Computed Effects of Heat Transfer on the Transonic Flow over an Aerofoil

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The effects of the model surface to freestream adiabatic temperature ratio (T_w/T_{ad}) on subsonic flows at zero pressure gradient and transonic flow over a NACA0012 aerofoil are evaluated using a computational fluid dynamic approach. The analysis, based on the thin layer Navier–Stokes equations with a Baldwin–Lomax turbulence model, indicated that surface heat transfer has significant effects on both subsonic and transonic flows confirming some of the experimental data available. The results have implications in wind-tunnel testing at nonadiabatic surface conditions.

Nomenclature

C_d	= aerodynamic drag coefficient
C_f	= skin friction coefficient
C_l	= aerodynamic lift coefficient
C_p	= pressure coefficient
c	= model chord length
H	= boundary layer shape factor, δ^*/θ
k	= thermal conductivity
M, M_e	= freestream, and boundary-layer edge Mach number, respectively
Re, Re_θ	= Reynolds number based on chord length and boundary-layer momentum thickness, respectively
St	= Stanton number
T_w, T_{ad}	= surface, adiabatic recovery temperature
U, U_e	= free stream, boundary-layer edge velocity
x, y	= coordinates from the leading edge of model
y^+	= normalized distance based on friction velocity
α	= angle-of-airflow incidence
δ	= boundary-layer thickness
δ^*	= boundary-layer displacement thickness
θ	= boundary-layer momentum thickness
μ	= viscosity
ρ	= density

I. Introduction

CONVECTIVE heat transfer can have significant influence on the aerodynamics of flow over bodies. Experiments on boundary layers with heat transfer have shown that the boundary-layer development[1] (laminar or turbulent) and boundary-layer transition[2,3] and separation on a model surface[4–7] are sensitive to heat transfer between the model and the flowfield. It can be inferred from these experiments that cooling $(T_w/T_{ad} < 1)$ postpones boundary-layer transition, and for a turbulent boundary layer produces a fuller velocity profile, increased skin friction, and decreased boundary-layer thickness and shape factor. In the case of separated flows, cooling also reduces the length of separation bubble. Heating the model $(T_w/T_{ad} > 1)$ has the opposite effects. For transonic flow with shock interaction, heat transfer can have a significant and favorable effect on the shock wave/boundary-layer interaction.[4,8,9] It is envisaged that in transonic flows model cooling can reduce the scale of shock induced separation and improve aerofoil trailing-edge pressure recovery.

For an aircraft in free flight the surface temperature of the aircraft is nearly equal to the recovery temperature of the freestream

and, therefore, the heat transfer effects mentioned may not be of any importance. In testing a model in short duration facilities and cryogenic wind tunnels, however, there could be significant differences between the model temperature and the freestream recovery temperature leading to spurious scale effects in measurements of both static and dynamic forces on a model. It has been estimated[10] that an increase in 1% in model-to-stream temperature ratio has an effect equivalent to that of $3\frac{1}{2}\%$ reduction in the Reynolds number. Mabey suggested[11] that some of the scale effects associated with the Reynolds number can be identified by cooling or heating the model relative to the adiabatic recovery temperature. Recent experiments[12] on biconvex aerofoils with cooling tend support these suggestions and also demonstrate that surface cooling has a large effect on dynamic measurements, such as buffet.

This understanding of the effects of heat transfer on flow over aerofoils is based on a limited amount of experimental data, hypothesis, and inference.

The aim of this paper is to achieve a further understanding of the effect of heat transfer on subsonic flow over a flat plate and transonic flow over aerofoils by a computational fluid dynamics (CFD) approach. The effect of heat transfer on wind-tunnel testing of aerofoils and the possibility of identifying scale effects associated with Reynolds number by model cooling or heating are discussed. The parameter chosen for the computational analysis is the ratio model surface temperature to adiabatic recovery temperature where the surface temperature was assumed constant over the complete model. Therefore, the conclusions from the analysis may not be applicable to cases the surface temperature varies along the model surface.

II. CFD Description and Validation

The computational fluid dynamic code was based on two-dimensional time-averaged, thin-layer Navier–Stokes equations with a Baldwin–Lomax turbulence model.[13] Sutherland's law was used for viscosity, and the Prandtl analogy was employed for thermal conductivity. A Cartesian grid with 150 × 50 grid cells with denser cells near the surface was used for the computation of flow over a flat plate. The initial normal spacing was set at 1×10^{-3} chords, corresponding to a value of $y^+ \leq 5$, which satisfies the thin-layer conditions. The outer boundary was located at 10 chord lengths. The grid used for aerofoil calculations was a C grid (Fig. 1) with 300 × 60 grid cells. The outer far field for these calculations was located at 50 chord lengths from the aerofoil surface.

The numerical scheme was the implicit MacCormack predictor corrector finite volume method with cell centered mesh and Gauss–Seidel time relaxation.[14] The upwind flux vector splitting method of van Leer[15] was used for attenuating the numerical oscillations in the vicinity of shock waves. The continuously differentiable flux limiter proposed by Mulder and van Leer[16] was used to eliminate undershoots and overshoots in the shock region. Far-field boundary conditions were imposed implicitly. The surface boundary
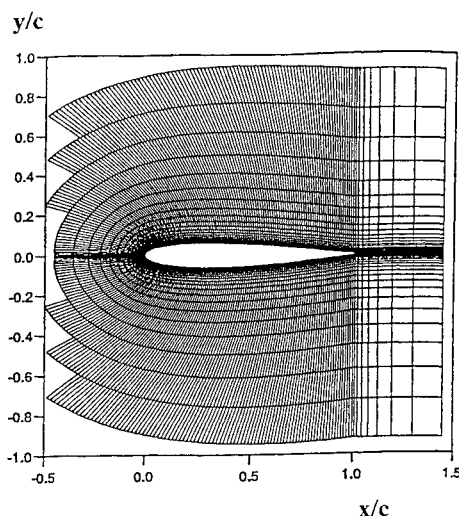
Fig. 1   C grid.

conditions were imposed implicitly. The surface boundary conditions were implicit for adiabatic wall conditions and explicit for nonadiabatic wall conditions. Flow variables at the wake cut were calculated as the average linear extrapolation from above and below the cut.

