A Numerical Investigation on Transonic Flow around a Biconvex Circular Arc Airfoil in a Channel

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Abstract *Shock wave boundary layer interactions (SWBLI) are not only fundamental research topics of aerodynamics but are observed in practical high-speed internal flows. Shock induced oscillations (SIO), aerodynamics instabilities (buffet), high cycle fatigue failure (HCF), nonsynchronous vibration (NSV), flutter and so on are detrimental consequences of this unsteady interaction. In the present study, a numerical computation has been performed to investigate the transonic flow around a biconvex circular arc airfoil in a two dimensional channel. Reynolds averaged Navier-Stokes equations with k-ω shear stress transport (SST) two equation turbulence model has been applied for computational analysis. The behavior of the flow field has been studied from pressure ratio (ratio of back pressure to inlet total pressure) of 0.75 to 0.65 with decreasing the downstream pressure. Several points have been set over both the upper and lower surfaces of the airfoil to investigate the static pressure history with time. The result shows that the flow field, shockwave type and its movement as well as frequency of movement vary with pressure ratio. For computational validation, obtained results have been compared with available experimental data.*

Key Words: Transonic flow, Shock wave, Boundary layer separation.

1. Introduction

When an airfoil is subjected to transonic flow, shockwave is generated on the airfoil surfaces. This shockwave lead to a rapid rise in drag. The flow separation occurs due to the sudden pressure rise across the shockwave. This separation is known as shock induced boundary layer separation. McDevitt *et al.* (1976) observed an experimental and theoretical study of transonic flow over a 18% thick airfoil. The results stated that shock-boundary-layer interaction phenomena are strongly dependent on Mach number and Reynolds number. Hendrik and Tijdeman (1977) had described the behavior of the transonic flow around an oscillating airfoil. In this study, they worked with the exploratory wind-tunnel experiments in high-subsonic and transonic flows. The interaction of steady and unsteady flow fields

and periodic motion of the shock are focused in the study. Levy (1978) had described an experimental and computational investigation of the steady and unsteady transonic flow fields at different Mach number. Tijdeman and Seebass (1980) had added different information in understanding the transonic flow past oscillating airfoils. They suggested a 3-D simulation to get more realistic results for low aspect ratio wings. Gronland *et al.* (1998) had discussed the accuracy obtained in predictions of unsteady transonic flows by a modern CFD method. The result shows that modern CFD methods can indeed predict the complex buffet phenomena with reasonable accuracy. However, the mechanism of self-excited shock wave oscillation around an airfoil in internal flow is still not fully clear.

This study investigates the flow over a biconvex circular arc airfoil in a channel at different pressure ratio which is defined as ratio of back pressure to inlet total pressure. The pressure ratio is varied by decreasing the back pressure while keeping the inlet total pressure constant. The flow behavior, aerodynamic characteristics are discussed for all the flow conditions.

2. Numerical Method and Boundary Conditions

The governing equations for the present problem is the unsteady Navier-Stokes equations with energy equation. To model the turbulence in the flow field, a two equation *k-ω* SST turbulence equation is used. The governing equations are discretized with finite volume method. During spatial discretization, second-order upwind condition was used for flow, turbulent kinetic energy, *k* and specific dissipation rate, *ω*. Second-order time accurate computation was performed with time step of 10^{-6}. The viscosity was considered to vary according to Sutherland's law.

The airfoil is 12% thick with chord length of 48 mm. the study is carried for chord Reynolds number 5.8×10^5 and angle of attack $0°$. The flow conditions are varied based on the pressure ratio, p_b/p_{01} which is defined as the ratio of back pressure to inlet total pressure.

Pressure inlet and pressure outlet boundary conditions are used at inlet and outlet, respectively. The inlet total pressure was kept constant at 101325 Pa and outlet

pressure is varied for different cases. The airfoil and the channel surfaces are considered as no-slip wall.
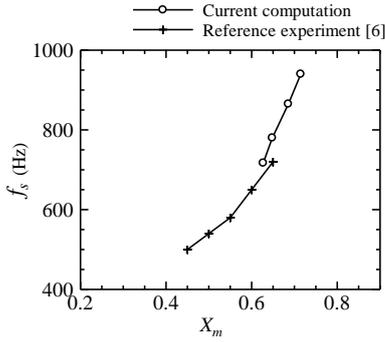

