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PASSIVE FLOW SEPARATION CONTROL OF AN AIRFOIL BY CONTINUOUS SUCTION AND INJECTION

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ABSTRACT

From a fluid dynamist's point of view, the performance of an aircraft is essentially controlled by the development of the boundary layer on its surface and its interaction with the mean flow. This interaction decides the pressure distribution on the airfoil surface, and subsequently the aerodynamic loads on the wing In this research an experimental study has been accomplished to determine the effects of suction and injection in the aerodynamic characteristics of a specific airfoil NACA 4220. The purpose of this research is to develop a flow separation control mechanism that could remove retarded air by suction and energize the tired air by injection using single cylinder buster to increase the lift force of airfoil. It is concluded that the suction and injection can significantly increase the lift coefficient and decrease the skin friction. The design mechanism shows that uniform and more powerful fluid flow could be generated along the slot of the airfoil The device is an excellent candidate to control flow separation, where the required frequency is changing with aircraft speed and angle of attack.. In the specific case studied here, flow separation occurs at 8° angle of attack in regular surface but for controlled surface with suction and injection flow separation occurs after 13^{0} angle of attack. This study can be a benchmark for the future numerical and experimental studies. Significant improvement was obtained in the lift coefficient for moderate to high angles of attack. But the effect decreases with the further increase of angle of attack, possibly due to less effective interaction between the disturbance and the shear layer.

Keywords: Flow Separation, Suction and Injection, Airfoil, Single Cylinder Buster

1. INTRODUCTION

All solid objects traveling through a fluid (or alternately a stationary object exposed to a moving fluid) acquire a boundary layer of fluid around them where viscous forces occur in the layer of fluid close to the solid surface. Boundary layers can be either laminar or turbulent. A reasonable assessment of whether the boundary layer will be laminar or turbulent can be made by calculating the Reynolds number of the local flow conditions. Flow separation occurs when the boundary layer travels far enough against an adverse pressure gradient that the speed of the boundary layer falls almost to zero. The fluid flow becomes detached from the surface of the object, and instead takes the forms of eddies and vortices. In aerodynamics, flow separation can often result in increased drag, particularly pressure drag which is caused by the pressure differential between the front and rear surfaces of the objects as it travels to the fluid. For this region mass effort and research has gone into the design of aerodynamic and hydrodynamic surfaces which delay flow separation and keep the local flow attached for as long as possible. When a real fluid flows past a solid boundary a layer of fluid which comes in contact with the boundary surface adheres to it on account of viscosity. Since this layer of fluid cannot slip away from the boundary surface it attains the same velocity as that of the boundary. In other words at the boundary surface there is no relative motion between the fluid and the boundary. If the boundary is stationary, the fluid velocity at the boundary surface will be zero. The boundary layer thickness is considerable affected by the pressure gradient in the direction of flow. If the pressure gradient is zero, then the boundary layer continues to grow in thickness along a flat plate. With negative pressure gradient, the boundary layer thickens rapidly. The adverse pressure gradient plus the boundary shear decreases the momentum in the boundary layer, and if they both act over a sufficient distance they cause the fluid in the boundary layer to come to rest. In this position the flow separation is started. Also when the velocity gradient reaches to zero then the flow becomes to separate. So when the momentum of the layers near the surface is reduced to zero by the combined action of pressure and viscous forces then separation occur. So boundary layer separates under adverse pressure gradients as well as zero velocity gradients. Fluid flow separation can be controlled by various ways such as motion of the solid wall, slit suction, tangential blowing and suction, continuous suction and blowing by external disturbance, providing bumpy the surface, using oscillating camber such as piezoelectric actuator etc. The other techniques are the using surface injection in a multistage compressor, steady and pulsed jets, oscillatory fluid injection, dielectric barrier discharge plasma actuator, synthesis jet, vortex generator smart acoustically active. Among them suction and injection was used in the present work to control flow the separation.

Suction is the flow of a fluid into a partial vacuum or region of low pressure. The pressure gradient between this region and the ambient pressure will propel matter toward the low pressure area. Suction is popularly thought of as an attractive effect, which is incorrect since vacuums do not innately attract matter. Dust being "sucked" into a vacuum cleaner is actually being pushed in by high pressure air on the outside of the cleaner. The higher pressure of the surrounding fluid can push matter into a vacuum but a vacuum cannot attract matter.

