Experiment - 9

Lift and Drag measurement on symmetrical airfoil NACA0015 at different angles of attack

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1. Aim :

- To measure value of Lift and Drag.
- To compare Lift Coefficients and Drag Coefficients with different Angles of Attack and Airspeed(or Reynolds No.)

2. Apparatus :

Required apparatus for performing this experiment are:

- Manometer
- C15-10 Armfield tunnel
- Pitot-static Probe
- Fan
- Symmetrical Airfoil model

3. Theory :

Thin Airfoil NACA0015 The NACA 0015 is the airfoil from the 4-digit series of NACA airfoils.The airfoil is symmetrical and 00 denotes it has no chamber. The 15 indicates that the airfoil has a 15% thickness to chord length ratio.The parameters of the airfoil are followings:

- Chord length of the airfoil(C) = 65 mm = 0.065 m
- Span length(L) = 150 mm = 0.15 m

Stagnation and Static Pressure: **Stagnation pressure**(P_{stag}) is the pressure at the stagnation points in the fluid flow.

Static pressure(P_∞) is the actual thermodynamic pressure of a flow.

$$P_{stag} = P_\infty + \frac{\rho v^2}{2}$$

Reynolds Number : The **Reynolds Number** is the ratio of inertial forces to viscous forces within a fluid that is subjected to relative internal moment due to variation of velocities.

$$R_e = \frac{\rho V_\infty L}{\mu}$$

Lift Force(L): **Lift** is the force on the body in a direction normal to the flow direction. The lift will only be presented if the fluid incorporates a circulatory flow about the body. As the velocity above the body is increased, the static pressure is reduced and the velocity beneath is decreased gives an increase in static pressure.

$$L = \int_{-\infty}^{\infty} \rho v(V_\infty - v)\, dx$$

Coefficients of Lift(C_l): The **Lift Coefficients** contains the effects of object shape, air viscosity, and compressibility. Lift coefficients are calculated by :

$$C_l = \frac{L}{(\rho V_\infty^2 A)/2}$$

Drag Force(D): The **Drag** on a body in an oncoming flow is the force on the body in a direction parallel to the flow. For small angles of attack, the lift force is high and the drag force is low. Drag is calculated by:

$$D = \int_{-\infty}^{\infty} \rho v(V_\infty - v)\, dy$$

Coefficients of Drag: **Drag Coefficients** caused due to skin friction and Drag. The drag coefficient is calculated by :

$$C_d = \frac{D}{(\rho V_\infty^2 A)/2}$$

ρ = Density of Air = 1.225 Kg/m^3 , V_∞ = Velocity of Air, A = $(C \times L)$ = Area of Airfoil

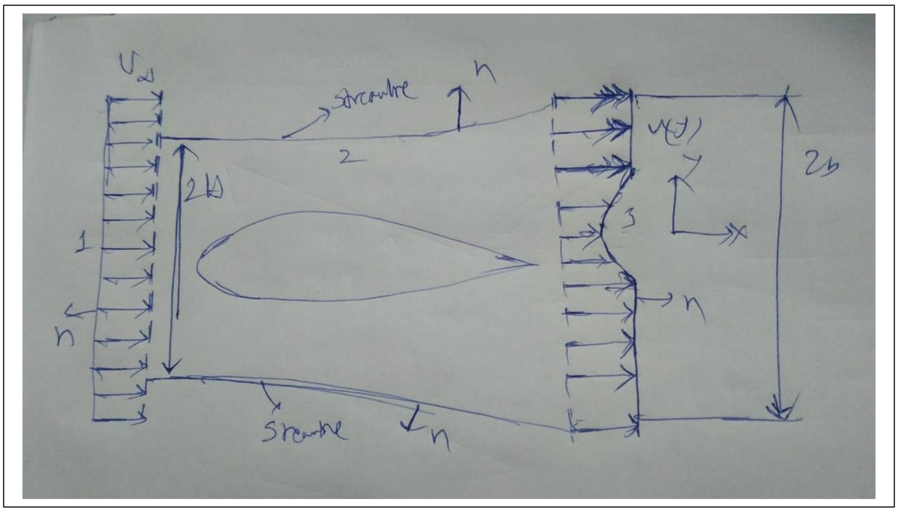

Figure 1: Wake and Velocity profile behind a symmetric airfoil

4. Procedure :

1. In wind tunnel test section is set.
2. Pitot-static probe is connected to manometer.
3. Fan speed is fixed.
4. Required readings are taken.

