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CFD Analysis of Dynamics of Interaction of Shock Wave -Vortex Core over Flapped Wing of Supersonic Aircraft at different angle of attack & Mach number

Mukesh Didwania, Kamal Kishore Khatri

Abstract: ***The objective of this paper was to analysis the condition for the appearance of the many types of interaction of a vortex core with shock wave over a flapped wing of a supersonic aircraft. A five digit NACA 23012 aerofoil was selected for this work. Structured Mesh was generated by Quadrilaterals Method. Steady-state density based implicit solver and K-ω SST turbulent model was selected. Q criterion method with vorticity magnitude was used to calculate the vortex core. NACA aerofoil Scaled model was manufactured by using NACA profile for experimental work and CFD results were validated by pressure coefficient calculated by wind tunnel setup. Finally, concluded that weak interaction with no vortex breakdown was observed at M= 1.4 and a strong interaction with a bubble-like vortex breakdown formed at M= 1.8 and It found that when a shock wave interact with vortex core, disturbance is generated, which expands along the shock wave and deformed into many small vortices. The flow field is compressed behind the curved shock wave which is reason of acoustic waves. This principle are related to the shock–turbulence interaction which is one of major source of noise. Also concluded that initially at low angle of attack,it observed a strong organized flow field in the downstream region which is due to less strength of the shock. The development of a transmitted shock wave across the vortex core was observed because of shock scattering phenomenon. The moderate breakdown of the vorticity field that occurs after a very strong shock at M =1.4 also observed and the breakdown was more intense when increased Mach No. up to 1.8. Weak and strong interaction region were observed and three stages of interaction found by the flow field over aerofoil at high Mach No. =1.8.***

Keywords: Angle of attack, Deflection Angle, Mach No., Aerofoil, Lift, Drag, Vortex, Shock wave, CFD, NACA.

I. INTRODUCTION

Wings have an airfoil and streamlined cross-sectional shape which produces lift. The purpose of the flaps is to generate more lift at slower air speed, which enables the airplane to fly at a greatly reduced speed with a lower risk of stalling. It has significantly effect on aerodynamics properties at high speed of aircraft also [6].

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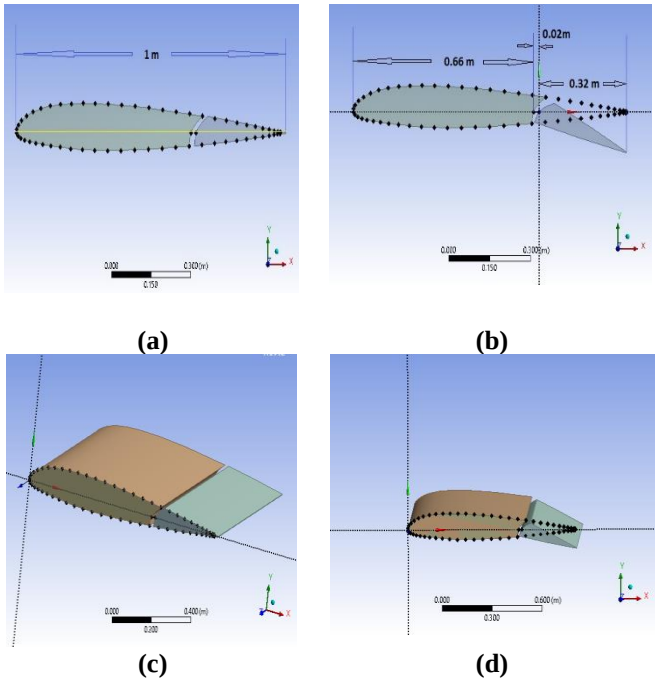
A. NACA Aerofoil: The NACA Five-Digit Series was used because it uses the same thickness forms as the Four-Digit Series but the mean camber line is defined differently which allowed for camber to be concentrated near the leading edge. A reflexed camber line was designed to produce zero pitching moment, but has generally not been used. [7]

B. Statement of Problem: When a shock wave interact with vortex core region, disturbance is generated andthis vortex cores expands and spreads along the shock and deformed. The flow fieldis compressed behind the curved shock wave and breakdown in small vortices andbreakdown of vortices formed acoustic waves. Interaction of a supersonic vortex with an oblique shock wave results Strong acoustic levels & wake turbulence and it effects the aerodynamic performance at high speed fluid flows through flap wings of an aircraft. The hazardous aspects of wingtip vortices were discussed in the reference of turbulence. The main focus of thisanalysis was onthe shock wave –wake turbulence interaction, which is another sources of noise. So theresearch will be a good solution to reduce these noise and optimized the angle of attack (AOA) with flap deflection (DA)and Mach No. (M)Where the vortex will still attached to the flap so it was important to propose a research on investigation of the dynamics of shock wave-vortex core interaction over flapped wing of supersonic aircraft by CFD Code.

C. Objective: objective of this research was to investigate (1)effect of Drag, Lift and pressure coefficient over aerofoil surface. (2) The effect of the shock waves and vortex core on the flow field(3)the condition for generation of the many types of shock wave- vortex core interaction and its reflection from surface at high Mach number, different AOA and different flap deflection angle through Flapped Wing by a CFD tool.

II. GEOMETRY, MESHING & SOLUTION

A. Geometry& Computational Domain:NACA 23012 was selected because of its best design. It has lift coefficient of 0.15 × 2 = 0.3. Its maximum camber point is at 3 × 0.05 = 0.15 of the chord. That is, the max camber occurs at the point that is 15% of the chord behind the leading edge. The digit 0 in the middle tells that this airfoil has a simple camber line. The trailing edge does not curve up. The thickest point of the airfoil is 12% of the chord. [7] Flap Span/ Wing Span = 1 and Trailing Edge Flap Type is slotted flap. Geometry & Computational domain was prepared in Ansys Work bench.



(a) (b)

(c) (d)

Figure 1. (a) Geometry of NACA aerofoil without flap (b) Geometry of NACA aerofoil with flap (c) Computational model at Deflection anlgle = 0 (d) Computational model at Deflection anlgle =20

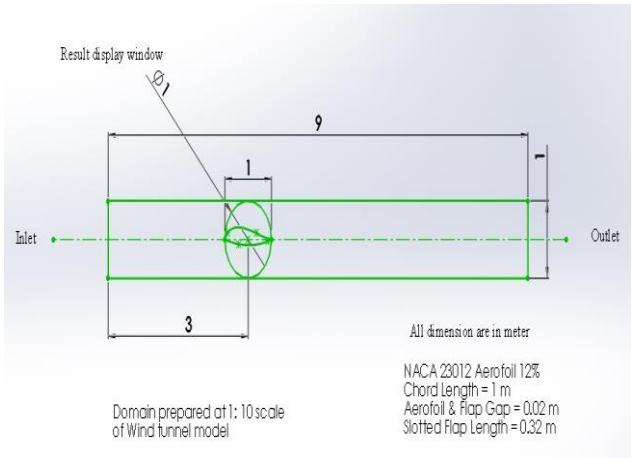
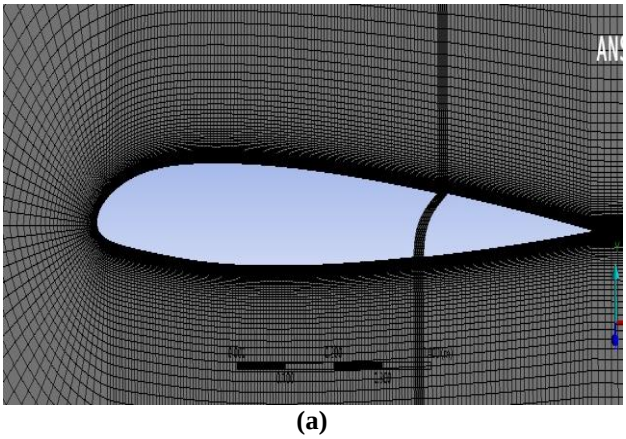
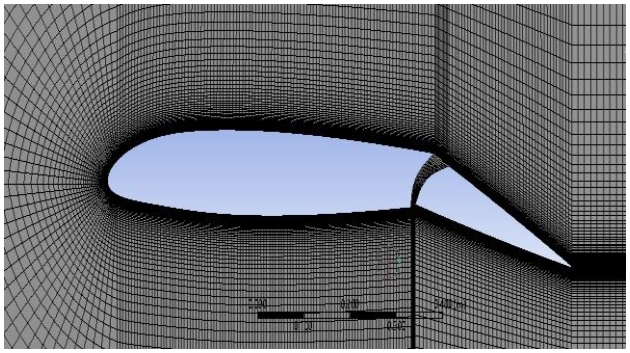