The effect of suction consists of decelerated fluid particles from the boundary layer before they are given a change to cause separation. A new boundary layer which is again capable of overcoming a certain adverse pressure gradient is allowed to form in the region behind the slit. With a suitable arrangement of the slits and under favorable conditions separation can be prevented completely. Simultaneously, the amount of pressure drag is greatly reduced owing to the absence separation. By applying suction, considerably greater pressure increases on the upper side of the airfoil are obtained at large angles of incidence, and consequently much larger maximum lift values. In more recent times, suction was applied to reduce drag. By the use of suitable arrangement of suction slits, it is possible to shift the point of transition in the boundary layer in the downstream direction; this causes the drag co-efficient to decrease, because laminar drag is substantially smaller than turbulent drag. The effect of the delay in transition caused by suction is to reduce the boundary layer thickness which then becomes less prone to turning turbulent. The velocity profile in a boundary layer with suction, being fuller, have forms which are less likely to induce turbulence compared with those in laminar boundary layers without suction and of equal thickness.

A further way of preventing separation consists of supplying energy to the particles in the fluid which are low in energy in the boundary layer .This can be achieved by tangentially blowing higher velocity fluid outside to inside of the body. The danger of separation is removed by the supply of kinetic energy to the boundary layer. The effectiveness of wing flaps can be greatly improved if fluid is tangentially blown out just in front of the flap. If the intensity of the blown jet is high enough, even the lift predicted by potential theory can be surpassed .The so-called jet flap effect then causes super circulation. Thus flow separation can be controlled by tangential blowing or injection.

Separation of flow can be controlled by continuous blowing or injection. If the wall is permeable and can therefore let the fluid through, the boundary layer can be controlled by this process. In this kind of blowing the wall shear stress and therefore the friction drag can be reduced by blowing or injection. The most important application of blowing is in so called transpiration cooling. If a different fluid is injected, a binary boundary layer occurs. As well as velocity and temperature fields, the boundary layer also has a concentration field. The stability of the boundary layer and the transition to turbulence are also considerably influenced by continuous blowing or injection.

Flow separation is mostly an undesirable phenomenon and boundary layer control is an important technique for flow separation problems on airfoils and in diffusers. An improvement of the efficiency of diffusers can be obtained by the application of boundary layer suction to the diverging passage. Flow separation control providing significant increase in the lift coefficient. For this case lift should be high and drag should be low, which increases efficiency. Separation of fluid affects the temperature dependent physical properties, cooling and heating effect. So it is necessary to control flow separation. Flow separation control maximizes energy conservation, operational efficiency, and safety and required to proper heat distribution.

2. METHODOLOGY

To control the flow, passive or active devices are used. Passive control devices are those, which are not energy consumptive. They mainly affect the flow by the geometry of the airfoil. In contrast, active control devices use energy such as surface suction or injection. The boundary layer suction is to prevent separation of either of laminar or turbulent layers. The suction removes the retarded air close to the surface, it will remove the cause of separation, and this aspect leads to its use to obtain high lift coefficients from various airfoil configurations. The suction of air from the boundary layer flow into the surface of the body, causes the tired air near the surface being removed and a new boundary layer is started to reform downstream of the suction point with a consequent reduction in drag (Schertz, J.A., 1984). Generally, in surface injection, a secondary fluid is injected through miniature openings or slots on the surface. In this separation control, which is due to the complete loss of energy of the air flowing immediately adjacent to the surface, is to energize this tired air by means of blowing a thin, high speed jet into it and improve efficiency that means reduce total consumption of energy by increase lift with decreasing drag force.

In order to study the effect of suction and injection, four inclined internal slots and holes are created in the airfoil. Fluid is sucked from the leading edge slot by a single cylinder piston mechanism through a circular pipe from main flow by suction. The low energy fluid in the boundary layer is removed by suction before it can separate. For injection the sucked air in suction process is used. For this fluid is injected at the trailing edge slot by the single cylinder piston mechanism through a circular pipe to main flow by injection. The low energy fluid in the trailing is energized by injection. So the sucked air is directly used to the separation point of the trailing edge and create a new boundary layer which control the flow separation.

