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AERODYNAMIC ANALYSIS OF SUPERCRITICAL NACA SC (2)-0714 AIRFOIL USING CFD

Ravikumar T[1], Dr. S B Prakash[2]

[1,2] Department of Thermal Power engineering VTU PG Center, Mysore-29
VTU Belgaum, Karnataka, india

A B S T R A C T

The various angle of attack of an airfoil has a great impact on the aerodynamic performance. Therefore, the optimization of airfoil by considering angle of attack is necessary. The distinctive airfoil shape, base on the concept of local supersonic flow with isentropic recompression, is characterized by a large leading-edge radius, condensed curvature over the middle region of the upper surface, and substantial aft camber. The present work emphasizes the computational study of flow separation over Supercritical Airfoil NACA SC (2)-0714 at different angle of attack (0^0,5º, 10º, 12º, 15º, 17º, 20º and 22.5º) using CFD (Computational fluid dynamics) simulation. Parameters which are observed are Pressure drag, Drag and Lift coefficient at Mach number 0.6. CFD gives 99% accurate results at various angle of attack from 0º to 22.5º. Variations of flow separation are plotted in form of contour for 0.36×10^6 Reynolds number. The computational analysis concluded that no flow separation is seen at 0º and 12º angle of attack due to low pressure gradient, but flow separation started at angle of attack 15º and it increased as angle of attack increasing i.e. 17º, 20º, and 22.5º due to the effect of pressure gradient. It also shows those parameters like maximum pressure, maximum velocity, and pressure drag are increasing as angle of attack increases and the maximum value is found at 22.5º. The Drag and Lift coefficient also increases up to 20º and optimized at 20º.

Keywords: Flow Over Airfoil, Lift And Drag Coefficient, CFD Analysis, Angle Of Attack

I. INTRODUCTION

It is a fact of common experience that a body in motion through a fluid experience a resulting force which, in the majority cases is mainly a resistance to the motion. A class of body exists, However for which the component of the resultant force normal to the direction to the motion is many time greater than the component resisting the motion, and the possibility of the flight of an airplane depends on the use of the body of this class for wing structure [1]. Airfoil is such an aerodynamic shape that when it moves through air, the air is come apart and passes above and below the wing. The wing's upper surface is formed so the air rushing over the top speeds up and stretches out. This reduces the air pressure over the wing. The air flowing below the wing moves in a comparatively straighter line, so its speed and air pressure remainder the same. Since high air pressure always moves in the direction of low air pressure, the air below the wing pushes upward toward the air above the wing. The wing is in the middle, and the whole wing is "lifted." The faster an airplane moves, the more lift there is. And when the force of lift is greater than the force of gravity, the airplane is able to fly.

1.1 Nomenclature of an airfoil

An airfoil is a body of such a shape that when it is placed in an airstreams, it produces an aerodynamic force. This force is used for different purposes such as the cross sections of wings, propeller blades, windmill blades, compressor and turbine blades in a jet engine, and hydrofoils are examples of airfoils. The basic geometry of an airfoil is shown in Figure 1

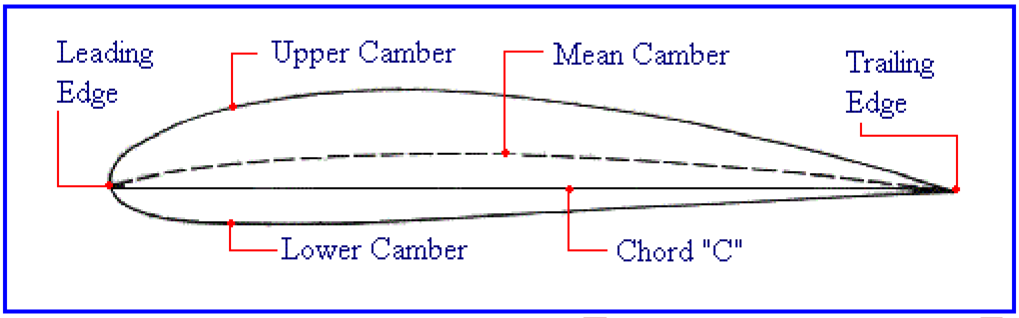

Fig 1: Basic nomenclature of an airfoil

The **leading edge** is the point at the front of the airfoil that has maximum curvature. The **trailing edge** is defined similarly as the point of maximum curvature at the rear of the airfoil. The **chord line** is a straight line connecting the leading and trailing edges of the airfoil. The chord length or simply chord is the length of the **chord** line and is the characteristic dimension of the airfoil section [2].