Figure 2. Ansys Computational domain of setup

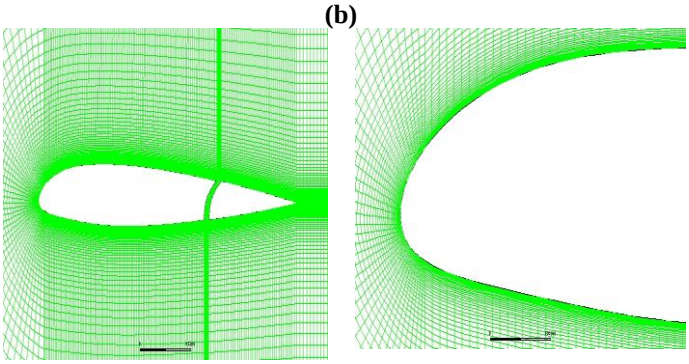
B. Meshing: Structured mesh was generated with 75879 Nodes and 75000 Elements by Quadrilaterals Method in Ansys CFD code. Viscous sublayer resolving approach used to resolve boundary layer for cell wall distance at aerofoil surface.



(a)



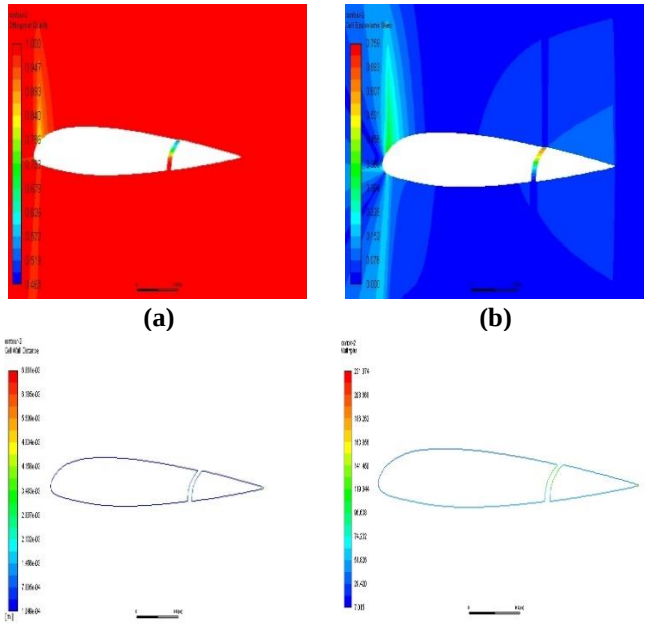
(b)



(c) (d)

Figure3. (a) Meshing of Computational Domain without deflection(b) Meshing of Computational Domain with deflection(c) Mesh displayed in fluent (d) Cell wall distance- Viscous sublayer resolving approch

C. Mesh quality: Mesh quality checked by (a) Orthogonal quality (b) Cell Skewness (c) Cell Wall distance (d) Wall Y+ Value on Aerofoil and (e) maximum aspect ratio. So for this Research Minimum Orthogonal Quality was 4.64981e-01 which was very good because Poor quality is 0 and best quality is 1for qualitative meshing and Maximum Aspect Ratio was 54.4975 which was not high. Skewness is less than 0.075 (for Quadrilaterals mesh should not exceed 0.900) and cell wall distance is less than 7.805e-04. Wall y+ value found less than 74.232 which is good condition because for best quality of boundary layer, it should not exceed 300.



(a) (b)

(c) (d)

Figure4.Mesh Quality check (a) Orthogonal quality (b) Cell Skewness (c) Cell Wall distance (d) Wall Y^+ Value on Aerofoil

D. Boundary Condition and Material: Pressure far field and Pressure outlet boundary conditions was selected for inlet, aerofoil surface and outlet of computational domain.Lift, Drag monitor and AOA for inlet Boundary Condition were set byforce vector values as X & Y componentsof Flow Direction were Cos α&Sin αrespectively. The CFD code data were compared with test data of wind-tunnel so it was compulsory to define static pressure as input data at inlet boundary condition and defined the angle of attack for inlet boundary conditionand Air as a fluid and aluminum as an aerofoil material was applied for computation domain.

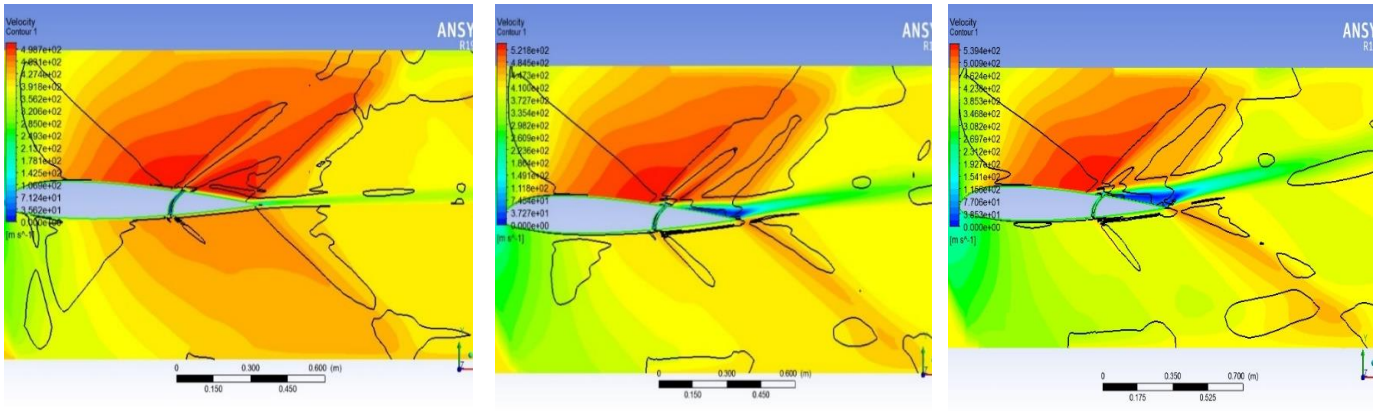
E. Solver: Ansys Fluent was used for the analysis of Interaction. Study-state density based implicit solver and kω- SST turbulent model was selected because of compressible aerodynamic flow.
Total 20 Cases were simulated in Ansys fluent CFD code