Calculations for the flat plate were made with an initial 20 iterations on a coarse grid and 60 iterations on a fine grid. For the aerofoil 50 iterations were performed on the coarser grid before the converged solution was transferred to the fine grid. The full convergence to a steady-state solution was obtained after approximately 250 iterations on the finer grid.

The code was validated against standard cases for NACA 0012 and RAE 2822 aerofoils. Figure 2 shows typical validations. Figure 2a shows the code validation for pressure distribution for RAE 2822 at $M = 0.73$, $Re = 6.5 \times 10^6$, $\alpha = 2.79$ deg and adiabatic surface conditions. Navier D refers to the design coordinates. Navier M and circles refer to the measured coordinates and experimental data of Cook et al.,[17] respectively. Transition was fixed at 3% chord on both the upper and lower surfaces. The predicted (M) aerofoil pressure distribution compares favorably with the experimental results. Small differences in the pressure distribution between the experimental data and computation near the leading edge and the shock interaction region have also been observed by other authors. The differences in C_p near the leading edge may be attributed to the experimental errors associated with the transition tripping. The differences in C_p in the shock interaction region is due to the inability of the turbulence model to predict correctly the shock/boundary-layer interaction. The computed values of δ^* and θ on the upper surface of the aerofoil for the test case also is found to compare favorably with the experimental results.

Figure 2b shows the pressure distribution on a NACA 0012 aerofoil at $M = 0.7$, $Re = 9 \times 10^6$ and $\alpha = 2.57$ deg with transition fixed at 5% chord, a case corresponding to a fully attached and subcritical flow over an aerofoil. The agreement between the computed and experimental results is very good.

Figure 2c shows a severe test for the code; pressure distribution on a NACA 0012 aerofoil at $M = 0.79$, $Re = 9 \times 10^6$ and $\alpha = 2.57$ deg. Transition was fixed at 5% chord. The discrepancy between the experimental and computational results for shock position and the pressure distribution in the shock wave/boundary-layer interaction region is unacceptable and also has been observed by Maksymiuk and Pulliam[18] and others. These have been attributed to the inadequacy of the turbulence models for transonic flows with strong interactions. However, the CFD model is thought to be adequate for incipient separations.

Figure 2d shows a comparison between the present predictions and experimental data of Maksymiuk and Pulliam[18] (ARC2D) and predictions of Harris[19] for the variation of drag with Mach number for a NACA 0012 aerofoil at $Re = 9 \times 10^6$ with transition fix at 5%

chord. The comparison between experimental data and predictions in the transonic Mach number range is reasonable.

The lift-drag polar for the NACA 0012 aerofoil at $M = 0.7$ and $Re = 9 \times 10^6$ based on the present computational results are compared with the experimental and computed results of Harris[19] and experimental results,[18] respectively, in Fig. 2e. The transition was fixed at 5% chord for all of these results. The present computed results are comparable with the other results.

A validation of the code for heat transfer with attached air flows is shown in Fig. 2f. In this figure the computed and experimental values[20] of the Stanton number are plotted against Re_θ for a flat plate at $M = 1.69$, $Re = 10^6$ and $T_w/T_{ad} = 1.65$. The Spalding–Chi[21] correlation based on computed values of the Stanton number and skin friction are also included in this figure. The agreements between the computed results and the experimental data are good at higher values of Re_θ indicating that the numerical code can predict heat transfer with reasonable accuracy at higher Re_θ values ($Re_\theta > 6000$).

III.   Heat Transfer Effects on Flat Plate Boundary layer

Results of the computation of subsonic flow over a flat plate (zero pressure gradient) are shown in Figs. 3 and 4. Figure 3 refers to a laminar boundary layer on a flat plate at $M = 0.5$, $Re = 8 \times 10^3$ and $T_w/T_{ad} = 0.6$, 1.0, and 1.4. The effects of heat transfer on a laminar boundary-layer at zero pressure gradient are appreciable.

Surface cooling increases the velocities (Fig. 3a), density (Fig. 3b) and momentum (Fig. 3c) of the air flow within the boundary layer. The increase in velocities and density are significant. Surface heating has the opposite effect to that of cooling. The trends in velocity profiles are similar to those measured by Van Driest.[1] It has also been observed by Back and Cuffel[22] that the momentum of the boundary layer near the surface increases with cooling and decreases with heating.

