DRAG REDUCTION AND BUFFETING ALLEVIATION IN TRANSONIC PERIODIC FLOW OVER BICONVEX AEROFOILS

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Keywords: *Transonic Flows, Shock Oscillations, Flow Control, Heat Transfer*

Abstract

Using CFD analyses, the effects of flow control techniques, as the contour bump and the surface cooling concepts, are separately investigated in transonic periodic flow over 14% and 18% biconvex aerofoils. For these aerofoils both cooling and bump contour, which was located in the vicinity of the mean shock position on the lower surface of the aerofoil, have a positive influence on drag reduction and buffeting alleviation. The Tijdeman's Type A, B and C shock oscillations are well identified on the aerofoil.

Symbols and Formulae

c   model chord length
C_d aerodynamic drag coefficient
C_f skin friction coefficient
C_l aerodynamic lift coefficient
C_p pressure coefficient
C_m aerodynamic moment coefficient
 f   frequency (Hz)
 k   reduced frequency ($2\pi f c / U$)
M_1 Mach number just upstream of the shock
M_∞ free stream Mach number
Re   free stream Reynolds Number
T_w, T   surface, adiabatic recovery temperature
U   free stream velocity
y^+ normalized distance based on friction velocity (wall unit)
α   angle of airflow incidence

1 Introduction

The complex phenomena present in the shock-boundary layer interaction region on an aerofoil contribute to increase viscous and shock wave losses, being also, in transonic regime, the main cause of buffet onset. The transonic shock-induced oscillation (SIO) triggered by flow separation behind the shock wave, tends to grow to bounded amplitude, which can reach levels large enough to cause irreversible structural failure. In order to minimize these negative effects induced by buffeting and to decrease the total drag of the aerofoil different control devices have been proposed. These devices are either passive or active and include sub-layer mechanical devices, bumps, surface cooling devices, vortex generators, boundary layer suction/blowing, continuous or pulse skewed air jets and synthetic jets. Some of the methods of flow control, using adaptive wing technology, have been recently reviewed by Stanewsky and Delery [1, 2, 3].

The aim of this paper is to predict numerically buffeting onset, investigating the effect of bump control technique and surface cooling device (T_w/T<1) on aerodynamic characteristics of biconvex aerofoil.

The heat transfer between the aerofoil and the flow field has an important influence on the laminar or turbulent boundary layer development [4], setting out characteristic boundary-layer separations and turbulent transitions [5, 6, 7], having also a significant effect on the shock wave boundary layer interaction [8, 9]. Furthermore, cooling has a

significant effect on skin friction at the surface of the aerofoil (Fig.1). In supersonic regions cooling decreases the viscosity and therefore the skin friction whereas at subsonic speeds, cooling increases the velocity gradients and hence skin friction. In transonic periodic flow cooling alleviates buffeting and reduces viscous drag, decreasing skin friction in supersonic regions.
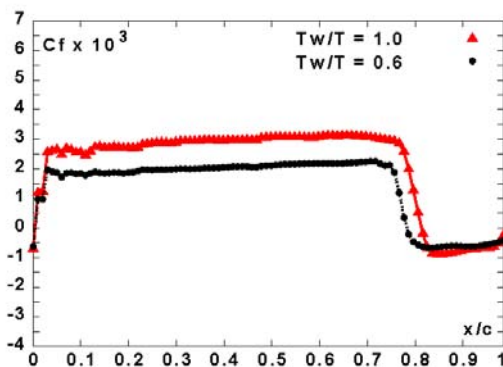
Fig. 1 Effects of cooling on transonic shock boundary-layer interaction over 14% biconvex aerofoil. Skin friction distribution (M=0.83, Re=9×10^6, α=0°).

The concept of Passive Control of Shock / Boundary Layer interaction, consists of a porous surface and a cavity or plenum underneath located in the region of shock boundary layer interaction. It is understood that the static pressure rise across the shock wave will result in a flow through the cavity from downstream to upstream of the shock wave.

This is equivalent to a combination of suction downstream and blowing upstream of the shock. The blowing upstream of the shock produces thickening of the boundary layer approaching the shock, which in turn results in a system of weaker shocks and an extended interaction region. This reduces the wave drag. The suction downstream of the shock can reduce viscous pressure drag.

The porous surface and the cavity can also alleviate pressure fluctuations associated with shock wave boundary layer interaction. The benefit of the passive control is limited by the fact that rapid thickening of the boundary layer upstream of the shock results in increased viscous losses in spite of any positive effect of suction downstream of the shock interaction. In

principle the concept of passive control, which produces a re-circulating flow in the region of control, is equivalent to having a contoured bump on the surface.

Indeed, DASA-Airbus in introducing a bump into an A340-type hybrid laminar flow wing and assuming a typical mission for such an aircraft of 600 flights per year, has determined that fuel savings of up to 2.11% at a cruise Mach number of 0.84 can be achieved [10]. The reduction of drag is essentially due to the reduction in wave drag but with no increase in viscous drag. As for most of the control techniques, the bump is also able to alleviate buffeting. However, the effects of a contoured bump on an aerofoil on buffet associated with transonic periodic flow, has yet to be understood and this is addressed in this paper.

The paper presents computational fluid dynamics analysis of transonic periodic flow over basic and adaptive 14 and 18 percent biconvex aerofoil with a contour bump located at the mean shock position on lower surface of the aerofoil. In the case of basic aerofoils, a cooling of aerofoil surface of T_w/T=0.6 for 14% biconvex aerofoil and T_w/T=0.9 for 18% biconvex aerofoil have been assumed. The mechanism of shock oscillations is further reviewed.

2. Transonic periodic flow over biconvex aerofoils. Numerical and experimental results

Generally, the experimental studies identified three kinds of SIO that could appear on aerofoils. These SIO are known as Tijdeman's types A, B and C SIO [11]. Type A is of relatively small amplitude and almost sinusoidal motion. Type B is of relatively larger amplitude and characterized by the tendency of the shock wave to disappear during a part of the cycle. In the case of Type C SIO the shock propagates only upstream. On a biconvex 18% aerofoil, McDevitt [12] discovered experimentally, in accelerated flow regime, a narrow domain of periodic motion, between 0.76 and 0.78 Mach, where Type C SIO occurs almost everywhere in the domain, except at the highest Mach number

and at non zero incidence, where type A SIO can also occur (Fig. 2).