Table 1.Cases to be investigated by CFD code

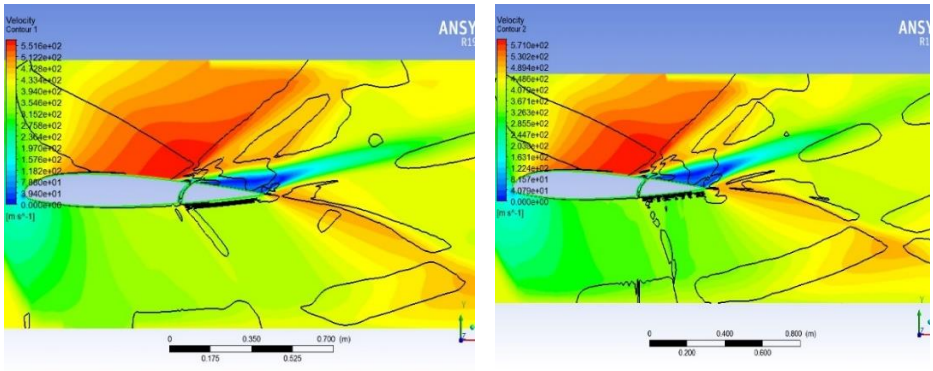
Flap Deflection Angle (δ)	Mach No (M)	AOA (α)	Flap Deflection Angle (δ)	Mach No (M)	AOA (α)
0	1.4	3	20	1.4	3
		8			8
		12			12
		15			15
		20			20
	1.8	3		1.8	3
		8			8
		12			12
		15			15
		20			20

III. RESULTS AND DISCUSSION

A. Comparison of Shock-Vortex core interaction at different angle of attack & Mach No. : Vortex core region plotted by Q criterion method with vorticity magnitude. Q criterion function was used to calculate vortex core region.

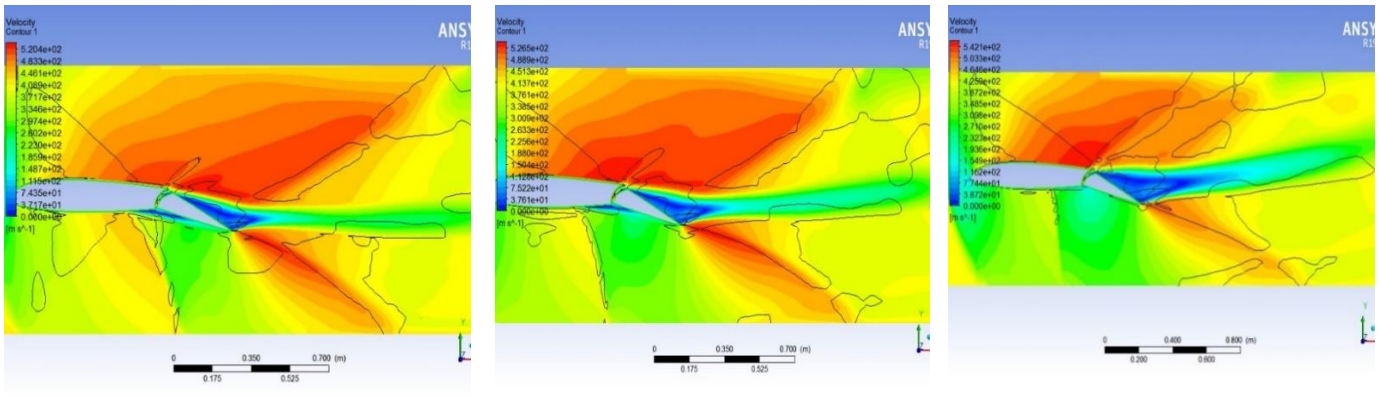


(a) (b) (c)



(d) (e)

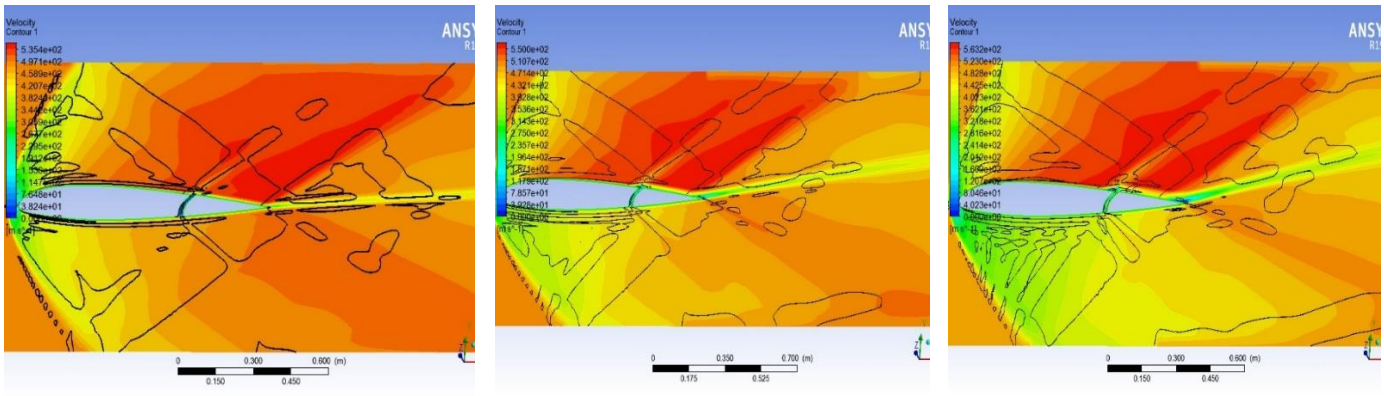
Figure 5. Interaction of Velocity & Vortex core Region contour at DA = 0 & M = 1.4 of (a) AOA=3 (b) AOA=8 (c) AOA = 12 (d) AOA = 15 (e) AOA = 20



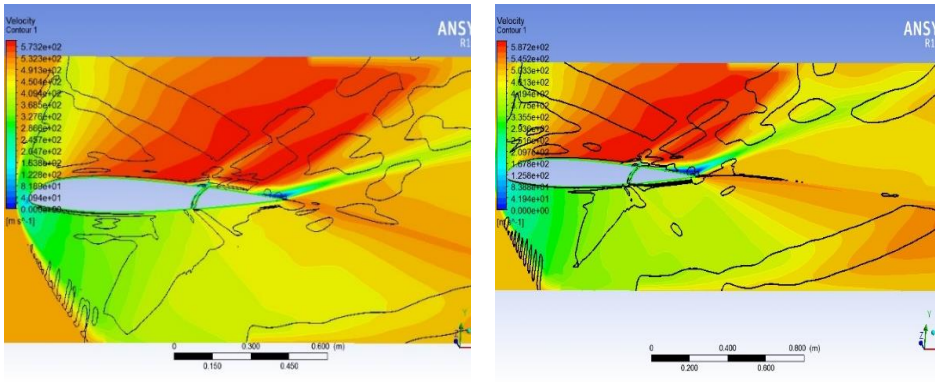
(a) (b) (c)

(d) (e)

Figure 6. Interaction of Velocity & Vortex core Region contour at DA = 20 & M = 1.4 of (a) AOA=3 (b) AOA=8 (c) AOA = 12 (d) AOA = 15 (e) AOA = 20