The experiment is conducted with a model wing constructed with a base profile of a NACA 4220. Each model has a recess cut in the upper surface, into which a sub-sonic flow separation control mechanism that could generate suction and injection fluid flow over an airfoil. The data of pressure difference was taken by the digital manometer. The pressure co-efficient is measured by following equation-

Co-efficient of Pressure, $$C_p = \frac{P - P_\infty}{q_\infty}$$

$$C_p = \frac{P - P_\infty}{\frac{1}{2}\rho_\infty U_\infty^2}$$

From above equation, the value of Cp is found and Lift and drag co-efficient are calculated by integrating the pressures over of the wing.

Co- efficient of Lift, $$C_L = \frac{1}{c}\int_0^c (Cp_l - Cp_u)dx$$

Co- efficient of Drag, $$C_D = \frac{1}{c}\int_0^c (Cp_l - Cp_u)dy$$

In the case with suction and injection seven different angles of attack, 0, 4, 8, 10, 12, 15, and 20 are taken into consideration and three different fluid flow velocities are 2.98 m/s, 3.48 m/s, 4.47m/s.

3. MODEL CONSTRUCTION

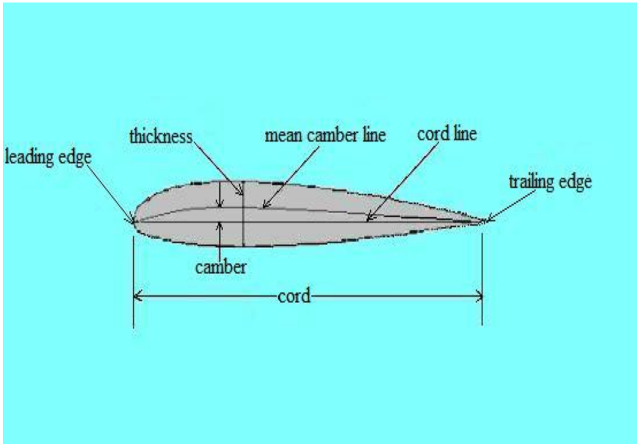

Fig 1. A typical airfoil

The cambered airfoil sections of all NACA families considered herein are obtained by combining a mean line and a thickness distribution. The necessary geometric data and some theoretical aerodynamic data for the mean lines and thickness distributions obtained from the supplementary figures by the methods described for each family of airfoils. The process for combining a mean line and a thickness distribution to obtain the desired cambered airfoil section is shown in figure below.

The cross sectional shape obtained by the intersection of the wing with the perpendicular plane is called an airfoil. The major design feature of an airfoil is the mean cambered line, which is the locus of points halfway between the upper and lower surfaces as measured perpendicular to the mean cambered line itself. The most forward and rearward points of the mean cambered line are the leading and trailing edges respectively. The straight line connecting the leading and trailing edges is the chord line of the airfoil and the precise distance from the leading to the trailing edge measured along the chord line is simply designated the chord of the airfoil, given by the symbol C. The camber is the maximum distance between the mean camber line and the chord line, measured perpendicular to the chord line. The camber, the shape of the mean camber line and to a lesser extent, the thickness distribution of the airfoil essentially controls the lift and moment characteristics of the airfoil. If X_u and Y_u represent respectively the abscissa and ordinate of a typical point of the upper surface of the airfoil and y_t is the ordinate of the symmetrical thickness distribution at chord wise position X_1, the upper surface coordinates are given by the flowing relations:

$$X_u = x - y_t \sin\theta$$
$$Y_u = Y_c + y_t \cos\theta$$

The corresponding expressions for the lower surface co ordinates are

$$X_1 = x + Y_t \cos\theta$$
$$Y_1 = Y_c - Y_t \cos\theta$$

As is θ very small, $\sin\theta = 0$, $\cos\theta = 1$

$$X_u = x \qquad Y_u = y_c + y_t s$$
$$X_1 = x \qquad Y_1 = y_c - y_t$$

The center for the leading edge radius is found by drawing a line through the end of the chord at the leading edge with the slope equal to the slope of the mean line at that point and laying off a distance from the leading edge along this line equal to the leading edge radius. This method of construction causes the cambered airfoils to project slightly forward of the leading edge point. Because the slope at the leading edge is theoretically infinite for the mean lines having a theoretically finite load at the leading edge, the slope of the radius through the end of the chord

For such mean lines is usually taken as the slope of the mean line at x /c = 0. 005. This procedure is justified by the manner in which the slope increases to the theoretically infinite value as x/c are reached. Large values of the slope are thus limited to values of x/c very close to 0 and may be neglected in practical airfoil design.

