AE 244

Assignment 1

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NACA Airfoil

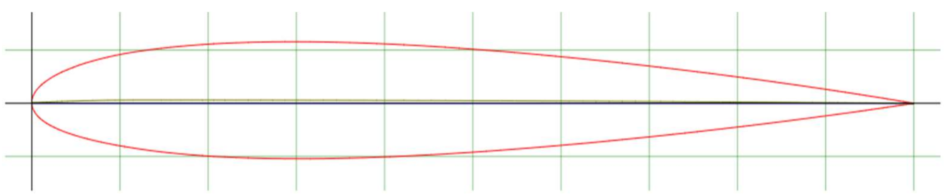
As per the given convention, the NACA airfoil parameters are:

- Maximum camber 0.3%
- Thickness 11%
- Maximum camber position 12.5%
- Chord length 1m

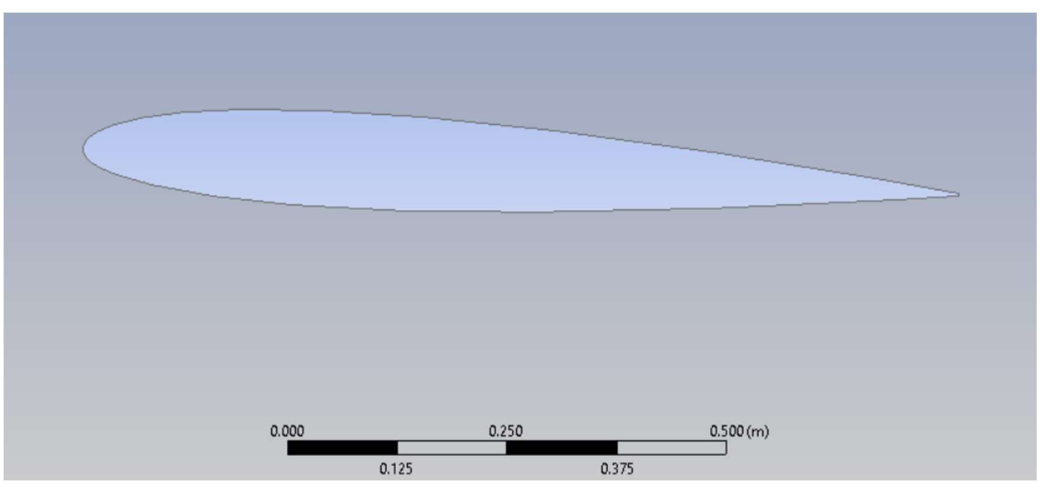
The testing environment (atmosphere) conditions as per ISA are as follows:

- Free stream velocity 30m/s
- Density 1.225 kg/m^3
- Dynamic Viscosity 1.789E-5 Ns/m^2