Figures 3d–3h refer to the values of the boundary-layer thickness, displacement thickness, momentum thickness, shape factor, and skin friction, respectively. In these figures the values are normalized with respect to the corresponding values at adiabatic wall conditions and are plotted against temperature ratio for the described case. The general effect of cooling is to decrease δ (Fig. 3d), and δ^* (Fig. 3e), H (Fig. 3g), and increase θ (Fig. 3f) and C_f (Fig. 3h). Typically, for a laminar boundary layer at zero pressure gradient, a 10% change in T_w/T_{ad} produces 1% change in C_f.
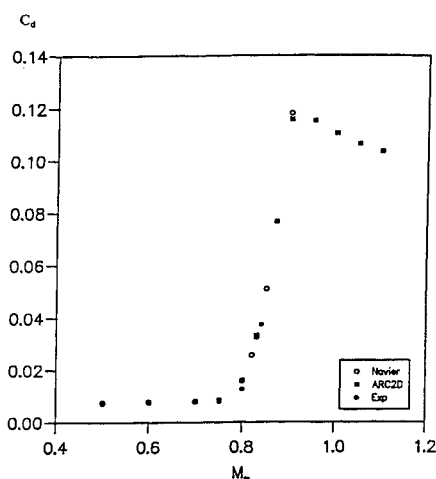
The effect of heat transfer on a turbulent boundary layer at zero pressure gradient is typified in Fig. 4. The results shown here are for the boundary-layer velocity (Fig. 4a), density (Fig. 4b) and momentum (Fig. 4c) profiles, and skin friction (Fig. 4d) at $M = 0.5$, $Re = 1 \times 10^6$, and $T_w/T_{ad} = 0.6$, 1, and 1.4. The effects of heat transfer on the velocity profile (Fig. 4a) and skin friction (Fig. 4d) are not as large for a turbulent boundary layer as that noticed for laminar boundary layers. Typically a 10% change in T_w/T_{ad} produces 1/2% change in C_f. As was the case for the laminar boundary layer, however, the effect of heat transfer on the density and momentum profiles are large.

The observed results here can be traced to the effect produced by the temperature of the wall relative to the adiabatic temperature on the viscosity, density, and energy exchange of the air near the surface. To illustrate the reasons for the effects observed, the cold wall case is considered. A cold wall would result in a transfer of energy within the boundary layer and toward the wall resulting in an increase in the velocities near the wall and velocity gradients normal to the surface. The air near the cold wall also has a relatively higher density. Primarily, it is the increase in density and location near the wall which leads to a large increase in mass and momentum flow. The displacement thickness is primarily controlled by the mass flow near the wall and, therefore, decreases with cooling. The fuller velocity profile leads to a decrease in the shape factor.
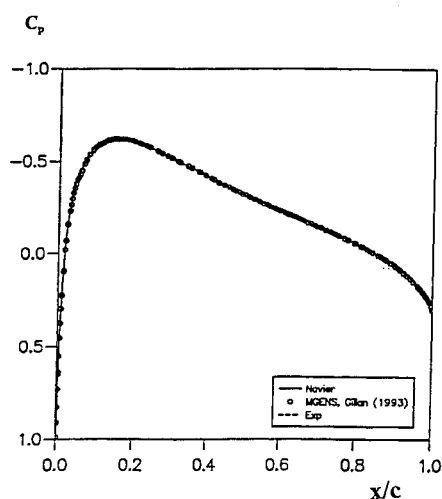
The skin friction C_f is proportional to $\mu (du/dy)$. Model cooling decreases μ and increases du/dy. The fact that C_f increases with a reduction in wall temperature indicates that for a zero pressure
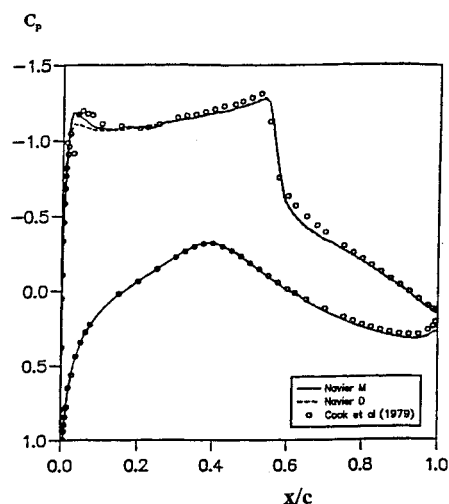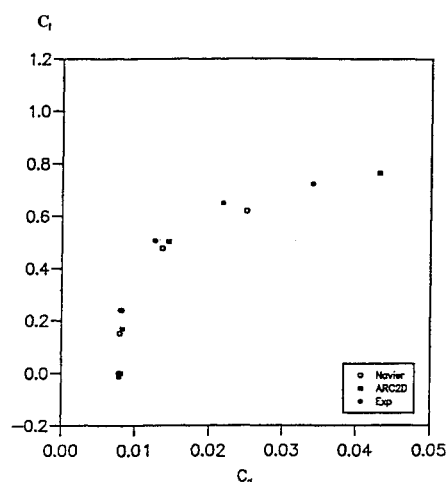
a) C_p distribution, RAE 2822, $M_\infty = 0.73$, $Re = 6.5 \times 10^6$, $\alpha = 2.79$ deg, transition fix 3% chord
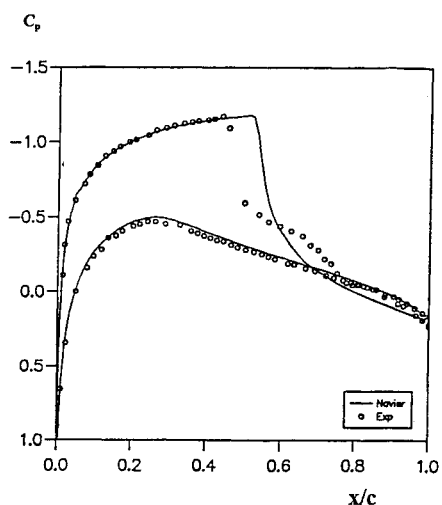
b) C_p distribution, NACA 0012, $M_\infty = 0.7$, $Re = 9 \times 10^6$, $\alpha = 0$ deg, transition fix 5% chord

c) C_p distribution, NACA 0012, $M_\infty = 0.79$, $Re = 9 \times 10^6$, $\alpha = 2.57$ deg, transition fix 5% chord

