ISSN 2393-8471

International Journal of Recent Research in Civil and Mechanical Engineering (IJRRCME)
Vol. 2, Issue 1, pp: (268-277), Month: April 2015 – September 2015, Available at: www.paperpublications.org

Airfoil Terminology, Its Theory and Variations As Well As Relations with Its Operational Lift Force and Drag Force in Ambient Conditions

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Abstract: **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. 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 remains 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.**

Keywords: **Airfoil, lift force, Drag Force.**

1. INTRODUCTION

An airfoil-shaped body moved through a fluid produces an aerodynamic force. The component of this force perpendicular to the direction of motion is called lift. The component parallel to the direction of motion is called drag. Subsonic flight airfoils have a characteristic shape with a rounded leading edge, followed by a sharp trailing edge, often with a symmetric curvature of upper and lower surfaces. Foils of similar function designed with water as the working fluid are called hydrofoils.

The lift on an airfoil is primarily the result of its angle of attack and shape. When oriented at a suitable angle, the airfoil deflects the oncoming air, resulting in a force on the airfoil in the direction opposite to the deflection.

This force is known as aerodynamic force and can be resolved into two components: lift and drag. Most foil shapes require a positive angle of attack to generate lift, but cambered airfoils can generate lift at zero angle of attack.

This "turning" of the air in the vicinity of the airfoil creates curved streamlines which results in lower pressure on one side and higher pressure on the other. This pressure difference is accompanied by a velocity difference, via Bernoulli's principle, so the resulting flow field about the airfoil has a higher average velocity on the upper surface than on the lower surface. The lift force can be related directly to the average top/bottom velocity difference without computing the pressure by using the concept of circulation and the Kutta-Joukowski theorem.

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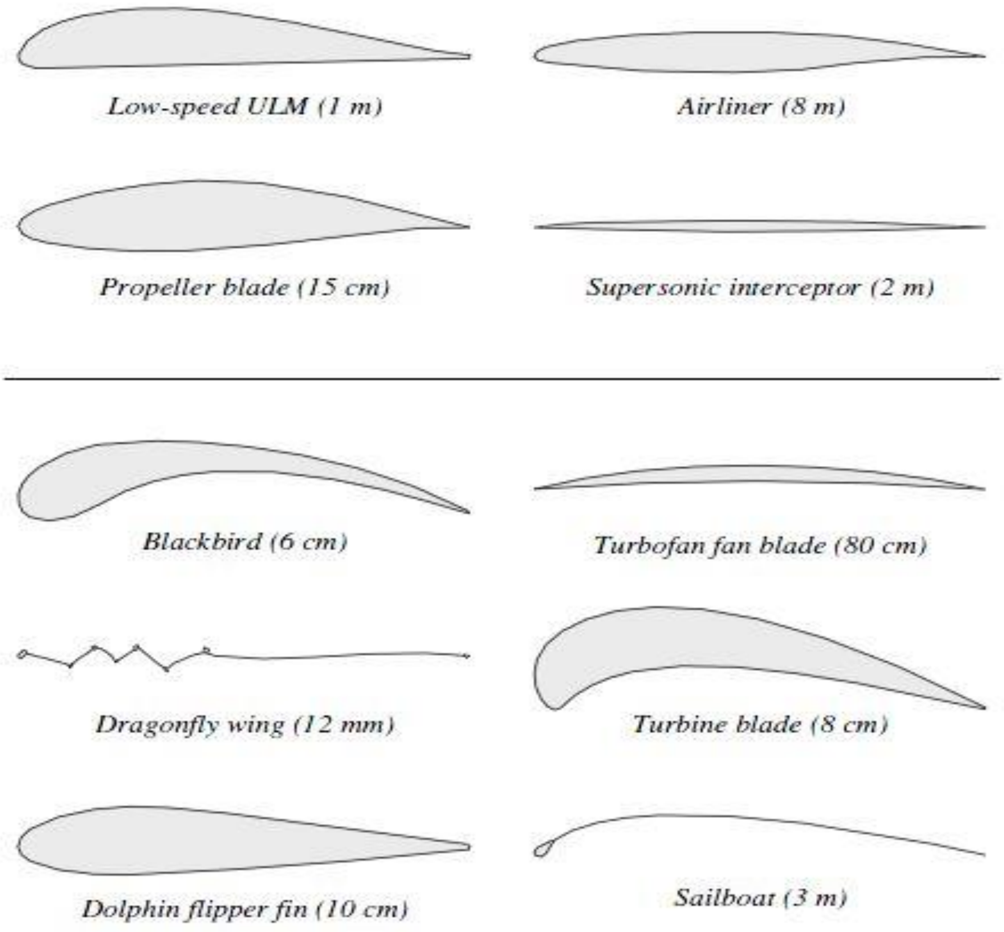