Airfoil Plot

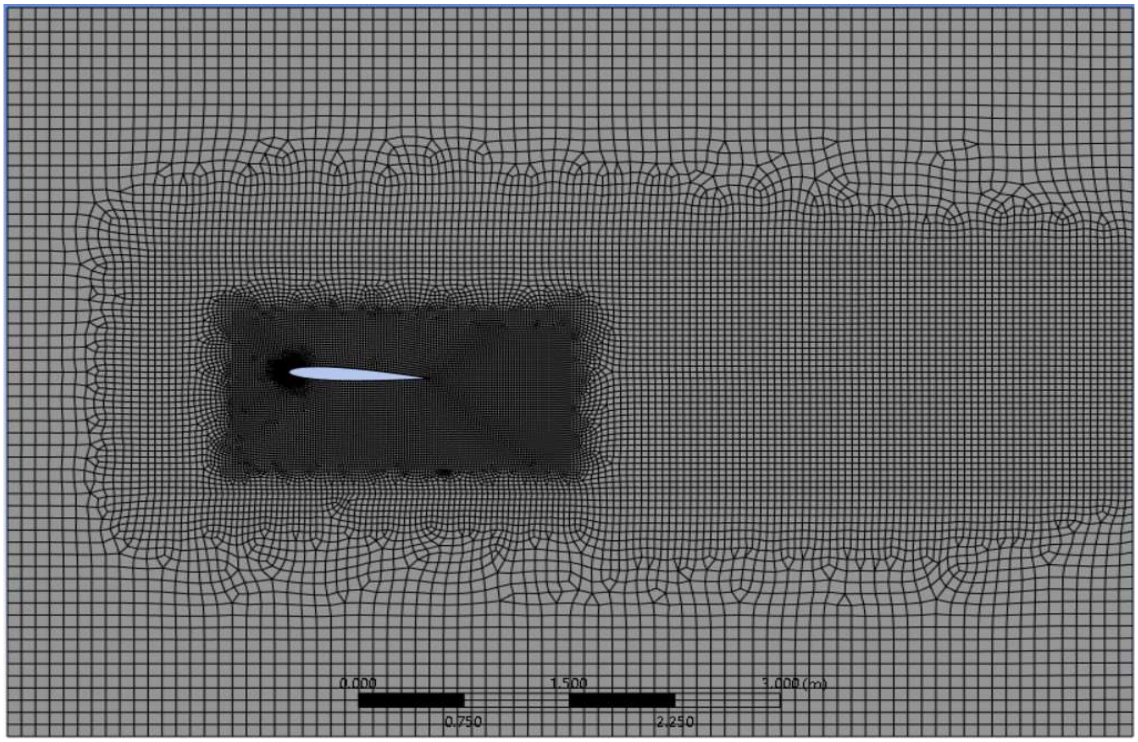


Airfoil CAD Model

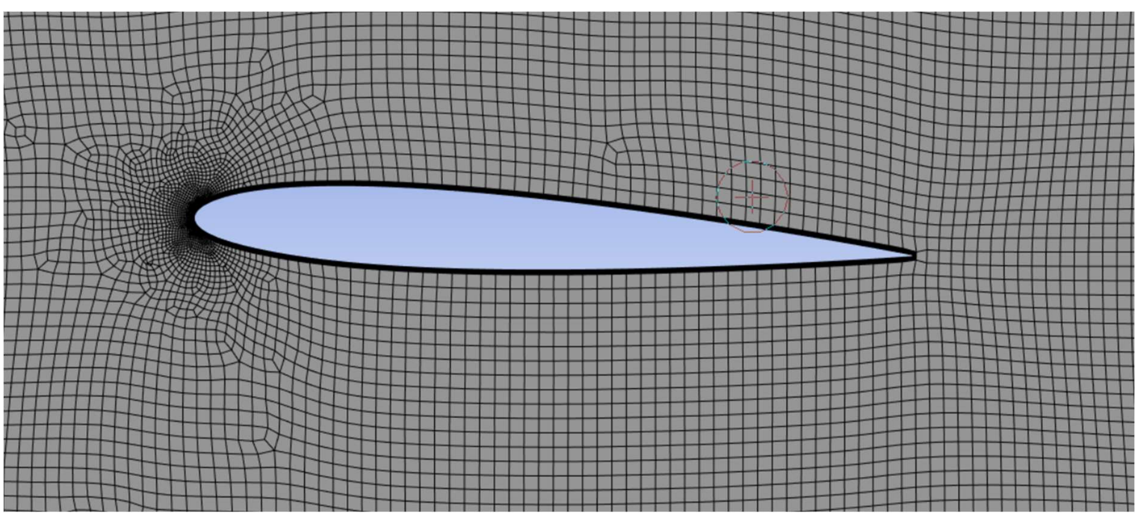


Airfoil Mesh

Far view (of complete fluid domain)

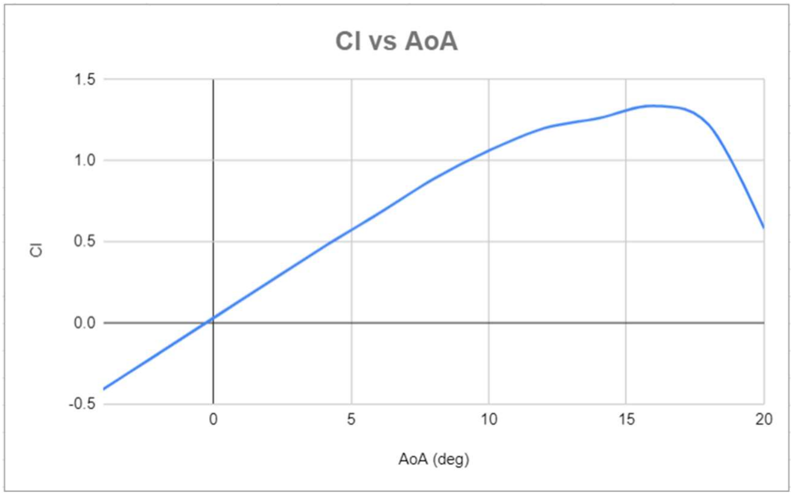


Close View (of region close to the airfoil)