1.2 Supercritical Airfoil

Mach numbers while retaining acceptable low-speed maximum lift and stalls characteristics and focused on a concept referred to as the supercritical airfoil. This characteristic airfoil shape, based on the concept of resident supersonic flow with isentropic recompression, was considered by a large leading-edge radius, condensed curvature over the middle region of the upper surface, and considerable aft camber. The early phase of this effort was successful in significantly extending drag-rise Mach numbers beyond those of conventional airfoils such as the National Advisory Committee for Aeronautics (NACA) 6-series airfoils. These early supercritical airfoils (denoted by the SC (phase 1) prefix), however, experienced a gradual increase in drag at Mach numbers just preceding drag divergence (referred to as drag creep). This regular buildup of drag was largely associated with an intermediate off-design second velocity peak (an acceleration of the flow over the rear upper-surface portion of the airfoil just before the final recompression at the trailing edge) and relatively weak shock waves above the upper surface. Developments to these early, phase 1 airfoils resulted in airfoils with significantly reduced drag creep features. These early, phase 1 airfoils and the improved phase 1 airfoils were developed before adequate theoretical analysis codes were available and resulted from iterative contour modifications during wind-tunnel testing. The process consisted of estimating experimental pressure distributions at design and off-design conditions and physically altering the airfoil profiles to yield the best drag characteristics over a range of experimental test conditions [4].

1.3 Angle of Attack

If you stretch your arm out through the window of car that is moving at a good speed, you can feel your arm pushed backward. If you hold your arm straight with your hand parallel to the road, and change the angle slightly, you can suddenly feel that it is drown upwards. The hand and arm work like the wing of an airplane and with the right angle (of attack) you can feel a strong lift force [3].

AOA is the angle between the oncoming air or relative wind and a reference line on the airplane or wing. Sometimes the reference line is a line connecting the leading edge and trailing edge at some average point on a wing. Most commercial jet airplanes use the fuselage center line or longitudinal axis as the reference line. It makes no difference what the difference line is as long as it used as consistently. As the nose of the wing turns up, AOA increases, and lift increases. Drag goes up also, but not as quickly as lift as shown in figure 2. During take-off an airplane builds up to a certain speed and then the pilot "rotates" the plane that is, the pilot manipulates the controls so that the nose of the plane comes up and, at some AOA, and the wings generate enough lift to take the plane into the air. Since an airplane wing is fixed to the fuselage, the whole plane has to rotate to increase the wing's angle of attack. Front wings on racecars are fabricated so the angle of attack is easily adjustable to vary the amount of down force needed to balance the car for the driver [2].

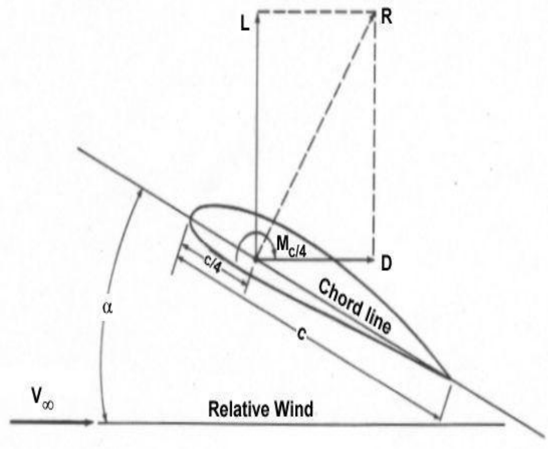

Fig 2 Angle of attack

1.4 Inputs and Boundary Condition

It involves inlet, outlet & wall boundary the velocity components are calculated for each angle attack case as follows. All the outermost boundaries are considered as the '**Pressure Far Field**' boundary conditions in Fluent.