Figure: 1 Examples of airfoils in nature and within various vehicles. Though not strictly an airfoil, the dolphin flipper obeys the same principles in a different fluid medium

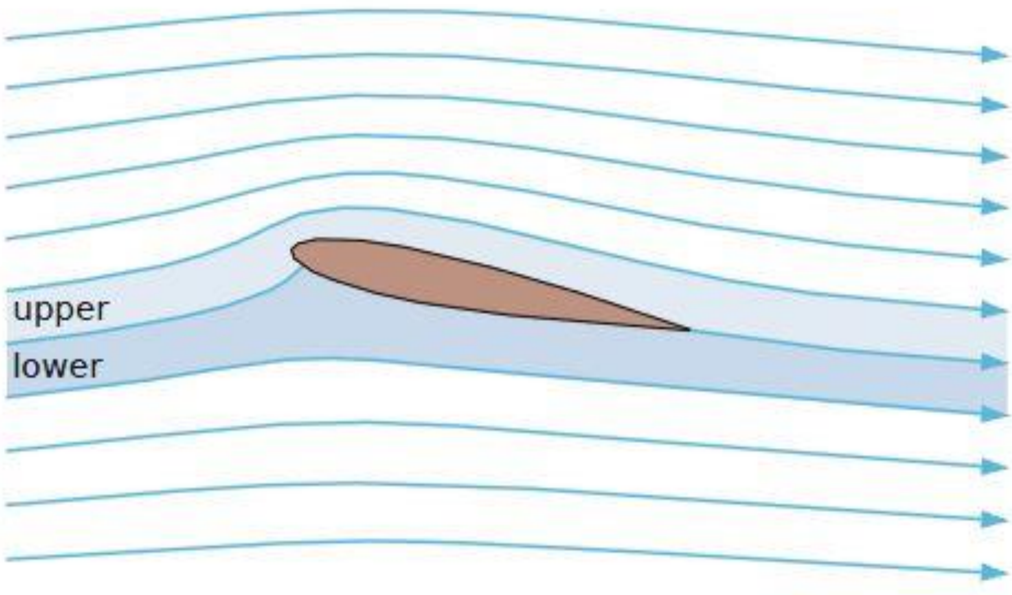

Figure: 2 Streamlines around a NACA 0012 airfoil at moderate angle of attack

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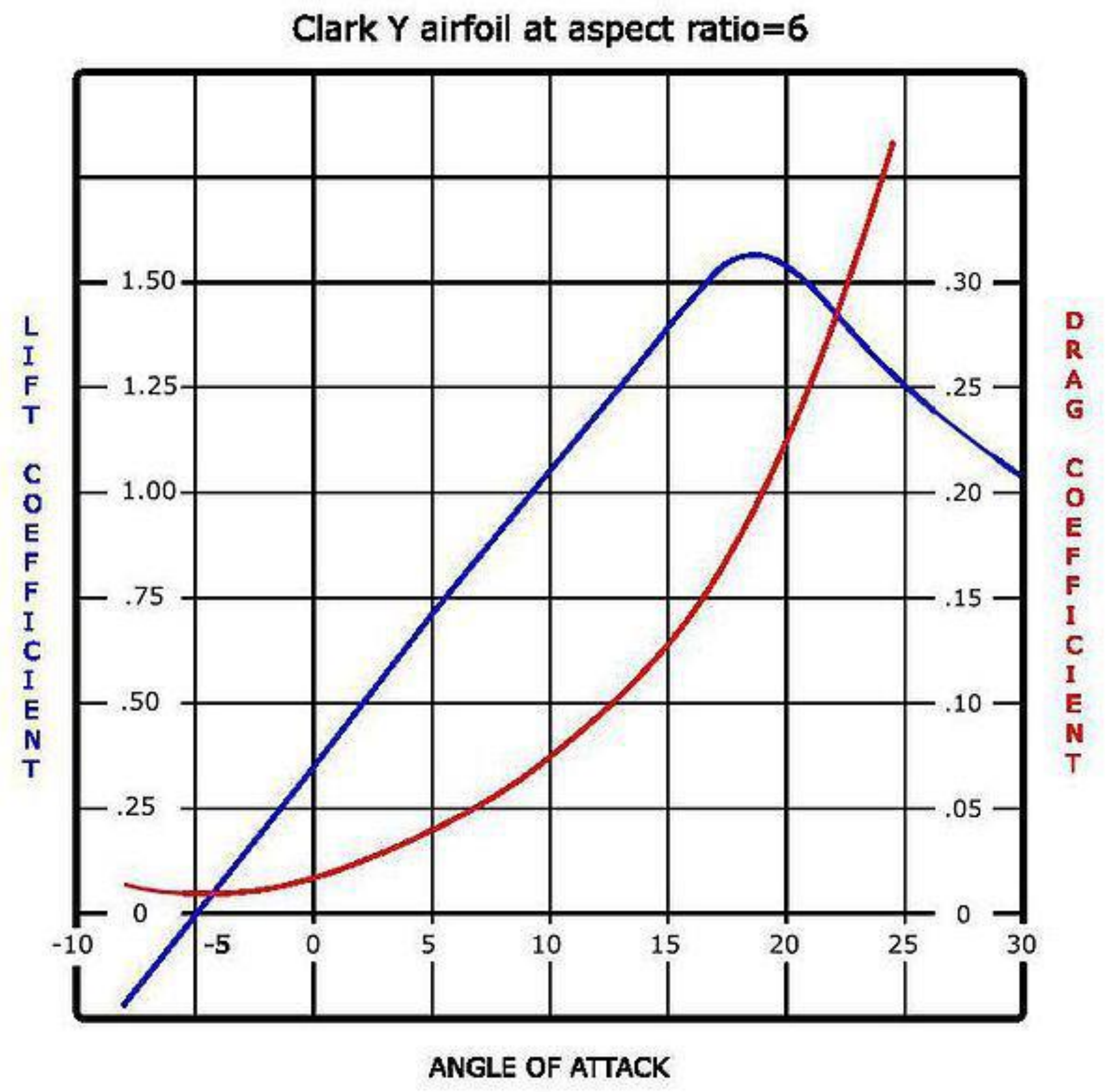

Figure: 3 Lift and Drag curves for a typical airfoil

2. THEORY OF AIRFOIL

Thin airfoil theory is a simple theory of airfoils that relates angle of attack to lift for incompressible, in viscid flows. It was devised by German-American mathematician Max Munk and further refined by British aerodynamicist Hermann Glauert and others in the 1920s. The theory idealizes the flow around an airfoil as two dimensional flows around a thin airfoil. It can be imagined as addressing an airfoil of zero thickness and infinite wingspan.

Thin airfoil theory was particularly notable in its day because it provided a sound theoretical basis for the following important properties of airfoils in two-dimensional Flow:

(1) On a symmetric airfoil, the center of pressure and Aerodynamic center lies exactly one quarter of the chord behind the leading edge

(2) On a cambered airfoil, the aerodynamic center lies exactly One quarter of the chord behind the leading edge

(3) The slope of the lift coefficient versus angle of attack Line is 2π units per radian As a consequence of (3), the section lift coefficient of a Symmetric airfoil of infinite wingspan is:

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$$c_L = 2\pi\alpha$$

Where c_L is the section lift coefficient α *is* the angle of attack in radians, measured Relative to the chord line.