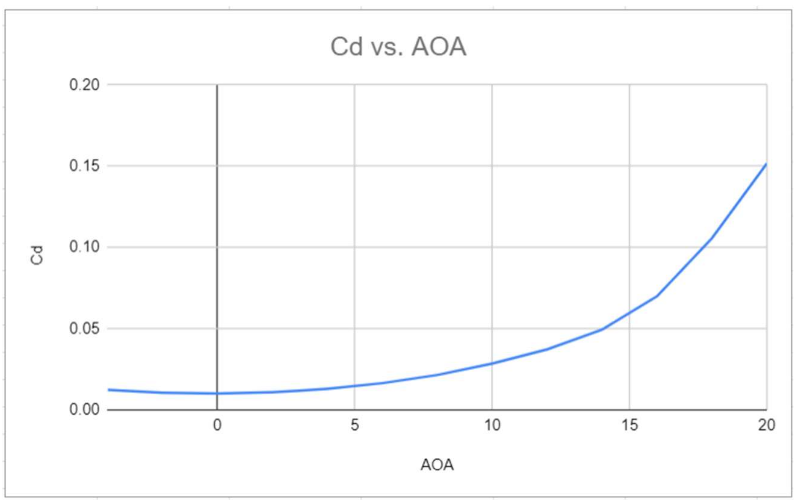


Simulation Results

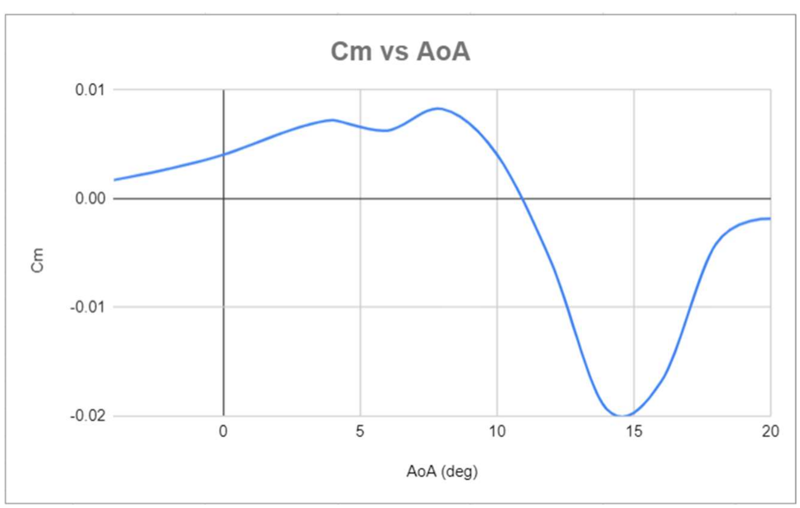
$C_l - \alpha$ Curve



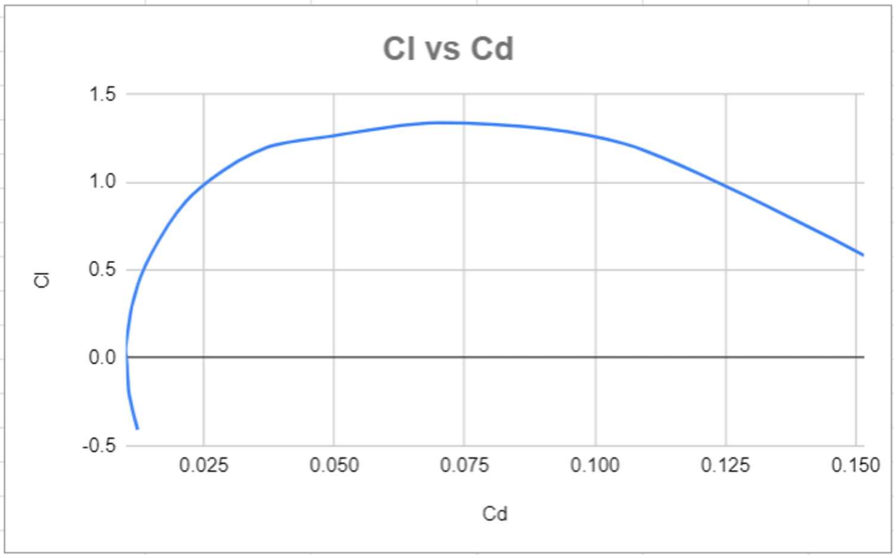
$C_d - \alpha$ Curve



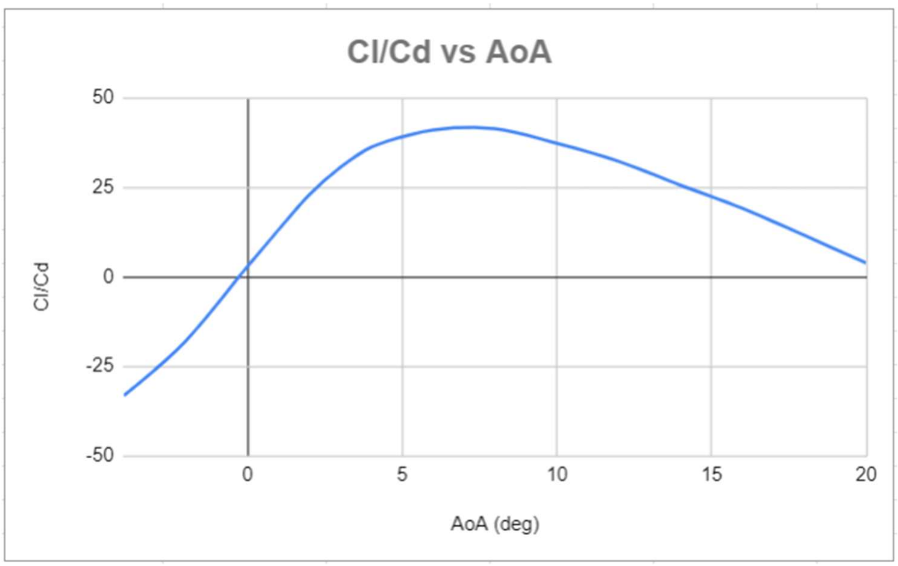
$C_m - \alpha$ Curve



$C_l - C_d$ Curve



$C_l/C_d - \alpha$ Curve



Key Parameters

- $C_l - \alpha$ Curve
  - Slope: 6.328 rad^{-1} (very close to 2π & hence good)
  - Y-intercept: 0.029
  - Stall Angle: 16^o
  - Max C_l: 1.34
- $C_l/C_d - \alpha$ Curve
  - Max C_l/C_d: 41.36

Inferences & Explanation of the Curves

C_l – α Curve

C_l increases with α linearly until stall occurs. At stall point, C_l is maximum. After stall, C_l decreases.

With larger camber or angle of attack, the incoming flow experiences larger change in direction. The previously horizontal flow (say) gets a downward component, dependent on the direction normal to trailing edge. Due to this change in momentum, a reaction force in the upward direction is applied on the airfoil. This upward force is Lift. Hence, we can now conclude that the larger camber or angle of attack imposes a larger downward component of velocity of air flow which consequently leads to larger lift. Ideally, the slope of C_l – α curve is 2π.

An alternate explanation for this trend is given using differential pressure distribution above and below the airfoil surfaces. Due to camber in the airfoil, the incoming air compresses a bit and hence pressure increases. Meanwhile, above the airfoil, the situation is reverses. The pressure reduces as we go up the airfoil till a certain point. Since pressure is higher below, and lower above the airfoil, a net force acts on the airfoil in the upward direction. Now, if angle of attack increases, the pressure rises below, and reduces above. Hence, the lift increases.

As the angle of attack increases, something interesting happens. The flow above and below the airfoil, which was initially attached, starts separating with large enough angle of attack. This flow separation introduces turbulence in the flow. After flow separation, lift stops increasing with angle of attack. This situation is called 'Stall'.

C_d – α Curve

Drag increases with angle of attack. Higher the α, higher the drag becomes.

Drag is of 2 types – Skin friction drag & Profile drag. Skin friction drag acts tangential to the surface of airfoil and induces a pulling force on the airfoil. This force as a summation over entire surface acts majorly along the flow direction. Profile drag (or Pressure drag) acts normal to the surface of airfoil. The pressure accompanied by the airflow applies a force on the airfoil as it passes through it. This force summed over entire surface has 2 components- lift and drag. Among the 2 types of drag, Profile drag is the dominating one. Skin friction drag is not a bid concern when compared with profile drag. The Profile drag is dependent on the total area normal to the flow. The larger this area is, the larger is the magnitude of this drag.