For type C shock oscillations there is no downstream motion of the shock wave and this phenomenon repeats periodically alternating between the upper and lower surfaces. On the other hand, on biconvex 14% aerofoil, Gibb [13] identified experimentally a domain of SIO (Fig. 2), between 0.82 and 0.86 mach, where Type B SIO appears almost everywhere in the domain, except at the highest Mach number where type A SIO can also occur.
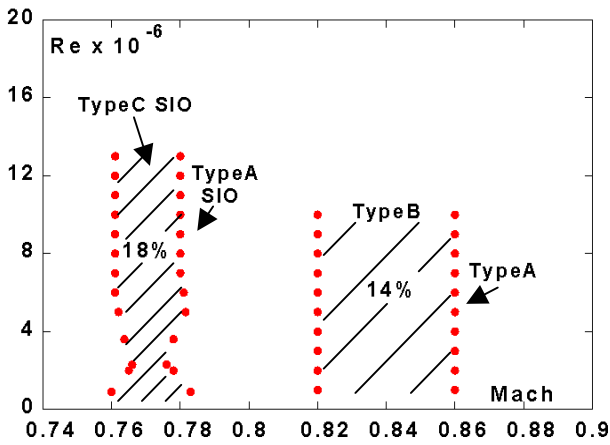
Fig. 2 Regions of periodic transonic flow over biconvex aerofoils in accelerated flow regime.

Theoretically, the buffet onset boundary for an aerofoil is shown to be a Poincare-Hopf bifurcation. In fact a narrow range of Mach numbers exists where the stationary solution of the Navier-Stokes equations becomes unstable and tends to a stable periodic solution, which can be one of Tijdeman's types A, B or C SIO. Having this result of existence and uniqueness of the Navier-Stokes periodic solution, the unsteady mass-averaged Navier-Stokes equations in two dimensional conservation form are integrated forward in time until the periodic solution is obtained asymptotically after a sufficient number of time steps. The presence of a relatively strong shockwave in transonic flow has a significant influence on the development of the turbulence field in the region of the shock, as well as the mean flow properties mainly downstream of the shock discontinuity. The standard turbulence models require special

modifications to handle the high-pressure gradients in a physically realistic way. The RANS based on one or two equations turbulence models usually predict a shock location well downstream of the experiment once certain shock strength is reached. This is mainly due to an excess of eddy viscosity resulting from these models. On the other hand, the algebraic models have the tendency to predict early separation of the boundary layer, the eddy viscosity being under evaluated. In order to improve the prediction of eddy viscosity some new turbulence models using a non-linear formulation of the wall region anisotropy have been proposed. For algebraic models, the modified Baldwin-Lomax turbulent model [14] leads to eddy viscosity predictions that are in good agreement with experiment for transonic flows with relatively weak interactions.
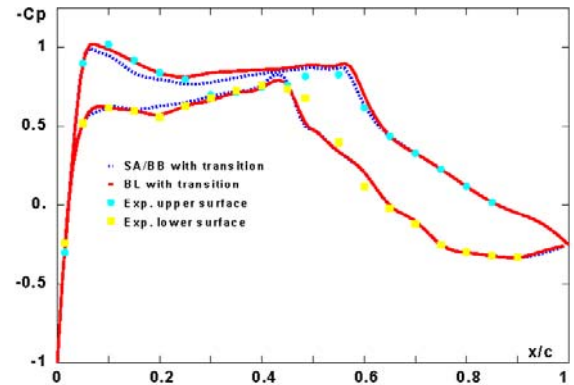

Fig. 3 Steady pressure distributions for BL, SA and BB turbulence models, for an NLR 7301 aerofoil at M=0.753, α=-0.08°.

These results can be slightly improved using the Johnson-King model [15], when an ordinary differential equation for the maximum Reynolds shear stress is solved, thus adjusting to a better value the eddy viscosity level. For two-equations $k-\varepsilon$ model, the insensitivity to adverse pressure gradients or the incorrect behavior in stagnation regions are corrected by local assumption on the form of the eddy viscosity expression or the form of production term for the turbulent kinetic energy. To this regard, the Menter's Shear-Stress-Transport (SST) model [16] significantly improves the

adverse pressure gradient performance. Moreover, using the physically correct assumption of the proportionality between the principal shear stress in a shear layer and the turbulent kinetic energy, the two-equations models can be transformed to one-equation models. It is concluded that one-equation models are at least equivalent in performance to two-equations models for boundary layer applications. Figure 3 shows a comparison

between the experiment and the computations using the modified Baldwin-Lomax, one-equation Spalart-Allmaras and Baldwin-Barth turbulence models for an NLR 7301 Airfoil [17]. The one-equation models lead to a good improvement of prediction on the suction surface. However, in transonic periodic flow for relatively weak interaction, as is the case in the present analysis, the both models are thought to yield quite comparable results.
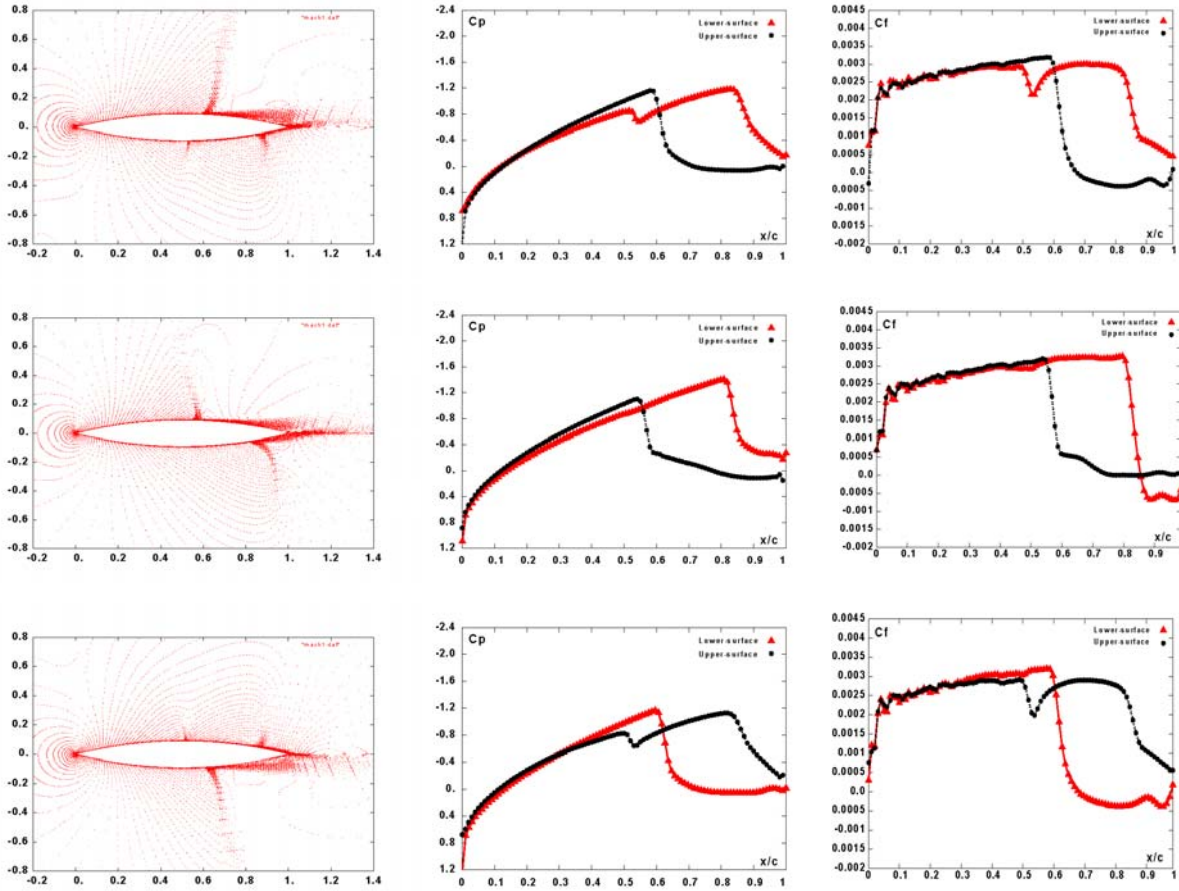
Fig. 4 The type C shock oscillation on the 18% thick biconvex aerofoil: Contours of Mach number, Pressure distribution and skin friction distribution at discrete time steps (M=0.76, Re=10×10^6, α=0°, k=0.485)