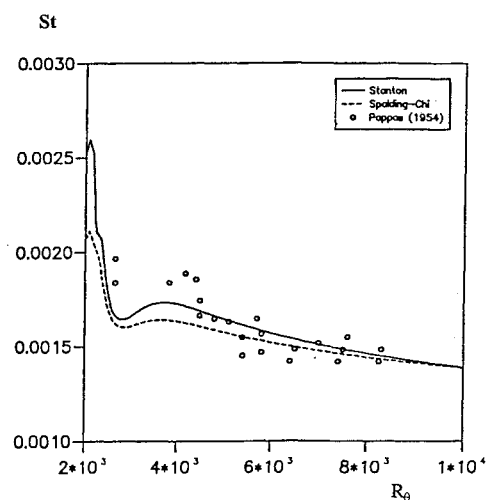d) Variation of C_d with M_∞, NACA 0012 $Re = 9 \times 10^6$, and $\alpha =$ deg

e) Drag polar, NACA 0012, $M = 0.7$, and $Re = 9 \times 10^6$

f) Validation for heat transfer for a flat plate, $M = 1.69$, $Re = 10^6$, and $T_w/T_{ad} = 1.69$

Fig. 2   Code validation.

gradient boundary layer the effect of cooling on the velocity gradients near the wall are larger than those due to viscosity.

The exchange of momentum and energy in a laminar boundary layer at adiabatic wall conditions is due to molecular motion. The velocity profile is not as full as the turbulent profile where the exchange of momentum and energy takes place by turbulent eddies. The effect of model cooling on the laminar velocity profile is to

make it fuller leading to large changes in the boundary-layer profile and its integral values.

The effect of heat transfer on a turbulent velocity profile and, therefore, the velocity gradients near the wall is relatively small due to the fact that the energy changes produced by heat transfer are only a small proportion of the energy exchange produced by the turbulent motion.

a) Velocity profiles
b) Density profiles
c) Momentum profiles

d) Boundary-layer thickness
e) Displacement thickness
f) Momentum thickness

g) Shape factors
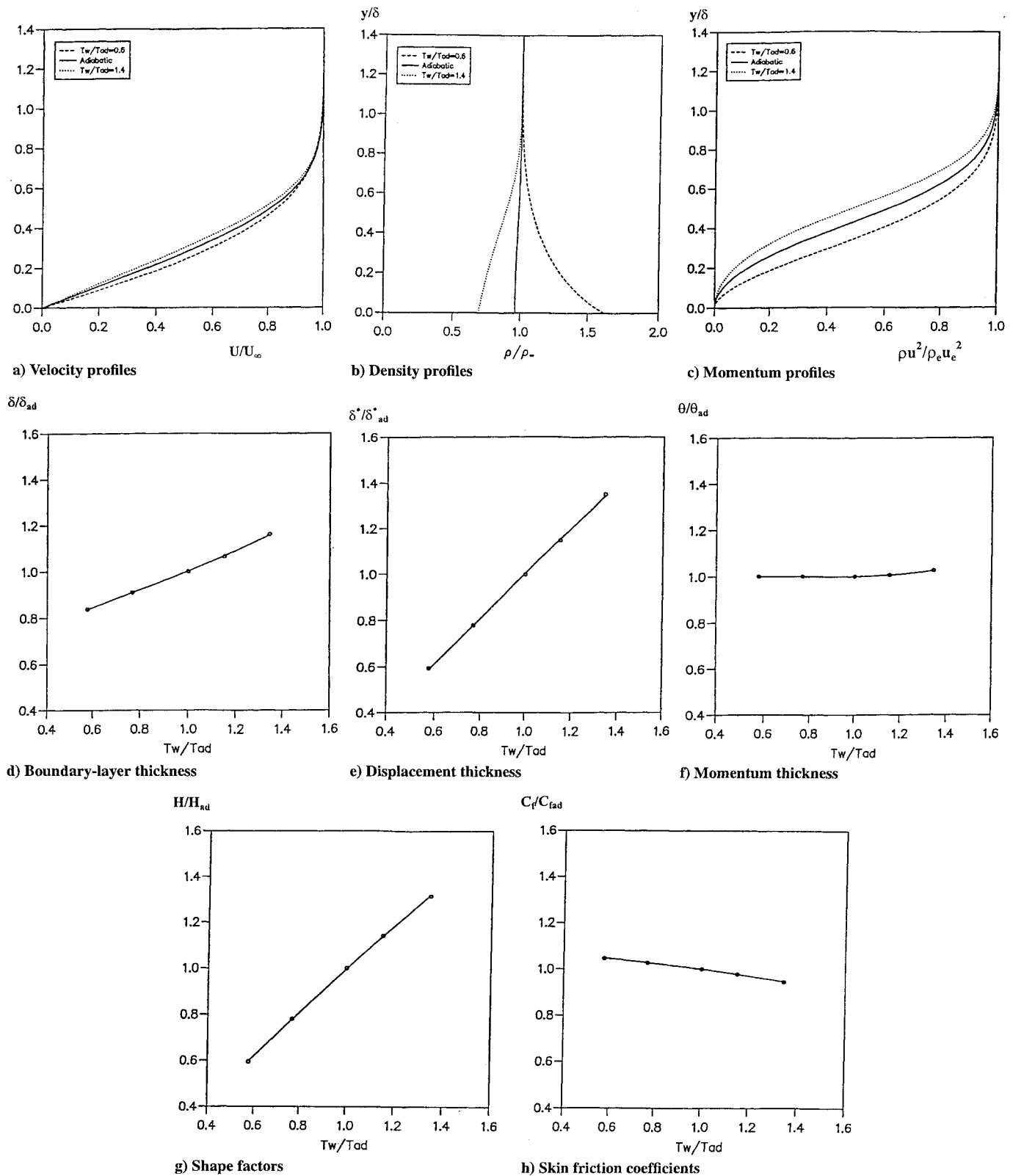h) Skin friction coefficients

Fig. 3 Effect of heat transfer on laminar boundary layer at zero pressure gradient.