5. Observation :

5.1 Required table for Drag Coefficients and Lift Coefficients :

Angle of Attack(α)	For velocity = 15 m/s				For Velocity = 20 m/s			
	Drag(D)	Lift(L)	C_d	C_l	Drag(D)	Lift(L)	C_d	C_l
2	0.02	0.38	0.015	0.283	0.05	0.76	0.037	0.566
4	0.04	0.54	0.030	0.402	0.06	0.93	0.045	0.692
6	0.05	0.63	0.037	0.469	0.07	1.12	0.052	0.834
8	0.07	0.75	0.052	0.558	0.09	1.36	0.067	1.012
10	0.12	0.8	0.089	0.595	0.12	1.56	0.089	1.161
12	0.18	0.75	0.134	0.558	0.17	1.58	0.134	1.176

5.2 Plot of Lift Coefficients vs Angle of Attack

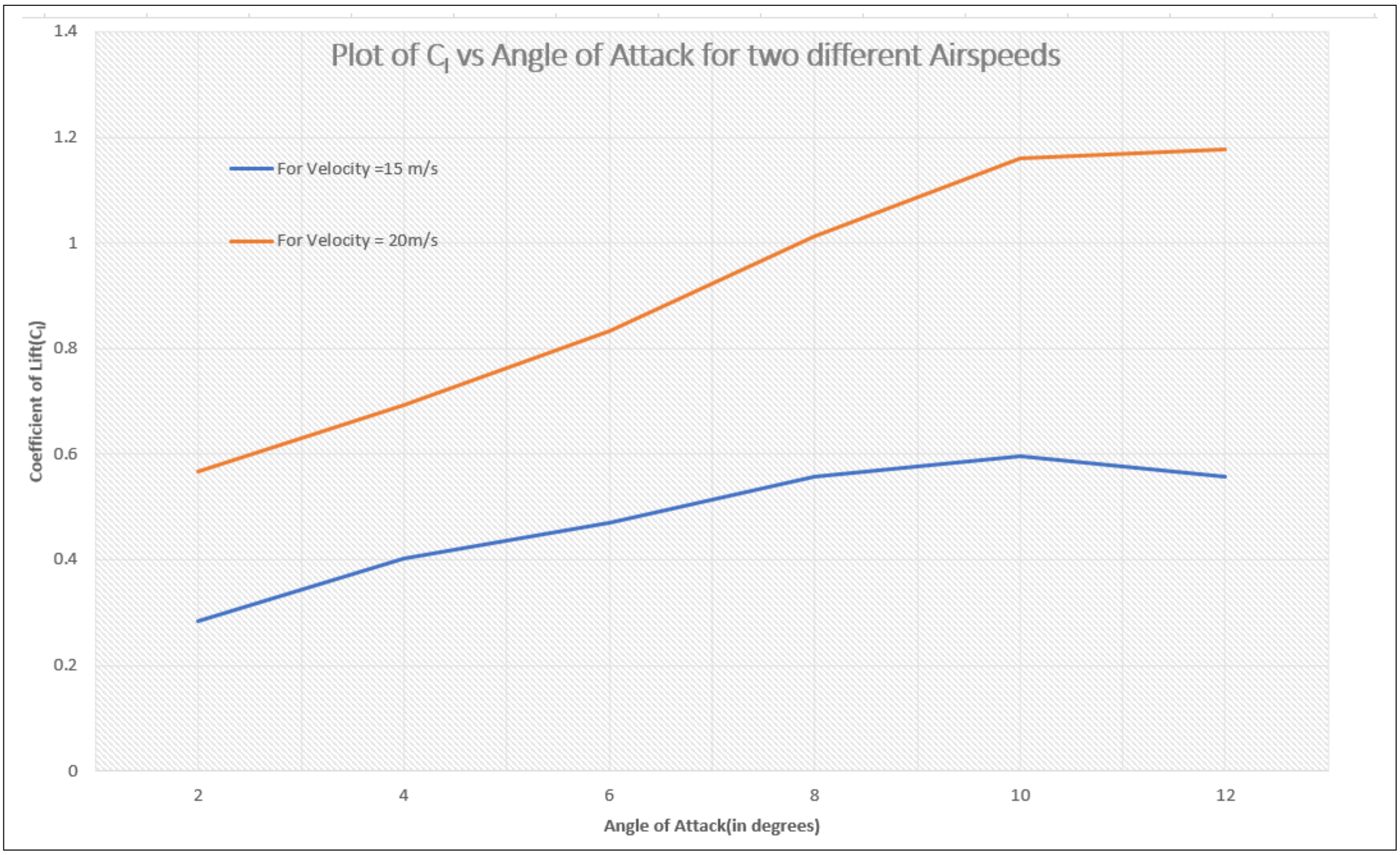

Figure 2: Plot of Coefficient of Lift VS Angle of Attack

5.3 Plot of Coefficient of Lift vs Coefficient of Drag :

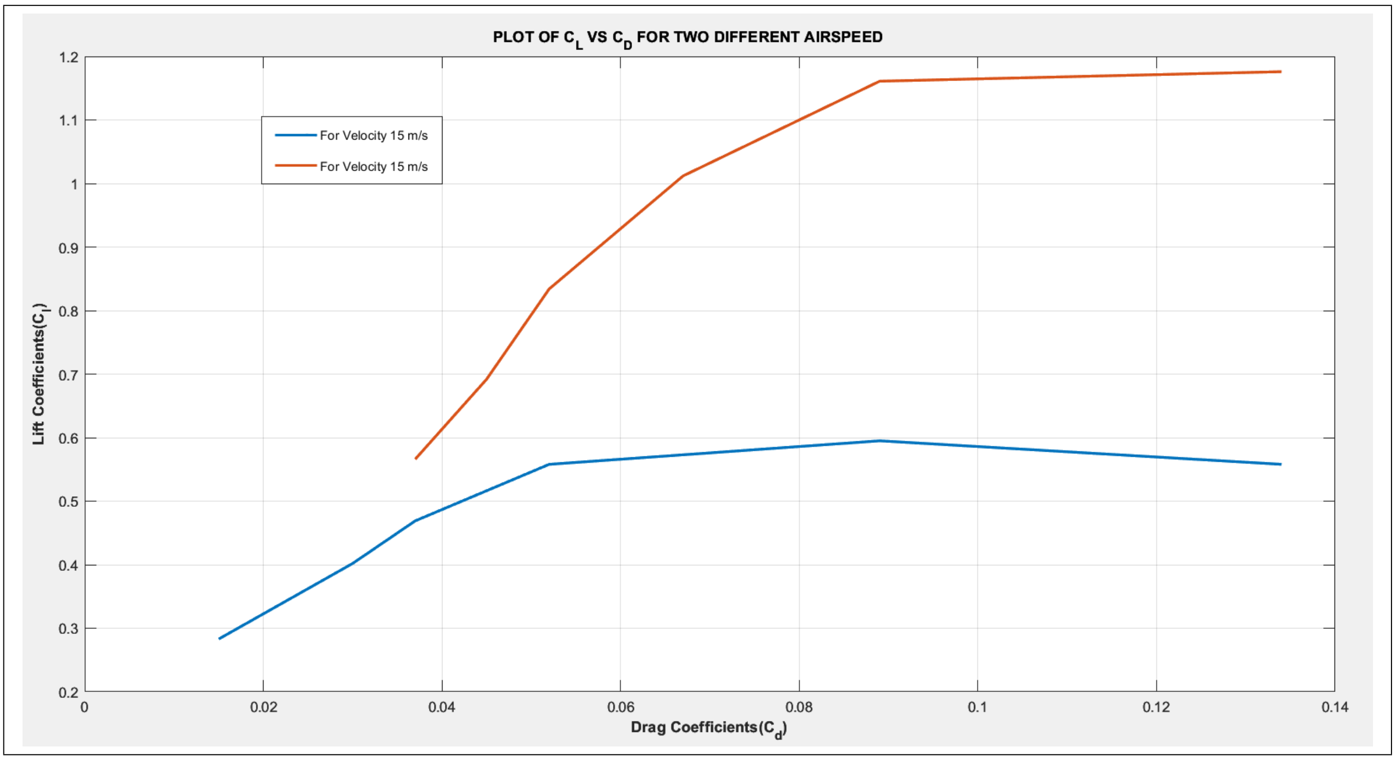

Figure 3: Variation of Coefficients of Lift with Coefficient of Drag for Two different Velocities

6. Calculations :

Density of air(ρ) = 1.225 $\frac{Kg}{m^3}$

6.1 Calculation of Drag Coefficients :

For Angle of Attack = 4 Degrees :
Velocity = 15 m/s :
From experimental observation,
Drag(D) = 0.04 N
Area of Airfoil(A) = Chord Length(C)× Span Length(L) = 0.065×0.15 = 9.75×10^{-3}

$$\Rightarrow C_d = \frac{2D}{\rho V_\infty^2 A}$$

$$\Rightarrow C_d = \frac{2 \times 0.04}{1.225 \times 15^2 \times 9.75 \times 10^{-3}}$$

$$\Rightarrow \boxed{C_d = 0.030}$$

6.2 Calculation of Lift Coefficients :

For Angle of Attack = 4 Degrees :