(The above expression is also applicable to a cambered Airfoil where α is the angle of attack measured relative to The zero-lift line instead of the chord line.)

Also as a consequence of (3), the section lift coefficient Of a cambered airfoil of infinite wingspan is:

$$c_L = c_{L_0} + 2\pi\alpha$$

Thin airfoil theory does not account for the stall of the Airfoil, which usually occurs at an angle of attack between 10° and 15° for typical airfoils.

4 Derivation of thin airfoil theory: The airfoil is modeled as a thin lifting mean-line (camber line). The mean-line, y(x), is considered to produce a distribution of vorticity (*s*) along the line, s. By the

Kutta condition, the vorticity is zero at the trailing edge. Since the airfoil is thin, x (chord position) can be used instead of s, and all angles can be approximated as small. From the Biot–Savart law, this vorticity produces a flow

field *w*(*x*) where

$$w(x) = \frac{1}{(2\pi)} \int_0^c \frac{\gamma(x')}{(x - x')} dx'$$

Where *x* is the location where induced velocity is produced, *x′* is the location of the vortex element producing the velocity and *c* is the chord length of the airfoil.

Since there is no flow normal to the curved surface of the airfoil, *w*(*x*) balances that from the component of main flow *V* , which is locally normal to the plate – the main flow is locally inclined to the plate by an angle *α-dy*/*dx*. That is:

3. 4 DERIVATION OF THIN AIRFOIL THEORY

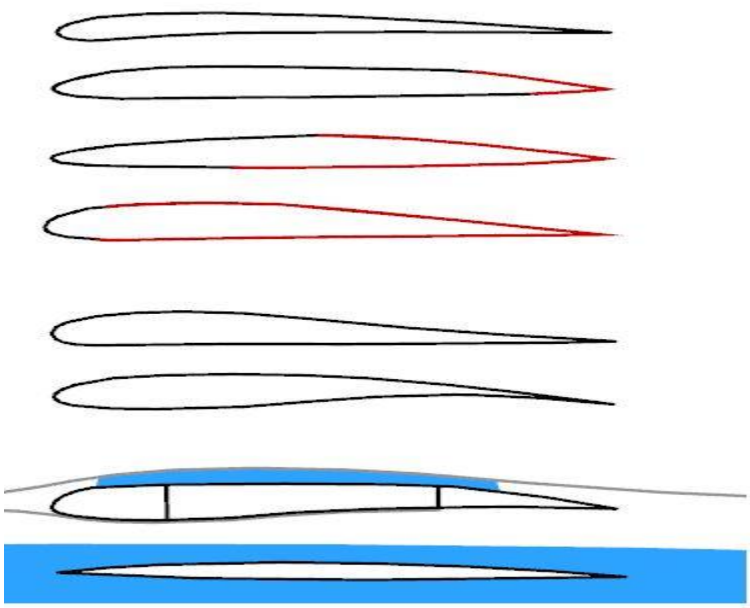

Figure: 4. 4 Derivation of thin airfoil theory

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From top to bottom:

• Laminar flow airfoil for a RC park flyer

• Laminar flow airfoil for a RC pylon racer

• Laminar flow airfoil for a manned propeller aircraft

• Laminar flow at a jet airliner airfoil

• Stable airfoil used for flying wings

• Aft loaded airfoil allowing for a large main spar and late stall

• Transonic supercritical airfoil

• Supersonic leading edge airfoil

Colors:

Black = laminar flow,

Red = turbulent flow,

Grey = subsonic stream,

Blue = supersonic flow volume

$$V\,(\alpha - dy/dx) = w(x) = \frac{1}{(2\pi)}\int_0^c \frac{\gamma(x')}{(x-x')}dx'$$

This integral equation can by solved for (*x*) , after replacing x by

$$x = c(1-\cos(\theta))/2$$

as a Fourier series in $A_n \sin(n\theta)$ with a modified lead term $A_0(1+\cos(\theta))/\sin(\theta)$

