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Flow Visualization on Naca2421 Airfoil with and without Piezo Electric Effects

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Abstract
In this study the flow over NACA 2421 Aerofoil has been experimentally studied using smoke visualization technique. The flow has been visualized with and without the piezo electric material at various angles of attack ranging from -15 to 15 degrees. The results have been encouraging since the point of detachment has been delayed significantly with the effect of piezo electric technique. It is suggested that this promising technique be exploited further using other types of airfoils including supercritical.

1.1. Background
Typical airspeeds at high angles of attack for micro air vehicles in the transitional Reynolds number (Re) regime, where flow separation appears on the suction surface of the wings. It is known that delaying and eliminating or controlling such separationare important aircraft design issues [1, 2, 3] but measuring and predicting them are challenging problems. Suppression of boundary layer [4] in experiments will paradoxically require finer measurement of sensitive and smaller scale flows. But the suppressed flow eliminates the flow separation and the lift increases and the drag reduces.

1.2. Aim
Natural laminar flow techniques push the boundary layer transition aft by re-shaping the aerofoil or fuselage so that its thickest point is more aft and less thick. This reduces the velocities in the leading part and the same Reynolds number is achieved with a greater length. At lower Reynolds numbers, such as those seen with model aircraft, it is relatively easy to maintain laminar flow. This gives low skin friction, which is desirable. However, the same velocity profile which gives the laminar boundary layer has low skin frictional so causes it to be severely affected by adverse pressure gradients.
As the pressure begins to recover over the rear part of the wing chord, a laminar boundary layer will tend to separate from the surface. Such flow separation causes a large increase in the pressure drag. In these cases, it can be advantageous to deliberately trip the boundary layer into turbulence at a point prior to the location of laminar separation.
The fuller velocity profile of the turbulent boundary layer allows it to sustain the adverse pressure gradient without separating. Thus, although the skin friction is increased, overall drag is decreased. This is the principle behind the dimpling on golf balls, as well as vortex generators onaircraft. Special wing sections have also been designed which tail or the pressure recovery so laminar separation is reduced or even eliminated issues [5, 6, 7]. This represents an optimum compromise between the pressure drag from flow separation and skin friction from induced turbulence.

1.3. Objectives
To carry out experimental work on NACA 2421 unsymmetrical airfoil section to increase the lift and reduce the drag by studying the suppression using piezo electric method. The thickness of the velocityboundary layer is normally defined as the distance from the solid body at which the viscous flow velocity is 99% of the free stream velocity. An alternative definition, the displacement thickness, recognizes that the boundary layer represents a deficit in massflow compared to in viscid flow with slip at the wall. It is the distance by which the wall needs tobe displaced in the inviscidflow to give the same total mass flow as the viscous case.

1.4. Pressure Measurement
Pressure = Force/Area
The total pressure (P_O) of a flow is measured by the Pitot tube. The difference between total pressure and static pressure (P_O-P) i.e velocity of the flow is measured by the Pitot static tube.
$V = \sqrt{2}(P_O-P_S)/\rho$
$P_O-P_S = 1/2\ \rho V^2$
$C_p= P_i-Ps/ P_o-P_s$
C_p= Coefficient of pressure
P_i = Model surface pressure
P_s = Static pressure
P_o = Total pressure

1.5. Boundary Layer
When the air entraps in any material the flow at the surface adheres to the surface because of friction between the gas and the solid material, that is, right at the surface, the velocity is zero, and there is a thin region of retarded flow in the vicinity of the surface. This region of viscous flow which has been retarded owing to friction at the surface is called boundary layer.

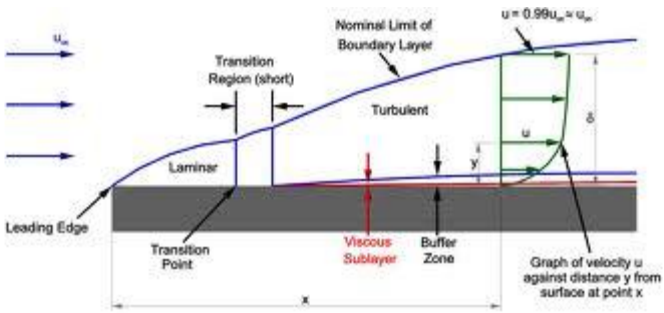

Figure.1. Formation of Boundary Layer over a flat plate

The boundary layer thickness grows as the flow moves over the body, that is, more of the flow is affected by friction as the distance along the surface increases.We define the thickness of this boundary layer as the distance from the wall to the point where the velocity is 99% of the "free stream" velocity.