A two-dimensional, time-averaged, thin-layer Navier-Stokes code capable of computing flows over an aerofoil with moving grid was developed for these investigations. The code includes heat transfer effects and a moving grid option in order to investigate the effect on periodic flow of a trailing edge splitter plate motion, a flap motion or a pitching aerofoil. The implicit code solves the time-averaged thin-layer Navier-Stokes equations using an

upwind implicit MacCormack [18] predictor/corrector cell-centered finite-volume method in conjunction with Gauss-Seidel line relaxation iterative procedure. The treatment of turbulence is made by a modified version of the Baldwin-Lomax turbulence model. To attenuate the numerical oscillation in the vicinity of shock waves both a continuous differentiable flux limiter, proposed by Mulder [19], and the flux splitting method of Van Leer

[20] are used. Diffusive terms are calculated using a central-difference strategy and the flow variables at the wake cut were calculated as the average linear extrapolation from above and below the cut. The development of a large trailing edge displacement thickness during shock oscillations and its effect on the direct viscous computational methods was accounted for in the grid generation. A 320×64 C-grid, with 256 cells placed on the aerofoil surface, has been employed throughout this study. The

minimum normal grid spacing was reduced to 5×10^{-6} chords, ensuring a value of $y^+ < 5$ everywhere on the aerofoil surface, which ensured adequate resolution of the viscous shear layer [21]. Far field boundary and surface boundary conditions were imposed implicitly at 50 chord lengths. The code was successfully applied to the computation of buffet over the NACA0012 aerofoil and for some of the AGARD test cases [22, 23].
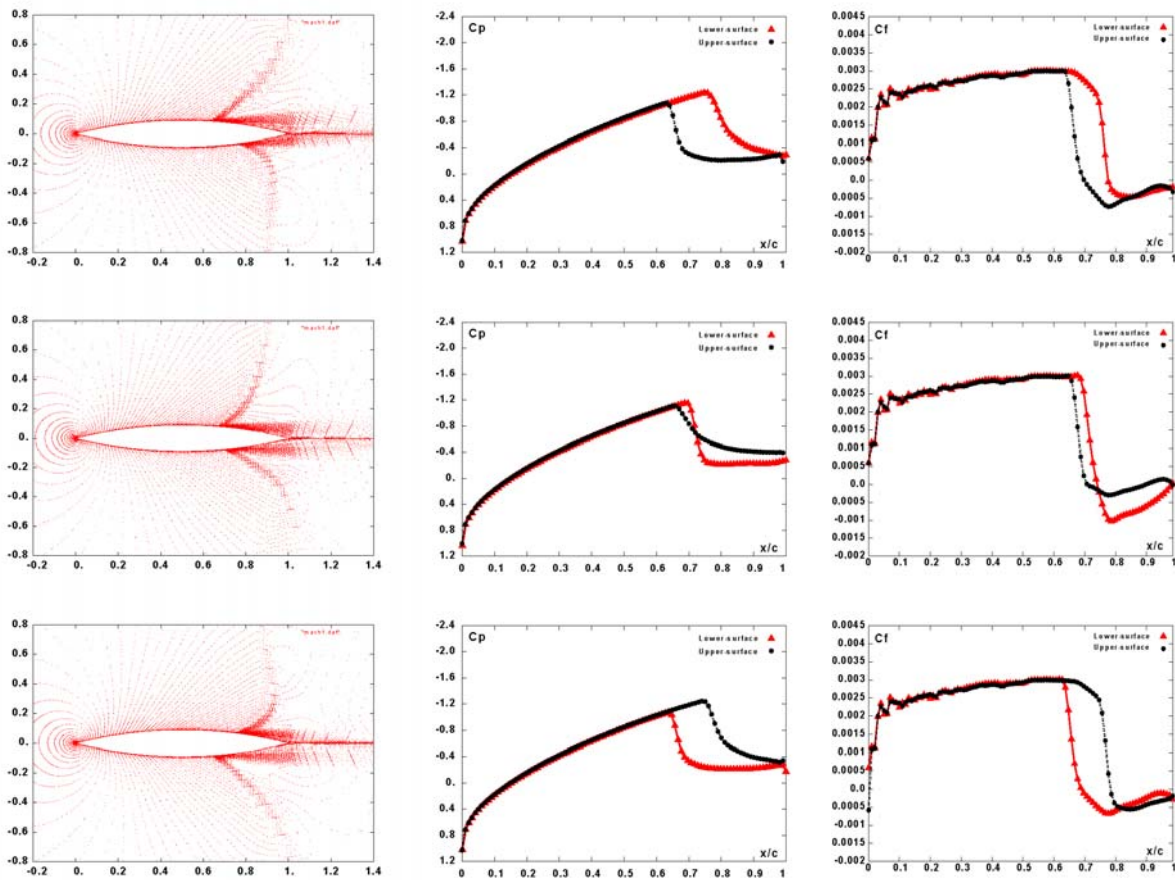
Fig. 5 The type A shock oscillation on the 18% thick biconvex aerofoil: Contours of Mach number, Pressure distribution and skin friction distribution at discrete time steps (M=0.78, Re=10×10^6, α=0.3°, k=0.47).