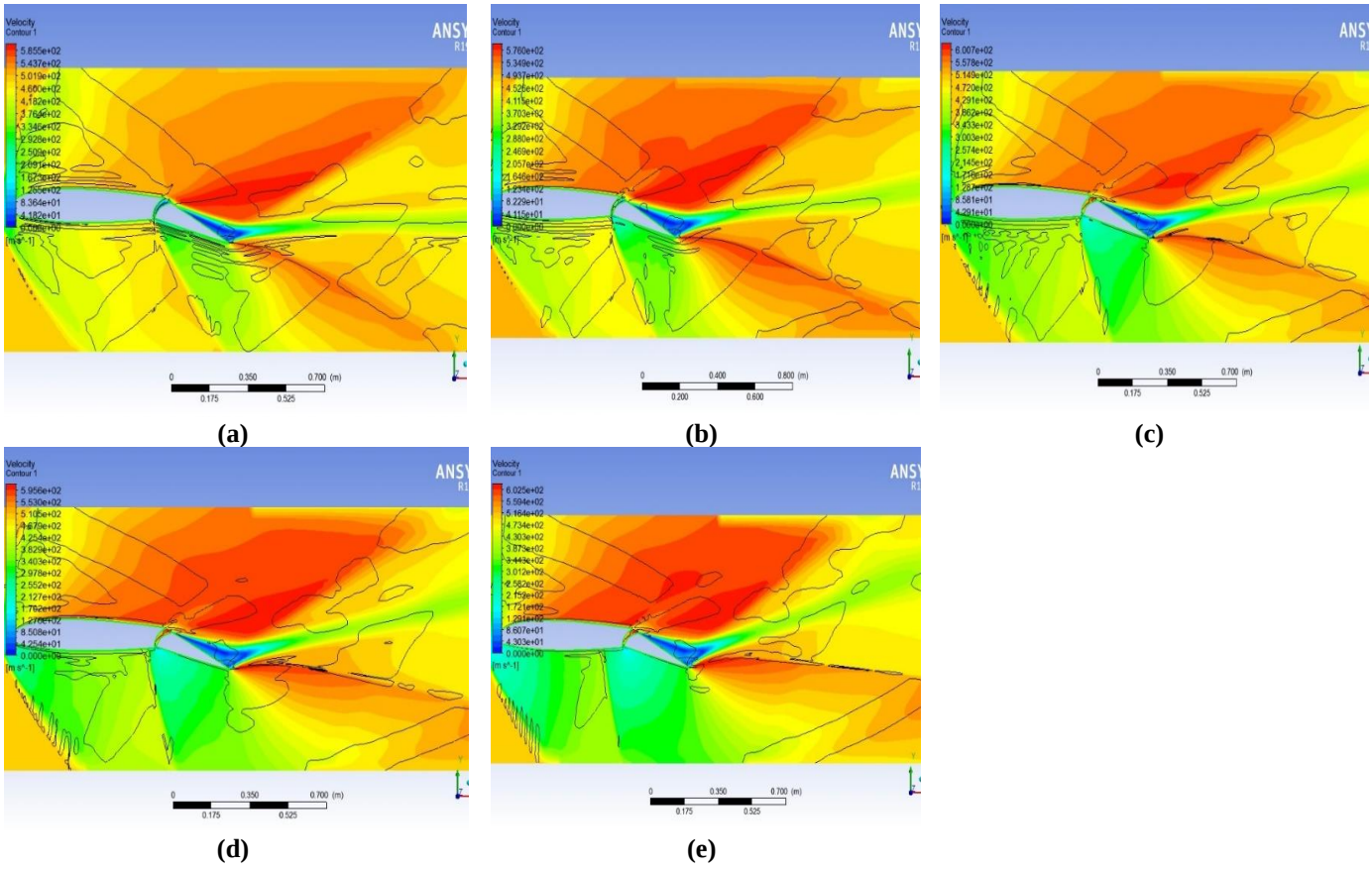


(a) (b) (c)



(d) (e)

Figure 7. Interaction of Velocity & Vortex core Region contour at DA = 0 & M = 1.8 of (a) AOA=3 (b) AOA=8 (c) AOA = 12 (d) AOA = 15 (e) AOA = 20



(a) (b) (c)

(d) (e)

Figure 8. Interaction of Velocity & Vortex core Region contour at DA = 20 & M = 1.8 of (a) AOA=3 (b) AOA=8 (c) AOA = 12 (d) AOA = 15 (e) AOA = 20

B. Discussion: Figure 5 show that at supersonic speed the shock-vortex interaction will start over upper surface of aerofoil. Vortex –oblique shock wave interaction was available at trailing edge. If we increase AOA, both shock wave and Vortex core will decrease so Vortex–oblique shock wave interaction also decreases. A close subsonic flow region formed behind the shock wave, in which there is no observation of return flow regions. Because of compression, the vortex deformed along the shock wave observed. It is called moderate interaction and the vortex core passing the shock wave is separated into many weakly connected vortices. Figure 6show that if we use flap at M = 1.4, vortex core region decreases so weak vortex–oblique shock wave interaction found because vortex core is transferring from trailing to leading edge but oblique shock is reducing by AOA increasing and still interaction is available at trailing edge. Figure 7 show that if we increase incidence Mach No. from 1.4 to 1.8, then Vortex –oblique shock wave interaction is found through the surface of aerofoil and Vortex –expansion shock wave interaction is also available. Even after increasing AOA still these Interaction is continuing because at this condition both Vortex core and Shock wave are generating at high level. Here we can see vortex with compression is reaching at leading edge and Vortex–bow shock wave interaction found. Because of strong interaction, vortex drag point is observed around the vortex axis in figure 7 and behind this vortex drag, the subsonic flow region is observed and in this subsonic region, a strong re-

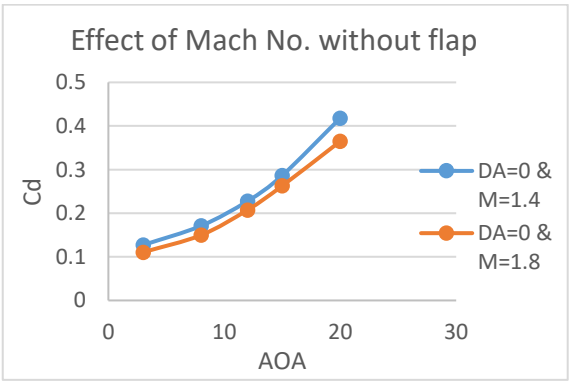
circulation flow region is formed (figure 6 d & e). Behind the shock wave, the vortices splitting into many unsymmetrical small vortices with the subsonic flow regions. It can be clearly seen in figure 7 (d & e). Figure 8 show that if we use flap at high Mach No. as 1.8 then there is High possibility of Vortex –oblique shock wave interaction but less Vortex –expansion shock wave interaction is available. Even after increasing AOA, Vortex-expansion shock wave interaction is disappearing and converting into compression waves.Initial vortex is deforming in many deformed vortices because of using flap at this stage.

By figure 5 & 7 it can clearly see that without flap at supersonic speed when breakdown of vortex core occurs with increasing angle of attack and at downstream from the shock, a free stagnation point is formed which is following by a reversed flow region with a bubble-like flow regions (figure 7 (c & d)). The scattered (burst) part of the vortex core region grows until the bubble reaches a stable position. The shock waves were remains normal near the vortex but were oblique further away from it.

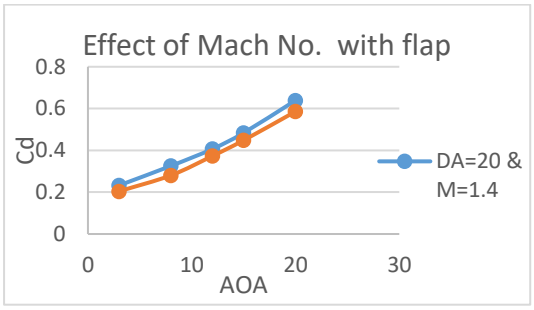
The results showed that the strong reflected shock wave generated by interaction of initial vortex and incoming shock wavecan interact with the deformed vortex (figure 5). In the second stage of interaction, the deformed vortex is compressing into a circular shape (figure 6) and in the third stage of interaction, it is compressing into an elliptical shape again. (Figure 7)

The result show that a slender (less area) vortex propagating across a normal shock does not necessarily leads to breakdown (figure 5 a). By increasing the free stream Mach number or the vortex strengths, the shock-vortex core interaction leads to free stagnation point nearby the vortex core axis, which can be understood as the initialization of vortex breakdown (figure 7 a). Many small bubble-like vortex structures are formed below a stagnation point at leading edge of aerofoil. (Figure 7 b, c & d).