DISPLACEMENT THICHNESS (δ^*)

The distance measured perpendicular to the boundary by which the main stream is displaced to an account of formation of boundary layer.

$$\delta^* = \int_0^\delta \left(1 - \left(\frac{u}{u_0}\right)\right) dy$$

u - Local velocity
u_0 - Free stream velocity

MOMENTUM THICKNESS (θ)

The distance measured by which the boundary should be displaced to compensate for reduction in momentum of the flowing fluid on the account of boundary layer formation.

$$\theta = \int_0^\delta \left(\frac{u}{u_0}\right)\left(1 - \left(\frac{u}{u_0}\right)\right) dy$$

1.6. AEROFOIL

Normal flow over aerofoil (a wing cross-section) is shown in the figure below with the boundary layers greatly exaggerated.

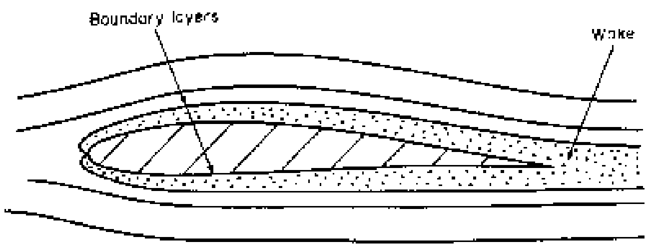

Figure.2. Stream Lines Around Airfoil Section

The velocity increases as air it flows over the wing. The pressure distribution is similar to that shown below so transverse lift force occurs.

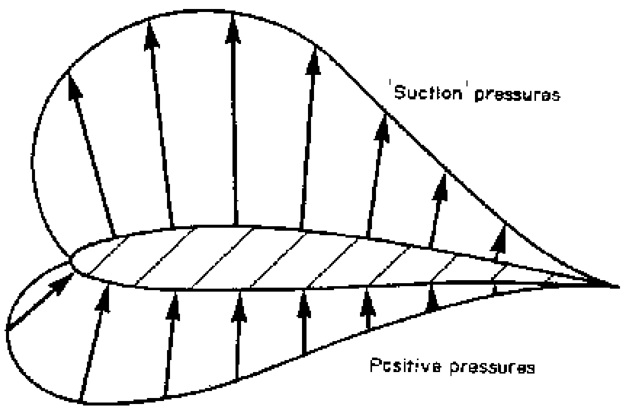

Figure.3. Pressure Distribution

If the angle of the wing becomes too great and boundary layer separation occurs on the top of the aerofoil the pressure pattern will change dramatically. This phenomenon is known as **stalling**.

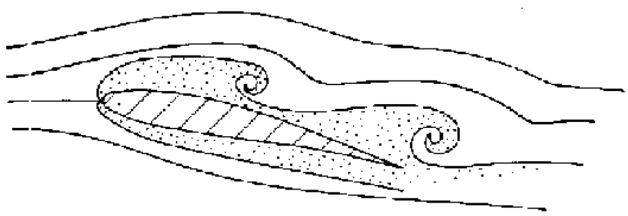

Figure.4.Airfoil Section at High angle of attack

When stalling occurs, all, or most, of the 'suction' pressure is lost, and the plane will suddenly drop from the sky! The only solution to this is to put the plane into a dive to regain the boundary layer. A transverse lift force is then exerted on the wing which gives the pilot some control and allows the plane to be pulled out of the dive.

Fortunately there are some mechanisms for preventing stalling. They all rely on preventing the boundary layer from separating in the first place.

1. Arranging the engine intakes so that they draw slow air from the boundary layer at the rear of the wing though small holes help to keep the boundary layer close to the wing. Greater pressure gradients can be maintained before separation take place.
2. Slower moving air on the upper surface can be increased in speed by bringing air from the high pressure area on the bottom of the wing through slots. Pressure will decrease on the top so the adverse pressure gradient which would cause the boundary layer separation reduces.

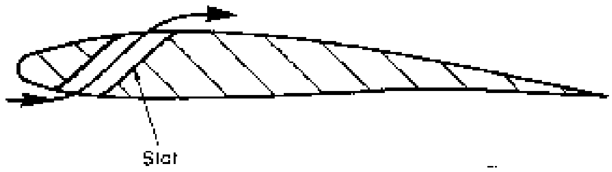

Figure.5. Slot in the Airfoil

3. Putting a flap on the end of the wing and tilting it before separation occurs increases the velocity over the top of the wing, again reducing the pressure and chance of separation occurring.

