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Boundary Layer Supression Using Bump Surface in Airfoil

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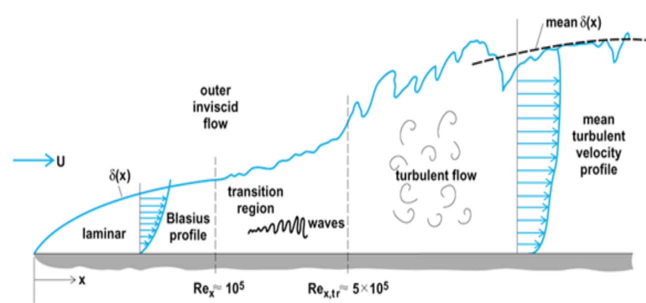
***Abstract*-This project deals with an experimental study of bumps on NACA 4412 airfoil. Results from previous studies shows that aerodynamically changing the profile of the airfoil by placing bumps cause the flow to reattach to the surface of the airfoil at higher angle of attacks and also avoids the flow separation which if present it may reduce the efficient performance of the airfoil. However factors such as height of the bump, width of the bump, distance between two bumps and location of the bump on the airfoil possess distinct performance. The objective of this project work is to increase the lift coefficient by placing three bumps at a distance of 1.2mm from the trailing edge on the upper surface of the airfoil.**

Keywords: **Airfoil, Angle of Attack, Boundary Layer, Reynolds Number, Flow Separation, Bump Surface.**

1. INTRODUCTION

The objective of the investigation reported herein was to increase the lift coefficient for NACA 4412 airfoil at lower Reynolds number. The lift coefficient is increased by controlling the boundary layer. There are several methods for controlling the boundary layer of air on the wing of an aircraft such as motion of the solid wall, slit suction, tangential blowing and suction, continuous suction and blowing by external disturbances, providing bumpy surface, surface roughness etc. Among them here the surface roughness method is used to control flow the flow separation. Due to this manufacturing constraint, the NACA 4412, a relatively thick airfoil, was selected. The objective of this project is to reattach the flow by placing the bumps on the upper surface of the airfoil.This project deals about the effect of bump surface on the aircraft wing and also will give an over view of the results that is obtained for increase in lift.

The concept of boundary layer is crucial to the understanding of the flow around an airfoil. A boundary layer is the layer of fluid in the immediate vicinity of bounding surface was the effects of viscosity are significant. The boundary layer develops up to a certain portion of the plate from the leading edge. Irrespective of the incoming stream flow pattern, the flow in the boundary layer exhibits all the characteristics of laminar flow, which is known as laminar boundary layer. If the chord length of the airfoil is long, the laminar boundary layer becomes unstableand then turbulent boundary layer is formed.



Boundarylayer equations:

The deduction of the boundary layer equations was one of the most important advances in fluid dynamics. Using an order of magnitude analysis, the well-known governing Navier–Stokes equations of viscous fluid flow can be greatly simplified within the boundary layer. Notably, the characteristic of the partial differential equations (PDE) becomes parabolic, rather than the elliptical form of the full Navier–Stokes equations. This greatly simplifies the solution of the equations. By making the boundary layer approximation, the flow is divided into an inviscid portion (which is easy to solve by a number of methods) and the boundary layer, which is governed by an easier to solve PDE. The continuity and Navier–Stokes equations for a two-dimensional steady incompressible flow in Cartesian coordinates are given by

$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0 \quad \ldots\ldots\ldots\ldots \quad 1.1$$

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = -\frac{1}{\rho}\frac{\partial p}{\partial x} + \upsilon\left(\frac{\partial^2 u}{\partial x^2} + \frac{\partial^2 u}{\partial y^2}\right)\ldots\ldots \quad 1.2$$

$$u\frac{\partial v}{\partial x} + v\frac{\partial v}{\partial y} = -\frac{1}{\rho}\frac{\partial p}{\partial y} + \upsilon\left(\frac{\partial^2 v}{\partial x^2} + \frac{\partial^2 v}{\partial y^2}\right)\ldots\ldots \quad 1.3$$

Where u and v are the velocity components, ρ is the density, P is the pressure, and υ is the kinematic viscosity of the fluid at a point.

