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Experimental Evaluation of Aerodynamics Characteristics of a Baseline Airfoil

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ABSTRACT : *A wind tunnel test of baseline airfoil NACA 0015 model was conducted in the Wind tunnel wall test section of the Department of Mechanical Engineering at KUET, Bangladesh . The primary goal of the test was to measure airfoil aerodynamic characteristics over a wide range of Angle of Attack (AOA) mainly from Zero degree to 20 degree AOA and with a wind tunnel fixed free stream velocity of 12m/s and at Re = 1.89×10^5. The pressure distribution in both upper and lower camber surface was calculated with the help of digital pressure manometer. After analysis the value of C_l and C_d was found around 1.3 and 0.31 respectively.*

KEYWORDS: *Base line airfoil NACA 0015, Wind tunnel test, Aerodynamic Characteristics, AOA, Reynolds Number*

I. INTRODUCTION

The cross-sectional shape obtained by the intersection of the wing with the perpendicular plane is called an airfoil. 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 [1]. Symmetric airfoils are used in many applications including aircraft vertical stabilizers, submarine fins, rotary and some fixed wings [2][3]. Here in this paper the chosen airfoil was tested to find the reliability of aerodynamics characteristics and to understand the nature of difficulties arises in a newly setup wind tunnel [4][5]. This evaluation of aerodynamic properties is measured and recorded so that future research work on active flow separation control especially by CFJ method will be convenient and comparable to each other but it's not our concern now [6]. The Chosen Aerofoil NACA 0015 model has chord length of 30 cm and span of 50 cm and pressure were measured at 31 point along chord length in each camber surface by digital pressure manometer with a free stream of velocity 12m/s. Once getting pressure distribution, all other propertied like Coefficient of pressure C_p, lift Coefficient C_l, Drag coefficient C_d, Drag Polar, C_l/ C_d were also calculated.

II. MODEL CONSTRUCTION AND METHODOLOGY

Designing NACA 0015 model by using surface profile equations.

For NACA 0015, Chord of the airfoil, c= 0.3 m

Maximum wing thickness, t= last two digit × % c=15×$\frac{1}{100}$×0.3 =0.04

Maximum camber, m= first digit × % c = 0 × $\frac{1}{100}$ × 0 = 0

Distance from leading edge to maximum wing thickness, p= second digit×10% c

$$= 0\times\frac{10}{100}\times 0.3 = 0$$

Maximum wing thickness, $y_t = t \times (1.4845\sqrt{x} - 0.6300\,x - 1.7580\,x^2 + 1.4215\,x^3 - 0.5075\,x^4)$

The mean chamber line,

$$y_c = \frac{m}{p^2}(2px - x^2) \quad \text{For } 0 < x < p$$

And, $\frac{dy_c}{dx} = \frac{2m}{p^2}(p - m)$

$$y_c = \frac{m}{(1-p)^2}\left[1 - 2p + 2px - x^2\right] \quad \text{For } p \le x \le c$$

And, $\frac{dy_c}{dx} = \frac{2m}{(1-p)^2}(p - x)$

Now, coding a C-program including above equation and the upper and lower surface equation and after compiling this program, a set of data were measured for the desired airfoil [7]. Plotting these data on any data plotting software gives the profile like below:

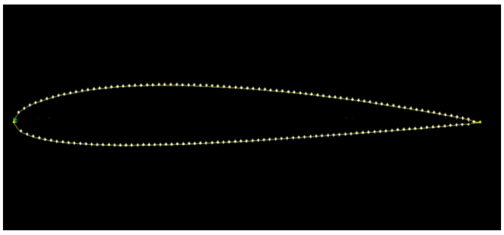

Fig.1: NACA 0015 airfoil profile

Fig.2: 3D Model view of Airfoil NACA 0015

After the construction, model were placed on an open loop Aerolab wind tunnel having test section geometry of 1m x 1m and has an operating speed from 0-40 m/s (0-145 miles per hour). This is made possible by a 10-horse power motor that drives a fan. After applying free stream velocity of 12 m/s, the value of pressure and hence the value of C_p also calculated at different AOA. AOA has changed manually by a clamp hooked assembly passing through the test wall section having a proper calibration of angle [8]. Following is the schematic view of total setup:

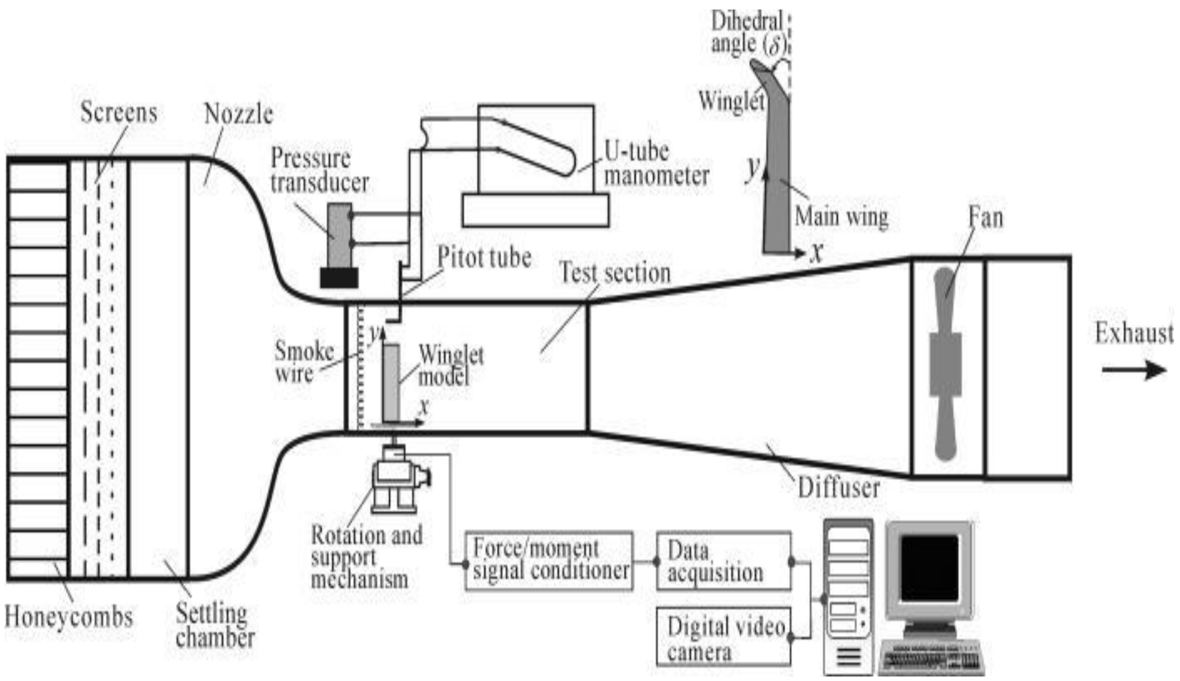

Fig.3: Schematic view of wind tunnel setup where test was conducted.

III. Figures and Tables

At AOA = 05 deg., 12 deg., 20 deg; the following value of C_p was witnessed.