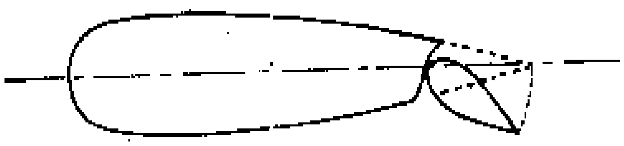

Figure.6. End Slot

The boundary layer is the portion of fluid adjacent to the surface of an object around which the fluid is flowing. The layer is the boundary between the object and the free-flowing fluid. Due to its contact or proximity to the object, the boundary layer is affected by the object and displays flow properties that are different from those of fluid flowing farther away from the object. The boundary area is that of viscous flow, which is subject to friction from the surface of the object and heat transfer from the object, and thus is ruled by its own set of equations derived from the Navier-Stokes equations.
The equations of fluid flow by dividing the flow field into two areas: one inside the boundary layer, dominated by viscosity and creating the majority of drag experienced by the boundary body; and one outside the boundary layer, where viscosity can be neglected without significant effects on the solution. This allows a closed-form solutionfor the flow in both areas, a significant simplification of the full Navier–Stokes equations.
The majority of the heat transfer to and from a body also takes place within the boundary layer, again allowing the equations to be simplified in the flow field outside the boundary layer. The pressure distribution throughout the boundary layer in the direction normal to the surface (such as an airfoil) remains constant throughout the boundary layer, and is the same as on the surface itself.
The thickness of the velocity boundary layer is normally defined as the distance from the solid body at which the viscous flow velocity is 99% of the freestream velocity (the surface velocity of an inviscid flow). An alternative definition, the displacement thickness, recognizes that the boundary layer represents a deficit in mass flow compared to inviscid flow with slip at the wall. It is the distance by which the wall would have to be displaced in the inviscid case to give the same total mass flow as the viscous case. The no-slip condition requires the flow velocity at the surface of a solid object be zero and the fluid temperature be equal to the temperature of the surface.
The flow velocity will then increase rapidly within the boundary layer, governed by the boundary layer equations, below.In high-performance designs, such as gliders and commercial aircraft, much attention is paid to controlling the behavior of the boundary layer to minimize drag. Two effects have to be considered. First, the boundary layer adds to the effective thickness of the body, through the displacement thickness, hence increasing the pressure drag. Secondly, the shear forces at the surface of the wing create skin friction drag.
At high Reynolds numbers, typical of full-sized aircraft, it is desirable to have a laminar boundary layer. This results in a lower skin friction due to the characteristic velocity profile of laminar flow. However, the boundary layer inevitably thickens and becomes less stable as the flow develops along the body, and eventually becomes turbulent, the process known as boundary layer transition.
One way of dealing with this problem is to suck the boundary layer away through a porous surface (see Boundary layer suction). This can reduce drag, but is usually impractical due to its mechanical complexity and the power required to move the air and dispose of it. Natural laminar flow techniques push the boundary layer transition aft by reshaping the aerofoil or fuselage so that its thickest point is more aft and less thick. This reduces the velocities in the leading part and the same Reynolds number is achieved with a greater length.
At lower Reynolds numbers, such as those seen with model aircraft, it is relatively easy to maintain laminar flow. This gives low skin friction, which is desirable. However, the same velocity profile which gives the laminar boundary layer its low skin friction also causes it to be badly affected by adverse pressure gradients. As the pressure begins to recover over the rear part of the wing chord, a laminar boundary layer will tend to separate from the surface. Such flow separation causes a large increase in the pressure drag, since it greatly increases the effective size of the wing section. In these cases, it can be advantageous to deliberately trip the boundary layer into turbulence at a point prior to the location of laminar separation, using a turbulator.
The fuller velocity profile of the turbulent boundary layer allows it to sustain the adverse pressure gradient without separating. Thus, although the skin friction is increased, overall drag is decreased. This is the principle behind the dimpling on golf balls, as well as vortex generators on aircraft. Special wing sections have also been designed which tailor the pressure recovery so laminar separation is reduced or even eliminated. This represents an optimum compromise between the pressure drag from flow separation and skin friction from induced turbulence.
When using half-models in wind tunnels, a peniche is sometimes used to reduce or eliminate the effect of the boundary layer.