For 18% biconvex aerofoil, numerical investigations (Fig. 4) show the wake deflections induced by type C shock oscillation. As the velocity of airflow and the velocity of shock are in opposite directions, the shock increases its strength and consequently induces further separation. In the same time the phase difference between type C shock oscillations on upper and lower surfaces will change the effective geometry, this deflects the

wake upward and downward respectively, similar to rapid deflections of a trailing edge flap. In the computational analysis the transition to turbulence was fixed on both the upper and lower surface at 3% chord. At Mach 0.76, Reynolds number of 10×10^6 and zero degrees incidence, the reduced frequency predicted by our computational results is 0.485, which compares favorably with experimental tests (0.476) [12, 24] and other computational

work (0.47) [25, 26]. This difference may be explained by the sensitivity of the predicted values of frequency of the SIO to the Mach number, grid spacing, wake and turbulent modeling [24, 27, 28].

As regards to type A shock oscillation (Fig. 5) the transonic periodic flow is initiated at a range of Mach number between 0.778 and 0.78, and non zero incidence by an unsteady disturbance, which is asymmetric and which increases the shock Mach number slightly to cause a shock induced separation, say on the upper surface. This would change the effective geometry (decrease in camber), which deflects the wave upward, similar to a rapid and upward deflection of a trailing edge flap. With an asymmetric wave and effective negative camber, the shock on the lower surface moves rearwards.

As the velocity of airflow and the velocity of shock are in the same direction, this will reduce the shock strength and keep the boundary layer attached on that surface. Meanwhile the effective change in geometry of the aerofoil should move the shock forward, which should initially increase the shock strength relative to the approaching airflow, but as the shock moves further forward it moves into upstream velocities which would result in a weaker shock and the boundary layer on the upper surface reattaches. This would result in a positive camber with a new shock on the lower surface moving forward. This cycle repeats itself. For small angles of incidence the reduced frequency of both type C and type A shock oscillation is roughly the same.
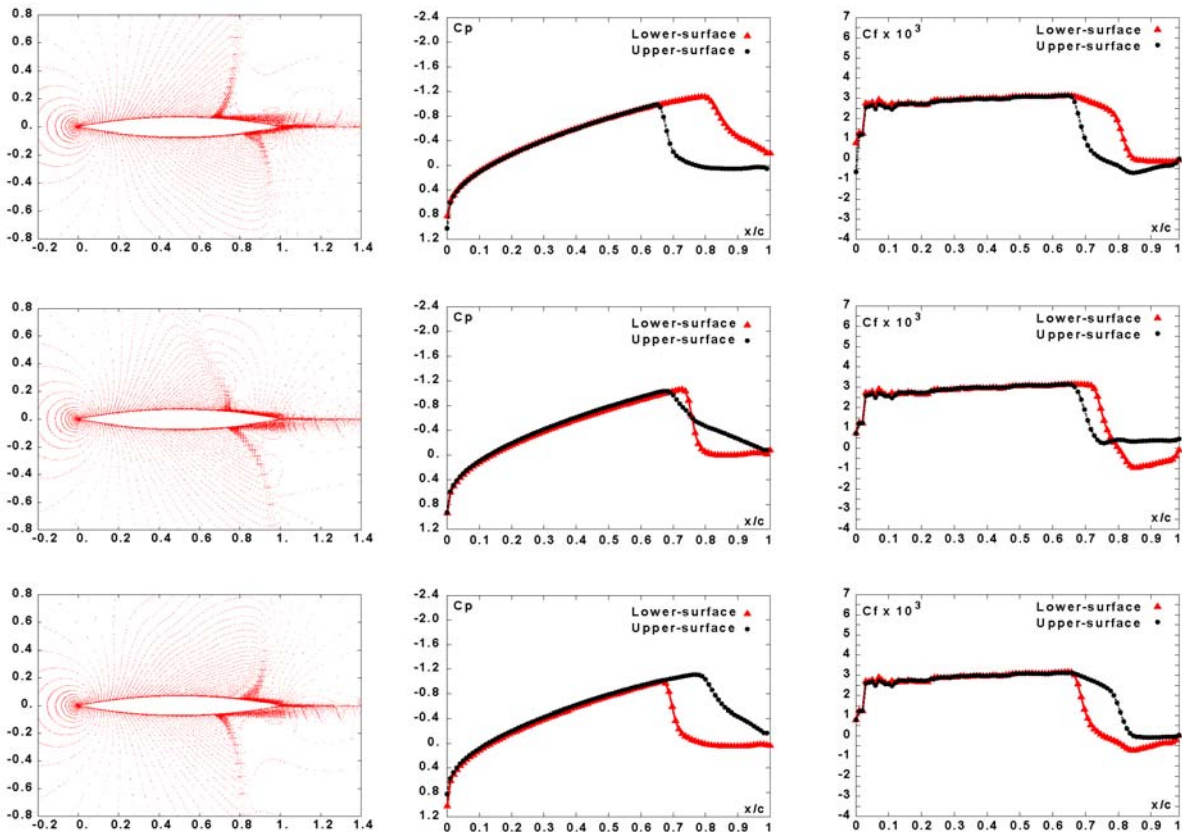
Fig. 6 The type B shock oscillation on the 14% thick biconvex aerofoil: Contours of Mach number, Pressure distribution and skin friction distribution at discrete time steps (M=0.83, Re=9×10^6, α=0°, k=0.57).

For biconvex 14% aerofoil the numerical results also compare well the experimental investigations. At M=0.83, Re 9 million and zero degrees incidence a Tijdeman's Type B SIO was numerically identified on the aerofoil (Fig. 6). During a part of the oscillation cycle

on upper or lower aerofoil surface, the shock wave intensity decreases considerably and the shock wave almost disappears on a face of aerofoil. As in the cases of types A and C SIO, the phase difference between type B shock oscillations on upper and lower surfaces will change the effective geometry, deflecting the wake upper and lower respectively, similar to rapid deflections of a trailing edge flap. For the same aerofoil, at the upper extremity of the periodic motion domain, at M=0.86, α=0.3, a Tijdeman's Type A SIO is identified on the aerofoil.

Generally, movement of the shock leads to the formation of pressure waves, which propagate downstream into the separated flow region [29]. On reaching the trailing edges the disturbances generate upstream moving waves. The waves interact with the shock wave and impart energy to maintain the limit cycle. The periodic motion on an aerofoil is sustained by the communication across the trailing edge and the frequency of the periodic motion is directly related to the time required for the signals to travel over the chord length. The necessary, but not sufficient, criteria for the periodic flow to occur is that the shock wave is strong enough to cause boundary layer separation.In the case of the free flight the cause of the inherent shock oscillations may be partially attributed to interaction with aircraft motion and angle of attack changes. For a three dimensional wing of varying chord the frequency of buffet excitation may be composed of frequencies attributed to the different chord lengths.

