Proceedings of the
International Conference on Mechanical Engineering 2007
(ICME2007) 29- 31 December 2007, Dhaka, Bangladesh

ICME07-FL-31

NUMERICAL ANALYSIS OF SHOCK AND BOUNDARY LAYER CONTROL OVER NACA0012 BY CONTOUR BUMP, SURFACE COOLING AND HEATING

Sukanta Bhattacharjee, M. Saifuddin Ahsan, Mithun Saha And Mamun Mohammad

Department of Mechanical Engineering, BUET, Dhaka-1000, Bangladesh.

ABSTRACT

In transonic flow conditions, the shock wave boundary layer interaction and flow separations on wing upper surface of a civil aircraft induce transonic drag divergence. In the context of adaptive wings for future transport aircraft the potential of a shock control bump for reducing wave drag is investigated numerically. The cause of wave drag are the strong shocks developing on the upper side of the wing creating an increase in entropy as well as an increase in boundary layer thickness eventually leading to separation. Bumps are employed to decrease the strength of the shock and adverse pressure gradient at the upper surface of NACA0012 airfoil, hence reduce the flow separation caused by the interaction of strong shock and boundary layer. Different bump locations and shapes are investigated to optimize the result form different approaches along with the effect of heat transfer through the aerofoil surface.

Keywords: Transonic drag, Shock Control bump, Surface heat transfer.

1. INTRODUCTION

Transonic aerodynamics is the focus of strong interest at the present time because it is known to encompass one of the most efficient regimes of flight. Most of todays long-range transport aircraft cruise within the transonic speed regime in order to maximize the operational range as well as the cruise speed. This flight regime is characterized by the appearance of shock waves, which terminate regions of supersonic flow. These shock waves cause a steep increase in wave drag with increasing Mach-number and angle of attack, the transsonic drag-rise, makes it in addition a very critical design aspect, since it basically limits the maximum economic speed of the aircraft. Flying an aircraft at slightly (higher) off-design Mach numbers means a large drag and thus fuel-burn penalty.Therefore, shock control is a promising mean for increasing aerodynamic efficiency, speed, as well as mission flexibility during cruise flight for future transport aircraft.

Two effects contribute to the transsonic drag-rise, the entropy increase in the shock itself and the thickening of the boundary layer and may be subsequent separation caused by the pressure field induced by the shock onto the boundary layer. The first approach to reduce wave drag would be to change the overall pressure distribution in such a way, that the formation of strong shocks is avoided while maintaining lift at a constant level. The second, more direct approach, is to locally control the shock boundary layer interaction.

Ashill, Fulker and Shires[1] proposed the so-called shock control bump (SCB), which is a local concave-convex-concave geometry modification of the clean airfoil near the foot point of the shock. At the concave upstream flank of the SCB isentropic compression waves are induced which reduce the pre-shock Mach-number and, thereby, the wave drag without destabilizing the boundary layer too much. Due to large variations of the shock position at transonic flight with changing Mach-number as well as lift coefficient, shock control bumps have to be adapted to the actual flow condition in order to realize an overall positive effect.

Along with bumps several devices have been studied [2],[3] to alleviate this negative affect like sub boundary layer mechanical devices, vortex generators, boundary layer suction/blowing, continuous or pulse skewed air jets and synthetic jets. However, taking into account the antagonistic mechanisms that are at work, it is difficult to find a control technique that could decrease the total drag, since one has to find a compromise between friction and wave dag losses.

The heat transfer between the airfoil and the flow field has an important influence on the laminar or turbulent-boundary layer development, setting out characteristic boundary-layer and turbulent transitions, having also a significant effect on the shock wave boundary-layer interaction. Furthermore cooling has a significant effect on skin friction at the surface of the aerofoil[4][5]. In supersonic regions cooling decreases the velocity and therefore the skin firction whereas at subsonic speed, cooling increases the velocity gradients and hence skin friction.

The paper presents numerical analysis of transonic flow over a NACA0012 aerofoil, with and without bump located at the mean shock position along with heat transfer through aerofoil surface, at Mach 0.7 and Reynolds number of $9x10^6$ (based on the chord length) with an angle of attack 3.2^o for evaluating the influence of SC bump on drag reduction. Considering the high aspect ratio of modern transport aircraft wings, a two dimensional approach seems to be adequate.