We have made software for the profile generation by using some basic equations of airfoil. They are as follows:
From leading edge to maximum wing thickness

$Y_c/c = f/c\,(1/X^2)\,[2x_1\,(X/C) - (X/C)^2]$
Valid for $0 \le X/C \le X_1$

From maximum wing thickness to trailing edge

$Y_c/c = f/c\,\{1/(1-X_1)^2\}[(1-2X_1) + 2X_1\,(x/c) - (x/c)^2]$
Valid for $x_1 \le x/c \le 1$

From maximum wing thickness to trailing edge

$Y_c/c = f/c\,\{1/(1-X_1)^2\}[(1-2X_1) + 2X_1\,(x/c) - (x/c)^2]$
Valid for $x_1 \le x/c \le 1$

Maximum wing thickness

$Y_t = 5t\,(0.2969\sqrt{x} - 0.126\,x - 0.3516x^2 + 0.2843\,x^3 - 0.1015\,x^4)$

Where, f = maximum camber
X_1 = distance from leading edge to maximum wing thickness
C = chord of the airfoil
t = maximum wing thickness
Y_c = vertical distance between mean camber line and chord line.

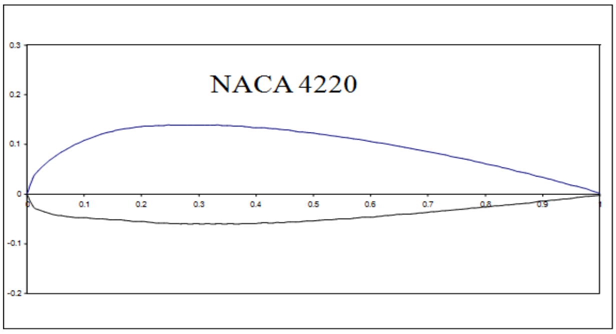

Fig 2. NACA 4220 Airfoil (Designed)

By applying Computer C- Programming Language the regular surface profile of the NACA 4220 model were made. The thickness and chord length of the model are 40 mm and 260 mm respectively. Thus the chord length based Reynolds number relevant at low flight speeds, which are a concern for the exploration of wing formation mechanism, is estimate to be about 10^5. The chord length of the model was determined to have Reynolds number of the same order. The span length of the model, relative to the chord length is one of the important design parameters. Obviously, it should be made as large as possible so that the weight of the model can be reduced. In the present experiment the span length was chosen to be 150 mm, a considerably large value, so as to minimize the effect of the model sides on the wing formation. To ensure the aerodynamic characteristics of an airfoil, it is important that the trailing edge of the model have a sharp edge form.

4. EXPERIMENAL SET-UP AND PROCEDURE

The experiments were conducted using wind tunnel. Figure: 3 shows a schematic of the experimental set up. A small sized model is appropriate to examine the aerodynamic characteristics for the experiments. If we desire to examine the aerodynamic characteristics of a large model, a large scale wind tunnel facility is necessary for testing or the inflatable wing must be drastically scaled down to match the usual wind tunnel size violating the Reynolds number analogy requirements.

Fig 3. Experimental Setup (Actual investigation)

The Experimental study is conducted on two cases in each angle of attack. The two cases are airfoil with surface suction and injection, and the base airfoil. The term base airfoil is referred to the airfoil without suction and injection.

For the complete testing the constructed model, subsonic wind tunnel and pressure measuring instrument were used as required apparatus. At the first step of the experimental procedure the constructed model was placed inside the testing section of the wind tunnel. For different angle of attack pressure was measured at different velocities of wind tunnel. The velocity of the wind tunnel was controlled by a regulator attached with the wind tunnel.