That is

$$\frac{\gamma(\theta)}{(2V)} = A_0\frac{(1+\cos(\theta))}{\sin(\theta)} + \sum A_n\ \sin(n\theta))$$

(These terms are known as the Glauert integral). The coefficients are given by

$$A_0 = \alpha - \frac{1}{\pi}\int_0^\pi (dy/dx)\ d\theta$$

and

$$A_n = \frac{2}{\pi}\int_0^\pi \cos(n\theta)(dy/dx)\ d\theta$$

By the Kutta–Joukowski theorem, the total lift force F is Proportional to

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$$\rho V \int_0^c \gamma(x)\, dx$$

and its moment M about the leading edge to

$$\rho V \int_0^c x\, \gamma(x)\, dx$$

The calculated Lift coefficient depends only on the first two terms of the Fourier series, as

$$C_L = 2\pi(A_0 + A_1/2)$$

The moment M about the leading edge depends only on $A0;A1$ and $A2$, as

$$C_M = -0.5\pi(A_0 + A_1 - A_2/2)$$

The moment about the 1/4 chord point will thus be,

$$C_M(1/4c) = -\pi/4(A_1 - A_2)$$

From this it follows that the center of pressure is aft of the 'quarter-chord' point 0.25 c, by

$$\Delta x/c = \pi/4((A_1 - A_2)/C_L)$$

The aerodynamic center, AC, is at the quarter-chord point. The AC is where the pitching moment M' does

not *vary* with angle of attack, i.e.,

$$\frac{\partial(C_{M'})}{\partial(C_L)} = 0$$

Coefficient of Drag and Coefficient of Lift:

The drag equation,

Fd = $\frac{1}{2}$ rn^2 Cd

so co efficient of drag is given by the, Cd= 2Fd / rn^2A

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 *cd* 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,

$$L = \frac{1}{2} rn\^2 \text{ A CL}$$

so coefficient of lift is given by the,

$$CL = 2L/rn\^2S = L/qs$$

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:

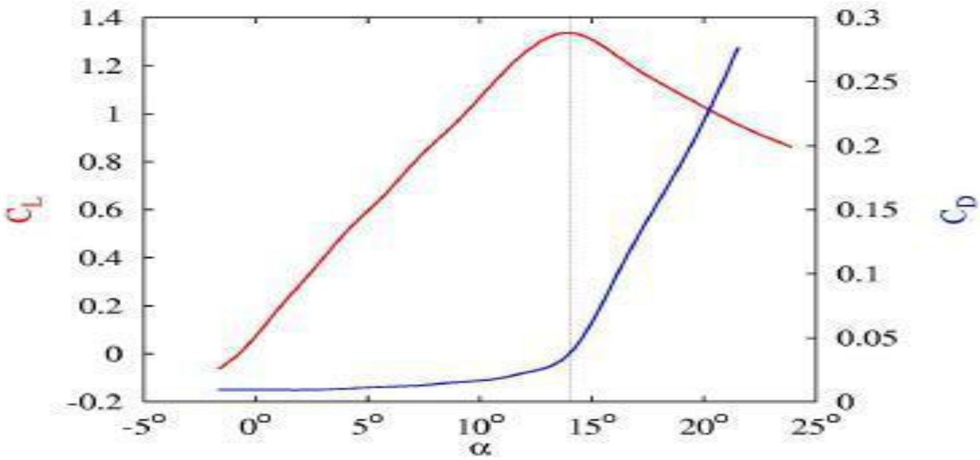

Figure 5: Relationship of Different Coefficients

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 a 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.