## 3 Drag alleviation and buffet control by surface cooling technique

The important effects of surface cooling on shock-boundary layer interaction and surface skin friction in transonic periodic flow suggest the possibility of influencing buffeting and viscous drag by surface cooling methods. The experimental and numerical investigations on this topic have been made in many research programs [30, 31, 32, 33]. This paper presents the heat transfer computations on 14% and 18% biconvex aerofoils. The numerical

investigations were performed with the explicit non-adiabatic boundary condition and using the adiabatic solution as initial condition.

For 14% biconvex aerofoil, the test conditions were M=0.83, Re=9×10⁶, α=0° with transition fixed on both the upper and lower surfaces at 3% chord. Computations on the aerofoil were performed for temperature ration of $T_w/T=0.6$. The resulting reduced frequency increases from 0.57 in adiabatic case to 0.645 with cooling. Figures 7, 8 and 9 show the buffeting alleviation and drag reduction (Fig. 9) obtained by surface cooling over a 14% biconvex aerofoil.
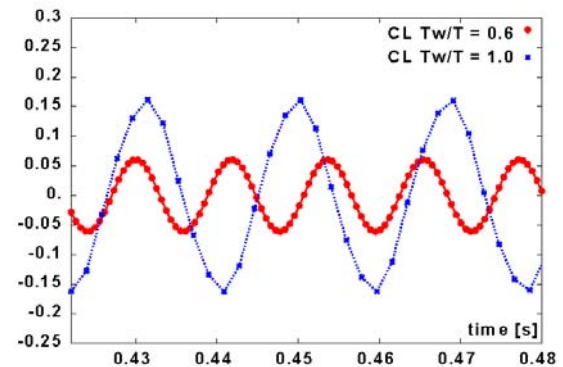
Figure 7 Lift alleviation during periodic motion by surface cooling over a 14% biconvex aerofoil (M=0.83, Re=9×10⁶, α=0°, k=0.645)
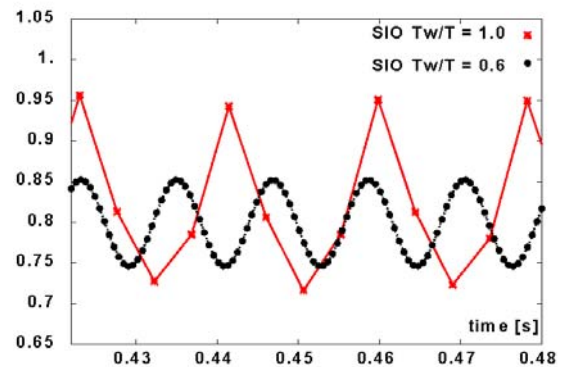

Fig. 8 Shock motion alleviation from type B to type A by surface cooling over a 14% biconvex aerofoil.

The cooling has produced a change in the type of shock oscillation (Fig. 8) from Tijdeman's type B (adiabatic conditions) to type A (surface cooling). For 18% biconvex aerofoil, the test conditions were M=0.76, Re=10×10⁶, α=0°, with a temperature ration of $T_w/T=0.9$ and

transition fixed as in the case of 14% biconvex aerofoil at 3% chord.
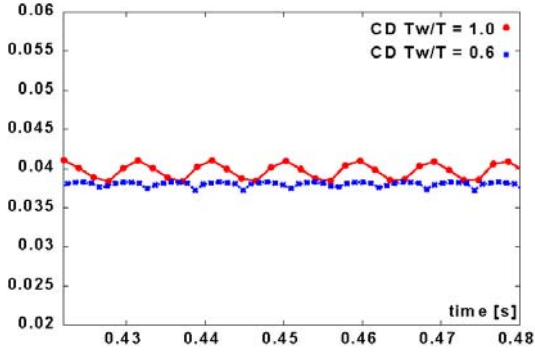
Fig. 9 Drag reduction by surface cooling over a 14% biconvex aerofoil (M=0.83, Re=9×10^6, α=0°, k=0.645).

For these conditions the resulting reduced frequency is roughly the same as in adiabatic case: 0.485. After cooling the Tijdeman's type C SIO from adiabatic case stays unchanged. However, the range of shock motion is reduced (Fig. 10) and buffeting is slightly alleviated (Figs. 11, 12).
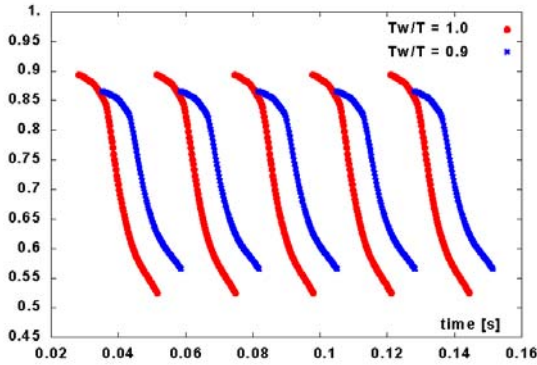

Fig. 10 Reduction of Type C SIO range on the upper surface of a 18% biconvex aerofoil (M=0.76, Re=10×10^6, α=0°, k=0.485).

On the other hand, the total drag is significantly reduced from an averaged value of 0.045 to 0.038. This could be explained by the viscous drag reduction through the skin friction, whereas the shock drag remains almost unchanged.

In spite of these optimistically results, the shock control methods by surface cooling are really efficient only in supersonic regime flow when the shock waves are not to strong. Therefore, seeing the present numerically

investigations, it seems that surface cooling is particularly efficient for drag reduction in transonic periodic flow.
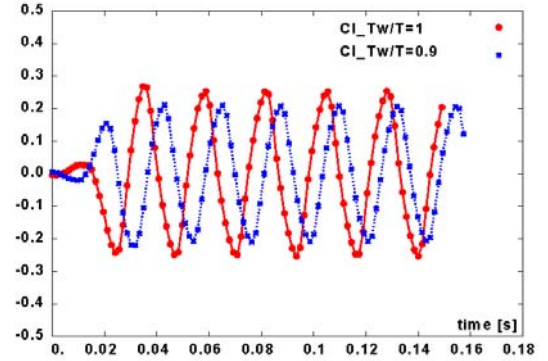
Fig. 11 Lift coefficient damping by surface cooling over a 18% biconvex aerofoil (M=0.76, Re=10×10^6, α=0°, k=0.485).
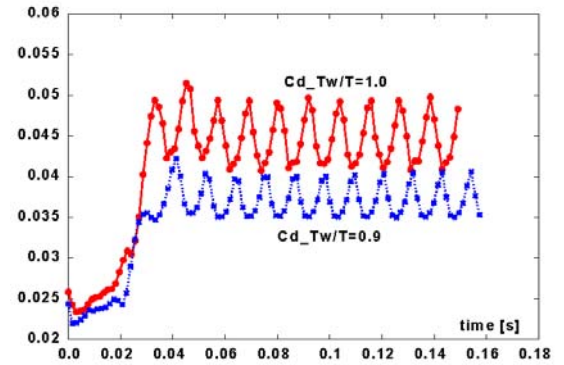

Figure 12 Drag reduction by surface cooling over a 18% biconvex aerofoil (M=0.76, Re=10×10^6, α=0°, k=0.485).