2. Unsymmetrical Airfoil

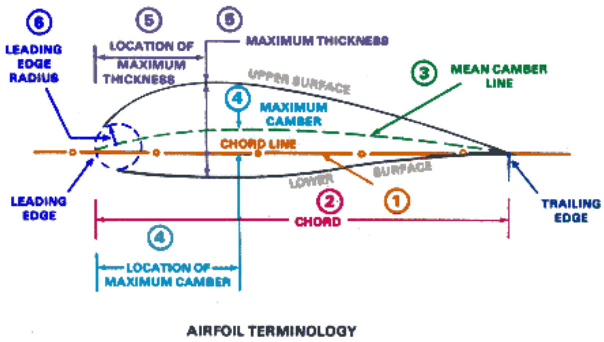

Figure.7. Unsymmetrical Airfoil

The unsymmetrical airfoil has a higher coefficient of lift than the symmetrical airfoil. On unsymmetrical airfoils, the top edge is shaped differently than the bottom edge, which changes the way air flows over it. This causes the air to move faster, which creates more lift. Some airfoils are curved differently on the top side than on the bottom. Those airfoils are unsymmetrical, because their two sides are differently shaped. Unsymmetrical airfoils are the most commonly used type for fixed-wing aircraft, but because the location of the center of pressure changes as the angle of attack changes, they are seldom used for rotary-wing aircraft.

2.1. NACA Series

1. First digit describing maximum camber as percentage of the chord.
2. Second digit describing the distance of maximum camber from the airfoil leading edge in tens of percents of the chord.
3. Last two digits describing maximum thickness of the airfoil as a percentage of the chord.

3. Flow Visualization in Wind Tunnel

3.1. Introduction

Flow visualization is an experimental means of examining the flow patterns around a body or over its surface. The flow is “visualized” by introducing dye, smoke or pigment into the flow in the area under investigation issues (10, 11, 12). The primary advantage of such a method is its ability to provide a description of the flow over a model without a complicated data reduction and analysis.

It is difficult to exaggerate the value of flow visualization. The ability to see flow pattern on a model often gives insight into a solution to an aerodynamic problem. Flow visualization can be divided into two broad categories the first is surface flow visualization when the visualization media is applied to the surface such as tufts and oil flow etc. The second type is off surface such as smoke and streams. There are basically four methods of recording the flow visualization test.

The first and the best but the least permanent method is for the scientist and the engineer to observe with his eyes. Because of the depth perception one can see a three dimensional picture. The other three common methods of recording the result of flow visualization are by film, either still or movie or television camera or video and magnetic tapes.

It must be realized that all of these methods are using a two dimensional medium to often record a three dimensional phenomenon. This is especially fine when using a smoke or helium bubbles to trace flow stream lines pass the model. All these three methods can be used in either black and white or colour. The photography methods while recording takes more time for developing and printing stills, when compared to video, but yield higher resolution.

3.2. Flow Visualization Techniques

The present section discusses flow visualization using smoke wire technique as it is important tool to understand the nature of flow field. Its proper utilization will provide reasonable information that will help in influencing the flow characteristics. This will eventually lead to better design modification leading to development of favorable pressures.

The visualization of the flow past the building models was made possible by using the above technique has developed and subsequently improved and used by Batill and Mueller issues (4, 5), and was further improved by Stahl and Mahmood issues (15, 16). The procedure involves brushing a thin wire, which then forms a large number of small droplets.

Heating the wire evaporates the oil with each droplet providing a fines treak-line in the flow. Under proper illumination the smoke appear bright which can be observed and photographed This technique basically requires a fine wire of about 0.1mm in diameter of Nickel Chromium steel and suitable oil, which can vaporize quickly, and a DC current to heat the wire.

This technique was utilized, as it is very useful in studying separation especially in smooth and nominal boundary turbulent flows. The smoke-wire was placed at the upstream side of the building model to see the separating bubble from frontal edge and ground vortices at normal incidence, for both sharp and round edge model. For oblique incidences of and the smoke-wire was placed ata suitable position to note the separation of shear layer in the immediate vicinity of the leading edge corner where high suction pressures are developing.

This technique provided a clear influence of rounding of roof edges on the shear layers that formed conical vortices. The influence due to change in the angle of incidence on the separation process for sharp and round edge models was found to be of interest. The smoke-wire could also be moved around the model to get useful information about the flow field.

3. Wind Tunnel

A wind tunnel is a device in which a jet of air or any other suitable gas of uniform properties across the cross section is produced. This is used for aerodynamic testing of the models under the given standard conditions.

Wind tunnels in general consist of driving section, settling chamber, accelerating unit (nozzle section), test section, diffuser and fan.

4.1 Specifications of the Wind Tunnel Used in this Investigation (Figure 8):

Type of tunnel	: Subsonic Suction Type
Testsection	: 600mm×600mm×2000m (Length)
Air speed	: Upto 50m/s
Contraction ratio	: 9:1
Overall size	: 1.2m× 2.2m × 7.0m
Power required	: AC 3 phase, 415 volts, 40A
Drive	: Axial flow fan driven variable speed DC Motor with Thyristor Controller.