C. Comparison of drag and Lift coefficient at different angle of attack & Mach No.

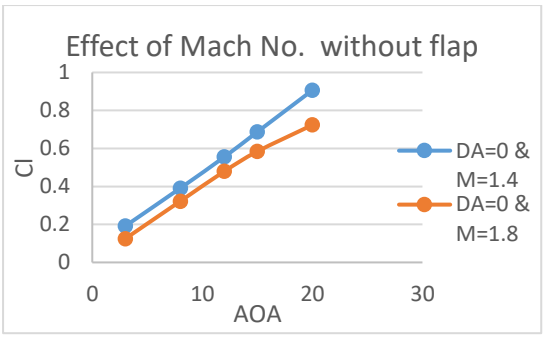


(a)

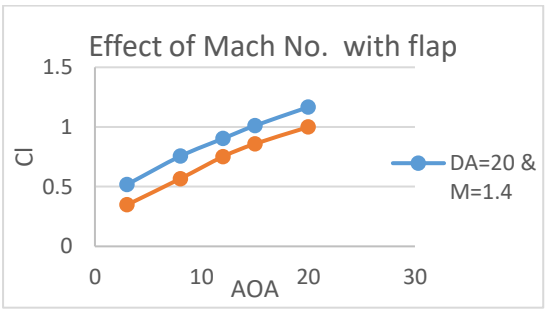


(b)

Figure9.Effect of Mach No. on Drag coefficient (a) without flap (b) with flap

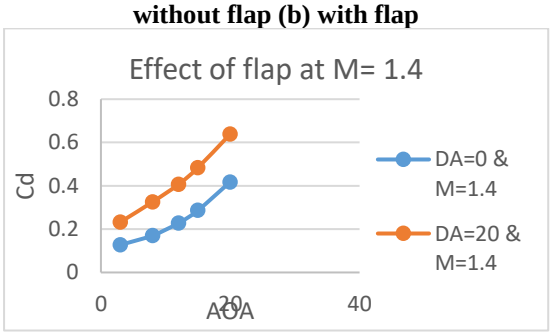


(a)

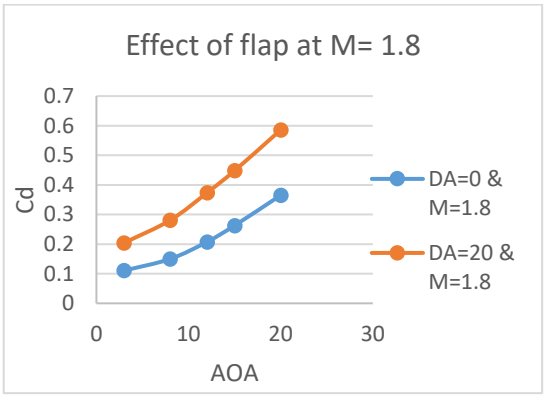


(b)

Figure10.Effect of Mach No. on Lift coefficient (a) without flap (b) with flap

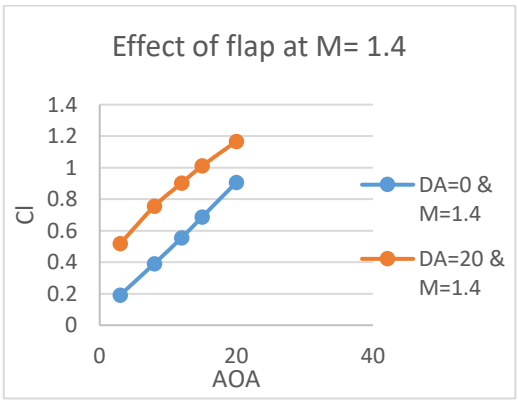


(a)

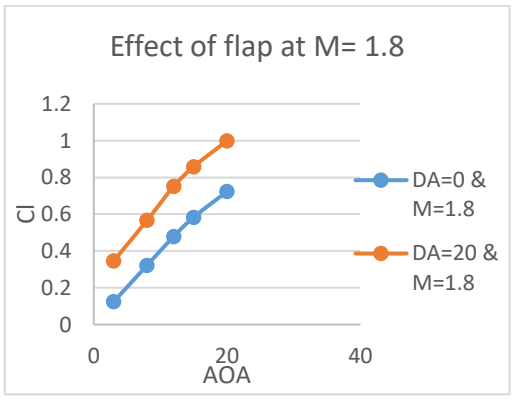


(b)

Figure 11. Effect of flap angle on drag coefficient at different Mach No. (a) M= 1.4 (b) M= 1.8

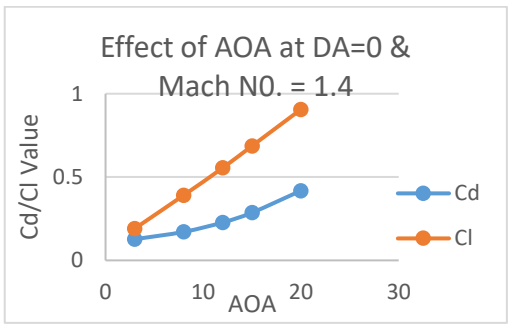


(a)

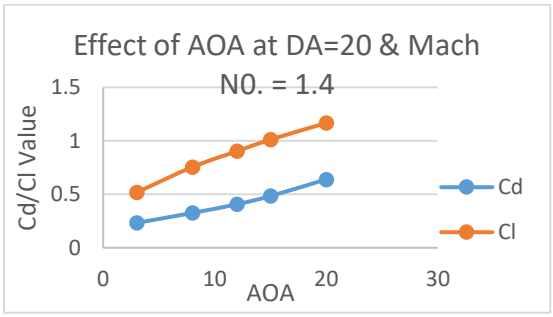


(b)

Figure 12. Effect of flap angle on Lift coefficient at different Mach No. (a) M= 1.4 (b) M= 1.8

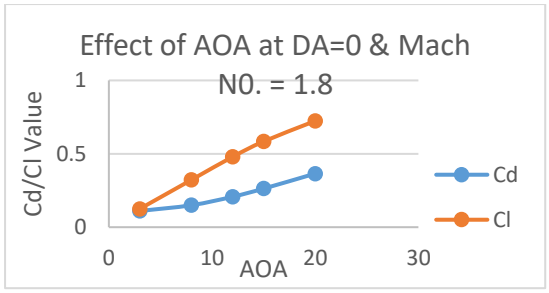


(a)

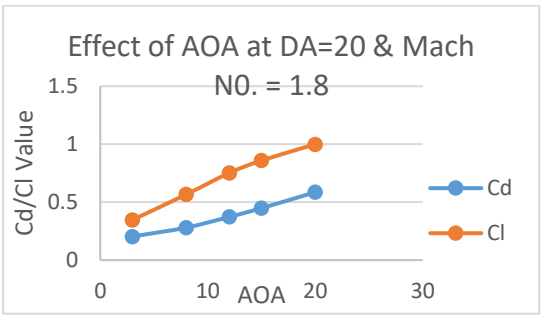


(b)

Figure 13. Relationship of Cd, Cl and AOA at M= 1.4 (a) DA=0 (b) DA=20



(a)