The approximation states that, for a sufficiently high Reynolds number the flow over a surface can be divided into an outer region of inviscid flow unaffected by viscosity (the majority of the flow), and a region close to the surface where viscosity is important (the boundary layer). Let u and v be stream wise and transverse (wall normal) velocities respectively inside the boundary layer.

Using scale analysis, it can be shown that the above equations of motion reduce within the boundary layer to become

$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0 \ldots\ldots\ldots \quad 1.4$$

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = -\frac{1}{\rho}\frac{\partial p}{\partial x} + \upsilon\frac{\partial^2 u}{\partial y^2} \ldots\ldots\ldots \quad 1.5$$

and if the fluid is incompressible (as liquids are under standard conditions)

$$\frac{1}{\rho}\frac{\partial p}{\partial y} = 0 \ldots\ldots\ldots \quad 1.6$$

The asymptotic analysis also shows that υ, the wall normal velocity, is small compared with u the stream wise velocity, and that variations in properties in the stream wise direction are generally much lower than those in the wall normal direction.

Since the static pressure p is independent of y, then pressure at the edge of the boundary layer is the pressure throughout the boundary layer at a given stream wise position. The external pressure may be obtained through an application of Bernoulli's equation. Let u_0 be the fluid velocity outside the boundary layer, where u and u_0 are both parallel. This gives upon substituting for p the following result

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = u_0\frac{\partial u_0}{\partial x} + \upsilon\frac{\partial^2 u}{\partial y^2} \ldots\ldots\ldots \quad 1.7$$

With the boundary condition

$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0 \ldots\ldots\ldots \quad 1.8$$

For a flow in which the static pressure p also does not change in the direction of the flow then

$$\frac{\partial p}{\partial x} = 0 \ldots\ldots\ldots \quad 1.9$$

So u_0 remains constant.Therefore, the equation of motion simplifies to become

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = \upsilon\frac{\partial^2 u}{\partial y^2} \ldots\ldots\ldots \quad 1.10$$

These approximations are used in a variety of practical flow problems of scientific and engineering interest. The above analysis isInstantaneous laminar or turbulent boundary layer, but is used mainly in laminar flow studies since the mean flow is also the instantaneous flow because there are no velocity fluctuations present.

Turbulent boundary layer

The treatment of turbulent boundary layers is far more difficult due to the time-dependent variation of the flow properties. One of the most widely used techniques in which turbulent flows are tackled is to apply Reynolds decomposition. Here the instantaneous flow properties are decomposed into a mean and fluctuating component. Applying this technique to the boundary layer equations gives the full turbulent boundary layer equations not often given in literature

$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0 \ldots\ldots\ldots \quad 1.11$$

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = -\frac{1}{\rho}\frac{\partial p}{\partial x} + \upsilon\left(\frac{\partial^2 u}{\partial x^2} + \frac{\partial^2 u}{\partial y^2}\right) - \frac{\partial u'v'}{\partial y} - \frac{\partial u'^2}{\partial x} \ldots \quad 1.12$$

$$u\frac{\partial v}{\partial x} + v\frac{\partial v}{\partial y} = -\frac{1}{\rho}\frac{\partial p}{\partial y} + \upsilon\left(\frac{\partial^2 v}{\partial x^2} + \frac{\partial^2 v}{\partial y^2}\right) - \frac{\partial u'v'}{\partial x} - \frac{\partial v'^2}{\partial y} \ldots \quad 1.13$$

Using the same order-of-magnitude analysis as for the instantaneous equations, these turbulent boundary layer equations generally reduce to become in their classical form

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = -\frac{1}{\rho}\frac{\partial p}{\partial x} + \upsilon\frac{\partial^2 u}{\partial y^2} - \frac{\partial u'v'}{\partial y} \ldots\ldots\ldots \quad 1.14$$

$$\frac{\partial p}{\partial y} = 0 \ldots\ldots\ldots \quad 1.15$$

The additional term $u'v'$ in the turbulent boundary layer equations is known as the Reynolds shear stress and is unknown a priori. The solution of the turbulent boundary layer equations therefore necessitates the use of a turbulence model, which aims to express the Reynolds shear stress in terms of known flow variables or derivatives. The lack of accuracy and generality of such models is a major obstacle in the successful prediction of turbulent flow properties in modern fluid dynamics. A laminar sub-layer exists in the turbulent zone; it occurs due to those fluid molecules which are still in the very proximity of the surface, where the shear stress is maximum and the velocity of fluid molecules is zero.