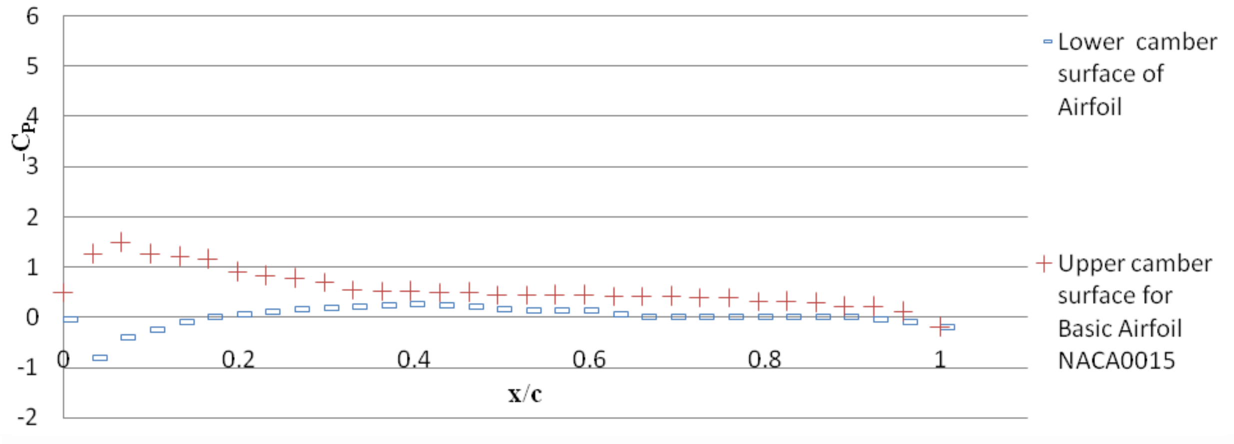

Fig.4: $-C_p$ vs. x/c at AoA= 05 deg.

From Fig.4 we can see that the stagnation point is indeed on the underside of the wing very near the front at $\frac{x}{c} = 0.01$. There are no flat areas of C_p which indicates that there is no boundary layer separation. The pressure distribution shows a smooth variation in both upper camber surface and lower camber surface. The maximum value of $-C_p$ in Baseline Airfoil are 1.5 and 0.25 at upper camber surface and lower camber surface respectively.

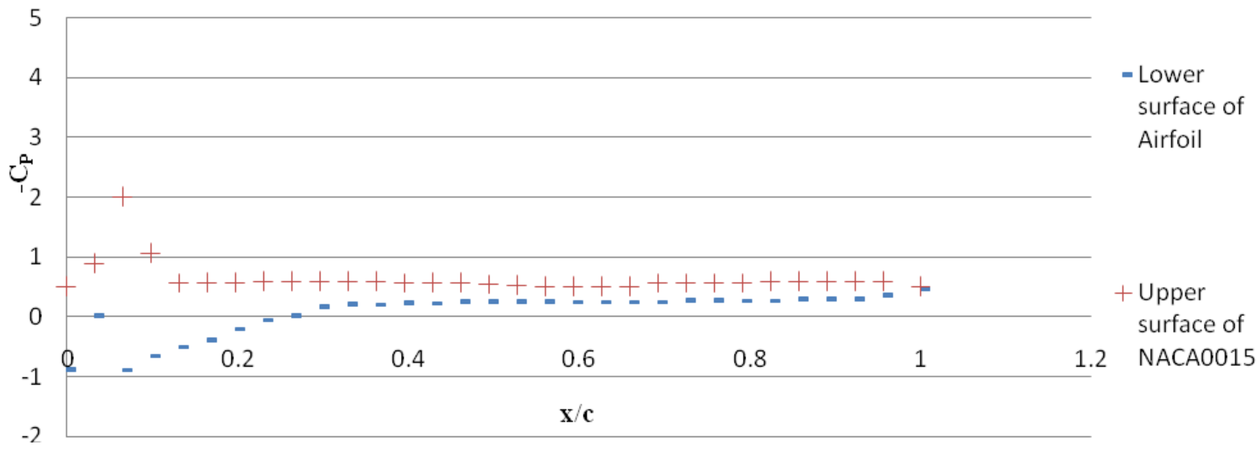

Fig.5: $-C_p$ vs. x/c at AoA= 12 deg.

From Fig.5 we can see that the stagnation point is indeed on the underside of the wing very near the front at $\frac{x}{c} = 0.022$. There are no flat areas of C_p which indicates that there is no boundary layer separation. The maximum value of $-C_p$ in Baseline Airfoil are 2.0 and 0.20 at upper camber surface and lower camber surface respectively. Since the value of pressure coefficient is increased this indicates that with further increase in AOA, the pressure in upper camber surface will be detached from the wall hence will offer boundary layer separation.