Velocity = 15 m/s :
From experimental observation,
Lift(L) = 0.54 N

Area of Airfoil(A) = Chord Length(C)$\times$ Span Length(L) = $0.065 \times 0.15 = 9.75 \times 10^{-3}$

$$\Rightarrow C_l = \frac{2L}{\rho V_\infty^2 A}$$

$$\Rightarrow C_l = \frac{2 \times 0.04}{1.225 \times 15^2 \times 9.75 \times 10^{-3}}$$

$$\Rightarrow \boxed{C_l = 0.402}$$

7. Sources of Error:

- Error due to instrumental defect.
- Error may occur in taking readings before flow becomes steady.
- Error due to environmental effect like temperature,pressure change.
- Error in measurement due to presence of zero error in parameters.
- Dimensional error may occurs .
- Parallex error may occur during set up of Angle of Attack.

8. Conclusion :

- For small Angles of Attack Lift force is high and Drag Force is low.
- After a certain Angles of Attack increases beyond a certain value, the lift force decreases and the drag forces increases.
- Both lift and drag coefficient increases as angle of attack is increased.
- After a certain Angle of attack(**StallAngle**) Lift Coefficients decreases. For air speed = 15 m/s Stall Angle is 10 °. Stall Angle is caused Due to transition of laminar to turbulence flow.
- Value of C_l/C_d gradually decreases on increasing Angles of Attack.
- For each Angles of Attack C_l is higher for flow with high Reynolds No.(or Airspeed).