Simulations of Supercritical Aerofoil at Different Angle of Attack With a Simple Aerofoil using Fluent

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***Abstract* —This In this project flow over supercritical aerofoil and simple aerofoil is compared at Mach number 0.6 parameters which are observed are pressure drag and strength of shockwave as they are one of the parameters which are prominent in transonic speed. These parameters decide the efficiency of the aerofoil. In this project NACA SC (02) 0714 and NACA 4412 aerofoil profiles is considered for analysis. Software tools used are GAMBIT and FLUENT. Gambit is used for preparing the geometry and meshing and FLUENT is used for analyzing the flow. Computational fluid dynamics is used because preparing a model of aerofoil is a lengthy and difficult process and wind tunnel capable of 0.6 Mach number is not available and difficult to produce accurate results In supercritical aerofoil, thickness of an aerofoil near trailing edge of lower surface is reduced, so that increase in pressure at lowers surface and helps in lift of an aircraft easily compared to simple aerofoil. At 15^{o} angle of attack, pressure drag is 12000 Pascal lower in case of supercritical aerofoil compared to simple aerofoil.**

Keywords— Fluent simulation; supercritical aerofoil; pressure drag; shock waves; Temperature distribution

I. INTRODUCTION

Transonic jet aircrafts fly at speed of 0.8 to 0.9 Mach number. At these speeds speed of air reaches speed of sound some were over the wing and compressibility effects start to show up. The free stream Mach number at which local sonic velocities develop is called critical Mach number. It is always better to increase the critical Mach number so that formation of shockwaves can be delayed. This can be done either by sweeping the wings but high sweep is not recommended in passenger aircrafts as there is loss in lift in subsonic speed and difficulties during constructions. So engineers thought [1] of developing an aerofoil which can perform this task without loss in lift and increase in drag. They increased the thickness of the leading edge and made the upper surface flat so that there is no formation of strong shockwave and curved trailing edge lower surface which incr

eases the pressure at lower surface and account's for lift. The Fig 1.1 shows sketch of a typical supercritical aerofoil [2].

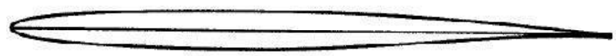

Fig 1: Supercritical Aerofoil.

A. *Features of supercritical aerofoil*

1. Trailing edge thickness

The design philosophy of the supercritical aerofoil required that the trailing-edge slopes of the upper and lower surfaces be equal. This requirement served to retard flow separation by reducing the pressure recovery gradient on the upper surface so that the pressure coefficients recovered to only slightly positive values at the trailing edge. Increasing the trailing-edge thickness of an interim 11-percent-thick supercritical aerofoil from 0 to 1.0 percent of the chord resulted in a significant decrease in wave drag at transonic Mach numbers [3];

2. Maximum thickness

For the thinner aerofoil, the onset of trailing-edge separation began at an approximately 0.1 higher normal-force coefficient at the higher test Mach numbers, and drag divergence Mach number at a normal-force coefficient of 0.7 was 0.01 higher. [3]

3. Aft upper surface curvature

The rear upper surface of the supercritical aerofoil is shaped to accelerate the flow following the shock wave in order to produce a near-sonic plateau at design conditions.[4]

4. Aerofoil data

There are two aerofoil's chosen for this analysis one Super critical aerofoil chosen for this project is NACA SC(2)0714 and other NACA 4412 which is conventional aerofoil. The specification of NACA SC (02) 0714 and simple aerofoil NACA 4412 [5] are shown in Table 1 and 2.

Table 1: Specification of supercritical aerofoil NACA SC (2) 0714.

Particulars	Dimensions with respect to chord length / chord line.
Thickness	13.9%
Camber	1.5%
Lower flatness	9.4%
Leading edge radius	2.9%
C_L max	1.442
Max. C_L angle	15 degree
Max L/D	27.881
Max L/D angle	4.5 degree
Stall angle	4.5 degree
Zero lift angle of attack	-5 degree
Material	Aluminum

Table 2: Specifications of simple aerofoil NACA 4412.

Particulars	Dimensions with respect to chord length / chord line.
Thickness	12%
Camber	4%
Lower flatness	76.1%
Leading edge radius	1.7%
C_L max	1.507
Max. C_L angle	11 degree
Max L/D	57.2
Max L/D angle	5.5 degree
Stall angle	6 degree
Zero lift angle of attack	4 degree
Material	Aluminum

II RESEARCH METHODOLOGY

A. Governing equation in CFD

The governing equations for computational fluid dynamics (CFD) are based on conservation of mass, momentum, and energy. FLUENT uses a finite volume method (FVM) to solve the governing equations. The FVM involves discretization and integration of the governing equation over the control volume. The following is a summary of the theory involved in the FLUENT analysis and is based on the FLUENT User's Manual [16].

The basic equations for steady-state laminar flow are conservation of mass and momentum. When heat transfer or compressibility is involved the energy equation is also required. The governing equations are, [5]

- Continuity Equation:

The continuity equation (3.1) expresses the conservation of matter. If matter flows away from a point, there must be a decrease in the quantity remaining. By definition, the continuity equation should be recognized as a statement of mass conservation. The continuity equation relates the speed of a fluid moving over an aerofoil.

$$\frac{\partial \rho}{\partial t} + \frac{\partial}{\partial x_j}[\rho u_j] = 0 \quad \text{........................(3.1)}$$

- Momentum equation:

The momentum equation (3.2) is statement of Newton's second law and relates the sum of the forces acting on an element of fluid to its acceleration or rate of change of momentum. The Newton's second law of motion F = ma, forms the basis of the momentum equation. In fluid mechanics it is not clear what mass of moving fluid we should use, such that we use different forms of equation. The Navier-Stokes equations are the fundamental partial differentials equations that describe the flow of incompressible fluids.