The process of airfoil design proceeds from a knowledge of the boundary layer properties and the relation between geometry and pressure distribution. The goal of an airfoil design varies. Some airfoils are designed to produce low drag (and may not be required to generate lift at all.) Some sections may need to produce low drag while producing a given amount of lift. In some cases, the drag doesn't really matter - it is maximum lift that is important. The section may be required to achieve this performance with a constraint on thickness, or pitching moment, or off-design performance, or other unusual constraints. Some of these are discussed further in the section on previous section of historical examples. One approach to airfoil design is to use an airfoil that was already designed by someone who knew what he or she was doing. This "design by authority" works well when the goals of a particular design problem happen to coincide with the goals of the original airfoil design. This is rarely the case, although sometimes existing airfoils are good enough. In these cases, airfoils may be chosen from catalogs such as Abbott and von Doenhoff's Theory of Wing Sections, Althaus' and Wortmann's Stuttgarter Profilkatalog, Althaus' Low Reynolds Number Airfoil catalog, or Selig's "Airfoils at Low Speeds". The advantage to this approach is that there is test data available. No surprises, such as a unexpected early stall, are likely. On the other hand, available tools are now sufficiently refined that one can be reasonably sure that the predicted

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performance can be achieved. The use of "designer airfoils" specifically tailored to the needs of a given project is now very common. This section of the notes deals with the process of custom airfoil design. Methods for airfoil design can be classified into two categories: direct and inverse design.

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. 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 is no pressure different Created between two faces of aerofoil at zero degree of an angle of attack.