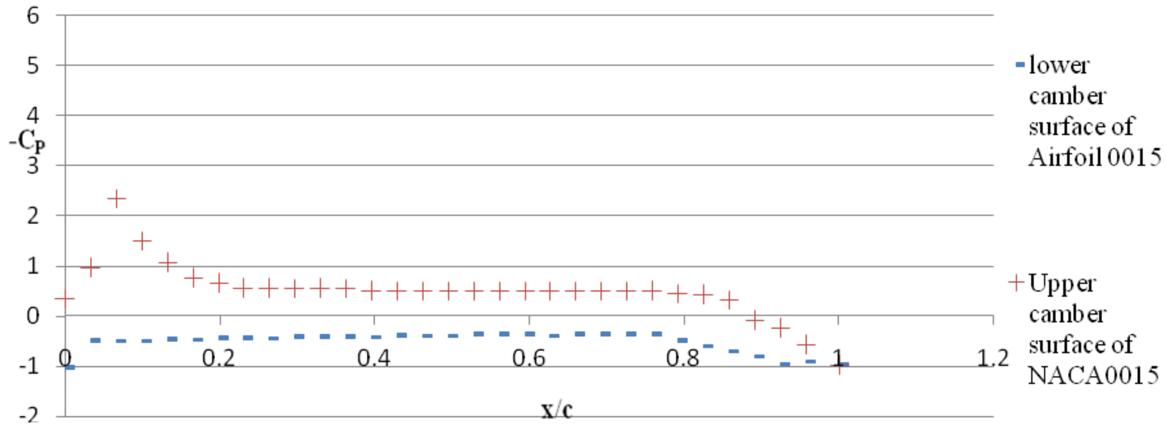

Fig.6: $-C_p$ vs. x/c at AoA= 20 deg.

From Fig.6 it seems that the value of Cp is smooth but at this stage boundary layer separation is occurred and the value of drag coefficient is increased with a plunged in lift coefficient.

The following table shows the value of C_l, C_d and C_l/C_d at different Angle of Attack (AOA) :

Angle of Attack(AOA)	C_l	C_d	C_l/C_d
0	0.01	0.01	1
5	0.6	0.02	30
10	1.3	0.08	16.25
12	1	0.105	9.52
20	0.9	0.25	3.6
25	0.85	0.31	2.74

Table 1: values of C_l, C_d and C_l/C_d at different Angle of Attack (AOA) of an airfoil NACA 0015.

The profile of C_l Vs. AOA is given below:

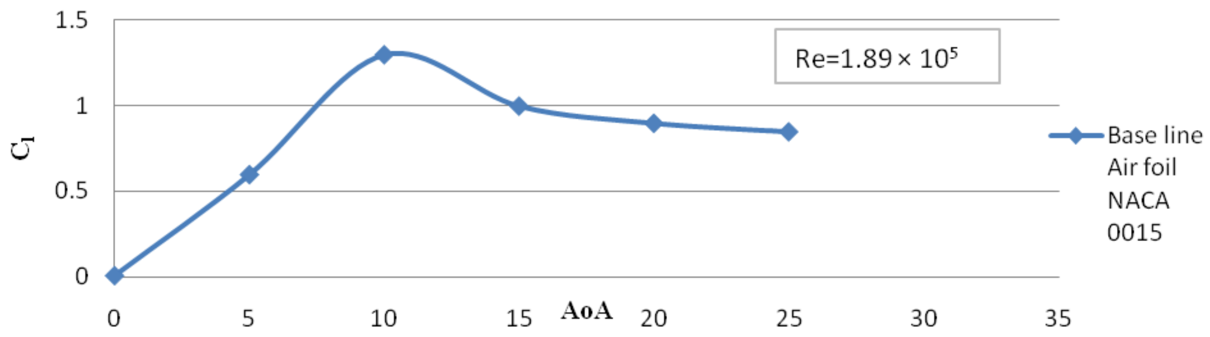

Fig.7: C_l Vs. AOA

From Fig.7 It is clear that $C_{l)max}$ = 1.3 at AOA = 10 deg.