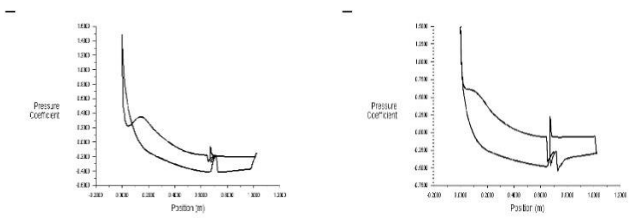


(b)

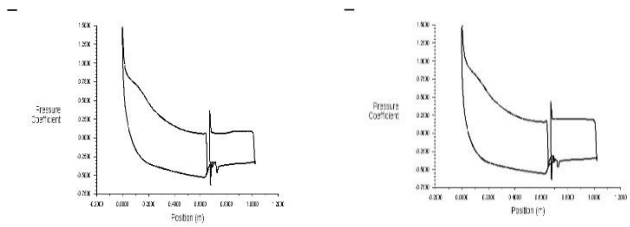
Figure 14. Relationship of Cd, Cl and AOA at M= 1.8 (a) DA=0 (b) DA=20

D. Discussion: Figure 9& 10show that there is increment in drag & lift coefficient more with Mach No. without using flap but significantly increment less with flap. By figure, 11 & 12 it can see that drag & Lift coefficient increases by using flap only. It shows that decrement in Cl can be eliminated by using flap.Overall by Figure 9&10 show that drag coefficient increases and lift coefficient deceases with Mach No increment from 1.4 to 1.8 but drag & lift coefficient both deceases with Mach No increment from 1.4 to 1.8.By figure 13&14 we can see the relationship between Cd, Cl and AOA for different Mach No. and Deflection angle. It shows that lift coefficient significantly increases more for all casesso it show the contrast of lift force with the drag force.

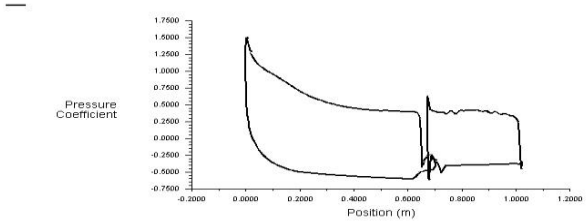
E. Comparison of pressure coefficient over aerofoil at different angle of attack & Mach No.



(a) **(b)**

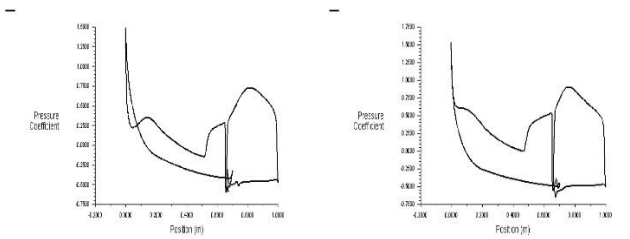


(c) **(d)**

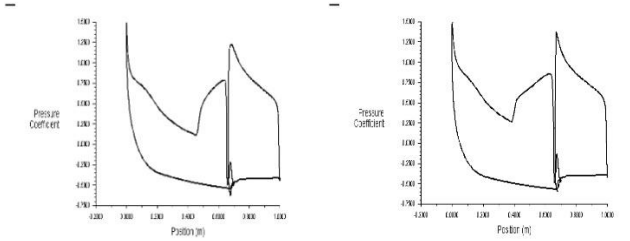


(e)

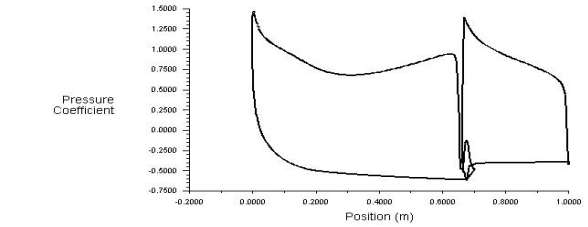
Figure 15: Pressure coefficient over Aerofoil at DA=0 & M = 1.4 for (a) AOA=3 (b) AOA = 8 (c) AOA=12 (d) AOA=15 (e) AOA=20



(a) **(b)**

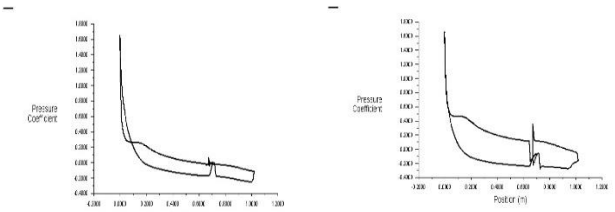


(c) **(d)**

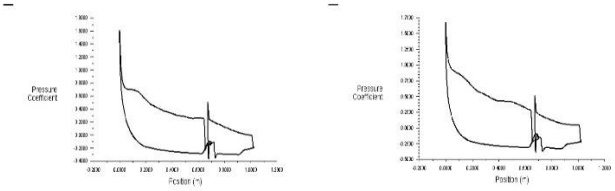


(e)

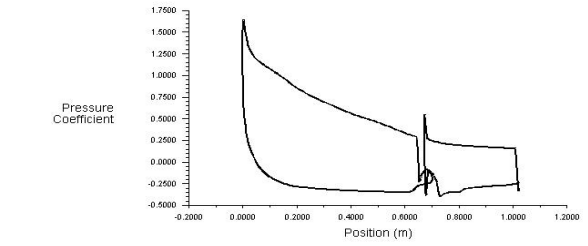
Figure 16: Pressure coefficient over Aerofoil at DA= 20 & M = 1.4 for (a) AOA=3 (b) AOA = 8 (c) AOA=12 (d) AOA=15 (e) AOA=20



(a) **(b)**

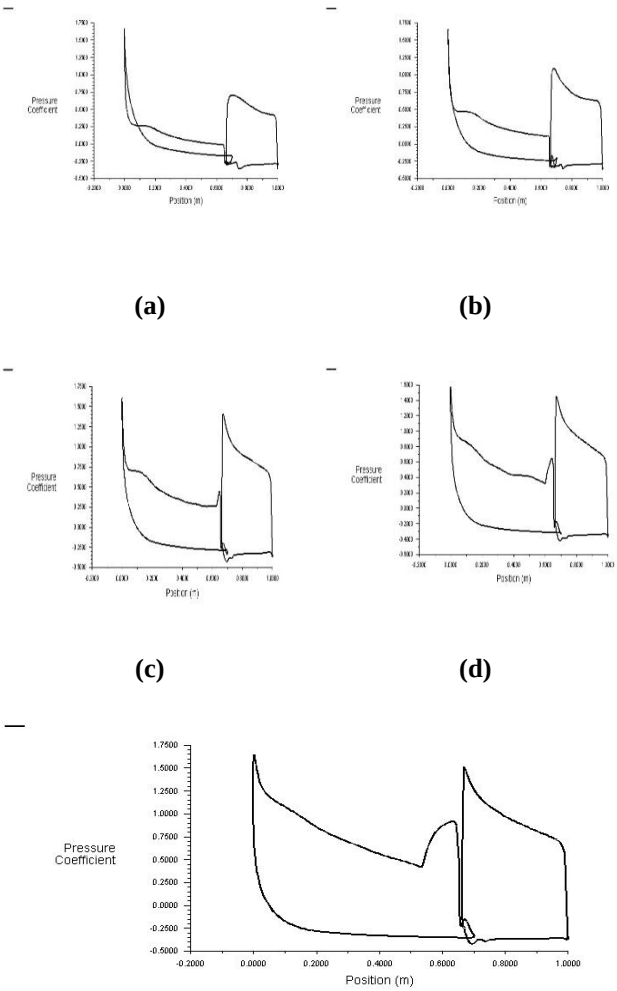


(c) **(d)**



(e)