4 Drag alleviation and buffet control by adaptive bump

The possibility of influencing the buffet boundaries and drag alleviation by a bump has been examined in many research programs [34, 35, 36]. The shape and position of a contour bump was optimised to reduce drag and alleviate buffeting for a supercritical DA VA2 aerofoil [37]. In order to delay buffet onset on the OAT15A supercritical aerofoil, one uses a bump 65/30/33/14 [38]. The four bump geometrical control parameters are the starting point of the bump on the aerofoil surface ($10^2 x_1/c$), its extent ($10^2 \Delta x/c$), and the local coordinates of a single Bezier point ($10^2 \alpha$, $10^3 \beta$), which define the bump height and shape (Fig 13).

The positive effects of a contoured bump on a transonic periodic flow appear also from the numerical analyses of adaptive 14% and 18% biconvex aerofoils. The test case chosen for these biconvex aerofoils were $M_\infty=0.82$, $Re=9\times10^6$, zero degrees incidence and $M_\infty=0.76$, $Re=10\times10^6$, zero degrees incidence respectively. The bump, which is located in both cases on the lower surface of the aerofoil, underneath the range of shock oscillation, reduces both buffeting and the value of the average drag during periodic motion.
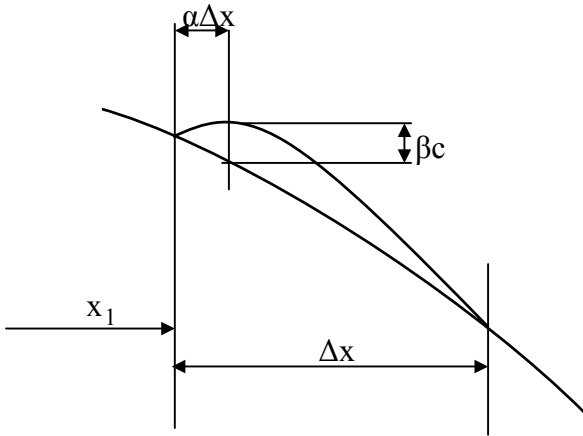

Fig. 13   Geometric bump parameters

For 18% biconvex aerofoil, the numerical results show an increase of the average value of C_l from zero to 0.035 (Fig. 14a) and a decrease of the average value of C_d from 0.045 to 0.038 (Fig 14b). As the type C shock oscillation on the upper surface of biconvex aerofoil shows no important change in characteristics of periodic motion, on the lower surface the bump changes completely the aerodynamic field of flow. The range of the type C shock oscillation is considerably diminished and a permanent shock wave of variable intensity appears upstream of the bump at $x/c=0.55$ (Fig. 15). The position of this permanent shock wave on the lower surface is the same as the position where the type C shock oscillation disappears on the upper surface.   The analysis of numerical results indicates also that the bump has a positive effect on drag and lift during a cycle of periodic motion. The time-averaged

variation of aerodynamic coefficient during a cycle of periodic motion may be computed using:

$$\frac{1}{T}\int_{-\frac{T}{2}}^{\frac{T}{2}} C_a(t)dt \quad ,$$

where C_a stands for the current aerodynamic coefficient. The bump increases the time averaged lift coefficient from zero to 0.078 (Fig 16a) and decreases the time averaged drag coefficient from 0.044 to 0.04 (Fig 16b). The numerical results show a tendency to reduce the frequency of adaptive wing from 0.485 to 0.474.
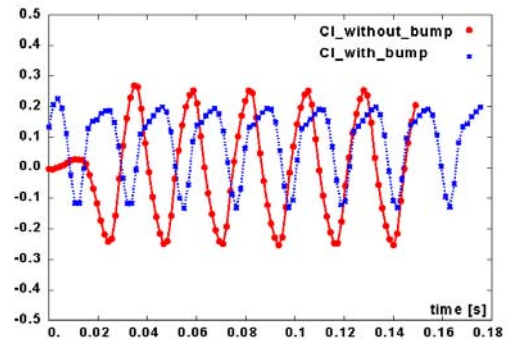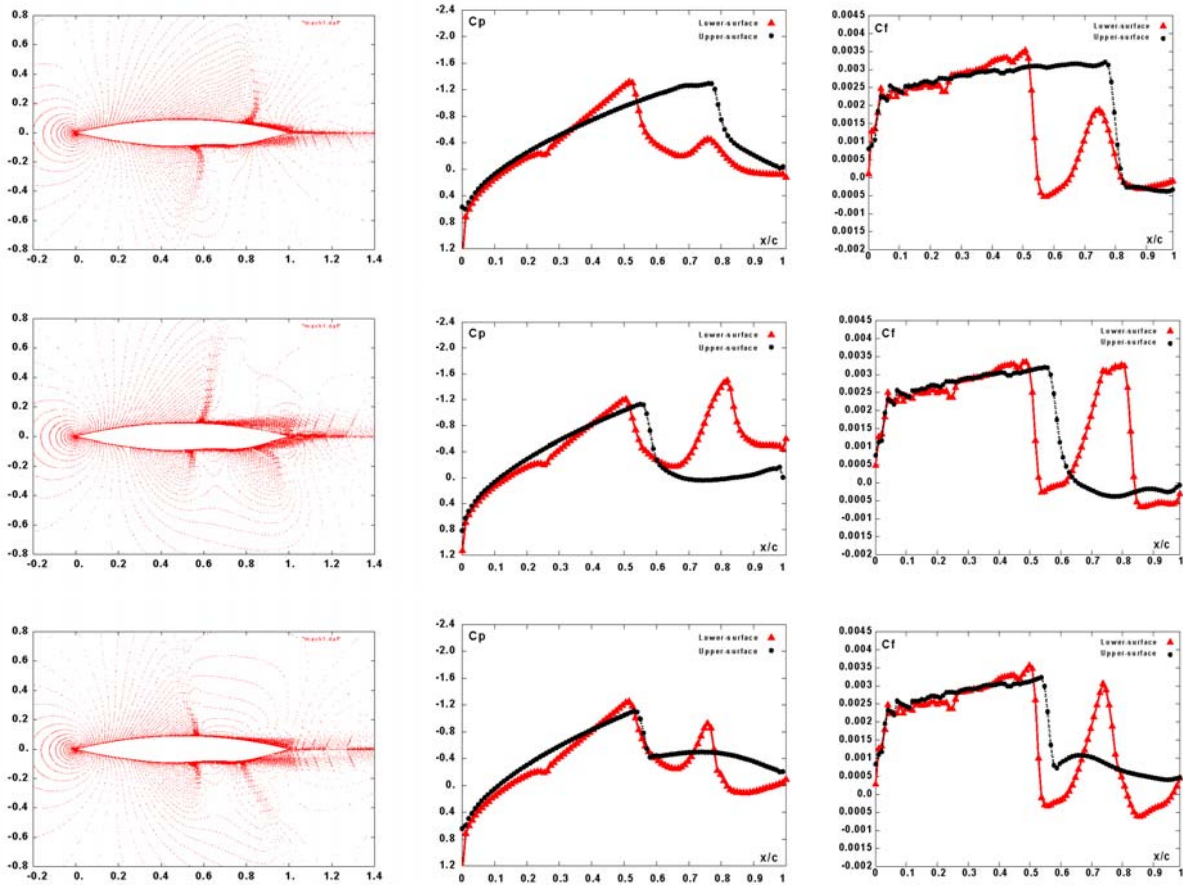