In general, some of the effects observed here due to surface cooling on a flat plate boundary layer are analogous to that of boundary-layer suction in the sense that boundary-layer suction also produces a transfer of energy in the boundary layer toward the surface leading to a fuller velocity profile, increased velocity gradients, increased skin friction, and decreased the shape factor. Unlike boundary-layer suction, however, cooling increases the density decreases the viscosity near the surface. This should increase the inertia forces and decrease viscous forces, respectively, resulting in an increase in the effective Reynolds number.

The effects of heating on the boundary layer are generally opposite to that of cooling.

IV. Heat Transfer Effects on Transonic Flow over an Aerofoil

The predicted effects of heat transfer on transonic flow over a NACA 0012 aerofoil can be observed from Figs. 5–7. Figure 5 refers to subcritical flow over the aerofoil at $M = 0.7$, $Re = 9 \times 10^6$ and $\alpha = 0$ deg, and for $T_w / T_{ad} = 0.6$, 1, and 1.4. Figure 6 refers to
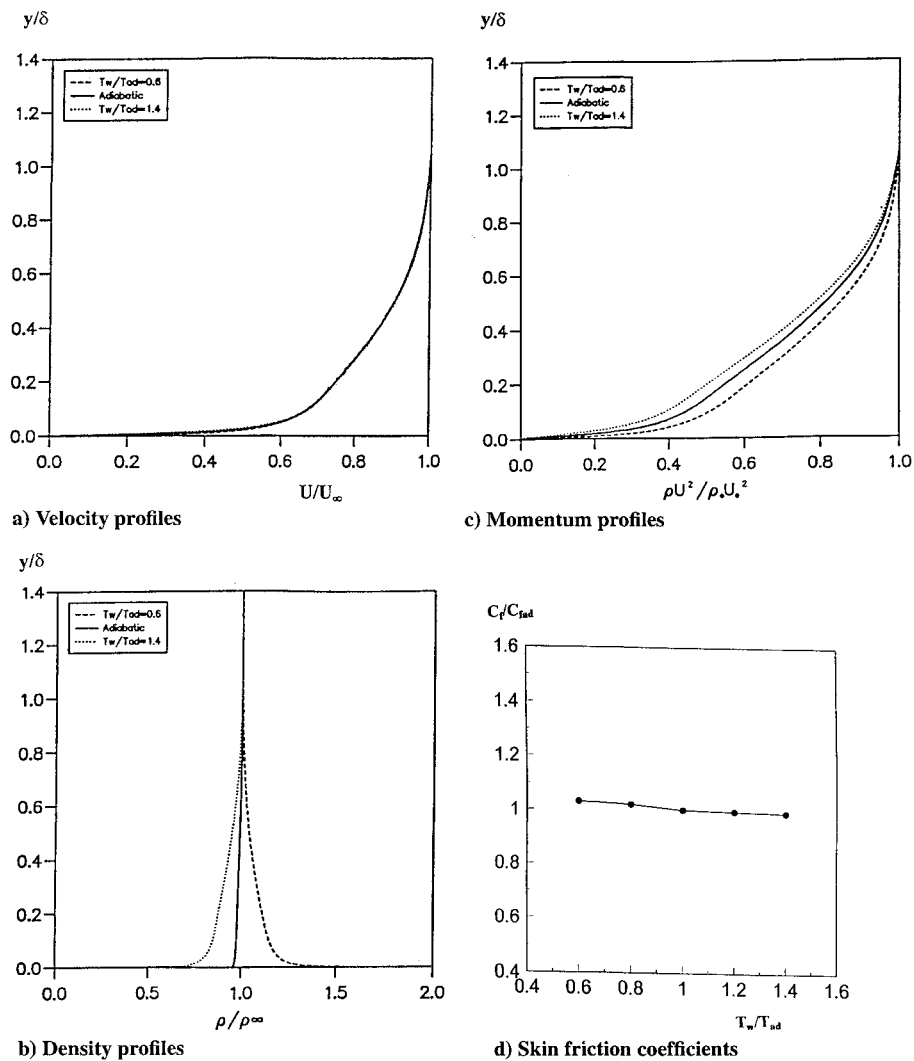
a) Velocity profiles

c) Momentum profiles

b) Density profiles

d) Skin friction coefficients

Fig. 4   Effect of heat transfer on turbulent boundary layer at zero pressure gradient.