Table 1: Input and Boundary conditions

No. Sl.	Input	Value
1.	Velocity of flow	0.6 Mach or 205.8 m/s
2.	Operating temperature	300 k
3.	Operating pressure	101325 Pa
4.	TurbulenceModel Transition	Spalart-Allmars
5.	Density of fluid	1.225 Kg/m^3
6.	Kinematic viscosity	1.4607×10^{-5}

7.	Reynolds number	3.5×10^6
9.	AOA	$0^0$and 22.5^0 respectively
10	Fluid	Air as an ideal

II. METHODOLOGY

2.1Model creation

We get the model dimensions through coordinates, from these coordinates get from NASA website, we created a model in CATIA V5 R20, and catia model as shown in figure 3

Table 2: Airfoil Data

Thickness	13.9%
Camber	1.5%
Trailing edge angle	16.7^o
Lower flatness	9.4%
Leading edge radius	2.9%

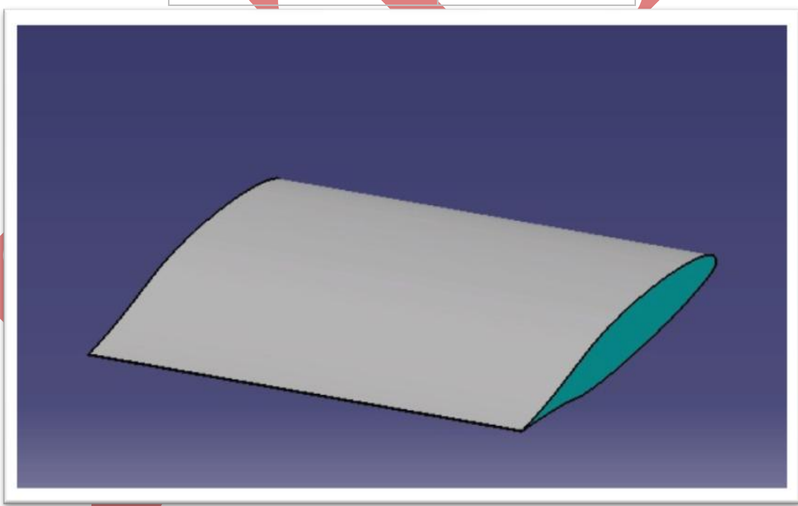

Fig 3: Airfoil NACA SC (2) 0714 Catia Model

2.2 Mesh generation

We meshed the above showsn computational domain using Hyper Mesh 11. We created the surface meshes with the trais and the volume mesh is created with tetras. The meshed model contains top surface, bottom surface, computational domain wall; pressure far filed wall and Pressure outlet surfaces. It has 404867 tetra elements and 55918 nodes. Meshed model shown in figure below.

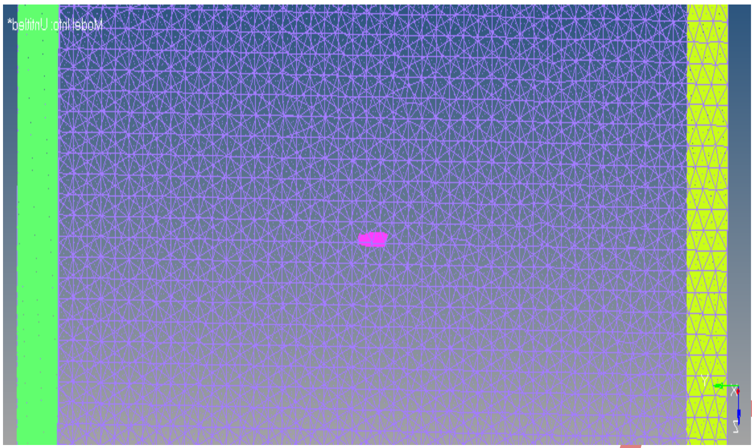

Fig 4 Airfoil NACA SC (2)0714 with outer body meshed profile

2.3 Solver step

Setting up of the solver is very important in any of the fluid flow problem; the solver setting indicates the method and also a procedure for solving (analysis) the problem. In FLUINT the solution can be obtained by many solving methods. All the methods will be working by considering the average of the fluctuation of the flow in-spite of considering the whole path of fluctuation. A first-order upwind discretization is used for the momentum equation and a first order upwind discretization is used for Modified Turbulent viscosity. The solver settings applied in Fluent for the simulations of Airfoil are tabulated in Table 3.

