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ANALYSIS OF THE 2-D SUBSONIC FLOW OVER A NACA-0012 AIRFOIL

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Abstract: In this report we have obtained the drag and lift forces using CFD which can also be determined through experiments using wind tunnel testing. In experimental setup, the design model has to be placed in the test section. This process is quite laborious & (surely) cost more than CFD techniques cost for the same. Thus we have gone through analytical method then it can be validated by experimental testing. The analysis of the two dimensional subsonic flow over a NACA 0012 airfoil at various angles of attack and operating at a Reynolds number of 3×E+06 is presented. The CFD simulation results show close agreement with those of the experiments, thus suggesting a reliable alternative to experimental method in determining drag and lift.

Keywords: Flow over airfoil; pressure coefficient; CFD analysis; Angle of attack.

INTRODUCTION

It is a fact of common experience that a body in motion through a fluid experiences a resultant force which, in most 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 split and passes above and below the wing. The wing's upper surface is shaped so the air rushing over the top speeds up and stretches out. This decreases the air pressure above the wing. The air flowing below the wing moves in a comparatively straighter line, so its speed and air pressure remain the same. Since high air pressure always moves toward 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.

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.

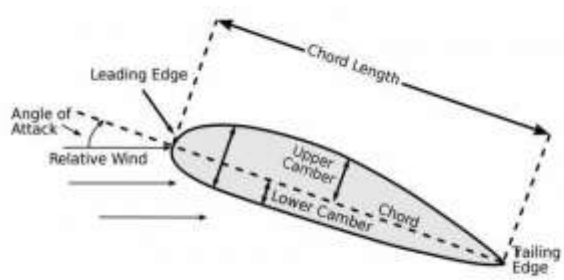

Figure 1: Basic nomenclature of an airfoil

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.

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 [2].

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. 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, 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.

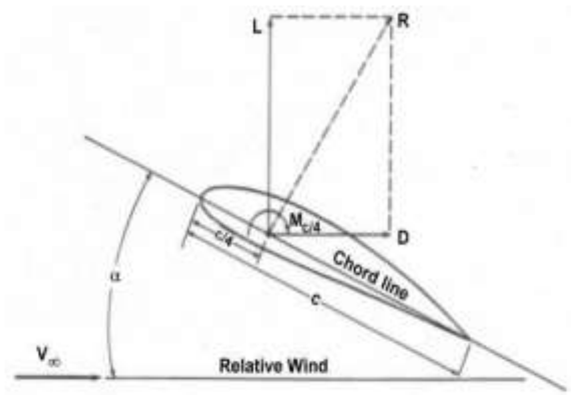

Figure 2: Angle of attack

COEFFICIENT OF DRAG AND COEFFICIENT OF LIFT

The drag equation,

$$F_d = \tfrac{1}{2}\rho v^2 c_d A$$

so co efficient of drag is given by the,

$$c_{\mathrm{d}} = \frac{2F_{\mathrm{d}}}{\rho v^2 A}$$

is essentially a statement that the drag force on any object is proportional to the density of the fluid and proportional to the square of the relative speed between the object and the fluid. In fluid dynamics the c_d is a dimensionless quantity that is used to quantify the drag or resistance of an object in a fluid environment such as air or water. It is used in the drag equation where a lower drag coefficient indicates the object will have less aerodynamic or drag. The drag coefficients always associated with a particular surface area. The drag coefficient of any object comprises the effects of the two basic contributors to fluid dynamics drag: skin friction and from drag. The drag coefficient of a lifting airfoil or hydrofoil also includes the effects of lift induced drag. The drag coefficient of a complete structure such as an aircraft also includes the effects of interference drag. The overall drag coefficient defined in the usual manner is The reference area depends on what type of drag coefficient is being measured. For automobiles and many other objects, the reference area is the projected frontal area of the vehicle. This may not necessarily be the cross sectional area of the vehicle, depending on where the cross section is taken and for an airfoil the surface area is a plane form area. The lift equation,

$$L = \tfrac{1}{2}\rho v^2 A C_L$$

so coefficient of lift is given by the,

$$C_{\mathrm{L}} = \frac{L}{\frac{1}{2}\rho v^2 S} = \frac{2L}{\rho v^2 S} = \frac{L}{qS}$$

A fluid flowing past the surface of a body exerts a force on it. Lift is the component of this force that is perpendicular to the oncoming flow direction. It contrasts with the drag force, which is the component of the surface force parallel to the flow direction. If the fluid is air, the force is called an aerodynamic force.