For the pressure measurement a digital manometer was placed outside of the wind tunnel test section. There were drilled holes (2 mm) vertically in every 10 mm distance of the model and vinyl tubes were placed in these holes. The vinyl tubes were connected between the pressure tubes, set to the model and the digital manometer. The values of surface pressure of the model were measured in accordance with various values of wind tunnel velocity and angle of attack.

5. RESULTS AND DISCUSSIONS

Wind tunnel measurements using the constructed airfoil model without suction and injection and suction and injection. The experiment was carried out to observe the change in the co efficient of pressure of the upper surface of the airfoil in different angle of attack (AOA) of suction and injection. Also various graphs have been plotted to examine the measured and calculated data nature.

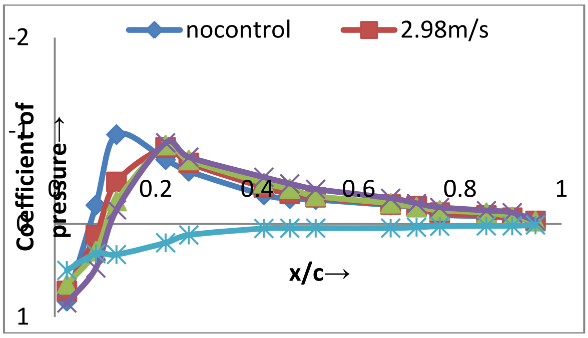

Fig 4. Pressure coefficient along the cord at α = 4 degree

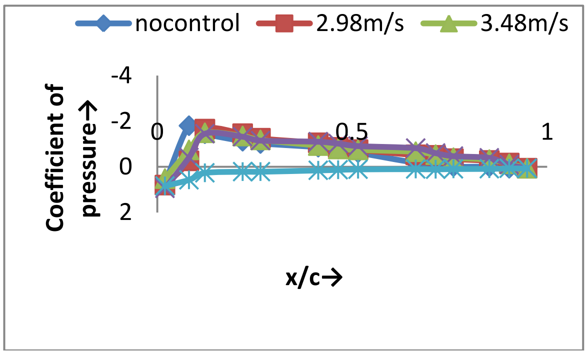

Fig 5. Pressure coefficient along cord at α = 12 degree

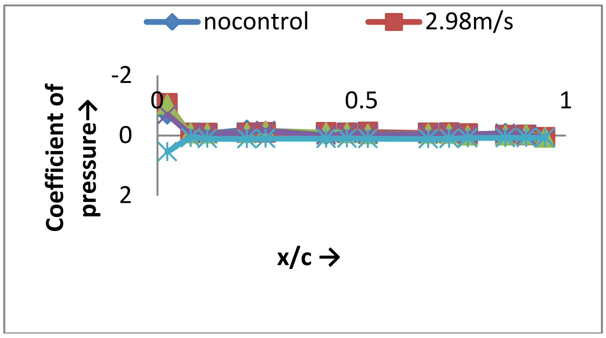

Fig 6. Pressure coefficient along cord at α = 20 degree

Pressure coefficients for the no control condition and test condition are shown in figure 4 to 6. The effect of the pressure co-efficient on the upstream of the slot is considerable. Downstream of the slot, the separation point is not significantly affected but the pressure in the vicinity of separation reduces. Just downstream of separation, there is a relatively sharp pressure drop, followed by a pressure recovery that crosses over the no control line with reattachment occurring further upstream. This results in a curious situation where control appears to promote separation close to the control location while simultaneously shortening the reattachment length. The pressure coefficient distributions around the airfoil NACA 4220 were calculated when the angle of attack changing from 0 degrees to 20 degrees**.** When the angle of attack is relatively small (i.e., < 8 degrees), the pressure near the nose of the NACA 4220 airfoil was found to decrease quickly along the upper surface of the airfoil, and reached its negative pressure coefficient peak rapidly, then, the static pressure was found to recover over the upper surface of the airfoil gradually and smoothly up to the trailing edge of the airfoil, which is a typical behavior of the static pressure distribution over the upper surface of an airfoil without any flow separation.