$$\frac{\partial}{\partial t}(\rho u_i) + \frac{\partial}{\partial x_j}[\rho u_i u_j + p\delta_{ij} - \tau_{ji}] = 0, \quad i = 1,2,3 \quad .(3.2)$$

- Energy equation:

The energy equation (3.3) demonstrates that, per unit volume, the change in energy of the fluid moving through a control volume is equal to the rate of heat transferred into the control volume plus the rate of work done by surface forces plus the rate of work done by gravity.

$$\frac{\partial}{\partial t}(\rho e_0) + \frac{\partial}{\partial x_j}[\rho u_j e_0 + u_j p + q_j - u_i \tau_{ij}] = 0 \quad .(3.3)$$

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2. Approach using FLUENT

The continuity and momentum equations, along with the realizable k-ε model with pressure gradients effects for turbulent flows, are solved using the FVM in FLUENT. A pressure based solver is used since the flow is incompressible and separation is caused by adverse pressure gradients.

B. Import edge

To specify the aerofoil geometry we will import a file containing a list of vertices along the surface and have GAMBIT join these vertices to create edge, corresponding to the surface of the aerofoil [17]. Fig 2 shows the importing edges of an aerofoil.

Main Menu >File >Input >ICEM input

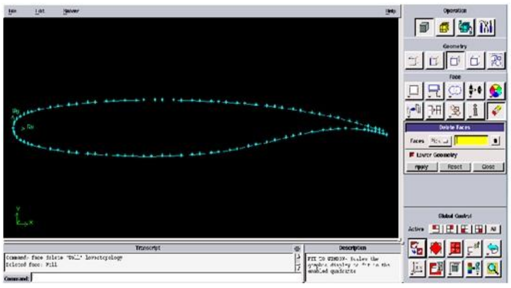

Fig 2: Import Edges

C. Crete farfield boundary

We will create the farfield boundary by creating vertices and joining them appropriately to form edges.

Operation Toolpad >GeometryCommand Button >Vertex Command Button >Create Vertex

Operation Toolpad >Geometry Command Button >Edge Command Button >Create Edge

Create edges AB, BC, CD, DA by selecting the vertices

- Create Face

We will create the face by selecting the edges AB, BC, CD, DA naming the face Farfield.

Operation Toolpad > Geometry Command Button > Face Command Button >Form Face

By selecting the aerofoil edges make an aerofoil face naming Aerofoil.

Before proceeding to the next step we will subtract the faces, subtracting face Aerofoil from Farfield.

Operation Toolpad > Geometry Command Button > Face Command Button

Click on the Boolean Operations Button and select Subtract Face Box select Farfield

in upper box and Aerofoil in lower box click apply.

D. Mesh geometry in Gambit

Mesh edges

Operation Toolpad >Mesh Command Button >Edge Command Button >Mesh Edges

Taking interval count 50 we mesh the edges AB, BC, CD, DA. Fig 3 shows meshing of aerofoil geometry.

- Mesh face

Operation Toolpad >Mesh Command Button >Face Command Button >Mesh Faces

Taking interval count 100 we mesh the face Farfield

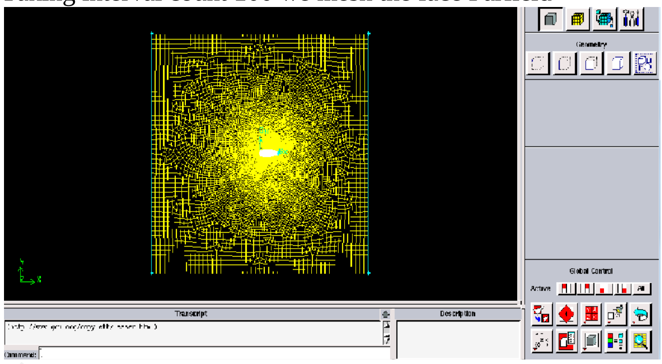

Fig 3: Meshing

E. Specify boundary types in Gambit

a. Define boundary types

Operation tool pad >Zone Command button >Specify boundary types

Under entity select edges and select AB, CD as Pressure Farfield, DA as velocity Inlet, BC as Pressure Outlet.

Save the work and Export Mesh.

Main Menu >File >Save

Main Menu >File >Export >Mesh

F. Set up problem in Fluent

Table 3 shows the properties of fluid that are given in FLUENT flow.

Import File[19]

Main Menu >File >Read Case

Check Grid

Main Menu >Grid >Check

Define Properties

Table 3: Properties of fluid

Fluid	ρ (kg/m^3)	μ (kg/m-s)	K (W/m-K)	C_p (kJ/kg-K)
Air	1.185	0.0000183	0.0261	1.004

Define >Model >Solver

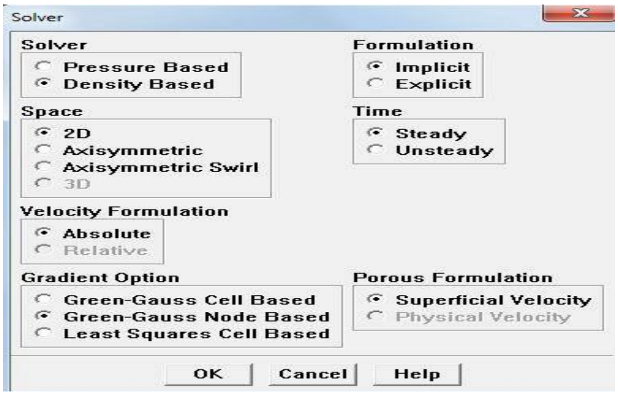

Fig 4: Model Solver

Under Solver select Density based Solver and in Gradient option select Green-Gauss node based.
Define >Model >Viscous
Fig 5 shows flow under viscous select K-epsilon[20]