As the angle of attack increases, the total area normal to the flow increases. This increase increases the profile drag and hence the total drag increases. This trend is observed in the curve as a monotonically increasing function.

Another thing that increases drag is turbulence. After stall occurs, the flow separation introduces turbulence in the airflow. The random, rapid vortices developed in the vortices increases the overall drag acting on the airfoil. Hence the drag increases drastically upon achieving flow separation.

C_m – α Curve

Moment of force is taken about the quarter chord point (c/4). Position moment means ACW rotation, i.e. nose rotating downwards. Similarly, negative moment means CW rotation, i.e. nose lifting upwards.

As inferred from the graph, the Moment is initially positive, then it decreases steeply into negative and remains in that region. This means that initially, the airfoil experiences downward bending at low angles of attack. Then is starts getting large moment in favour of rotating upwards at higher angle of attacks.

At low angle of attacks, the aerodynamic centre of the airfoil exists below the quarter chord point. Due to this, the net torque acting due to the aerodynamic forces (lift and drag) about the quarter chord point results into anticlockwise moment. Hence, the downward bending of nose, or positive moment. As the angle of attack increases, the aerodynamic centre starts moving closer to the quarter chord point and after a angle starts existing before that point. Due to this, the direction of torque reverses. Hence, moment becomes negative & nose lifts upwards.

C_l – C_d Curve

C_l vs C_d curve behaves like a parabolic curve. This behaviour is modelled as following:

$$C_D = C_{D,0} + K\,(C_L - C_{L,0})^2$$

$C_{D,0}$ is the minimum Drag

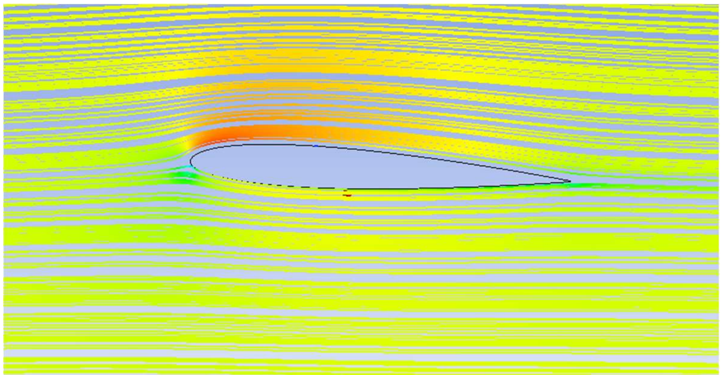
$C_{L,0}$ is the C_L value when drag is minimum

C_l/C_d can be obtained by drawing a line connecting a point on the curve and the origin. The slope of this line gives the C_l/C_d value. When this line becomes the tangent to the curve, then the C_l/C_d becomes maximum.

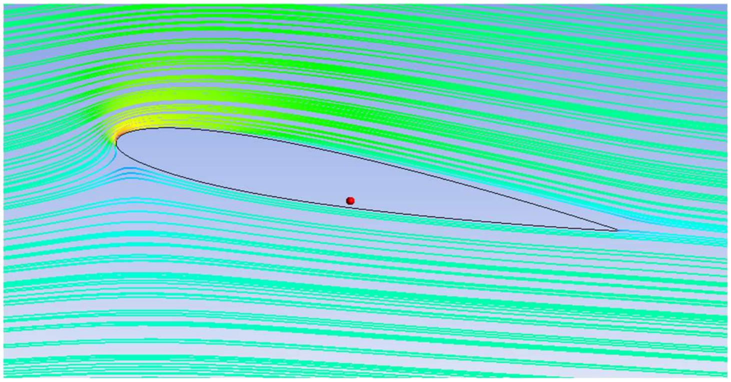
The parabolic behaviour is only valid up till stall point. After stall, C_l decreases, while C_d increases, leading to the curve approaching the C_D axis (x-axis). This behaviour can also be observed from the graph.