2. CFD MODELING

A finite volume method CFD code was used on both the original airfoil, as well as, the airfoil with SC bump.The model airfoil had a chord length of 1m. The computational domain was extended 11.5 and 21 chord lengths upstream and downstream of the blade, respectively, as shown in Fig.1. Fig.2 shows a closer view of the grids. 33550 quadrilateral cells and 34135 nodes were used.

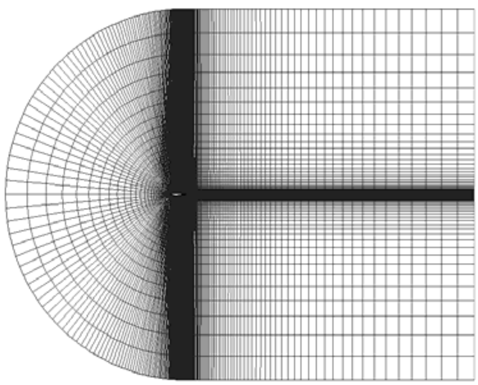

Fig 1: The computational domain.

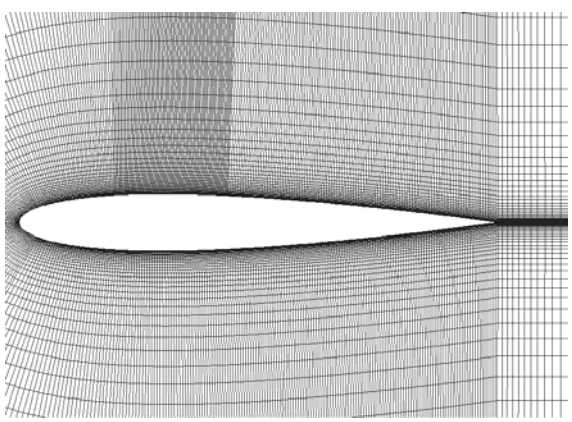

Fig 2: Extended view of the grid around aerofoil.

Density based implicit solver that solves continuity equation, momentum equation and energy equation was used because of high speed flow calculation and for better resolution of the shock wave. As turbulent model vorticity based 1 eqn Spalart-Allmaras method was used, which gives satisfactory result for shock wave boundary layer (SWBL) interaction. The viscosity of air is calculated with three coefficient sutherland method. Used boundary values for the calculations are given below.

2.1 BOUNDARY CONDITIONS

Condition	Value
Gauge Pressure (pascal)	101325
Mach Number	0.7
Temperature (k)	300
X-Component of Flow Direction	0.99844074
Y-Component of Flow Direction	0.055821501
Turbulent Specification Method	2
Modified Turbulent Viscosity (m2/s)	0.001
Turbulent Intensity (%)	0.1
Turbulent Length Scale (m)	1
Hydraulic Diameter (m)	1
Turbulent Viscosity Ratio	10
Wall Roughness Constant	0.5

For both flow and modified turbulent viscosity discretization second order upwind scheme was used. The CFL and URF values are increased gradually during the calculation for better stability and covergence of the solution.

3. THE SHOCK WAVE BOUNDARY LAYER INTERACTION (SWBLI)

In flows without boundary layers a shock wave would meet, be generated or reflected by a solid surface. In such flow the pressure at the surface would increase discontinuously across the shock. However, the preseance of a viscous boundary layer does not allow this to happen, as the inner part of the boundary layer has subsonic velocities and discontinuities are not possible.

In real flows, the interaction has a complicated structure due to the mixed flows regions with adjacent subsonic and supersonic regions. The viscous boundary layer is predominantly subsonic, which allows pressure disturbance to be transmitted in both upstream and down stream directions. The interaction generates large shear gradients normal to the wall and at the same time the low energy air is dragged downstream. With in the outer supersonic region the effect of viscosity is relatively small and the flow can be defined in terms of the shock equations[6]. The velocity contours created after the shock meets the upper suface boundary layer is shown in figure3, which also depicts the thick boundary layer after the shock. And the imminent result of all this is divergence of drag in this flight envelope. The huge increase of drag in transonic region is shown in figure 4. The total drag in this regime is the sum of form drag, viscous drag and wave drag. As the velocity increases the shock wave appears more down stream and when it completely leaves the solid surface the drag dramatically decreases as a result of no additional boundary layer thickening by the SWBLI which is shown in figure 4.