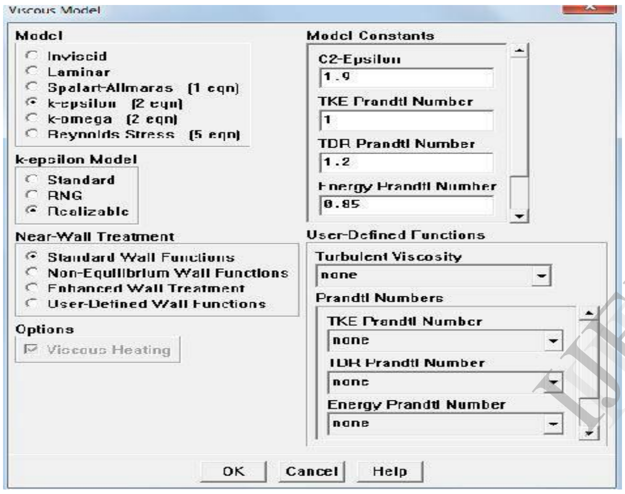

Fig 5: Model Viscous

Define >Model >Energy
Fig 5 shows defining boundary conditions for velocity inlet.
Turn On the Energy equation
Define >Materials
Make sure that air is selected under Fluid Material and set Density to Ideal Gas
Define >Operating Conditions
Set Operating Pressure to be 101325 Pascal
Define >Boundary Conditions
Fig 6 showa the applying boundary conditions

Set the Velocity Magnitude to be 250 m/sec i.e around 0.6 Mach

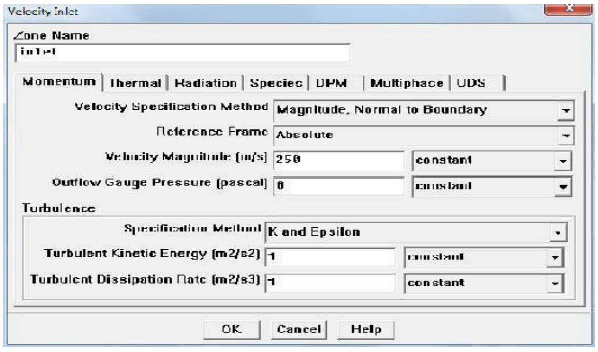

Fig 6: Defining Boundary condition

G. *Solver*

Solve >Control >Solution
Set Discretization to be Second Order Upwind for Flow, Turbulent Kinetic Energy,
Turbulent Dissipation Rate
Solve >Initialize >Initialize
Set Velocity_Inlet under compute form
Main Menu >File >Write >Case
Solve Iterate

IV RESULTS AND DISCUSSIONS

A. *Supercritical aerofoil at zero degree angle of attack*

1. Contours of static pressure

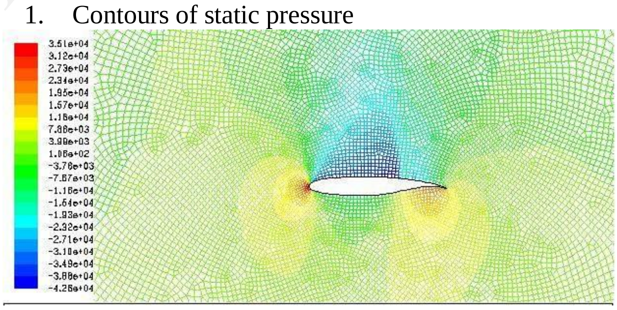

Fig 7: Contours of static pressure

Fig 7 shows static pressure contour at 0.6 Mach number. It can be observed that there is high pressure of 35100 Pascal and at trailing edge pressure is -18200 Pascal. Resultant pressure is 53300 Pascal.

2. Contours of dynamic pressure

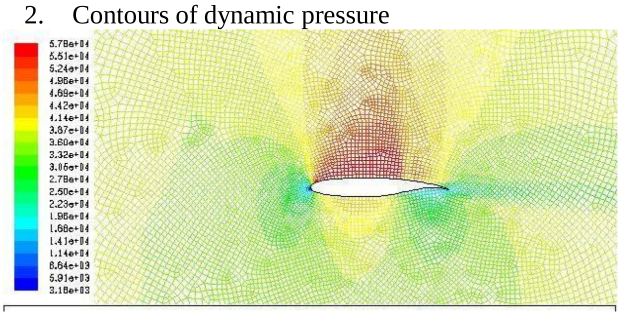

Fig 8: Contours of dynamic pressure

Fig 8 shows dynamic pressure contour at 0.6 Mach number. It can be observed that a weak shock is formed near the trailing edge of the aerofoil. And at the lower surface of the trailing edge high pressure region is there which compensates for lift loss due to flat upper surface.

3. Contours of static temperature

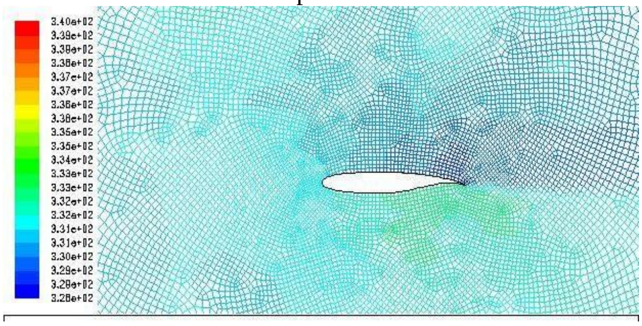

Fig 9: Contours of static temperature

Fig 9 shows static temperature contours at 0.6 Mach number. It can be observed that a temperature at leading edge is maximum about 340 K.