Figure 17: Pressure coefficient over Aerofoil at DA= 0 & M = 1.8 for (a) AOA=3 (b) AOA = 8 (c) AOA=12 (d) AOA=15 (e) AOA=20



(a) (b)

(c) (d)

(e)

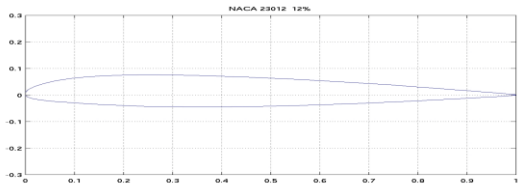
Figure18: Pressure coefficient over Aerofoil at DA= 20 & M = 1.8 for (a) AOA=3 (b) AOA = 8 (c) AOA=12 (d) AOA=15 (e) AOA=20

F. Discussion: It can be seen that the pressure coefficient varied mostly with different angle of attack. The pressure coefficient of the upper surface of aerofoil was negative and at lower surface of aerofoil it was positive, so the lift force of the airfoil is always act in the upward direction due to this pressure coefficient difference. If angle of attack is more, the difference of pressure coefficient between the lower and upper surface will be more. It can also see that the pressure coefficient difference is much larger on the front edge, while on the rear edge it was much lower for both aerofoil and flap, thus indicating that the lift force of the airfoil is mainly generated from the front edge.

Figure 15 &17 show that at high Mach No. and low AOA there is very less difference in pressure coefficient but if AOA increases, pressure coefficient difference of lower & upper surface of aerofoil will increase by increasing overall pressure and positive pressure coefficient is more over upper surface of aerofoil so at high speed, if AOA increases, overall pressure coefficient will increase at bottom surface. Figure 16 &18 show that if flap uses, pressure coefficient difference will increase at flap only by increasing AOA.Lift force observed very less at high Mach no but it suddenly increases at trailing edge initially at M=1.4 with flap and continuously increases with angle of attack at this condition.Pressure coefficient difference of bottom and top surfacereduces drastically if Mach No. Increases from M=1.4 to M=1.8 which is the indication of high strength shock wave and compression of shock wave and by increasing angle of attack lift forces can be increase at both conditions.

IV. EXPERIMENTAL WORK

Scaled Model of NACA aerofoil was manufactured by using this NACA profile for experimental work. [18]

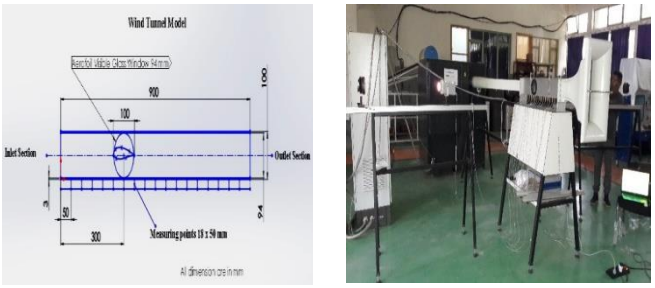


(a)[18]

(b) (c)

Figure19. (a) NACA aerofoil Profile (b) NACA aerofoil Model (c) NACA aerofoil Model with flap

A. Wind tunnel setup and validation: Wind tunnel setup was developed for 18 measurement points. Aerofoil model was fitted at 0.5 position which was center of aerofoil. 0.5 position means 5 cm from leading edge of aerofoil and leading edge was set at 0 position means 0 cm of the wind tunnel. Toward negative x direction negative values were set and towards positive x direction positive value were set. Inlet section of the wind tunnel was taken at -2.5 position means -25 cm from leading edge of aerofoil and outlet section was taken at 65 cm from leading edge of aerofoil. For experiment work measuring points were set at both upper and lower surface of tunnel and it started from outlet section it means outlet section point was taken as a first point and -0.2 position point was taken as a last point. Deflection angles of flap & AOA were set manually by using protector scale printing on glass window. Mach No. was fixed by Mach No. generators. For this experiment total two generators were used as M =1.4 and M = 1.8. CFD code results were validated by pressure coefficient calculated by wind tunnel setup at different Angle of Attack 3, 8, 12, 15& 20 at Mach No. = 1.4 and 1.8 for flap Angle 0 & 20.



(a) (b)

(c) (d)

Figure20. (a)Wind tunnel model (b) Wind tunnel setup (c) Mach no generator (d) Aerofoil location screen

B. Pressure coefficient calculate by CFD code were validated by wind tunnel setup.

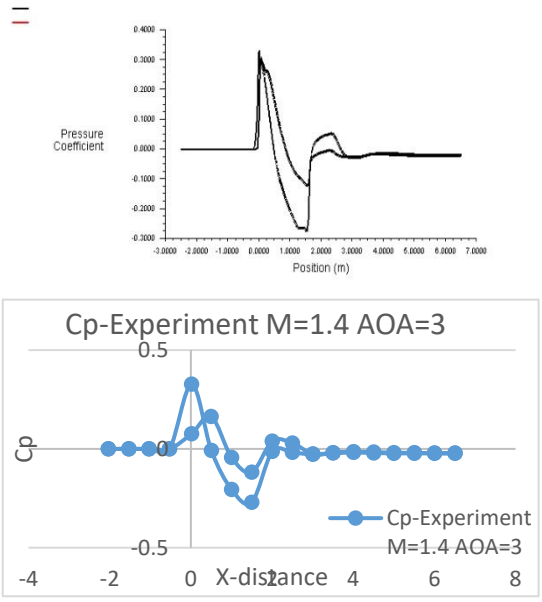


Figure 21. Comparison of CFD code's Pressure coefficient with experiment result for DA= 0, M = 1.4 & AOA = 3

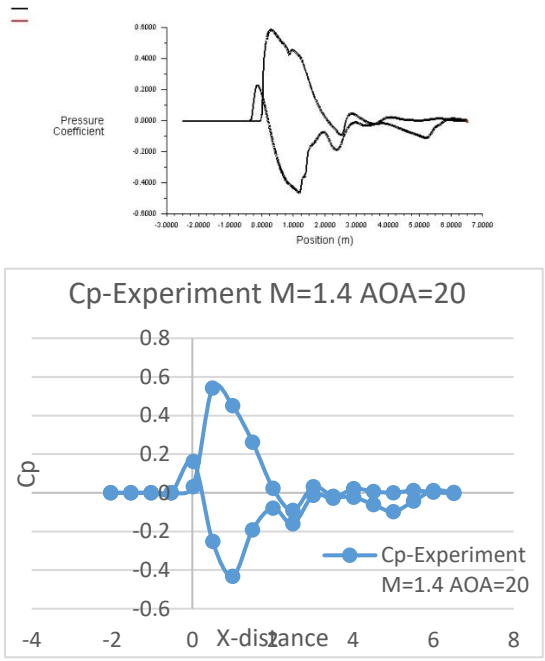


Figure 22. Comparison of CFD code's Pressure coefficient with experiment result for DA= 0, M = 1.4 & AOA = 20