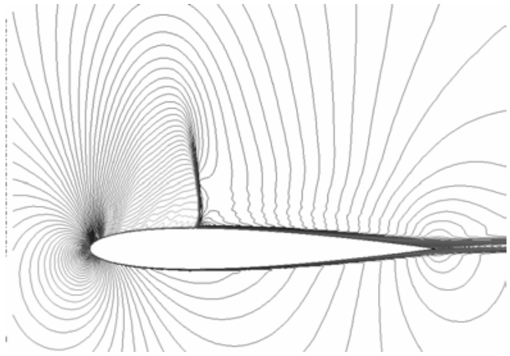

Fig 3: Velocity contour of the flow field on NACA0012 at Mach 0.7, 3.2 ^{o} angle of attack

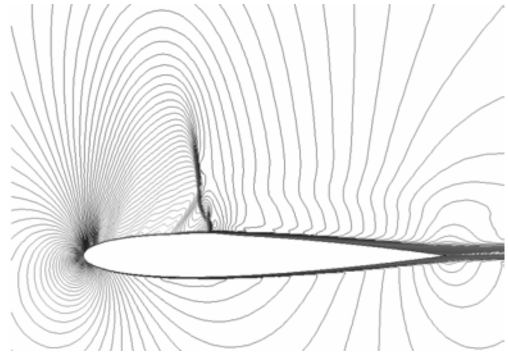

Fig 5: The effect of bump creates a shock that resembles lambda.

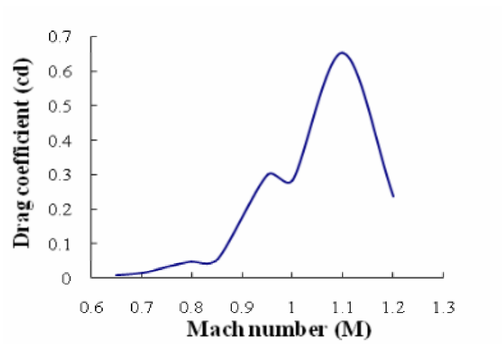

Fig 4: Drag divergence at transonic regime

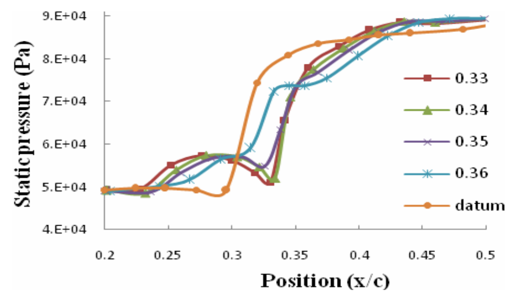

Fig 6: Distribution of pressure at upper surface for different bump crest location at 0.35 % bump height.

4. THE EFFECT OF BUMP PARAMETERS IN ALLEVIATING TOTAL DRAG

The shock control bumps map the mechanism of flow deceleration by isentropic compression waves of airfoils to a smaller scale, they also share their sensitivity to changing free stream conditions. During the optimization each SCB design is analysed for single flow conditions of 0.7 Mach and 3.2 ^{o} angle of attack.

The SC bump is designed so that it weaken the shock wave by creating a lambda type shock while does not increase the skin friction drag noticeably. The bump starts at a position upstream of the shock root and the crest sitiuated at just downstream of the shock root. The position of the bump and the lambda shock is shown in figure 5. The reduction of the wave drag and viscous drag is directly related with the bump height, length, shape, start position and crest position. The maximum 15.8 % reduction in total drag is so far achieved.