2. LITERATURE SURVEY

1. C.-P.-Huhne presented an experimental and numerical study on separation delay on airfoils with bumps. This paper deals with an investigation and parameter study of

bumps of a tail plane airfoil with deflected control flap. The effects of the bump to the flow cause them to prevent separation and hence increases the aerodynamic performance. This concept was adopted where they transferred the leading edge protuberances in front of a control flap to delay stall at highly deflected flaps. The goal of the research was to produce rotational moment needed for take-off with less deflected flaps and hence causes decreased drag by the configuration. At the moment this was limited by the trailing edge flow separation. The aim was to hence decrease flow separation by bump surface. The presence of bump creates a common flow up and common flow down behind the bump as a result of which a high energy flow is transferred to the boundary layer thereby delaying the flow separation. At first, the study investigated the possibility of delaying flow separation with bumps and secondly improving the aerodynamic performance with varied bump parameters such as bump height, position of bump in front of separation line. In addition to this, investigation was carried out with varied flap deflection and various free stream velocities. The experiment showed that the bump size and span wise distance of the bump are the two important parameters which influence maximum achievable lift with decreasing bump height and span wise distance affects the interaction between the vortices. However, bump height lesser than the boundary layer thickness provide no enough energy to be transferred by the vortices resulting in drop of effectiveness.

2. William E. Milholen II et al examined the application of contour bumps for transonic drag reduction.This examination is based on the effect of discrete contour bumps on reducing the transonic drag at off-design conditions on an airfoil. The off-design scenario considered is that of a speed increase, while maintaining a constant lift coefficient. This has been achieved by placing contour bumps on a cruise wing upper surface, as a means by which to spread the shock wave and thereby reducing the transonic drag. The research was carried out on a fully-turbulent flow conditions at a realistic flight chord Reynolds number of 30 million. The new baseline airfoil was designed by means of Computational Fluid Dynamics Methods. The new configuration was evaluated experimentally by minimizing wall interference. This study approaches two different application goals, a short term one and a long term one. The short term one is focused on the design of the surface contour bump limited to 20% of airfoil chord length, while the long term one is focused on the design and development of the surface contour bump covering the last 40% of the airfoil chord whose performance would typically replace the application of control flaps. The computational predictions indicated that the contour bumps generate significant drag reduction in the range of 12.0% to 15.0% at an off-design Mach number of 0.78. The drag reduction resulted by creating a weaker lambda-shaped shock wave pattern on the baseline airfoil. The maximum height of the contour bumps was in the order of 0.005c, with the crest of the contour bumps located 1.0% to 2.0% chord downstream of the normal shock wave.

3. ArvindSanthanakrishnan et al studied the effect of regular surface perturbations on flow over an airfoil. This paper presents an investigation on the effect of introducing large-scale roughness through static curvature modifications on the low speed flow over an airfoil. The surfaces of a standard Eppler 398 airfoil have been modified with regular perturbations or “bumps” of the order of 2%c for this purpose. While the actual E398 airfoil is not a suitable candidate for low Re cases due to extensive prevalence of boundary layer separation, it is expected that the bumps would exercise passive flow control by promoting early transition to turbulence, thereby reducing the extent of separation and improving the performance. The fluid dynamic mechanism behind this separation control methodology is not clearly understood, however. Smoke-wire flow visualization is performed for qualitative observation of the separation region in both the perturbed and unmodified airfoil geometry cases. At higher Re values, pressure probe measurements are made to quantify the wake momentum deficits. Unsteady 2D PIV measurements are employed to understand the near-wall flow field behavior. The size and strength of vertical structures formed in the separating shear layer are examined, along with measurements on the laminar separation bubble. All the experiments are conducted for chord based Re values ranging from 25, 000 to 500, 000.