RELATIONSHIP BETWEEN ANGLE OF ATTACK, COEFFICIENT OF DRAG AND COEFFICIENT OF LIFT.

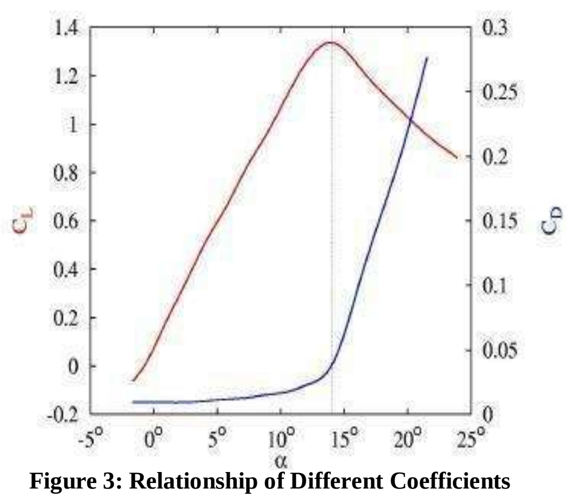

Figure 3: Relationship of Different Coefficients

The lift coefficient of a fixed-wing aircraft varies with angle of attack. Increasing angle of attack is associated with increasing lift coefficient up to the maximum lift coefficient, after which lift coefficient decreases. As the angle of attack of fixed-wing aircraft increases, separation of the airflow from the upper surface of the wing becomes more pronounced, leading to a reduction in the rate of increase of the lift coefficient. The figure shows a typical curve for a cambered straight wing. A symmetrical wing has zero lift at 0 degrees angle of attack. The lift curve is also influenced by wing platform. A swept wing has a lower, flatter curve with a higher critical angle.

Identically the value of drag coefficient is zero at the zero AOA and it increase slowly till the stall condition and at the time of stall as well as after stall it increase readily as shown in figure 3. Particular airspeed, the airspeed at which the aircraft stalls varies with the weight of the aircraft, the load factor, the center of gravity of the aircraft and other factors. However the aircraft always stalls at the same critical angle of attack. The critical or stalling angle of attack is typically around 15° for many airfoils.

INPUTS AND BOUNDARY CONDITION

The problem considers flow around the Aerospatiale a airfoil at 0º and 6ºangles of AOA. For that we take some initial inputs and boundary condition for our problems which are shown in the table 1.

S No	Input	Value
1	Velocity of flow	0.15 Mach or 51 m/s
2	Operating temperature	300 k
3	Operating pressure	101325 Pa.
4	Model	Transition sst (4th equation)
5	Density of fluid	1.225 Kg/m3
6	Kinematic viscosity	1.4607 × E-5
7	Reynolds number	3.5 × E+6
8	Length	1 m
9	AOA	0 degree and 6 degree Respectively
10	Fluid	Air as a ideal

Table 1: operating parameters

CFD ANALYSIS PROCESS

S No.	Steps	Process
1	Problem statement	Information about the flow
2	Mathematical model	Generate 3D model
3	Mesh generation	Nodes/cells, time instants
4	Space discretization	Coupled ODE/DAE systems
5	Time discretization	Algebraic system Ax=b
6	Iterative solver	Discrete function values
7	CFD software	Implementation, debugging
8	Simulation run	Parameters, stopping criteria
9	Post processing	Visualization, analysis of data
10	Verification	Model validation / adjustment
11	Saving case and Data	Save all the obtain data
12	Comparing	Comparing the outcome values with real practical values

Table 2: General procedure for CFD analysis

MESH GENERATION

In order to analyze fluid flow, flow domains are split into smaller sub domains. The governing equations are then discretized and solved inside each of these sub domains. The meshed area around the aerofoil is shown in below figure in which meshing accuracy is increasing as we are go towards the aerofoil.