4. Contours of velocity magnitude

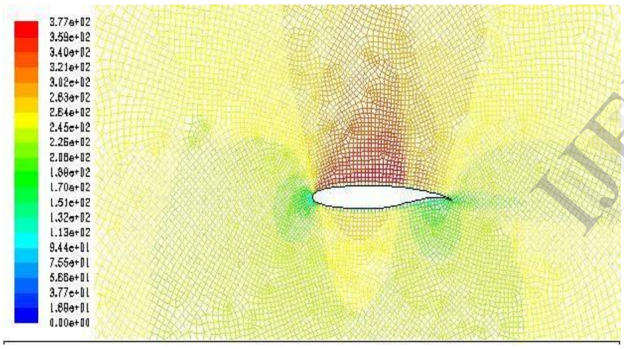

Fig 10: Contours of velocity magnitude

5. Velocity vectors

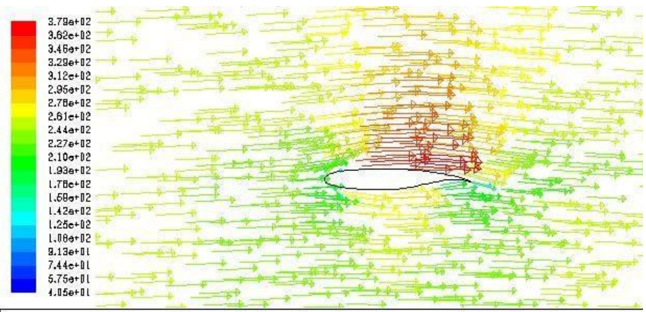

Fig 11: Velocity vectors

Fig 10 and 11 shows Velocity magnitude and Velocity vectors at center of pressure that is maximum camber point velocity is maximum around 379 m/s and minimum at leading edge and trailing edge.

B. Supercritical aerofoil at fifteen degree angle of attack

1. Contours of static pressure

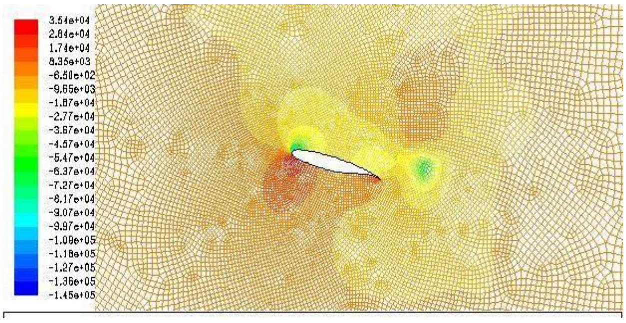

Fig 12: Contours of static pressure

Fig 12 shows static pressure contour at 0.6 Mach number. It can be observed that there is high pressure of 35100 Pascal and at trailing edge pressure is -27700 Pascal. Resultant pressure is 62800 Pascal.

2. Contours of dynamic pressure

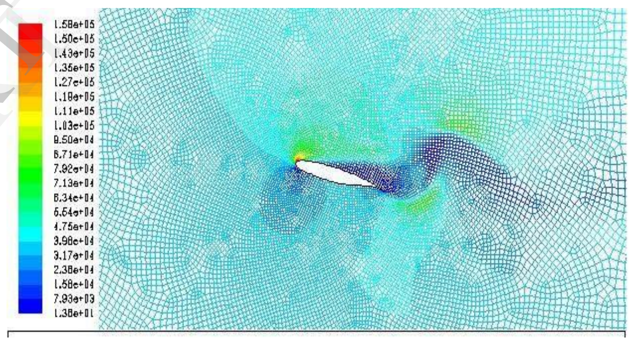

Fig 13: Contours of dynamic pressure

Fig 13 shows dynamic pressure contour at 0.6 Mach number. It can be observed that a weak shock is formed near the trailing edge of the aerofoil. And at the lower surface of the trailing edge high pressure region is there which compensates for lift loss due to flat upper surface.

3. Contours of static temperature

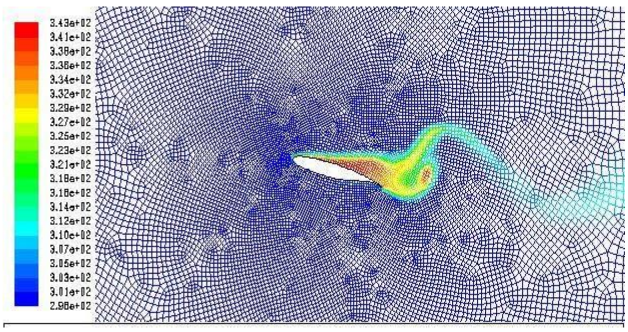

Fig 14: Contours of static temperature

Fig 14 shows temperature is maximum at center of pressure after the separation point from laminar to turbulent flow.

4. Contours of velocity magnitude

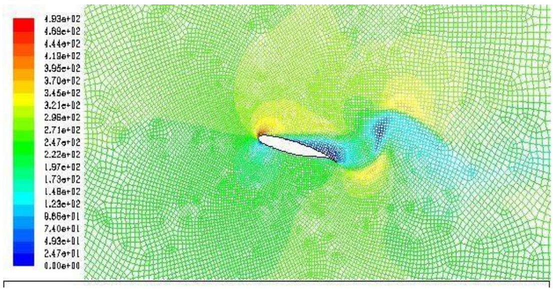

Fig 15: Contours of velocity magnitude

Fig 15 and 16 shows velocity magnitude and velocity vectors. It shows the flow separation after maximum camber point velocity maximum at leading edge 512 m/s

.