3. METHODOLOGY

The experiment began with the fabrication of NACA 4412.

- The model of NACA 4412 with chord length of 150mm was fabricated using teak wood according to its respective coordinates. The coordinates were obtained from airfoiltools.com

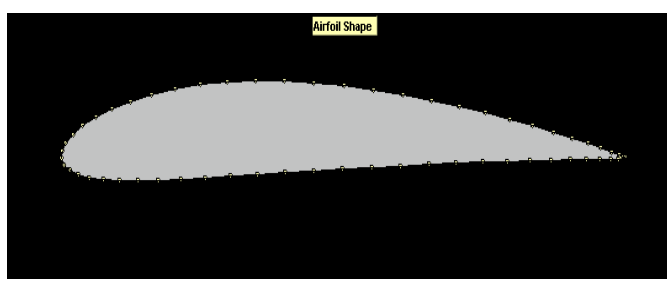


Fig-3.1

- For the purpose of inserting pressure ports into the model, this T-shaped portion having a thickness of half of that ofthe airfoil's thickness was removed using vertical milling machine.
- To insert the pressure ports, the model was drilled with hole diameter of 2mm. Number of holes drilled was 14 with seven on either sides of the airfoil

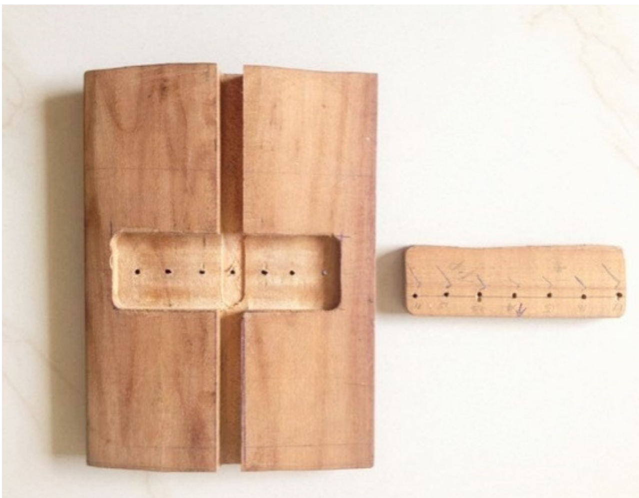


Fig-3.2

- These are fourteen steel tubes used as pressure ports. Inner diameter of the tube = 1mm Outer diameter of the tube = 2mm Insulated wires were inserted into each steel tubes to avoid the blockage of the tubes due to bending. The steel tubes were bent according to their location and the length of the hole on the airfoil. Pressure ports were inserted in to its respective holes and fixed to the airfoil using an adhesive. The insulated wires were removed from all the steel tubes. The separate wooden piece was fixed firmly to the bottom surface of the airfoil with the help of a C-clamp, and a strong adhesive was applied.
- A steel rod of half inch diameter was used to clamp all the pressure ports together and fix the airfoil to the wind tunnel for testing. This rod was attached to the airfoil using anabond.

Fig-3.3

- The gap between the separate wooden piece and the airfoil's bottom surface was filled with wood powder in order to bring the original shape of the airfoil. The extra portion of the steel tubes projecting out of the airfoil's surface was cut down.
- In order to obtain an even surface of the airfoil, metal paste and M seal were applied and the airfoil was smoothened using sand paper and emery paper. Finally it was coated with varnish.
- The bumps were made of balsa wood with a height of 2mm and width of 7mm. Also for the bump a coat of varnish was given to obtain a smooth surface.