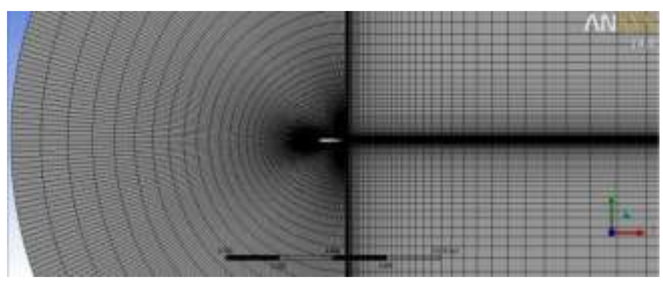

Figure 4: Meshed Region

CONTOURS OF STATIC PRESSURE OVER NACA 0012 AIRFOIL

The static pressure of the air is simply the weight per unit area of the air above the level under consideration. For instance, the weight of the column of air with a cross-sectional area of 1 ft-square and extending upward from sea level through the atmosphere is 2116 lb. The sea level static level is therefore 2116 psf. Static pressure is decrease as altitude is increased because there is less air weight above. At 18,000 ft altitude the static pressure is about half that at sea level [3]. The amalgamation of static pressure and dynamic pressure is known as total pressure.

For and angle of attack is zero degree we obtain that the contours of static pressure over an aerofoil is symmetrical for above and lower sections and the stagnation point is exactly at the nose of an aerofoil. Hence there are no pressure different created between two faces of aerofoil at zero degree of an angle of attack.

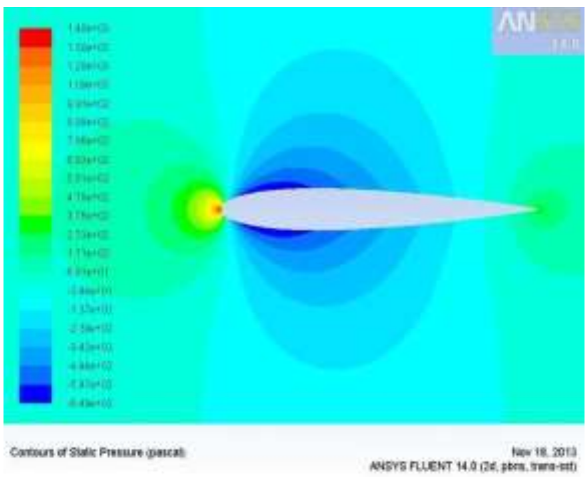

Figure 5: Contours of static pressure over NACA 0012 airfoil at 0 degree of AOA

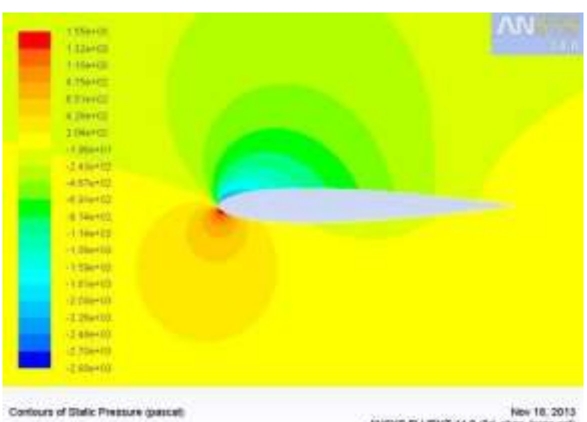

Figure 6: Contours of static pressure over NACA 0012 airfoil at 6 degree of AOA

For an angle of attack of 6 degrees, we see that the flow has a stagnation point just under the leading edge and hence producing lift as there is a low pressure region on the upper surface of the foil as shown in Figure 6. We can also observe that Bernoulli's principle is holding true; the velocity is high

(Denoted by the red contours) at the low pressure region and vice-versa. There is a region of high pressure at the leading edge (stagnation point) and region of low pressure on the upper surface of airfoil.