Figure 1. Frequency against mean shock position

3. Results and discussions

To validate present numerical simulation the computational results are compared with experimental results. The present computation is for 12% thick airfoil but the experimental results of Xiong et al. (2010) are for 18% airfoil. The frequency is plotted against mean shock position in figure 1 which shows that the computational and experimental result have the same trend of frequency variation with the shock position. At the lower pressure ratio (p_b/p_{01} = 0.68) results are very close. A slight discrepancy in frequency may be noticed at higher pressure ratio for the effect of span wise flow in real experimental study. The plot validates the results from the current computational analysis. The results and findings from this study are discussed below:

For higher pressure ratio (greater than 0.75) the flow field is steady and subsonic throughout the region. For pressure ratio 0.74 to 0.72 the a local supersonic flow region is observed. But the flow field shows the steady characteristics. If pressure ratio goes down 0.71 to 0.68, the low field becomes unsteady. The shock appeared on both upper and lower surfaces start to oscillate with time. For further lower pressure ratio the flow field becomes steady again with stronger shock wave. Figure 1 illustrates the sequential contour maps for pressure ratio 0.68. The dense region indicates the drastic rise in velocity of fluid particle as well as the presence of shock wave. Within the pressure ratio range (for which flow field is unsteady), the shock for higher pressure ratio is normal type. Then the shock wave becomes λ shaped which is a combination of normal and oblique shock wave.

For pressure ratio 0.68 the flow consists of a λ shaped unsteady shock wave. To investigate the behavior of the shock, the time period (T) of one full oscillation is divided into eight segments. From the sequential contour maps of the Mach number (figure 2) we can understand the unsteady characteristics of the shock wave.

Here time has been made dimensionless by dividing with time period (T). The frequency of the shock oscillation is not uniform over the entire surfaces of the airfoil. This can be understood from the pressure history over the
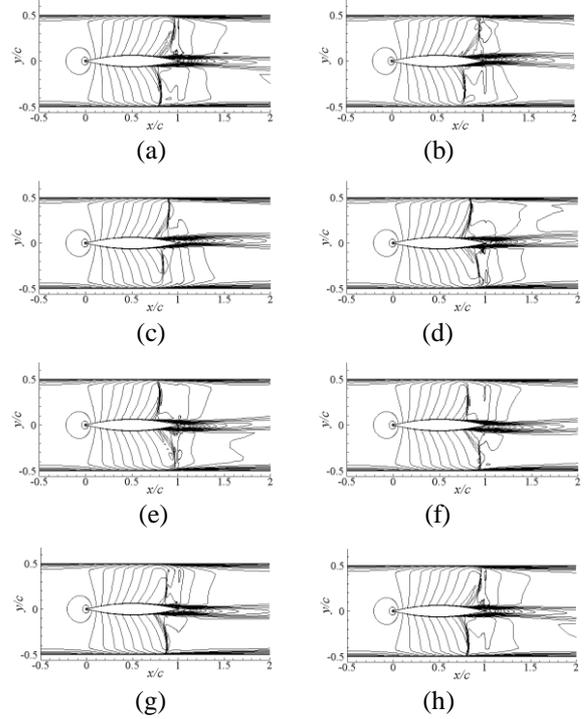

(a)   (b)

(c)   (d)

(e)   (f)

(g)   (h)

Figure 2. Sequential contour maps of Mach number during one cycle for p_b/p_{01}= 0.68; (a) t/T = 0, (b) t/T = 1/8, (c) t/T = 2/8, (d) t/T= 3/8, (e) t/T= 4/8, (f) t/T = 5/8, (g) t/T= 6/8, (h) t/T= 7/8.
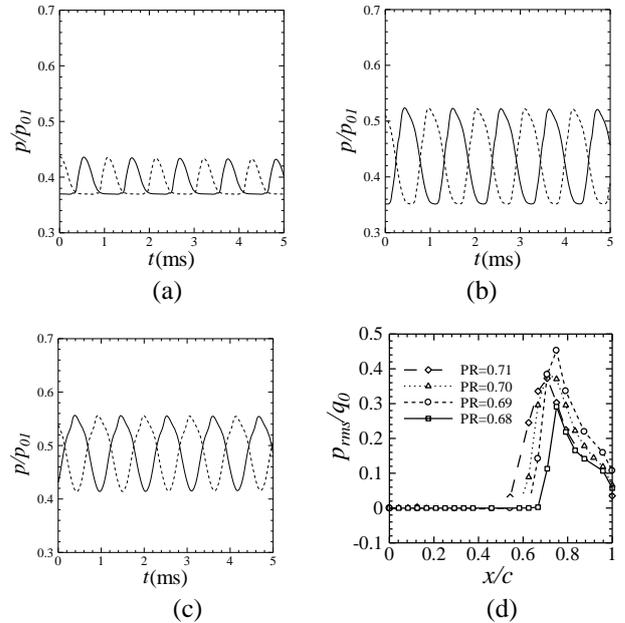

