding shock interactions. Some recent results from a study conducted in Russia are presented in Gnedin et al. [24] and Zheltovodov and Maksimov [28]. Gnedin et al. [24] present measured and computed heat transfer data for an interaction produced by two opposing 7° wedges in a Mach 3.95 flow. While the wedge angle is smaller and the Mach number higher than in the present study, it is worthwhile comparing the trends in the results. In [24] the heat transfer rates were measured using discrete sensors working on a calorimeter principle. The experimental data are thus limited in area covered and resolution compared with the present results. Gnedin et al. quote an experimental uncertainty in *St* of ±10 to 15%.

The results in Gnedin et al. show increasing Stanton numbers towards the wedges, as observed in the present

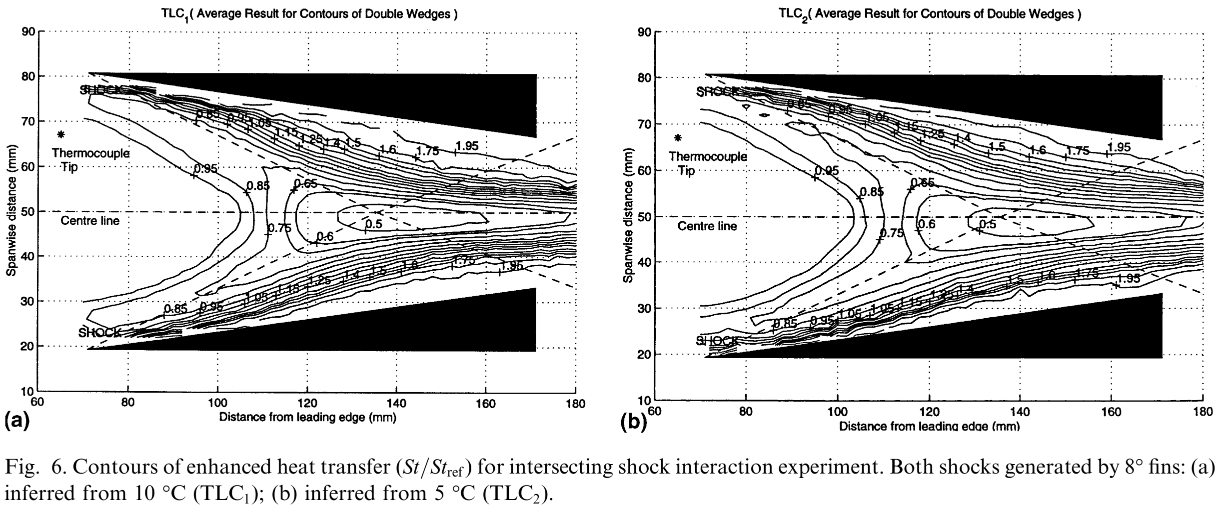

Fig. 6. Contours of enhanced heat transfer (St/St_{ref}) for intersecting shock interaction experiment. Both shocks generated by 8° fins: (a) inferred from 10 °C (TLC_1); (b) inferred from 5 °C (TLC_2).

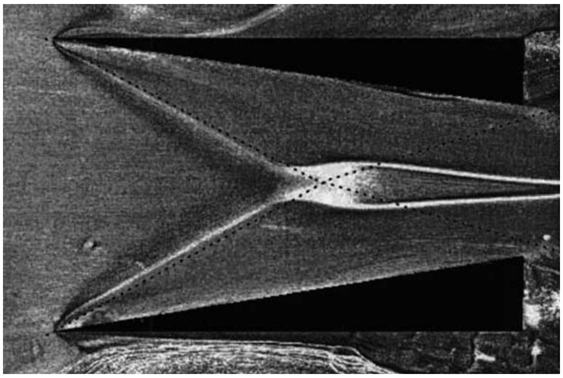

Fig. 7. Limiting streamline pattern for intersecting shock interaction experiment.

experiments. Over the region where results are available, their computed Stanton numbers agree with the experimentally measured values typically to within about 25%. The computations show the peak St occurring closest to the wedges at a level approximately twice the level of the undisturbed flow upstream of the interaction. A similar peak St enhancement was measured close to the wedges in the present tests.