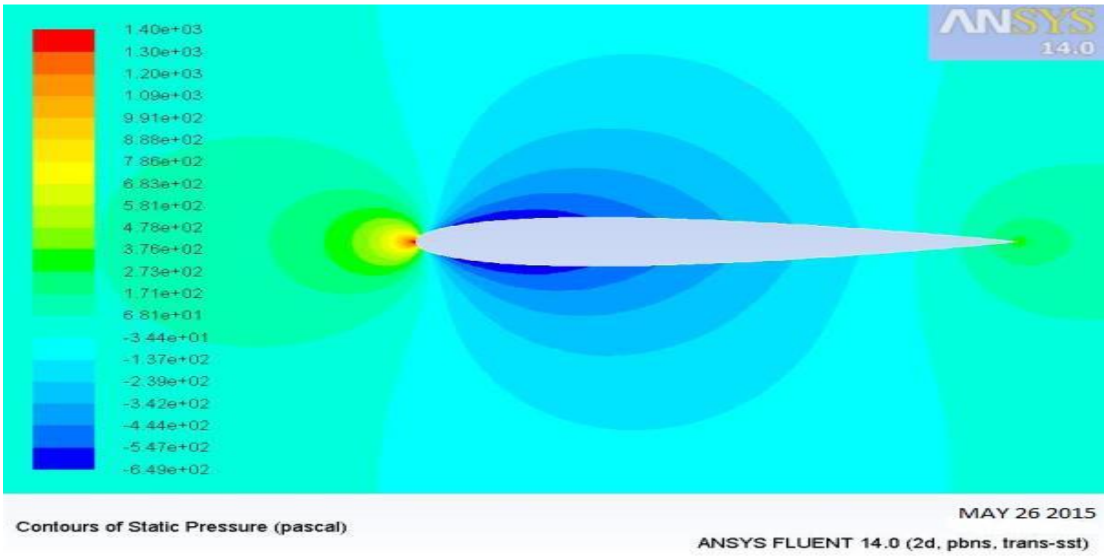

Figure 6: Contours of static pressure over NACA 0012 airfoil at 0 degree of AOA

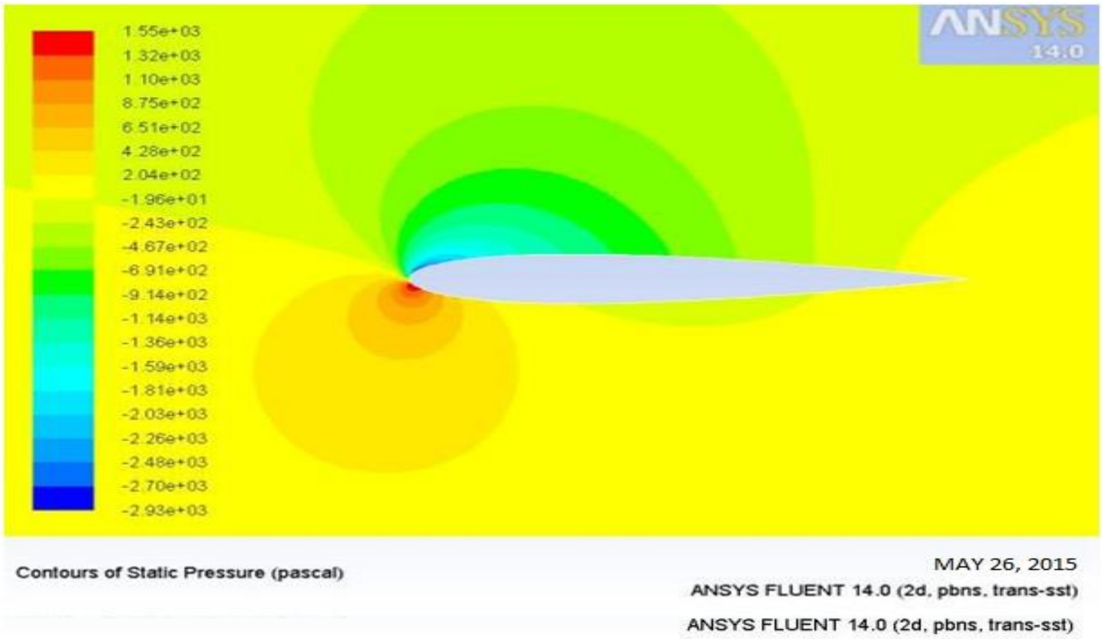

Figure 7: Contours of static pressure over NACA 0012 airfoil at 6 degree of AOA

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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 NACA0012 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

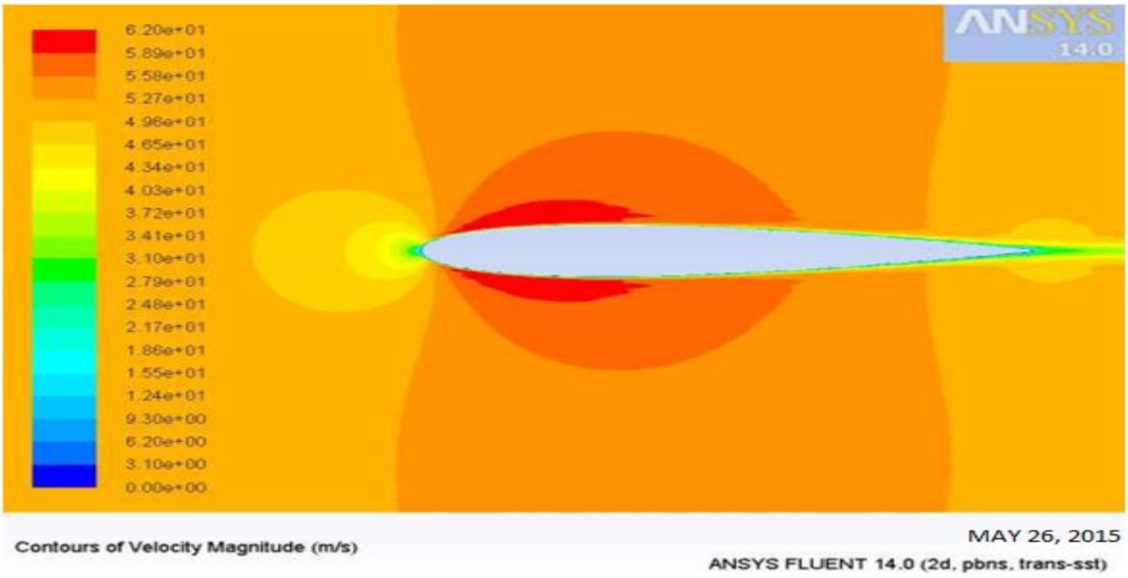

Figure 8: Contours of velocity magnitude over NACA 0012 airfoil at 0 degree of AOA