Figure 6 and 7 shows the adverse rise of pressure in the wall adjacent cells from 20% chord length to 50% chord length at the upper surface of the airfoil for both datum and SCB airfoil. At relatively backward position the rise of pressure is less steep then that of at reltively forward. The graphical information is presented here for 0.35% and 0.40% bump height. The gradual pressure rise has an impression of the lambda shock root. The more spread the lambda shock root the more the gradual rise of the pressure. A sheer pressure rise indicates strong shock hence more wave drag. After the implimentation of bump greater than 22 % reduction of wave drag is achieved which is shown in figure 8.

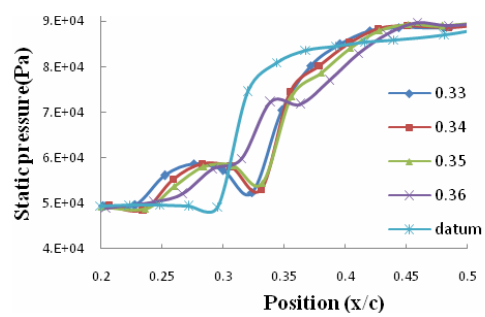

Fig 7: Distribution of pressure at upper surface for different bump crest location at 0.40 % bump height

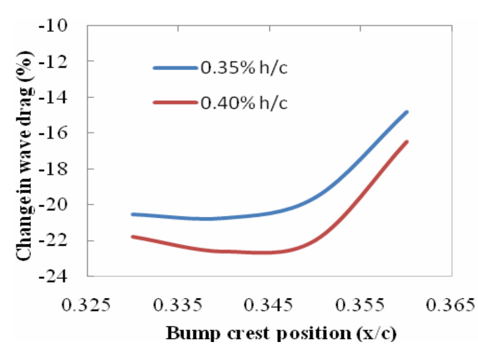

Fig 8: Wave drag reduction for different bump crest location.

Besides wave drag the thickening of boundary layer arise another problem of seer increase in viscous drag. At the interface of boundary layer the shock wave starts smearing as the subsonic boundary layer allowes pressure gradient to pass through it. The high pressure fluid after the shock tends to push the fluid downward and even reverse the fluid at higher mach number and / or higher angle of attack. Large pressure gradient indicated in figure 7 is a measure of flow separation tendency. After applying bump the the region of pressure rise is spreaded. It is obvious that the X wall shear stress in figure 9 for datum airfoil upper wall is near zero i.e. the fluid about to stop after the shock is improved when the bump is applied.

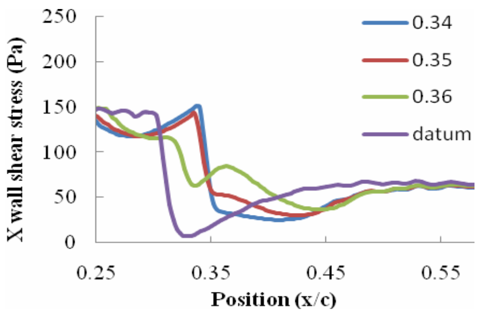

Fig 9: X wall shear stress in the upper wall for datum airfoil and for 0.40 % bump height for different location.

Properly designed bump can successfully handle wave drag which further decrease the thendency of flow seperation hence viscous drag without further increase in other drag components. This results in total drag deminished by more than 15 % and increase the efficiency i.e. cl/cd of the airfoil by more than 22 %. This is indicated in figure 10 and 11.

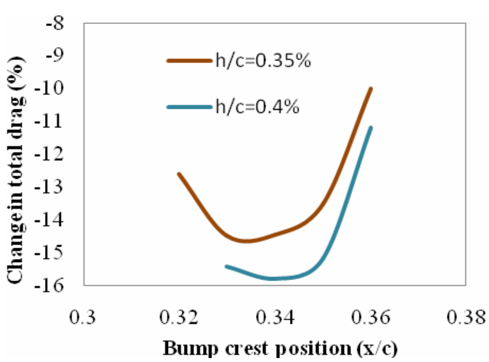

Fig 10: Total drag reduction for different bump crest location.