The profile of C_d Vs. AOA is given below:

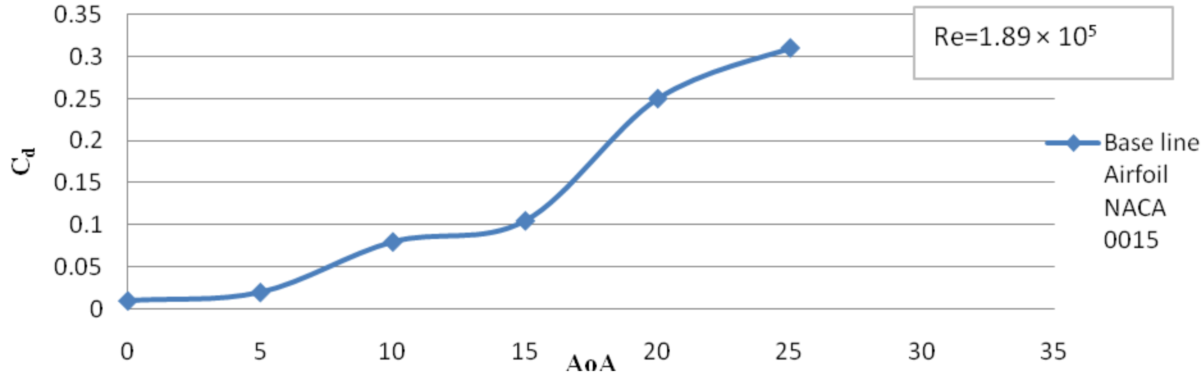

Fig.8: C_d vs. AOA

From Fig.8 It is clear that the value of C_d is increased as the AOA is increased.

The profile of C_l/ C_d vs. AOA is given below:

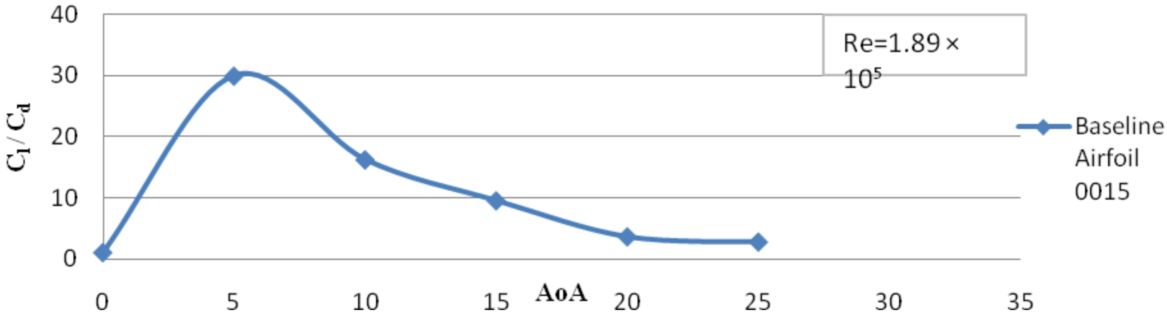

Fig.9: C_l/ C_d vs. AOA

From Fig.9 It is clear that the maximum value of C_l/ C_d is 30 and is gradually decreased as the value of AOA is increased.

The smoke flow visualization technique is used to observe the flow separation at different AOA. After 12 degree Angle of Attack the snap short of flow visualization is given below:

Fig 10: snap short of smoke flow visualization after 12 degree AOA.

IV. CONCLUSION

The NACA 0015 airfoil was analyzed for the lift, drag and moment coefficients as planned. From this subsonic wind tunnel test of an Airfoil NACA 0015 it is found that the boundary layer separation is occurred in between 15-20degree AOA and the maximum value of C_l is 1.3 at AOA = 10 deg. and $C_{d)\ max}$ = 0.31 at AOA = 25 deg. And this value of C_d is increased gradually with the increase in AOA. The optimum AOA of this NACA 0015 Airfoil is found to be around 12 deg. It is clear from this paper that boundary layer separation is not controlled or delayed as well as aerodynamic characteristics in not easy to enhance with this type of conventional way. With some modification, techniques like Active flow separation should be applied in future work to enhance the aerodynamic characteristics.

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