Figure.8.Wind Tunnel

4.2. Surface Flow Visualization Technique

The study of the flow near the surface of the square plate model was carried out by using the chalk powder suspended in Kerosene oil and sprayed on the model. Figure 9 shows the setup for surface flow visualization issues (7, 8, 9). When the flow passes the model the chalk and Kerosene mixture settles as per flow lines issues (13, 14).

Figure.9. Surface Flow Visualization Technique

Figure.10.Unsymmetrical Airfoil

Figure 10 shows the present model used for experiments. Figures 11 and 12 show the flow visualization at angle of attack of 0 degree with and without piezo electric effect.

Figure.11.Angle of Attack at 0 Degree Without Piezo Electric Effect

Figure.12.Angle of Attack at 0 Degree With Piezo Electric Effect

Figures 13 and 14 showflow visualization at angle of attack of 5 degree with and without piezo electric effect. Till about quarter chord line the flow is attached, after that the flow becomes detached, (i.e) after $1/4^{th}$ of the chord line the laminar flow is transited to the turbulent flow in the without piezo electric case.

Figure.13.Angle of Attack at 5 Degree Without Piezo Electric Effect

Figure.14.Angle of Attack at 5 Degree With Piezo Electric Effect

Figures 15 and 16 showthe flow visualization with the angle of attack of -5 degree with and without piezo electric effect.

Figure.15.Angle of Attack at -5 Degree Without Piezo Electric Effect

From these figures, it can be seen that the flow with piezo electric remains attached till about 50% of the chord, after that the flow is detached.

Figure.16.Angle of Attack at -5 Degree With Piezo Electric Effect

Figures 17 and 18 show that the effect of flow visualization with the angle of attack of 10 degree with and without piezo electric effect.

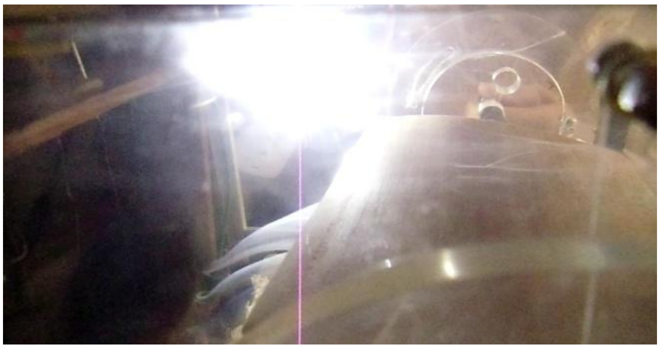

Figure.17.Angle of Attack at 10 Degree Without Piezo Electric Effect

Figure.18.Angle of Attack at 10 Degree With Piezo Electric Effect

Figures 19 and 20 show the effect of flow visualization with the angle of attack of -10 degree with and without piezo electric effect.

Figure.19.Angle of Attack at -10 Degree Without Piezo Electric Effect

Figure.20.Angle of Attack at -10 Degree With Piezo Electric Effect

Figures 21 and 22 show the effect of flow visualization with the angle of attack of 15 degree with and without piezo electric effect.

Figure.21.Angle of Attack at 15 Degree Without Piezo Electric Effect

Figure.22.Angle of Attack at 15 DegreeWithPiezo Electric Effect

Figures 23 and 24 show that the effect of flow visualization with the angle of attack of -15 degree with and without piezo electric effect. Here, the effect of piezo electric is not noticeable as the airfoil is already stalled.

Figure.23.Angle of Attack at -15 Degree Without Piezo Electric Effect

Figure.24.Angle of Attack at -15 Degree With Piezo Electric Effect

5. Advantages with Piezo

High lift is generated with low angle of attack.
Less fuel is consumed.
Performance of the aircraft is increased.
High thrust is generated.
Easy to suppress the flow.

6. Conclusion

The flow visualization with and without piezo electric effects were visualized. It is evident from the results that the point of detachment was delayed from near quarter chord location to the half chord location due to piezo electric effect. Thus this technique is very useful in improving the aerodynamic performance of the airfoil and is very promising.

7. Future Enhancements

In future, analysis of various NACA series airfoils will be used to increase the performance of the aircraft. Also supercritical airfoils will be analysed to increase the aerodynamic performance. The supersonic wind tunnels can be used to test the airfoil with piezoelectric strip in various locations with respect to its flow separation. Various types of smoke generation methods can be used to increase the density of the smoke. Advanced high grade piezo electric materials are to be used to suppress the flow.

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