One difference between the present results and those in [24] is in the region where the shocks intersect. Near the point where the shocks cross the Stanton number in the present experiments drops to about half the undisturbed upstream level. Results in [24] show a local minimum St level there but the level is only several percent lower than the undisturbed upstream value. This difference may be associated with the assumption in the present work that the ratio of adiabatic wall temperature to stagnation temperature (f of Eq. (5)) is constant over the interaction region. Results in [24] suggest that T_{aw} increases near shock intersection. If a higher value of f was used in analysis of the data in this region a higher value of St would be obtained. In fact if the data near shock intersection for the present conditions are processed assuming a recovery factor of 0.91 rather than 0.89, the ratio of St/St_{ref} in that region increases from 0.5 to 0.75. This demonstrates the sensitivity of Stanton number to the assumed value for r for a cold flow supersonic blowdown tunnel. It also shows the limitations of assuming a constant recovery factor to infer Stanton numbers in such flows.

Overall the results indicate that the present technique can be used for qualitative indication of heat transfer distributions in cold supersonic blowdown tunnels. The accuracy of heat transfer enhancement measurement using TLCs is estimated to be $\pm15\%$ when results are averaged over about five tests and a value for the recovery factor is assumed. The major contributor to this uncertainty is the accuracy of the temperature measurement using the TLC. The uncertainty in temperatures obtained from the TLCs in the current experiments is estimated to be ±0.25 K. The uncertainty in heat transfer enhancement resulting from this is highest where heat transfer is highest and the surface temperature of the plate changes most rapidly. For the lower temperature TLC at the highest heat transfer rates, the uncertainty in heat transfer enhancement due to the uncertainty in calibration can be as high as $\pm20\%$. With improved TLC calibration it should be possible to reduce the overall uncertainty in this quantity. Also the use of broad band rather than narrow-band liquid crystals could potentially make the measurement of recovery factor as well as Stanton number possible and remove the restrictive assumption of constant recovery factor. A broad-band TLC would enable the complete time history of temperature at each point on the surface to be obtained and the curve-fitting technique described in Section 3 could be used to find St and r. Tests on this technique are currently being made.

6. Conclusions

The use of a cold supersonic blowdown wind tunnel for heat transfer measurements has been investigated. Analysis of surface temperature signals indicates that it is possible to obtain recovery temperature and Stanton number from a single test. A decreasing total temperature during a test makes the identification of both recovery temperature and Stanton number more susceptible to errors. Experiments in a small blowdown tunnel at Mach 3 show that thermochromic liquid crystals can be used in such a flow to obtain a qualitative indication of Stanton number distribution. Quantitative measurements with an estimated accuracy of $\pm15\%$ have been made but with improved calibration of the TLC this could be reduced further.

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References

[1] H.T. Nagamatsu, Hypersonic experimental facilities, in: F.E. Goddard (Ed.), High Speed Problems of Aircraft and Experimental Methods, Oxford University Press, London, 1961.

[2] G.T. Roberts, P.H. Schuricht, N.R. Mudford, Heating enhancement caused by a transverse control jet in hypersonic flow, Shock Waves 8 (2) (1988) 105–112.