Figure 3. Time histories of static pressure at p_b/p_{01} = 0.68; (a) x/c=0.708, (b) x/c=0.75, (c) x/c=0.792 (d) RMS pressure fluctuation for different pressure ratio. The solid line and the dashed line represent the upper

and lower surface static pressures, respectively.

airfoil surfaces. Also the pressure fluctuation at different points on the airfoil is not same. The pressure fluctuations on both upper and lower surface of the airfoil are shown in figure 3.
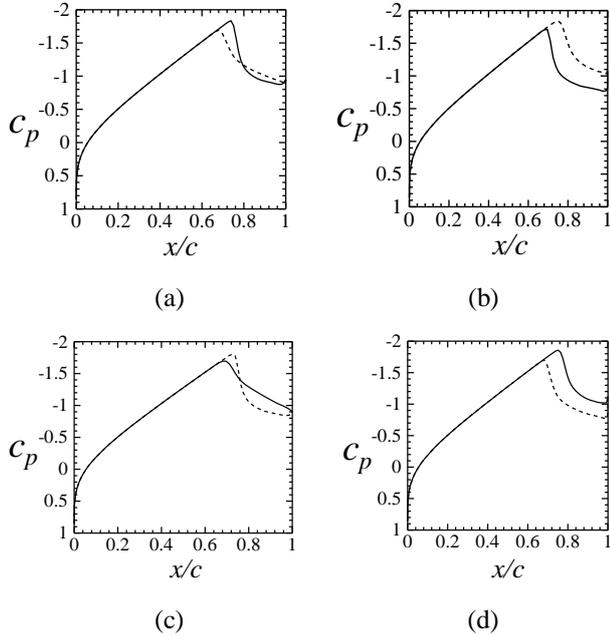

(a)

(b)

(c)

(d)

Figure 4. Distribution of c_p for p_b/p_{01}=0.68; (a) $t/T = 0$; (b) $t/T = 2/8$; (c) $t/T = 4/8$; (d) $t/T= 6/8$. The solid line and the dashed line represent the upper and lower surfaces, respectively.

From figure 3 it is abundant the pressure fluctuation at different points on airfoil is different. To get the pressure history over airfoil surfaces several points are selected on both the surfaces. Pressure fluctuation increases considerably after the half of the chord length. Also the frequency of pressure fluctuation depends on x/c. The RMS pressure fluctuation plot for different pressure ratio is shown in figure 3 (d). From figure it can be concluded that the maximum fluctuation varies with the pressure ratio and the position of maximum pressure fluctuation moves towards the trailing edge with decrease in pressure ratio. Also the frequency of the shock oscillation is different at different x/c. For this investigation the frequency is measured where the RMS of static pressure fluctuation is maximum. Here the pressure fluctuation is normalized by dividing with inlet dynamic pressure (q_0).

To understand the pressure history the pressure coefficient (c_p) is calculated from the static pressure over the airfoil. From the drastic change in pressure coefficient the presence of shock wave and the movement of shock can be realized. The pressure coefficients on both the surfaces are shown in figure 4 (after time duration t/T=1/4).