5. Velocity vector

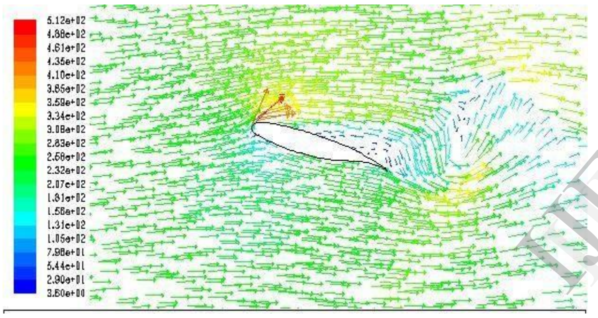

Fig 16: velocity vector

C. Supercritical aerofoil at thirty degree angle of attack

1. Contours of static pressure

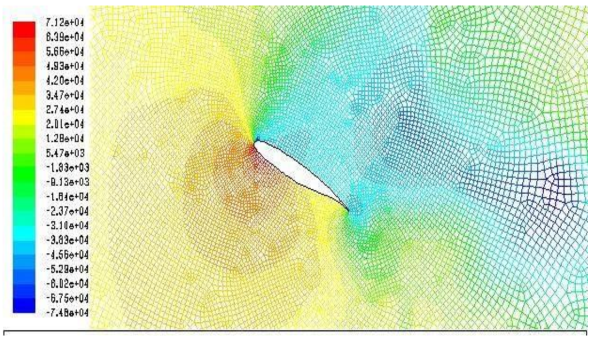

Fig 17: Contours of static pressure

Fig 17 shows static pressure contour at 0.6 Mach number. It can be observed that there is high pressure of 71200 Pascal and at trailing edge pressure is 35000 Pascal. Resultant pressure is 106200 Pascal.

2. Contours of dynamic pressure

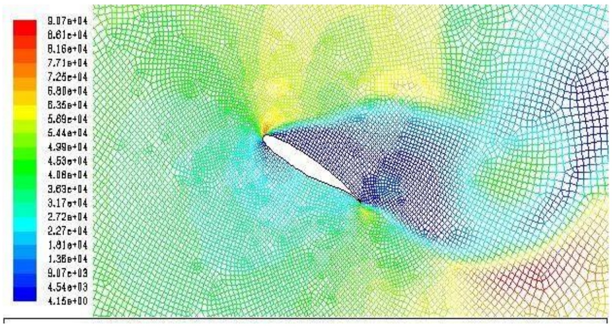

Fig 18: Contours of dynamic pressure

Fig 18 shows dynamic contour at 0.6 Mach number. It can be observed that a weak shock is formed near the trailing edge of the aerofoil. And at the lower surface of the trailing edge high pressure region is there which compensates for lift loss due to flat upper surface.

3. Contours of static temperature

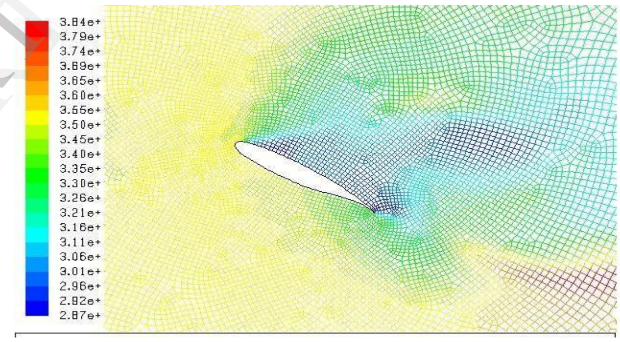

Fig 19: Contours of static temperature

Fig 4.13 shows effect on static temperature and it shows same result as static pressure. Formation of shockwave leads to rise in temperature near the leading edge 384 K.

4. Contours of velocity magnitude

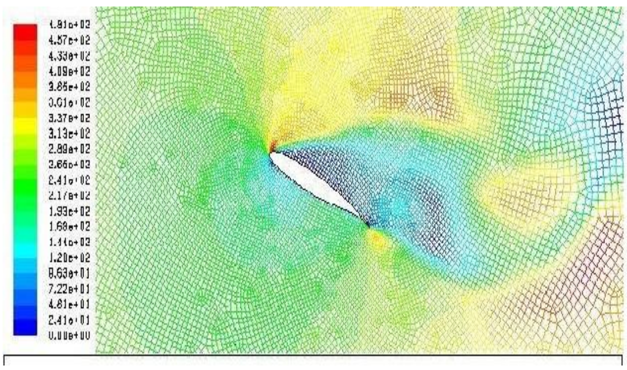

Fig 20: Contours of velocity magnitude

5. Velocity vectors

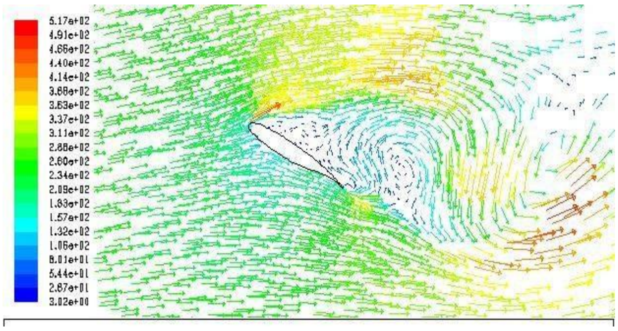

Fig 21: velocity vector

Fig 20 and 21 shows velocity magnitude and velocity vectors of supercritical aerofoil at 0.6 Mach number. It can been seen that flow separation starts at immediate to the leading edge and maximum at leading edge 517 m/s.

D. Simple aerofoil at zero degree

1. Contours of static pressure

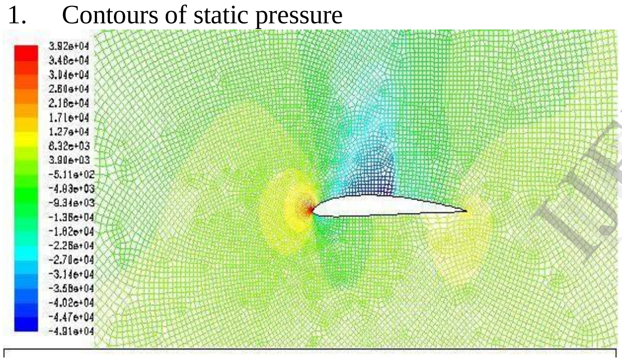

Fig 22: Contours of static pressure

Fig 22 shows static pressure contour at 0.6 Mach number. Iit can be observed that there is high pressure of 39200 Pascal and at trailing edge pressure is -18200 Pascal. Resultant pressure is 57400 Pascal.