Table 3 Solver Setting Applied For Airfoil

Pressure- Velocity Coupling	SIMPLE
Solver	Pressure Based
Formulation	Implicit
Pressure Discretization	Standard
Momentum Discretization	First order upwind
Modified Turbulent viscosity	First order upwind

III. RESULTS AND DISCUSSIONS

From the contours, we see that there is a region of high pressure at the leading edge (stagnation point) and region of low pressure on the upper surface of airfoil. From Bernoulli equation, we know that whenever there is high velocity, we have low pressure and vice versa.

3.1 0^0 Angle of attack

The figure 5 shows that the contours of static pressure at the airfoil wall and center domain of the airfoil. It shows that the maximum pressure and pressure appeared at leading edge is 29079.86pascal and the pressure appeared at trailing edge is 2390.27pascal and the Pressure Drag is measured by taking difference in static pressure at leading edge and trailing edge. More difference means more pressure drag. So the pressure drag is 26689.59pascal and the average pressure around the airfoil wall is -2847.358pascal. The static pressure at upper surface of the airfoil wall -5950.22pascal and at lower surface is -5950.22pascal so the Lift of the airfoil is not much high enough at this angle of attack.

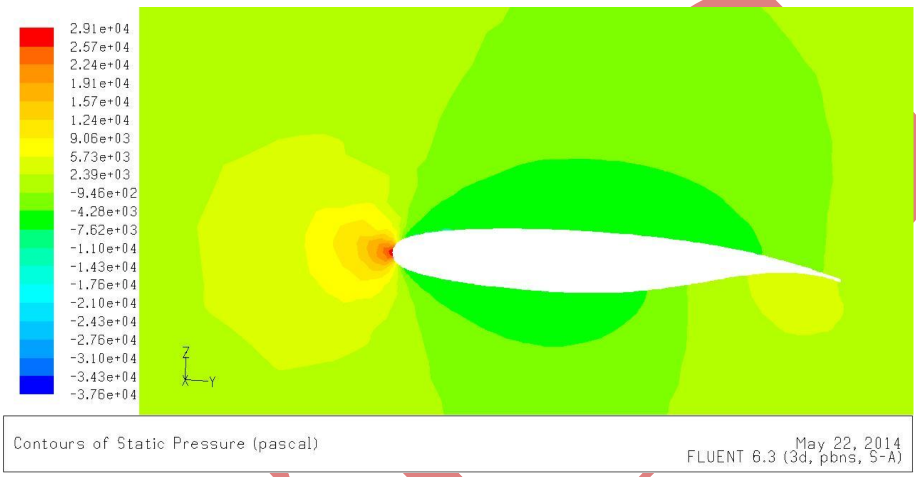

Fig 5 Static Pressure of airfoil at 0° Angle of attack

3.2 12^0 angle of attack

The figure 6 shows that the contours of static pressure at the airfoil wall and center domain of the airfoil. It shows that the maximum pressure is appeared in plane is 41358.3pascal and pressure appeared at leading edge is 27624.89pascal and the pressure appeared at trailing edge is -3765.74pascal so the Pressure Drag is 31390.63pascal and the average pressure around the airfoil wall is-2696.56pascal. The static pressure at upper surface of the airfoil wall -7689.6pascal and at lower surface is -3765.74pascal so the Lift of the airfoil is more than the drag.