The position of shock can be found either from the Mach contour or the pressure coefficient plots. The shock from the pressure coefficient plots easily. From the pressure coefficient and mach contour, it is clearly understood that the shock wave moves periodically and the shock type, movement of shock, frequency of shock movement also depend on the pressure ratio. The shock movement
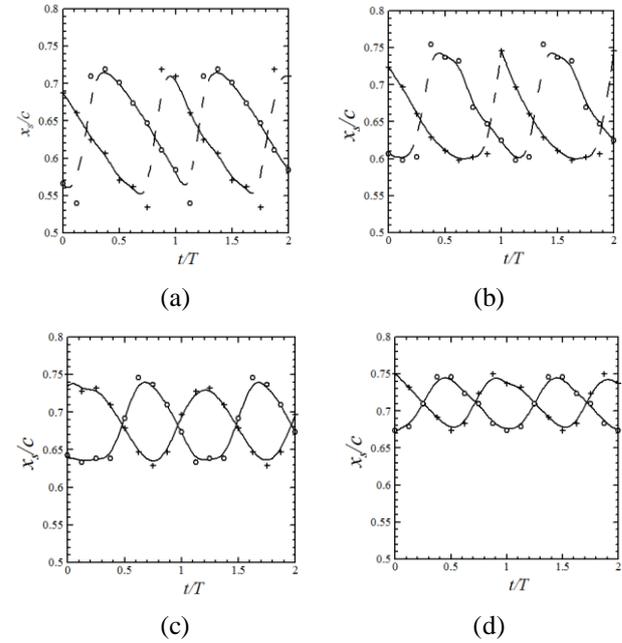

(a)

(b)

(c)

(d)

Figure 5. Time histories of shock position (a) $p_b/p_{01} = 0.68$, (b) $p_b/p_{01} = 0.69$, (c) $p_b/p_{01} = 0.70$, (d) $p_b/p_{01} = 0.71$.

for different pressure ratio is shown in figure 5. Here x_s is the shock position. The shock position is predicted from the average position of the drastic pressure change. The shock position is found

The shock position found from the pressure coefficient plots are plotted against the time in figure 5. The figure shows the movement of shocks on both surfaces for time equals twice of the time period. From figure 5 it can be concluded that shock on both upper and lower surface moves in opposite direction. They are almost always 180^0 in phase. The mean shock position moves toward the trailing edge (TE) and the shock starts to move within a narrower region while decreasing the pressure ratio

In figure 5 the solid line with circles and with cross represent movement of shock on upper and lower surface respectively. The dashed line represents no shock. For higher pressure ratio (0.71 and 0.70) the shock movement is discontinuous and there is no shock on lower surface when there is a shock on upper surface and vice versa. The shock on both upper and lower surface starts to move from trailing edge toward the leading edge. This type of shock oscillation is known as

Tijdeman (1977) type B. But for lower pressure ratio (0.69 and 0.68) the movement is continuous on both surfaces. For this cases the shock on both surfaces are present for all the time. The shock on upper surface moves toward the trailing edge while the lower surface shock moves toward the leading edge. This type of shock oscillation is known as Tijdeman (1977) type A. Meanwhile the movements of shock for all cases are periodic. The frequency of the shock oscillation is also calculated from the pressure history plot (where the RMS pressure fluctuation is maximum).
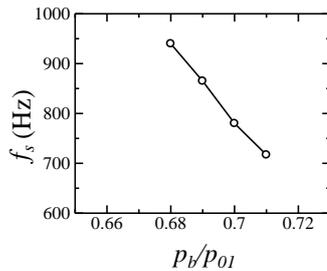

Figure 6. Variation of frequency with pressure ratio

The frequency increases with the decrease in pressure ratio. This relationship is illustrated in figure 6. The plot shows almost a linear relation of frequency with the pressure ratio. This is also confirmed by many relevant experimental studies. The frequency at pressure ratio 0.71 is 717 Hz. However, at pressure ratio 0.70, 0.69 and 0.68 observed frequencies are 780, 865 and 940 Hz, respectively.

4. Conclusions

A numerical computation is carried out to investigate the transonic flow around a circular arc airfoil in a two-dimensional channel. The pressure ratio is varied to get the detail picture of compressible flow field around the airfoil. From this study the following conclusions can be made:

i. The flow field characteristics vary with the pressure ratio and there is a steady-unsteady interaction in the flow field.
ii. Normal shock wave generated is generated at higher pressure ratio.
iii. λ- shock wave is seen at lower pressure ratio.
iv. The movement of shock is discontinuous Tijdeman type B at higher pressure ratio and continuous Tijdeman type A for lower pressure ratio.
v. The shock wave on upper and lower surfaces moves in opposite direction to each other for both types of oscillation. The shock oscillates within a narrower region at higher pressure ratio.
vi. The frequency of the shock oscillation increase with decrease in pressure ratio.

vii. The mean shock position moves toward the trailing edge with decrease in pressure ratio.

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