2. Contours of dynamic pressure

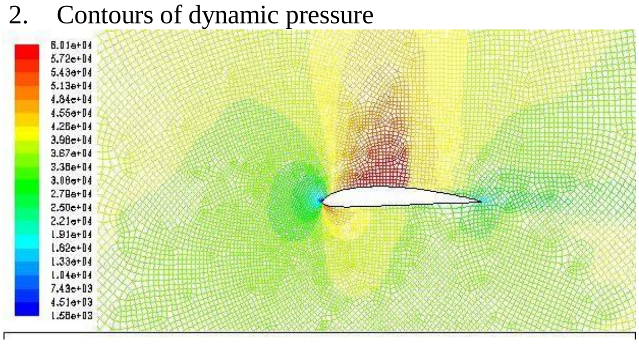

Fig 23: Contours of dynamic pressure

Fig 23 shows dynamic contour at 0.6 Mach number. It can be observed that a weak shock is formed near the trailing edge of the aerofoil. And at the lower surface of the trailing edge high pressure region is there which compensates for lift loss due to flat upper surface.

3. Contours of static temperature

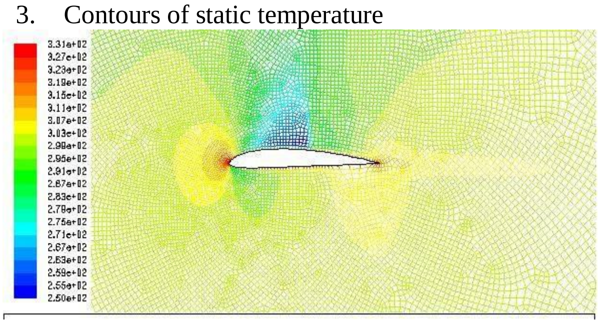

Fig 24: Contours of static temperature

Fig 24 shows effect on static temperature and it shows same result as static pressure and shows effect on dynamic temperature the formation of shockwave leads to rise in temperature at leading edge surface around 331 K.

4. Contours of velocity magnitude

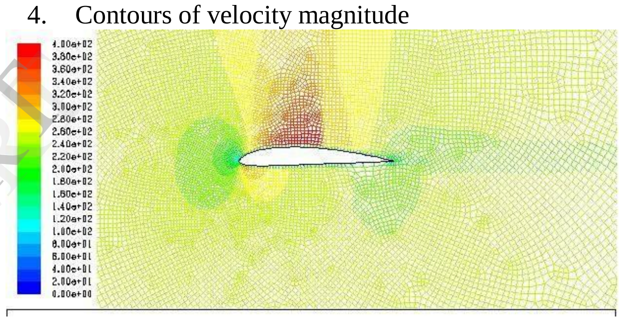

Fig 25: Contours of velocity magnitude

Fig 24 and Fig 25 shows velocity magnitude and velocity vectors of a simple aerofoil at 0.6 Mach number. Form Fig 4.20 can be observed that velocity is maximum at maximum camber point as high as 400 m/s greater than supercritical aerofoil 384 m/s at 0^{o} angle of attack.

5. Velocity vectors

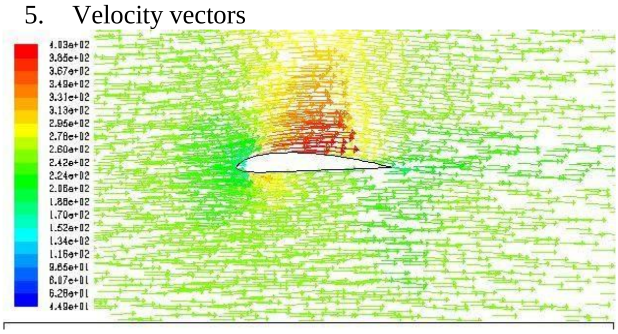

Fig 25: velocity vectors

E. Simple aerofoil at fifteen degree

1. Contours of static pressure

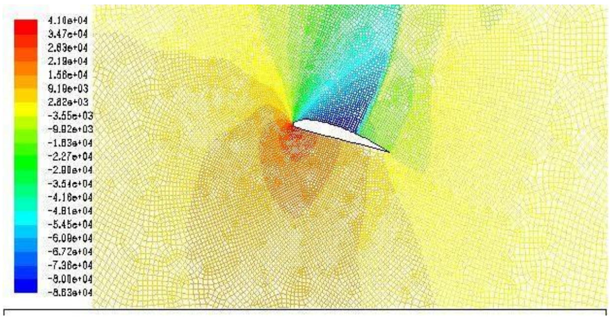

Fig 26: Contours of static pressure

Fig 26 shows static pressure contour at 0.6 Mach number and it can be observed that there is high pressure of 41000 Pascal and at trailing edge pressure is -35100 Pascal. Resultant pressure is 76100 Pascal.

2. Contours of dynamic pressure

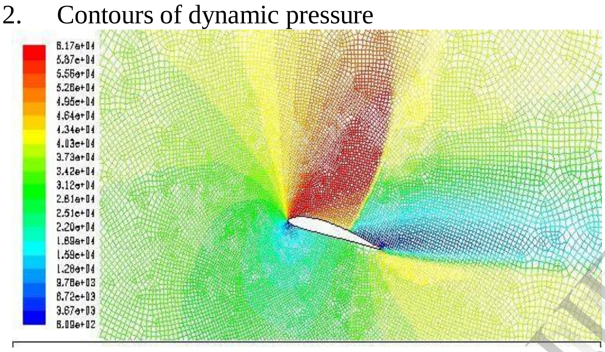

Fig 27: Contours of dynamic pressure

Fig 27 shows dynamic contour at 0.6 Mach number. It can be observed that a weak shock is formed near the trailing edge of the aerofoil. And at the lower surface of the trailing edge high pressure region is there which compensates for lift loss due to flat upper surface.