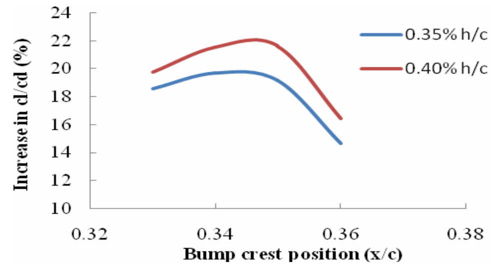

Fig 11: Increse in cl/cd with respect to datum cl/cd for different bump crest location.

5. SURFACE COOLING AND HEATING

The effects of surface cooling on shock-boundary layer interacrion and surface skin fricrion in transonic flow suggest the possibility of influencing viscous drag by surface cooling methods. The numerical investigation were performed with the explicit non-adiabatic boundary condition and using the adiabatic solution as initial condition. For NACA0012 airfoil, the test conditions were 0.7 Mach, 3.2^{o} angle of attack, with a temperature ration of T_w/T=0.9. For this condition total drag is reduced by 1.7 % form the average value. This could be explained by the viscous drag reduction through the skin friction, whereas the wave drag remains almost unchanged. For temperature ration of T_w/T=1.33 the total drag is increased about 1 %.

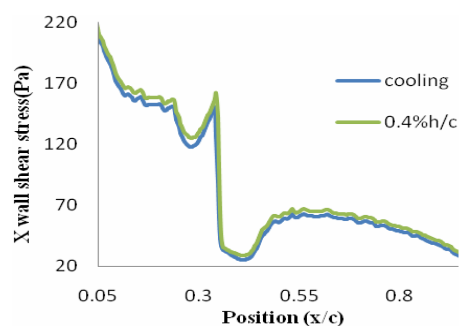

Fig 12: X wall shear stress for datum airfoil and surface cooled airfoil.

6. CONCLUSIONS

The controle of the shock wave boundary layer interaction by a bump located underneath the mean shock position, has proven to be an efficinet method of reducing drag on NACA0012. Different bump paremeters are studied for optimized result. The numerical result for SCB shows good mathing with experimental result for 2D case, while the heat transfer results are mathed simply for proportional basis. However, the adjustment of the bump in hight and position is needed for optimum performance gains over a given range of Mach numbers. Further work concerning the effect of bump may consider for 3D case in finite swept wing.

7. REFERENCES

1. P. R. Ashill, J. L. Fulker and A. Shires, 1992, "A Novel Technique for Controlling Shock Strength of Laminar-Flow Aerofoil Sections", Proceedings 1st European Forum on Laminar Flow Technology, Hamburg, Germany, DGLR, AAAF, RAeS, 1992, 175
2. P. Neittaanmaki, T. Rossi, S. Korotov, E. Onate, J.Periaux and D. Knorzer (eds.) Jyvaskyla, 24-28 july 2004. European Congress on Computational Methods in Applied Science and Engineering ECCOMAS 2004.
3. E. Stanewsky, J. Delery, J. Fulker and W. Gibler, editors, 1997. EUROSHOCK – Drag reduction by passive shock control. Vieweg, Braunschweg/Wiesbaden, 1997.
4. Mabey, D. G. , 1991. "Effects of heat transfer in aerodynamics and possible implications for wind tunnel tests" Progress in aerospace sciences, 27 (4), pp 267-303, 1991.
5. Raghunathan, S., Zarifi-Red, F., Mabey, D. G., "Effects of Model Cooling in Transonic Periodic Flow." AIAA. J., Vol. 30, No. 8, pp. 2030-2089, 1992.
6. Stanewsky, Egon, Delery, Jean, Fulker, John and de Matteris, Paulo.2002. Drag Reduction by Shock and Boundary layer control. Results of the Project EUROSHOCK II, Supported by the European Union, 1996 - 1999, Notes on Numerical Fluid Mechanics and Multidisciplinary Design, Volume 80, Springer, Berlin 2002.

8. NOMENCLATURE

Symbol	Meaning	Unit
M	Mach Number	
c	Chord Length	(m)
h	Height of Bump Crest	(m)
x	Position on Airfoil Surface	(m)
cl	Coefficient of Lift	
cd	Coefficient of Drag	
T_w	Wall temperature	(K)
T	Air temperature	(K)