a) C_p distribution
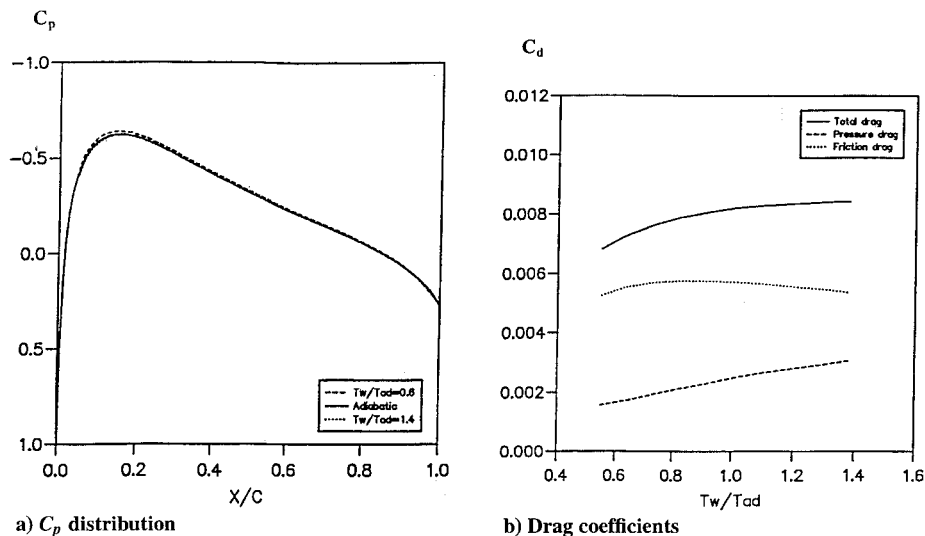
b) Drag coefficients

Fig. 5   Effect of heat transfer on a NACA 0012 profile at subcritical flow $M_\infty = 0.7$, $Re = 10^6$, $\alpha = 0$ deg, transition fix 5% chord.

supercritical flow with incipient shock induced separation at $M = 0.85$, $Re = 9 \times 10^6$, $\alpha = 0$ deg, and for $T_w/T_{ad} = 0.6$, 1, and 1.4. Transition was fixed at 5% chord for both cases.

At subcritical conditions heat transfer has only a small effect on the surface pressure distribution (Fig. 5a). The effect of cooling on the surface pressure distribution is to decrease the pressures and increase the local Mach numbers. The effect of heating on the

surface pressures is even smaller than produced by cooling. Similar observations at subcritical flows have been made by Inger et al.[8] The effect of heat transfer on drag is significant (Fig. s5b). At adiabatic conditions the pressure drag and skin friction drag are of the same order. Heat transfer has only a small effect on the skin friction drag but relatively large effect on the pressure drag. Cooling reduces the pressure drag and, therefore, the total
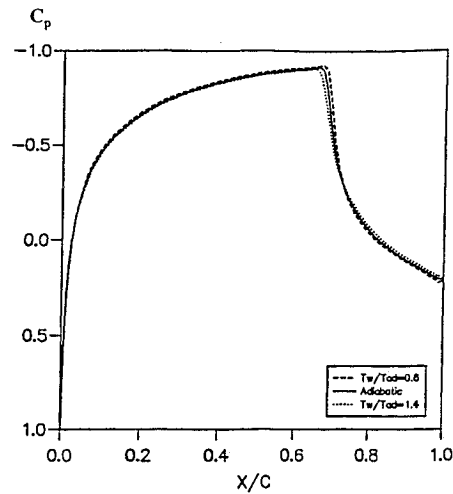
a) C_p distribution
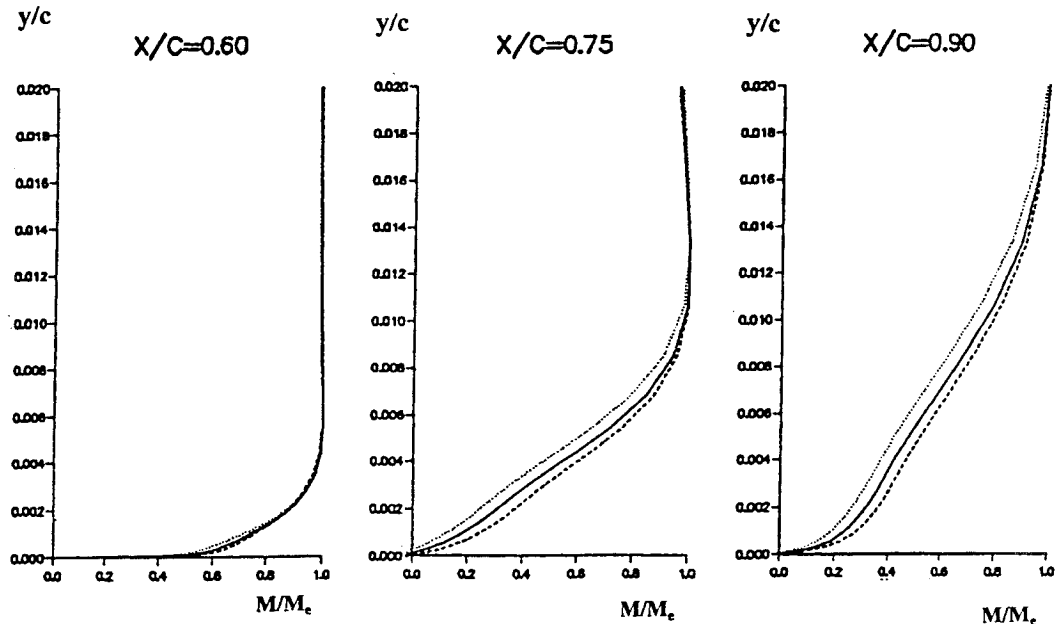
b) Mach number profiles
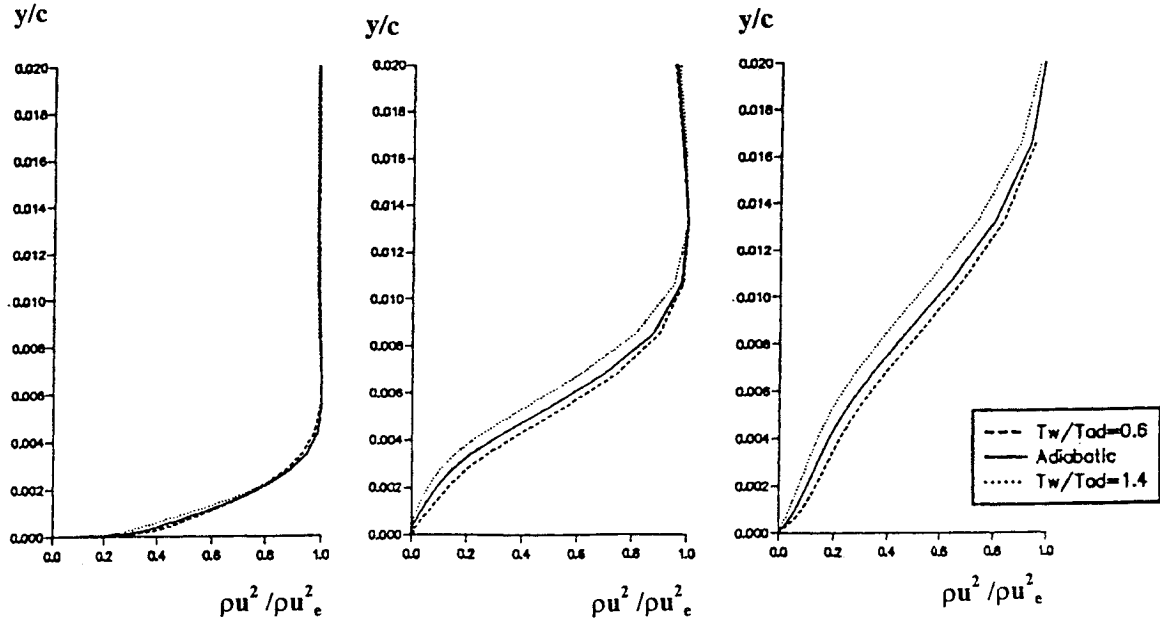
c) Momentum profiles