3. Contours of static temperature

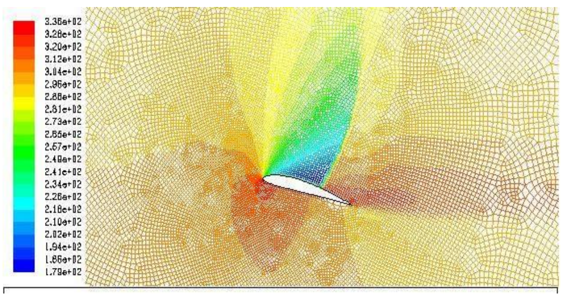

Fig 28: Contours of static temperature

Fig 28 shows effect on static temperature and it shows same result as static pressure. The formation of shockwave leads to rise in temperature. It shows effect on static temperature and it shows same result as static pressure. It also shows effect on the formation of shockwave leads to rise in temperature.

4. Contours of velocity magnitude

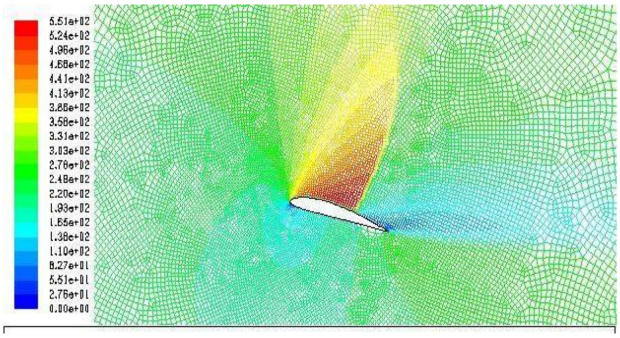

Fig 29: Contours of velocity magnitude

5. Velocity vectors

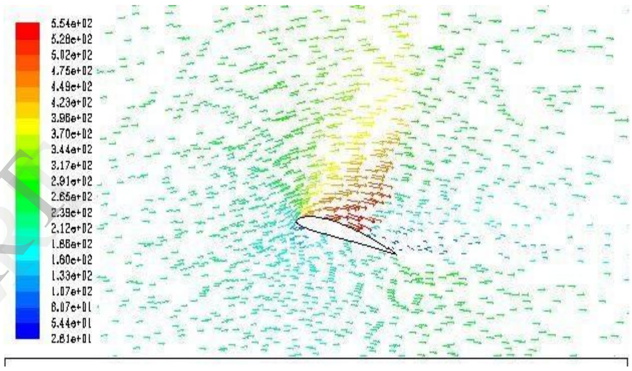

Fig 30: velocity vector

Fig 29 shows velocity magnitude and Fig 30 shows direction of velocity vectors the contours behave same as plots of dynamic pressure. And separation of flow after the trailing edge. Maximum velocity at upper surface of aerofoil about 550 m/s.

F. Simple aerofoil at thirty degree

1. Contours of static pressure

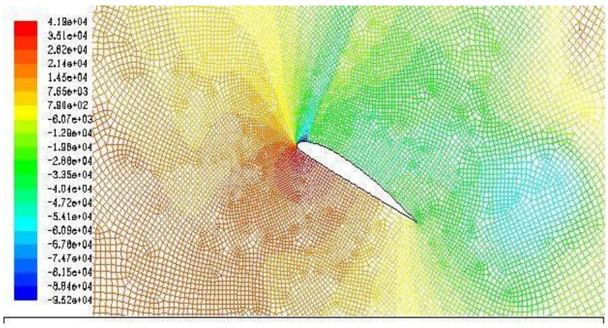

Fig 31: Contours of static pressure

Fig 31 shows static pressure contour at 0.6 Mach number. It can be observed that there is high pressure of 41900 Pascal and at trailing edge pressure is -28800 Pascal. Resultant pressure is 70700 Pascal.

2. Contours of dynamic pressure

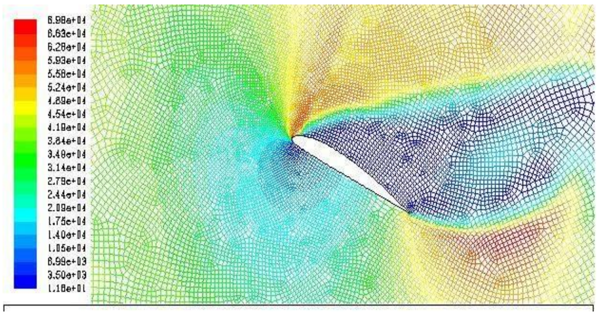

Fig 32: Contours of dynamic pressure

Fig 33 shows dynamic contour at 0.6 Mach number. It can be observed that a weak shock is formed near the trailing edge of the aerofoil. And at the lower surface of the trailing edge high pressure region is there which compensates for lift loss due to flat upper surface.

3. Contours of static temperature

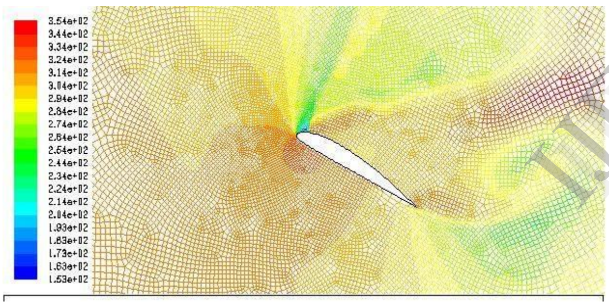

Fig 33: Contours of static temperature

Fig 33 shows effect on static temperature and it shows same result as static pressure. The formation of shockwave leads to rise in temperature at lower surface of leading edge 350 K.