a
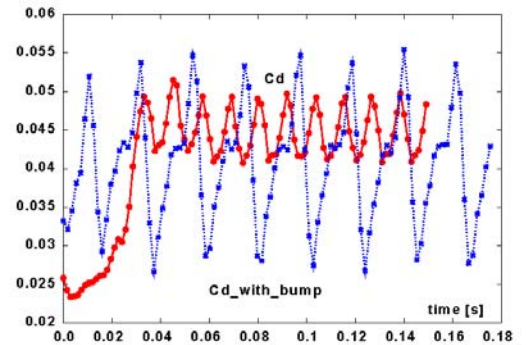
b

Fig. 14   Effect of a bump on periodic motion over an 18% thick biconvex aerofoil at $M_\infty=0.76$, $Re=10\times10^6$, $\alpha=0°$.

In the case of 14% biconvex aerofoil at $M_\infty=0.82$, $Re=9\times10^6$ and $\alpha=0°$, the same bump as in the other case was located on lower surface, underneath the mean shock position. The bump tends to alleviate slightly buffeting (Fig.17a), having also a significant positive effect on drag reduction (Fig. 17 b). In previous case the bump has tended to reduce frequency from 0.485 to 0.474. In this case the bump has

significantly reduced frequency from 0.52 (datum aerofoil) to 0.35 (adapted aerofoil). This change in frequency could be explained by the higher sensibility and better aerodynamic performances of thin aerofoils. The frequency change gives an additional dimension to the structural design of transonic wings for buffet. It could be also noticed that unlike supercritical aerofoil, where the reduced frequency is increased by the bump located on the upper surface underneath the shock [39, 40, 41], in the case of the biconvex aerofoil the bump, which is located on de lower surface, decreases the reduced frequency of the aerofoil. During a cycle of periodic motion the

Tijeman's type B SIO on the upper surface of the aerofoil suffers no important change. The shock wave tends to disappear during a part of the cycle of the type B SIO (Fig. 18). On the other hand, the bump changes completely the aerodynamic field of flow on the lower surface. As in the case of the 18% biconvex aerofoil the range of the type B SIO is bounded to the bump location, whereas a permanent shock wave of variable intensity appears upstream of the bump at x/c=0.68. This permanent shock wave interacts with type B SIO on the lower surface, leading to a spread lambda configuration and therefore to an increase of viscous drag and a decrease of shock wave loses respectively.
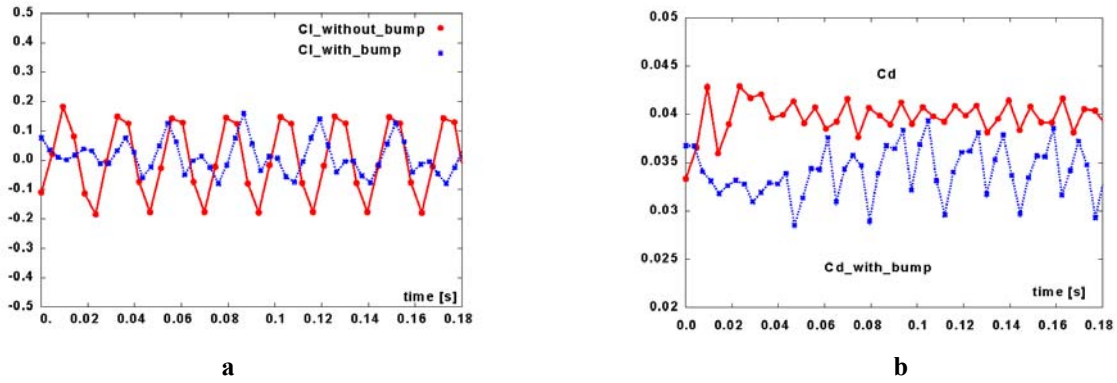

Fig. 15 Effect of a bump on periodic motion over an 18% thick biconvex aerofoil at M_∞=0.76, Re=10×10^6, α=0°, k=0.474: Mach contours, Pressure distribution and skin friction coefficient at several intervals during a period of oscillation.
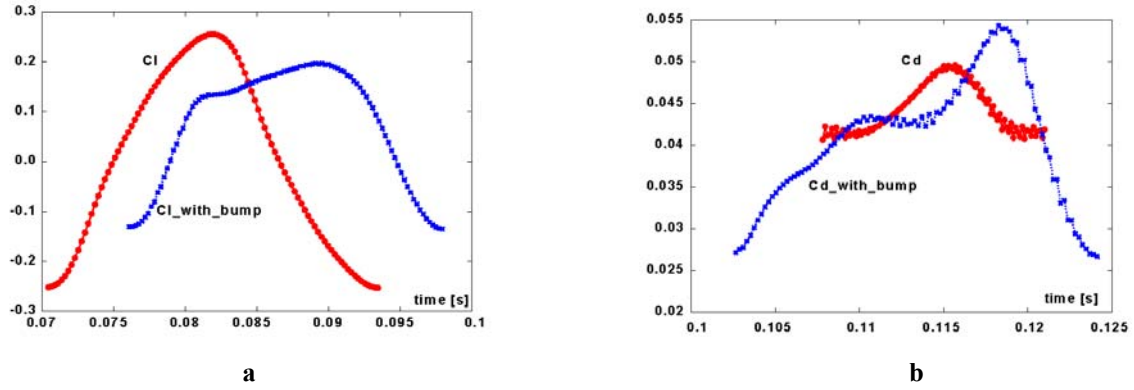
Fig. 16 a, b Aerodynamic coefficients during a cycle of periodic motion: **a** Time averaged lift increasing, **b** Time averaged drag reduction.