Fig. 6   Effect of heat transfer on steady transonic flow, NACA 0012, $M_\infty = 0.85$, $Re = 9 \times 10^6$, $\alpha = 0$ deg, transition fix 5% chord.

δ*/c

d) Displacement thickness

θ/c

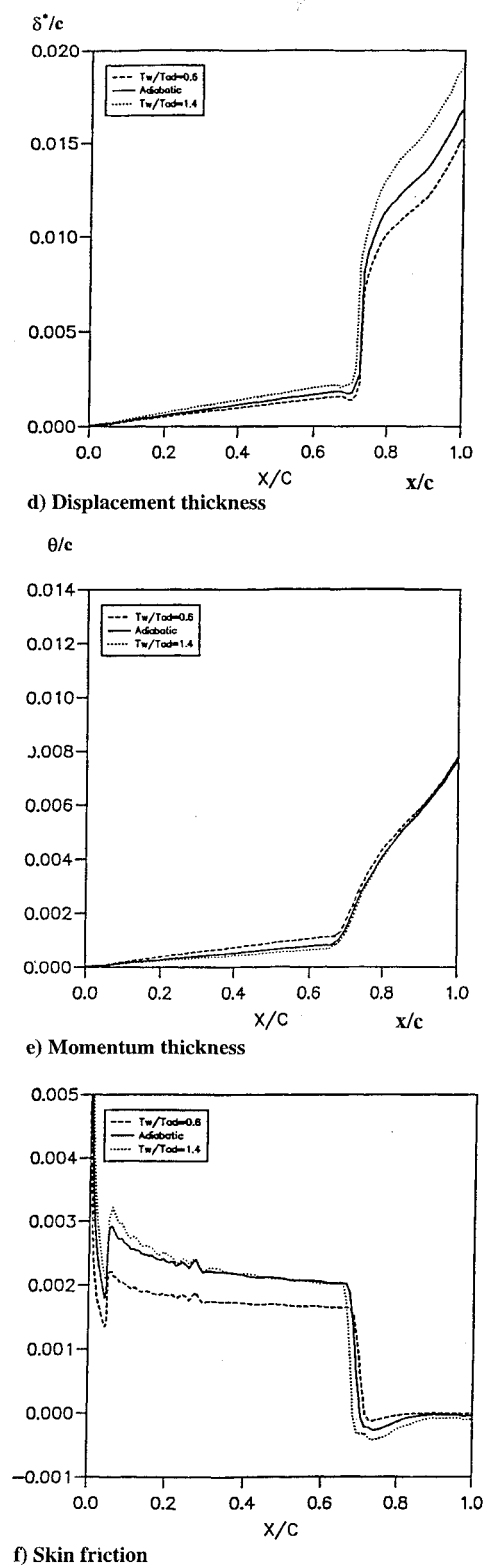e) Momentum thickness

f) Skin friction

Fig. 6 (Continued) Effect of heat transfer on steady transonic flow, NACA 0012, $M_\infty = 0.85$, $Re = 9 \times 10^6$, $\alpha = 0$ deg, transition fix 5% chord.

drag. Typically a 10% change in T_w/T_{ad} results in 2% change in C_d.