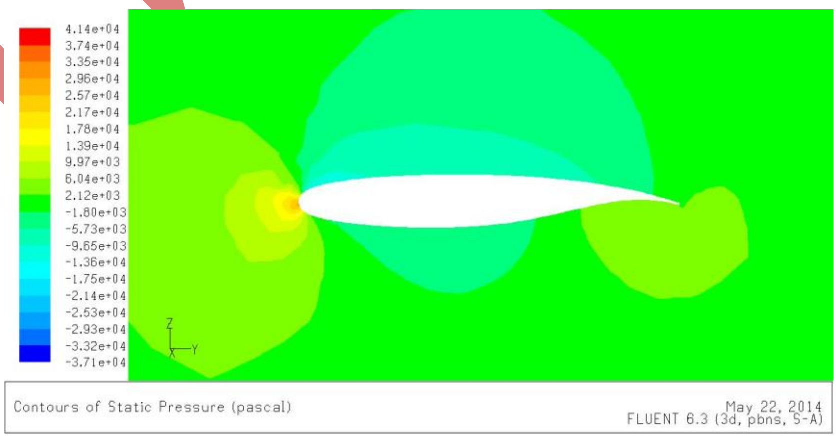

Fig6 Static Pressure at 12° Angle of attack

3.3 20° Angle of attack

The figure 7 shows that the contours of static pressure at the airfoil wall and center domain of the airfoil. It shows that the maximum pressure is appeared in plane is 69675.74pascal and pressure appeared at leading edge is 29648.28pascal and the pressure appeared at trailing edge is -4747.38pascal, the average pressure around the airfoil wall is -1881.45pascal and Pressure Drag is 34395.65pascal. The static pressure at lower surface of the airfoil wall 2963.31pascal and at upper surface is -7710.69pascal so the Lift of the airfoil is more and this angle is optimized.

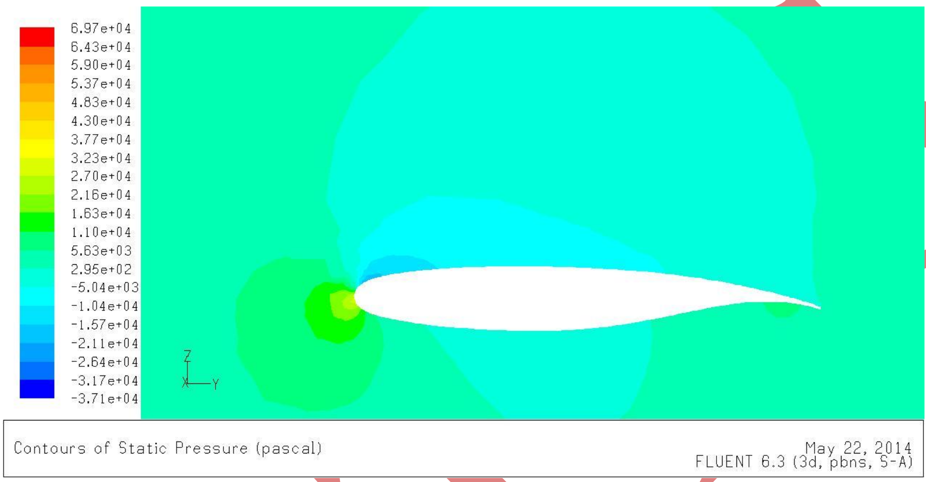

Fig7 Static Pressure at 20° Angle of attack

3.4 22.5^0 angle of attack

The figure 8 shows that the contours of static pressure at center domain of the airfoil. The maximum pressure is appeared in plane is 78052.59pascal and pressure appeared at leading edge is 29005.72pascal and the pressure appeared at trailing edge is 309.223pascal, so the Pressure Drag is also reduced as compare previous angle of attack, pressure drag is 28696.5pascal and the average pressure around the airfoil wall is-1881.45pascal. The static pressure at upper surface of the airfoil wall 3039.725pascal and at lower surface is 2789.5pascal so the Lift of the airfoil is decrease.