Fig. 17 a, b Effect of a bump on periodic motion over a 14% biconvex aerofoil at M_∞=0.82, Re=9×10^6, α=0°: **a** increase of the average lift, **b** decrease of the average drag.

Frame 1

Frame 2

Frame 3

Frame 4

Fig. 18 Effect of a bump on periodic motion over a 14% thick biconvex aerofoil at M_∞=0.82, Re=9×10⁶, α=0°, k=0.35: Mach contours, Pressure distribution and skin friction coefficient at several intervals during a period of oscillation.
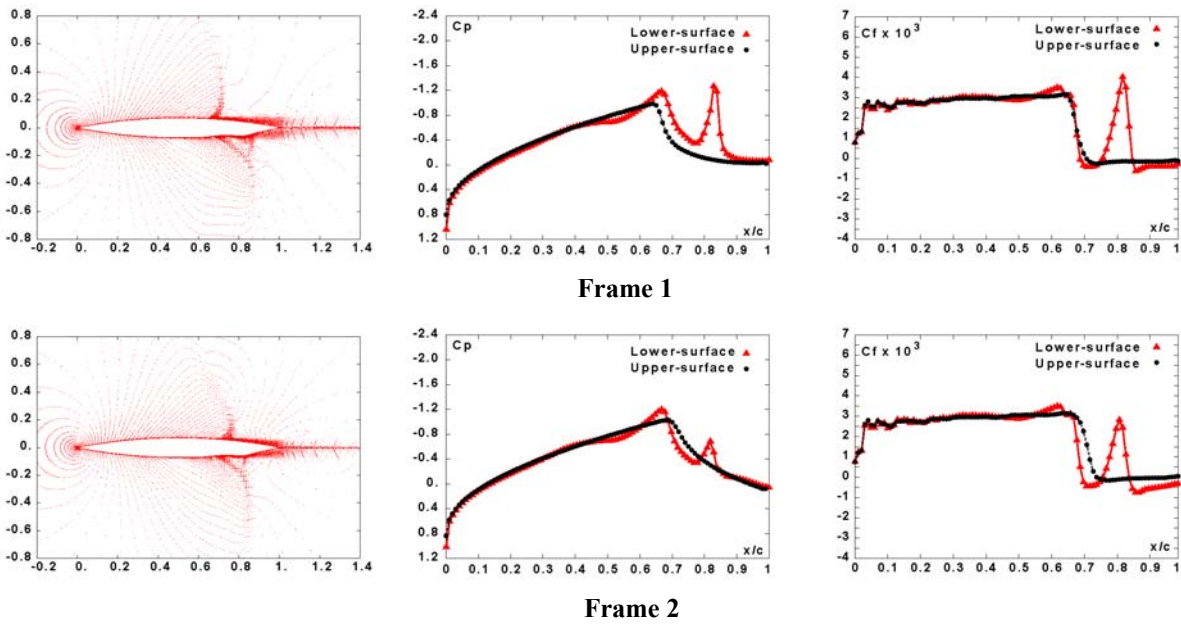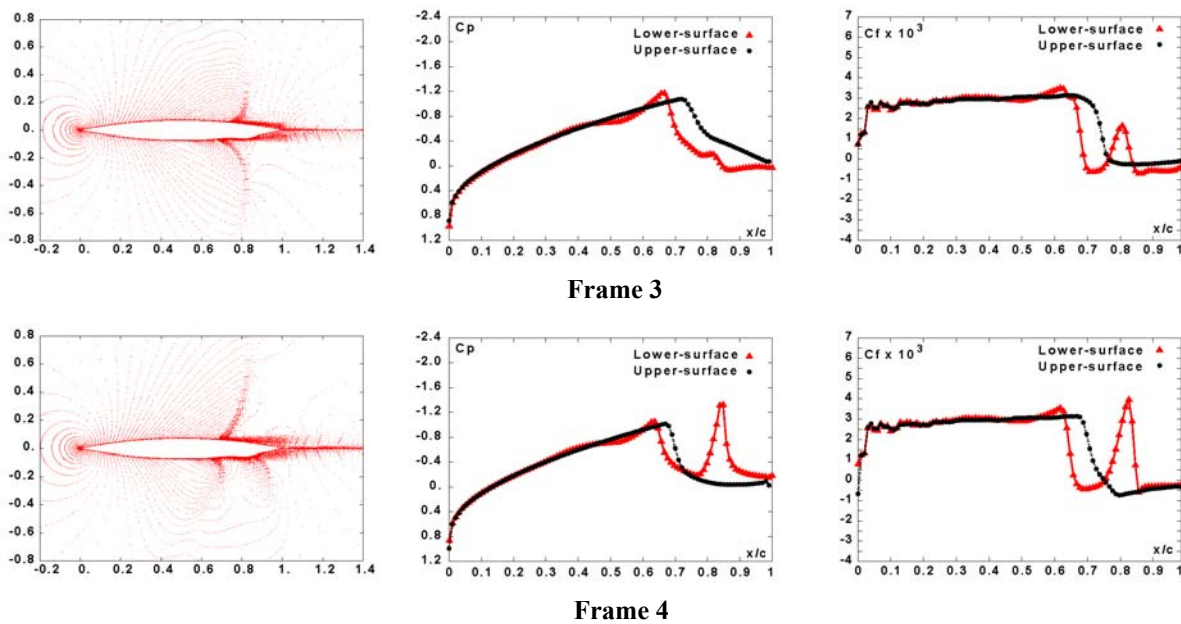
5 Concluding remarks

An investigation into the effect of flow control by a bump and surface cooling on transonic periodic flow over biconvex aerofoils was performed using Computational Fluid Dynamics. The Tijdeman's types A, B and C SIO have been well numerically identified on datum biconvex aerofoils in the range of transonic periodic flow found experimentally by McDevitt and Gibb. The computations on adapted 14% and 18% biconvex aerofoils as well as on cooling datum biconvex aerofoils indicate that both flow control techniques are effective in drag reduction and buffeting alleviation in transonic periodic flow.

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