4. Contours of velocity magnitude

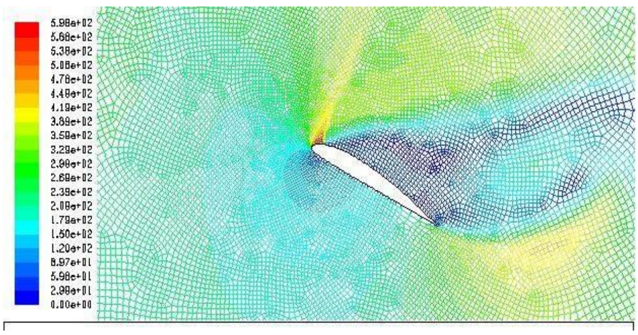

Fig 34: Contours of velocity magnitude

Fig 34 and 35 shows the velocity magnitude and velocity vectors distribution of a simple aerofoil at 0.6 Mach number at 30^{o} angle of attack. From Fig 35 can we observe that foe separation starts immediately to the leading and maximum velocity vector at leading edge only around 630 m/s which is much greater than supercritical aerofoil at 30^{o}.

5. Velocity vectors

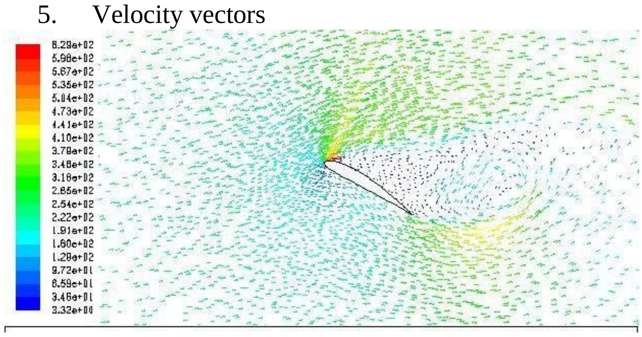

Fig 35: Velocity vector

Fig 35 shows velocity magnitude and 4.30 shows direction of velocity vectors the contours behave same as plots of dynamic pressure.

G. Comparison

1. Drag pressure versus Angle of attack

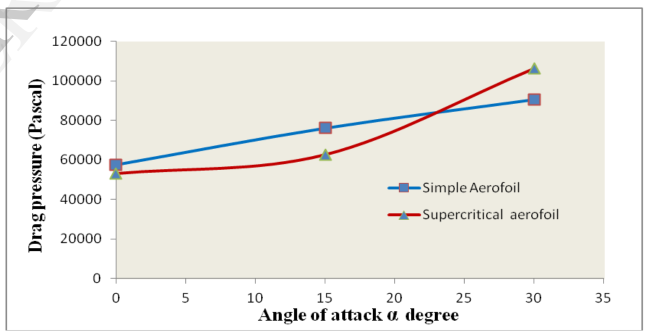

Fig 36: Drag pressure versus angle of attack for supercritical aerofoil and simple aerofoil.

Fig 36 shows that supercritical aerofoil had pressure drag less when compared to simple aerofoil at 0 degree and 15 degree angle of attack. And at 30 degree angle of attack pressure drag is greater in case of Supercritical aerofoil compared tosSimple aerofoil due to maximum surface area facing opposite to relative wind direction.

2. Velocity decrease versus Angle of attack
3.

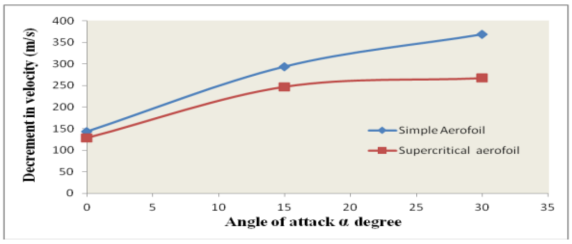

Fig 37: Decrease in Velocity versus Angle of attack for Supercritical aerofoil and Simple aerofoil.

Fig 37 shows magnitude of velocity decrease in the flow field in supercritical aerofoil is less when compared on a simple aerofoil in all cases.

4. Percentage decrement versus angle of attack

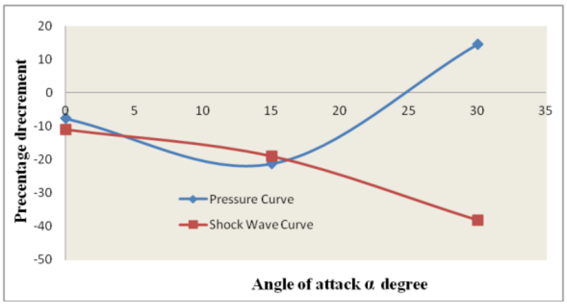

Fig 38: Percentage decrease in drag pressure and velocity in supercritical aerofoil when compared to simple aerofoil versus angle of attack.

V CONCLUSION

Summarized conclusions

- The modified supercritical aerofoil NACA SC (02) 0714 i.e., upper surface of a aerofoil 70 % of chord length is made flat or parallel to chord line. So it reduces decrease in velocity over an aerofoil and so strength of shock waves decreases.
- Less decrease or variation in velocity around aerofoil, less number of shock waves raises and poor shock waves.
- In supercritical aerofoil, thickness of an aerofoil near trailing edge of lower surface is reduced, so that increase in pressure at lowers surface and helps in lift of an aircraft easily compared to simple aerofoil.
- At 15^o angle of attack, pressure drag is much lower in case of supercritical aerofoil compared to simple aerofoil.
- And it limits the angle of attack up to 22^o to supercritical aerofoil because pressure drag increases drastically.

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