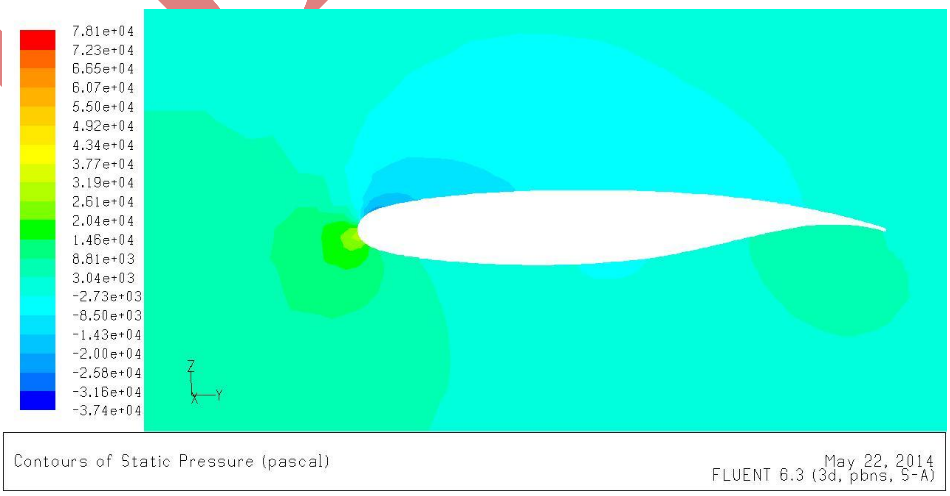

Fig8 Static Pressure at 22.5° Angle of attack

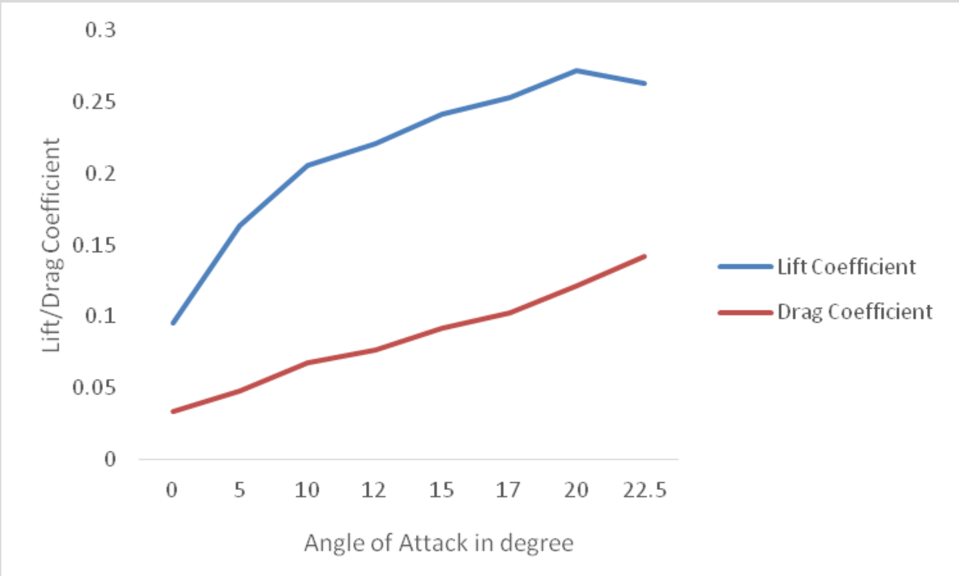

Fig9 Angle of attack v/s Lift and Drag coefficient

The figure 9 shows that the angle of attack v/s Lift and Drag coefficient plot. Drag coefficient will slightly increases with angle of attack but not like lift and the turbulence will be there at trailing edge but less as compared to leading edge. The pressure at trailing edge is compared to leading edge is very less and maximum velocity of the airfoil is less at trailing edge so drag will not increases as like lift but drag will also be there. After the 20^0 angle of attack pressure on upper surface of the airfoil at leading edge is more as compare to lower surface of the airfoil at leading edge so drag is increases gradually from 20^0 angle of attack. Lift coefficient will increases as the angle of attack increases because pressure on lower surface of the airfoil at leading edge is more as compare to pressure on upper surface and turbulence is very high on the lower surface of the airfoil at leading edge so Lift of the airfoil will occur. Lift of airfoil is increased up to 20^0 angle of attack, after this angle of attack lift coefficient decreases gradually because the pressure on lower surface of the airfoil is decreases than the upper surface of the airfoil.

IV. CONCLUSION

It is found that the static pressure on the lower surface of the airfoil at leading edge side is greater than static pressure on the upper surface of the airfoil so the lift of airfoil is increases gradually till 20^0 angle of attack, from this angle of attack the pressure on lower surface of the airfoil is reduced than upper surface so lift of the airfoil is decreasing and drag of the airfoil is increasing. Pressure Drag is calculated by taking difference in static pressure at leading edge and trailing edge. More difference means more pressure drag. At 20^0 Angle of attack pressure drag will increase and form 22.5^0 angle of attack pressure drag will decrease continuously.

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