Fig-3.4

4. PERFORMANCE MEASURES AND INVESTIGATION

At a velocity of 25.852m/s NACA 4412 without and with bump was tested in the low speed subsonic wind tunnel with the test section of 600×600×2000mm at different angles of attack (-15, -10, -5, 0, 5, 10, 15). From the manometer reading, pressures at each ports were noted down and pressure coefficient was calculated using,

Cp = (Pi - Ps) / (Po - Ps)…………4.1

Where, Cp = Pressure coefficient
Pi = Model Surface Pressure
Ps = Static Pressure
Po = Total Pressure

The wind tunnel testing of an airfoil without bump results, will get a clear idea about the aerodynamic characteristics of the airfoil. The experimental results can be used to locate the position of bump and can calculate how much lift has increased when bump is used. These bumps can be used to reattach the turbulent boundary layer at high angle of attack. Performance calculations has been carried out for various angle of attack. For each angle of attack Lift forces acting on airfoil is calculated by using the following formula

$$C_L = {}_0\int^{x/c} (C_{pl} - C_{pu})\, d(x/c) \ldots\ldots\ldots 4.2$$

Where,

C_{pl}= pressure coefficient on lower surface of the airfoil.
C_{pu}= pressure coefficient on upper surface of the airfoil.
X= distance of port from leading edge
C= chord length

C_L VS α tabular column and graphs
Velocity= 25.852m/s
Without bump

A	C_L
-15	0.19742
-10	0.04062
-5	0.0636
0	0.24435
5	0.31588
10	0.40614
15	0.66112

Table-4.1

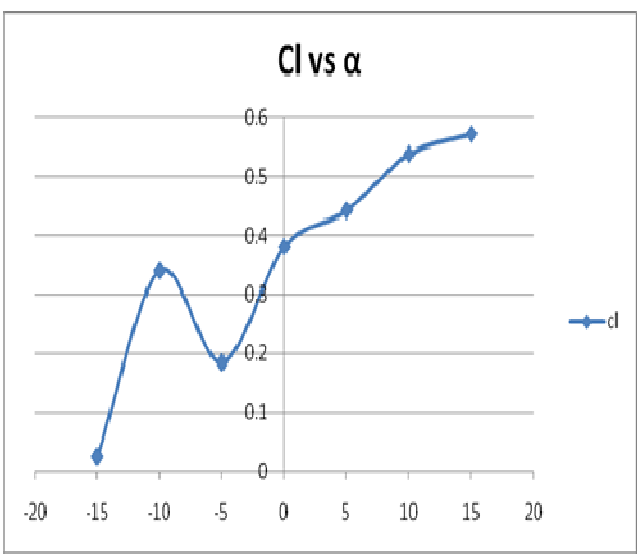


Fig-4.1

Table-4.2

α	C_L
15	0.570324
10	0.537472
5	0.441676
0	0.381065
-5	0.18488
-10	0.339713
-15	0.026731

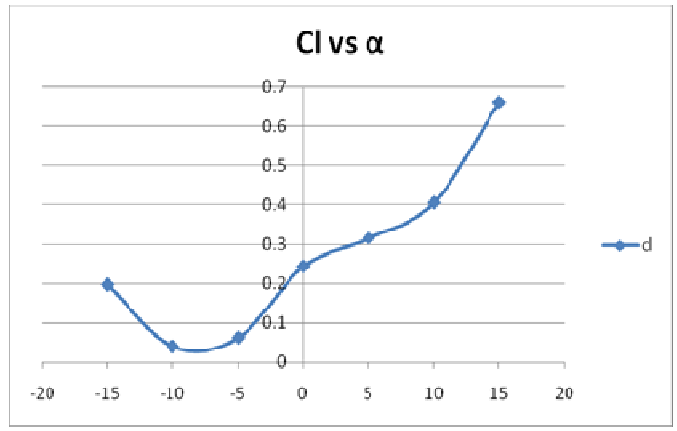


Fig-4.2

5. CONCLUSION

NACA 4412 airfoil without and with bump are tested and analysed and the aerodynamic charecteristic of the model for these two cases are obtained and compared with each other. From the results it is found that the bumps at the trailing edge are effective only at higher angles of attack. The lift is increased is by 15.92% at an angle of attack 15^0 and velocity of 25.852m/s.

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