CONTOURS OF VELOCITY MAGNITUDE OVER NACA 0012 AIRFOIL

As shown in figure 7 and 8 at the 0 degree of AOA the velocity contours are same as symmetrical and at 6 degree of AOA the stagnation point is slightly shift towards the trailing edge via bottom surface hence it will create low velocity region at lower side of the airfoil and higher velocity acceleration region at the upper side of the airfoil and according to principle of Bernoulli's upper surface will gain low pressure and lower surface will gain higher pressure. Hence value of coefficient of lift will increase and coefficient of drag will also increase but the increasing in drag is low compare to increasing in lift force.

In a symmetrical airfoil at no incidence, the distribution of velocity and thus the pressures along both surfaces would have been exactly the same, canceling each other to a resulting total lift force of zero

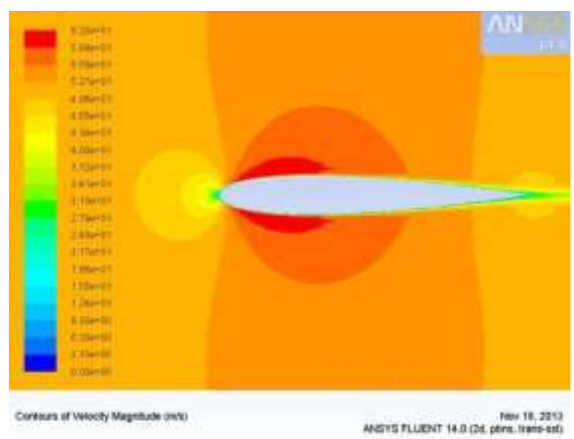

Figure 7: Contours of velocity magnitude over NACA 0012 airfoil at 0 degree of AOA

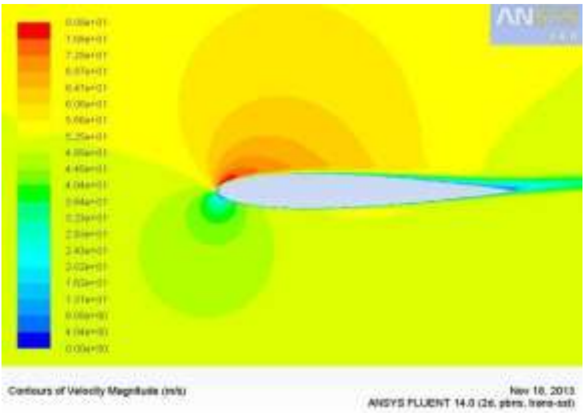

Figure 8: contours of velocity magnitude over NACA 0012 airfoil at 6 degree of AOA.

CONCLUSION

Based on the CFD analysis of the flow over NACA 0012 air foil we can conclude that at the zero degree of AOA there is no lift force generated and if we want to increase amount of lift force and value of lift coefficient then we have to increase the value of AOA. By doing that obviously amount of drag force and value of drag coefficient also increased but the amount of increment in drag force and drag coefficient is quite lower compare to lift force. The exact numerical values of the forces and coefficient are given as below.

RESULTANT VALUES AND GRAPHS

Graphs of coefficient of drag and coefficient of lift at AOA zero and six respectively verses number of iterations is shown below.