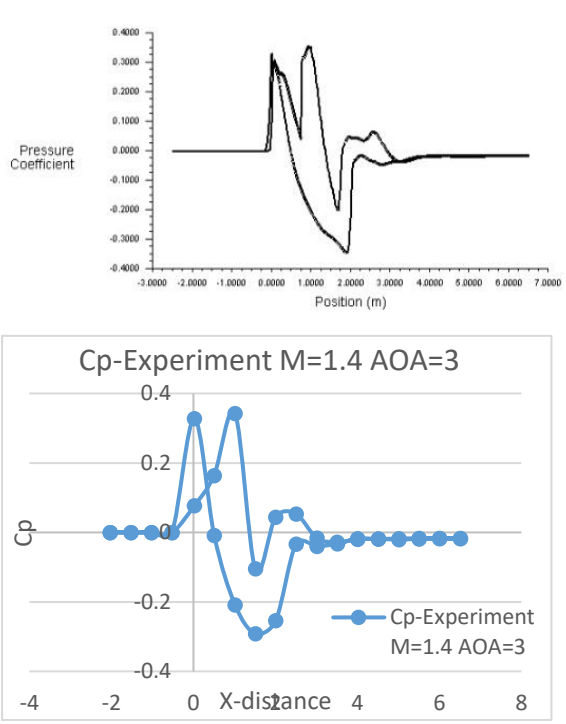


Figure 23. Comparison of CFD code's Pressure coefficient with experiment result for DA= 20, M = 1.4 & AOA = 3

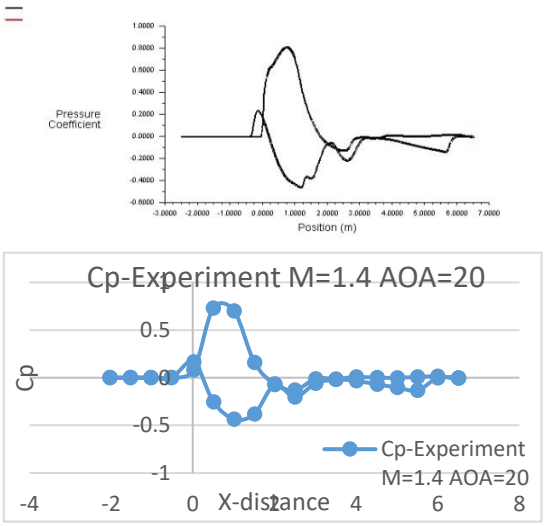


Figure 24. Comparison of CFD code's Pressure coefficient with experiment result for DA= 20, M = 1.4 & AOA = 20

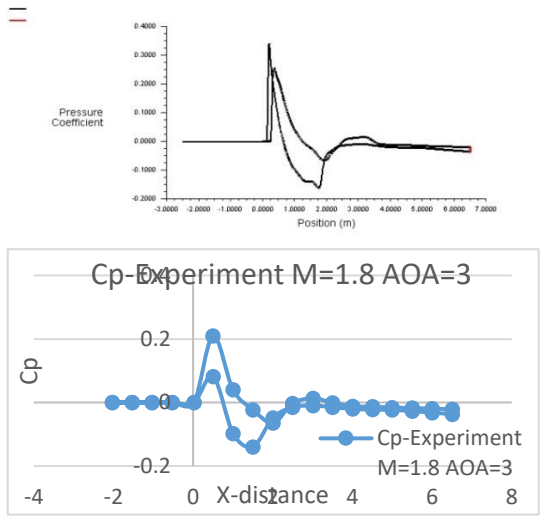


Figure 25. Comparison of CFD code's Pressure coefficient with experiment result for DA= 0, M = 1.8 & AOA = 3

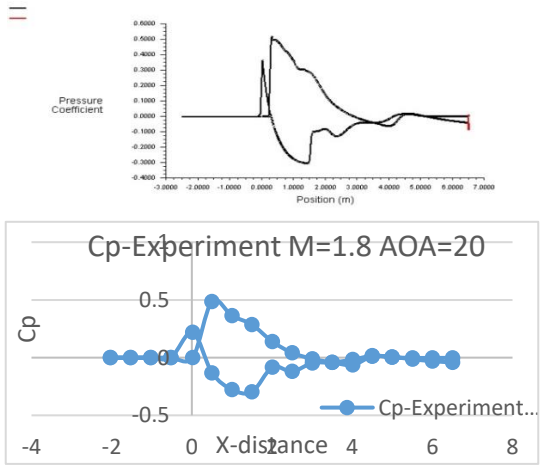


Figure 26. Comparison of CFD code's Pressure coefficient with experiment result for DA= 0, M = 1.8 & AOA = 20

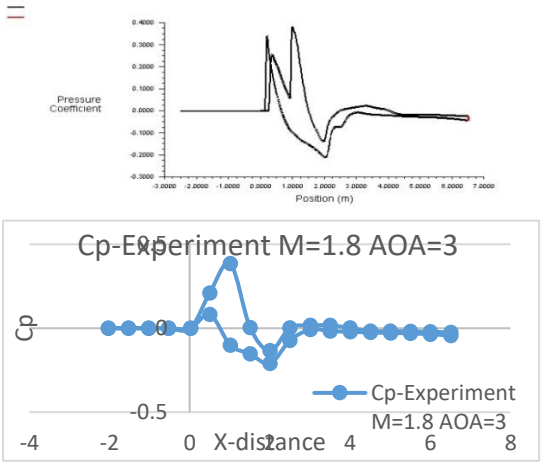


Figure 27. Comparison of CFD code's Pressure coefficient with experiment result for DA= 20, M = 1.8 & AOA = 3

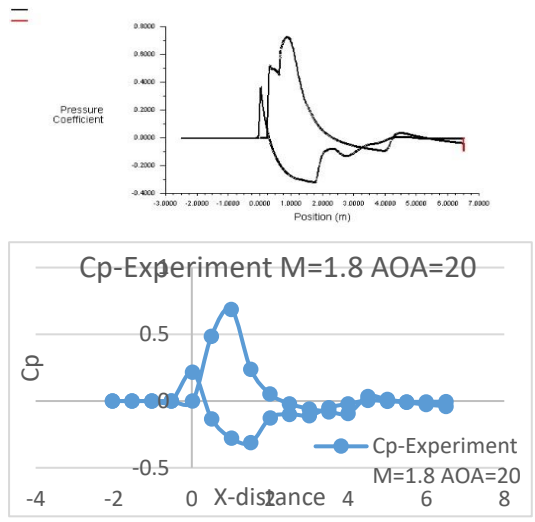


Figure 28. Comparison of CFD code's Pressure coefficient with experiment result for DA= 20, M = 1.8 & AOA = 20

Note: - all 20 results plotted by Experimental setup but Eight most related validation result presented only because of less space.

V. CONCLUSION

By result and discussion of Shock-vortex interaction it concluded that initially at low AOA it observed a strong organized flow field in the downstream region which is due to less strength of the shock. The development of a transmitted shock wave across the vortex core was observed because of shock scattering phenomenon. The moderate breakdown of the vorticity field that occurs after a very strong shock at M =1.4 also observed and the breakdown is more intense when increasing Mach No. up to 1.8. Vortex drag observed at M=1.8 and a strong re-circulation flow region is formed behind the vortex drag point in the subsonic region. Also observed that, the vortex splitting into many unsymmetrical small vortices with the subsonic flow regions behind the shock wave.

Vortex drag increases with AOA. Also concluded that without flap at supersonic speed (M=1.4 & M = 1.8) when breakdown of vortex core occurs with increasing angle of attack and at below the shock, a free stagnation point is formed which is following by a reversed flow region with a bubble-like flow regions. The scattered (burst) part of the vortex core region grows until the bubble reaches a stable position. The shock waves were remains normal near the vortex but were oblique further away from it. Weak and strong interaction region were observed and three stages of interaction found by the flow field over aerofoil at high Mach No. =1.8 and bubble like vortex formed. It means instead of M=1.4, M=1.8 is good condition for interaction. By experiment work, it concluded that The CFD simulation results showed close agreement with those of the experiments results.

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