4. RESULTANT VALUES AND GRAPHS

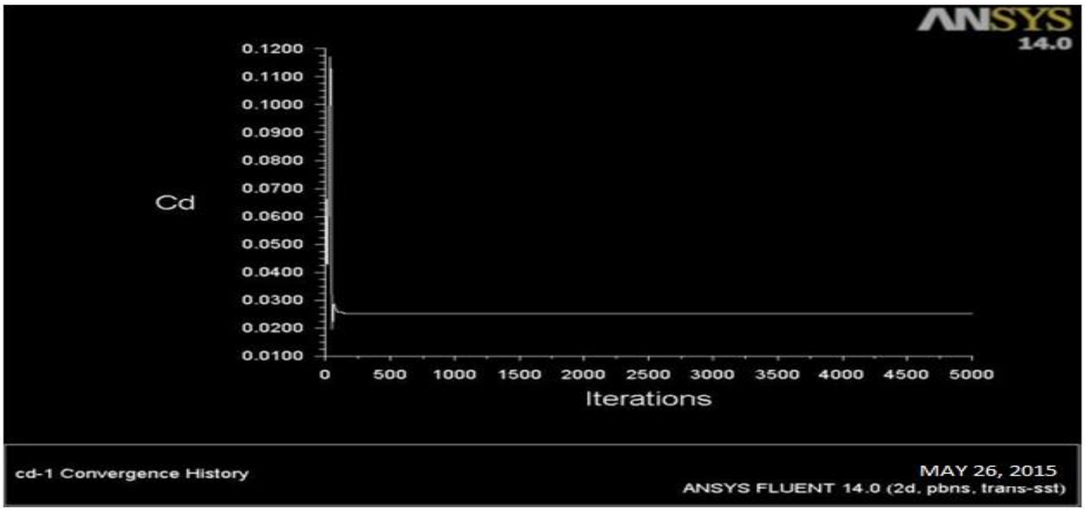

Figure 9: Graph of coefficient of drag at zero degree AOA

ISSN 2393-8471

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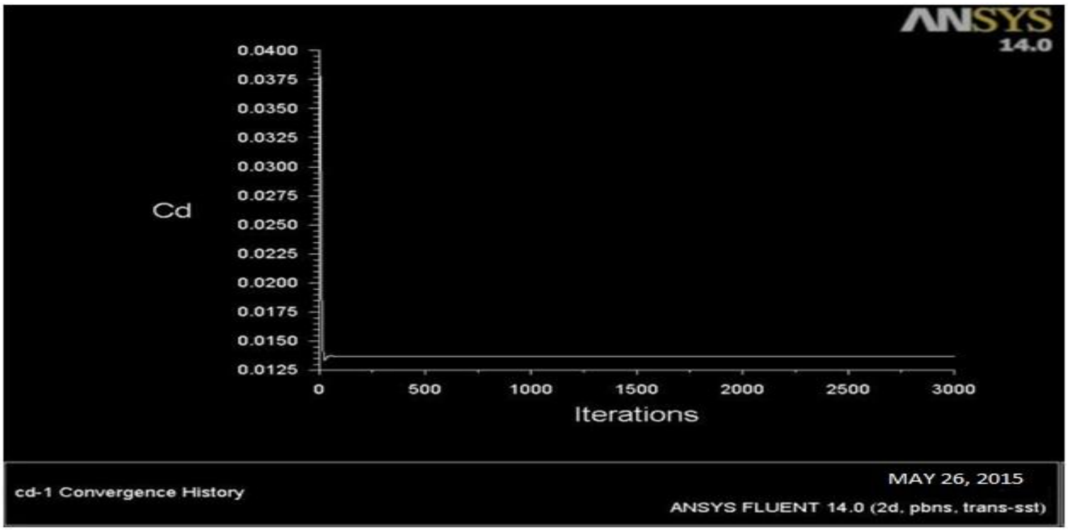

Figure 10: Graph of coefficient of drag at zero degree AOA

5. 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. Here we understand the all about the airfoil and from have some analysis we got some suitable result.

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