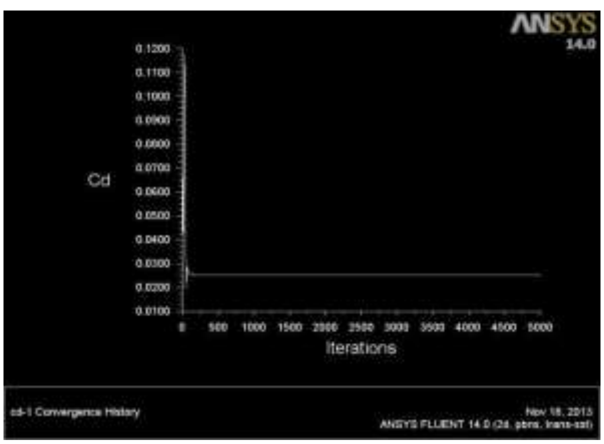

Figure 9: Graph of coefficient of drag at six degree of AOA.

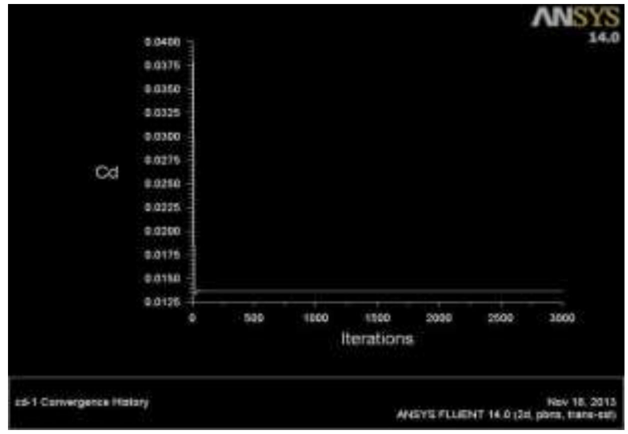

Figure 10: Graph of coefficient of drag at zero degree AOA.

Variables	0 degree of AOA	6 degree of AOA
Drag force	21.79 N	40.0502 N
Lift force	0.2487 N	888.7298 N
Drag coefficient	0.01373	0.02566
Lift coefficient	0.00015	0.56947

Table 3: Conclusion Table

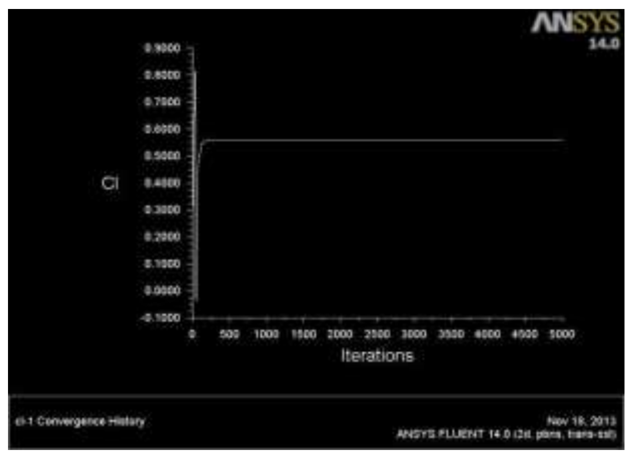

Figure 11: Graph of coefficient of Lift at six degree of AOA

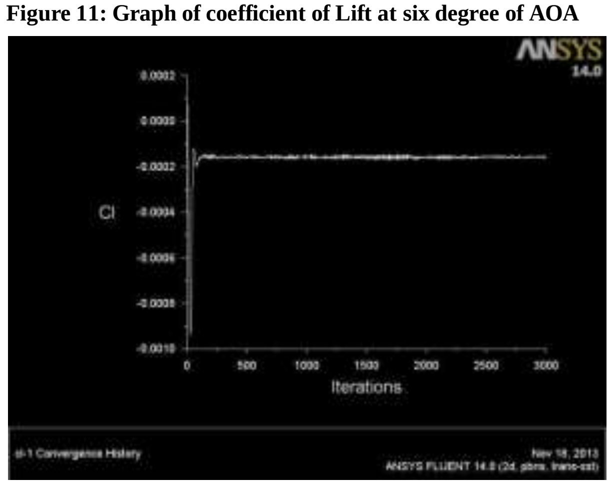

Figure 12: Graph of coefficient of Lift at zero degree of AOA.

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