[3] D. Kumar, J.L. Stollery, Hypersonic control flap effectiveness, Aeronaut. J. 100 (996) (1996) 197–208.
[4] D.J. Mee, C.P. Goyne, Turbulent spots in boundary layers in a free-piston shock-tunnel flow, Shock Waves 6 (6) (1996) 337–343.
[5] Y. Lee, G.S. Settles, C.C. Horstman, Heat transfer measurements and computations of swept-shock-wave/boundary-layer interactions, AIAA J. 32 (4) (1994) 726–734.
[6] R. Martinez-Botas, G.D. Lock, T.V. Jones, Heat transfer measurements in an annular cascade of transonic gas-turbine blades using the transient liquid crystal technique, J. Turbomach. 117 (1995) 425–431.
[7] D.L. Schultz, T.V. Jones, Heat-transfer measurements in short-duration hypersonic facilities, AGARDograph 165 (1973).
[8] D.J. Mee, P.T. Ireland, Full surface heat transfer measurements in supersonic flows using thermochromic liquid crystals, presented at the 1995 Scramjet Workshop, The University of Queensland, 18–21 September 1995.
[9] D.J. Mee, P.T. Ireland, Heat transfer measurements in a cold supersonic blowdown tunnel using thermochromic liquid crystals, in: A.F.P. Houwing et al. (Eds.), Proceedings of the 21st International Symposium on Shock Waves, Panther Publishing, Canberra, 1998, pp. 471–476.
[10] T.T. Evans, A.J. Smits, Measurements of the mean heat transfer in a shock wave – turbulent boundary layer interaction, Exp. Therm. Fluid Sci. 12 (1996) 87–97.
[11] E.J. Klein, Application of liquid crystals to boundary-layer flow visualization, AIAA Paper 68-0376, 1968.
[12] Z. Wang, P.T. Ireland, S.T. Kohler, J.W. Chew, Heat transfer measurements to a gas turbine cooling passage with inclined ribs, J. Turbomach. 120 (1) (1998) 63–69.
[13] D.J. Mee, S.B. Harrison, T.W. Walton, T.V. Jones, A comparison of liquid crystal techniques for transition detection, AIAA Paper 91-0062, 1991.
[14] H. Babinsky, J.A. Edwards, Automatic liquid crystal thermography for transient heat transfer measurements in hypersonic flow, Exp. Fluids 21 (1996) 227–236.
[15] M.A. Zanchetta, R. Hillier, Boundary-layer transition on slender blunt cones at hypersonic speeds, in: B. Sturtevant, J.E. Shepherd, H.G. Hornung (Eds.), Proceedings of the 20th International Symposium on Shock Waves, World Scientific, Singapore, 1996, pp. 699–704.
[16] K. Asai, H. Kanda, T. Kunimasu, T.S. Liu, J.P. Sullivan, Boundary-layer transition detection in a cryogenic wind tunnel using luminescent paint, J. Aircraft 34 (1) (1997) 34–42.
[17] L. Deluca, G. Cardone, D.A. Delachevalerie, A. Fonteneau, Viscous interaction phenomena in hypersonic wedge flow, AIAA J. 33 (12) (1995) 2293–2298.
[18] J.R. Micol, Aerothermodynamic measurement and prediction for modified orbiter at Mach 6 and 10, J. Spacecr. Rockets 32 (5) (1995) 737–748.
[19] T.V. Jones, Recent developments in transient heat transfer measurements, in: Proceedings of Eurotherm Seminar No. 9, July 1989, pp. 73–80.
[20] J.R. Hayes, R.D. Neumann, Introduction to the aerodynamic heating analysis of supersonic missiles, in: M. Hemsch, J. Nielsen (Eds.), Tactical Missile Aerodynamics, Progress in Astronautics and Aeronautics Series, vol. 142, AIAA, Washington, 1992.
[21] D.J. Mee, R.J. Stalker, J.L. Stollery, Glancing interactions between single and intersecting oblique shock waves and a turbulent boundary layer, J. Fluid Mech. 170 (1986) 411–433.
[22] D. Gaitonde, J.S. Shang, Skin-friction predictions in a crossing-shock turbulent interaction, J. Propulsion Power 13 (3) (1997) 342–348.
[23] N. Narayanswami, C.C. Horstman, D.D. Knight, Computation of crossing shock turbulent boundary-layer interaction at Mach 8.3, AIAA J. 31 (1993) 1369–1376.
[24] M. Gnedin, D. Knight, A. Zheltovodov, A. Maksimov, A. Shevchenko, S. Vorontsov, 3-D crossing shock wave-turbulent boundary layer interaction, AIAA Paper 96-2001, 1996.
[25] S.R. Sanderson, Shock wave interaction in hypervelocity flow, Ph.D. Thesis, California Institute of Technology, Pasadena, CA, 1995.
[26] W.H. Press, B.P. Flannery, S.A. Teukolsky, W.T. Vetterling, Numerical Recipes: The Art of Scientific Computing, Cambridge University Press, London, 1986.
[27] P.T. Ireland, T.V. Jones, The response time of a surface thermometer employing encapsulated thermochromic liquid crystals, J. Phys. E 20 (1987) 1195–1199.
[28] A.A. Zheltovodov, A.I. Maksimov, Symmetric and asymmetric crossing-shock-waves/turbulent boundary layer interactions, EOARD Report SPC 97-4045, 1997.