As the angle of attack increased to 8 ~ 10 degrees, one distinctive characteristic of the pressure coefficient profiles along the upper surface of the airfoil is the region of nearly constant pressure at 0.08 < *x/c* < 0.2. Such region in the pressure coefficient profiles would indicate the separation of the laminar boundary layer from the airfoil upper surface (i.e., flow separation occurred). The sudden increase in static pressure following the "plateau" serves to indicate the rapid transition of the separated laminar shear layer to turbulent flow, which would lead to the reattachment of the separated boundary layer and formation of a laminar separation bubble. The static pressure profile was found to recover gradually and smoothly at downstream region of *x/c* > 0.25 ~ 0.30, which is as the same as those cases with smaller angle of attack and no flow separation. It indicates that the reattachment point, where the separated boundary layer reattach to the airfoil upper surface (i.e., the rear end of the separation bubble) would be located in the neighborhood of *x/c* ≈ 0.25 ~ 0.30.The angle of attack becoming bigger than 15 degrees, the maximum absolute value of the pressure coefficient on airfoil upper surface near the leading edge was found to be only about 1.0, which is much smaller than that with smaller angle of attack (about 4.0).

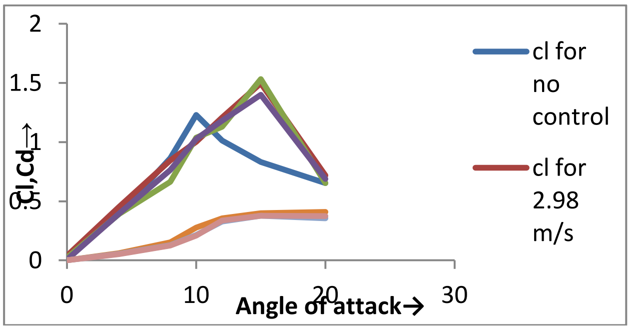

Fig 7. Lift Coefficient $C_{L,}$ Drag Coefficient C_D Vs Angle of attack

Lift coefficients and drag coefficients are shown in the figure 7 for the regular surface and control surface for different velocities and angle of attack. From the figure it is shown that the lift co-efficient increases with increasing angle of attack and after certain time lift co-efficient decreases with increasing angle of attack. The lift co-efficient curve is almost straight line for each frequency up to 5 degree angle of attack. Here flow is fully attached with airfoil surface. After 5 degree angle of attack a slight deflection is created and again increases lift co-efficient rapidly and reach its maximum point at surrounding 15 degree angle of attack. After the maximum point of lift co-efficient at angle of attack of 15 degree, the values of C_L get decreasing. The point from which the value starts to decrease is called stall. After stall point a dead flow region is created and flow is unable to re-attach with wing surface. Here flow is separated from the upper pressure surface.

For regular surface and controlled surface with different velocities the drag coefficients curves are near similar to each other. From the figure 7 the drag coefficient is increased with the increase of angle of attack. After the angle of attack 15 degree, flow is separated from the upper surface and creates a vortex. Consequently static pressure and drag force are increased with the decrease of lift force. Here flow is fully separated from wing surface. It is observed finally from the figure 7 that the curve for standard value of C_L against angle of attack for regular surface is always lower than controlled surface and the curve for standard value of C_D against angle of attack for regular surface is always upper than the controlled surface. So from figure 7 the lift co- efficient for controlled surface is increases with angle of attack higher than the regular surface lift co- efficient and drag co- efficient is decreases with angle of attack than the regular surface drag co-efficient.

6. CONCLUSION

An experimental study has been accomplished to determine the effects of suction and injection in the aerodynamic characteristics of a specific airfoil NACA 4220. It is concluded that the suction and injection can significantly increase the lift coefficient. The design mechanism shows that uniform and more powerful fluid flow could be generated along the slot of the airfoil. The device is an excellent candidate to control flow separation, where the required frequency is changing with aircraft speed and angle of attack. As drag at the turbulent boundary layer is far greater than that at the laminar boundary layer, the basic idea of drag reduction is focused on delaying the occurrence of transition, expanding the range of laminar flow at the object surface, and reduces friction drag at the turbulent boundary layer. In the specific case studied here, flow separation occurs at 8° angle of attack in regular surface but for controlled surface with suction and injection flow separation occurs after 15° angle of attack. This study can be a benchmark for the future numerical and experimental studies.

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