The pressure distribution on the surface of a NACA 0012 aerofoil at supercritical conditions (Fig. 6a) indicates that surface cooling moves the shock rearwards, increases the shock strength, and improves the pressure recovery at the trailing edge. The pressure gradients in the shock interaction region are increased. Cooling increases the Mach numbers (Fig. 6b) and momentum (Fig. 6c) within the boundary layer and momentum thickness (Fig. 6e). It decreases the
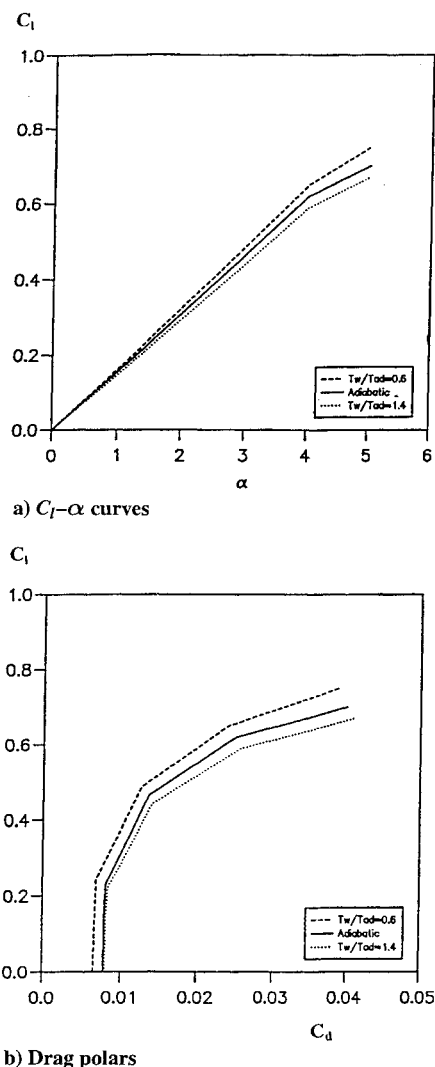
C_1

a) C_l–α curves

C_1

b) Drag polars

Fig. 7 Effect of heat transfer on aerodynamic force coefficient, NACA 0012, $M_\infty = 0.7$, $Re = 9 \times 10^6$. Transition fix 5% chord.

displacement thickness (Fig. 6d) and skin friction (Fig. 6f). Similar effects of cooling on transonic flows have been observed experimentally.

The effects observed here are due to the effect of heat transfer on the boundary-layer thickness and sonic height, which in transonic flows are of the same order. The rearward movement of the shock wave may be attributed to the reduced displacement thickness of the boundary layer approaching the shock wave. The increased shock strength and pressure gradients in the shock interaction region are due to a reduction in the sonic height and a corresponding reduction in the communication across the shock wave. These are evident by the Mach number profiles (Fig. 6b) and momentum profiles (Fig. 6c) at three chordwise positions, namely, upstream of the shock $x/c = 0.6$, downstream of the shock $x/c = 0.75$ and close to the trailing edge $x/c = 0.90$ and the variation of displacement thickness in the chordwise direction (Fig. 6d).

The skin friction values on the surface of the aerofoil surface (Fig. 6f) indicate that for the adiabatic case the flow is separated locally downstream of the shock wave. The effect of model cooling is to decrease C_f in the region of negative pressure gradients. In this region the effect of cooling on the mass flow profile and, therefore, the velocity profile is negligible, largely due to the transfer of energy toward the surface due to turbulent eddies far outweighing the heat transfer effects. (The velocity profile is very full at adiabatic surface conditions.) The major effect of cooling on the skin friction is through viscosity, which decreases with cooling, that is, C_f decreases with reduced surface temperature.

The fact that surface cooling improves the pressure recovery near the trailing edge of the aerofoil indicates that cooling also reduces

the height of the separation bubble, as can be observed from the δ^* values in the separated region. This may be attributed to the increase in momentum of the boundary layer approaching the shock wave (Fig. 6c).

Figure 7 shows the predicted effects of heat transfer on the lift and drag coefficients for a NACA 0012 aerofoil at $M = 0.7$, $Re = 9 \times 10^6$, $\alpha = 0$–5 deg and $T_w/T_{ad} = 0.6$, 1, and 1.4. For a given incidence, surface cooling increases, the lift coefficient, C_l (Fig. 7a), this being more noticeable at higher values of incidence. The results at high incidence, however, should be viewed with caution as the code is valid only for attached boundary layers and incipient separation. The increase in lift is associated with the increased velocities on the surface. Surface cooling also reduces the overall drag coefficients due the effects on the boundary layer discussed earlier. Thus, there is an increase in the lift-to-drag ratio (Fig. 7b) and improvement in the performance of the aerofoil. Typically at an incidence of 3 deg, a 10% change in T_w/T_{ad} results in 2% change in C_d and 1.5% change in C_l. These results are in general agreement with experimental results of Macha et al.[4] at low speeds.

All of the results presented here confirm that surface heat transfer can introduce spurious scale effects in wind-tunnel testing as observed by Green et al.[10] Further, the effects of cooling on the flow over an aerofoil are similar to those produced with an increase in Reynolds number. Therefore, it may be possible to identify scale effects associated with Reynolds number by cooling or heating the model relative to the adiabatic temperature.

V. Conclusions

The effect of heat transfer on steady subsonic and transonic flow may be summarized as follows.

Model cooling 1) produced a thinner boundary layer, 2) increased the momentum within the boundary layer, 3) increased shock Mach number and pressure gradients in the shock interaction region, 4) produced a rearward movement of the shock wave, 5) reduced the height of the separated flow, and 6) increased pressure recovery at the trailing edge.

Surface heating produced opposite effects to cooling.

The effect heat transfer on skinfriction is pressure gradient dependent.

Testing a model in a wind tunnel at nonadiabatic surface temperatures can lead to spurious scale effects.

Heat transfer may be a technique that could be used to identify scale effects on models and needs further investigations.

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