Streamline Plot

- $\alpha = 3^o$



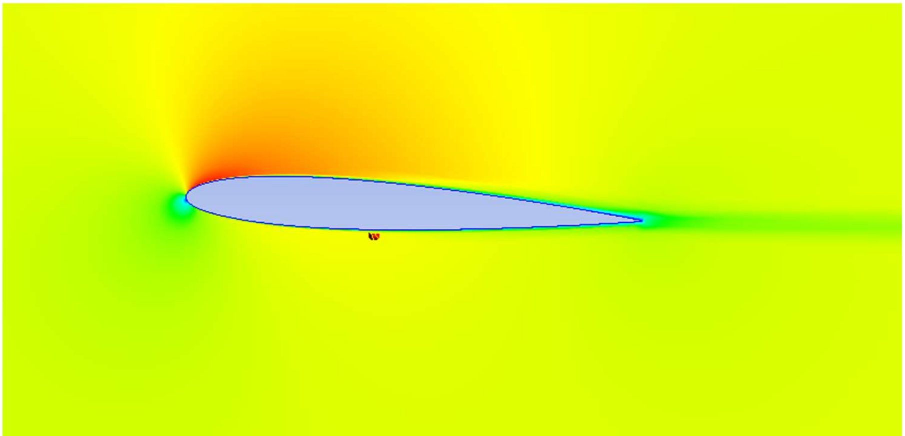
- $\alpha = 10^o$



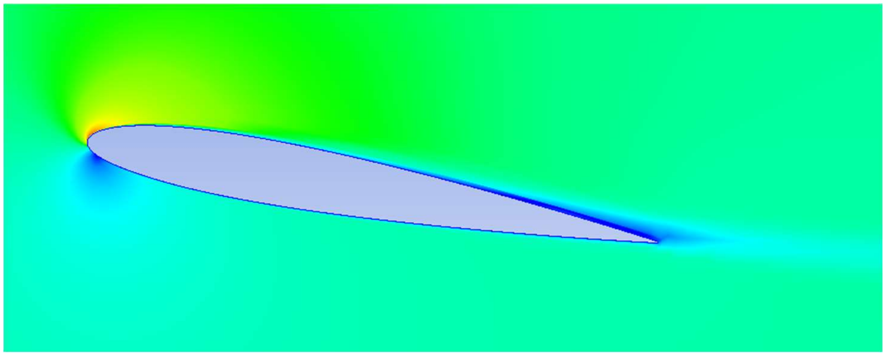
The flow at low α has nearly straight streamlines. The incoming flow comes head on and splits into flow above and below the airfoil. The flow at higher α has a difference in that the incoming flow has an upward component. It attacks the airfoil from a bit below the nose. The streamline that goes above the airfoil curves backwards and then follows along the surface. This phenomenon is mainly due to the pressure gradient that gets developed due to the introduction of the airfoil in the undisturbed flow.

Velocity Field Magnitude Contour Plot

- $\alpha = 3^o$



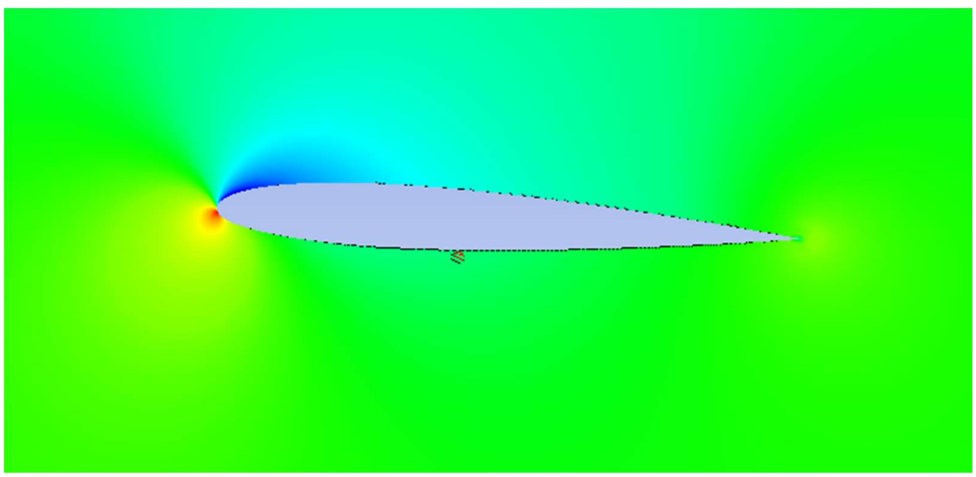
❖ $\alpha = 10^o$



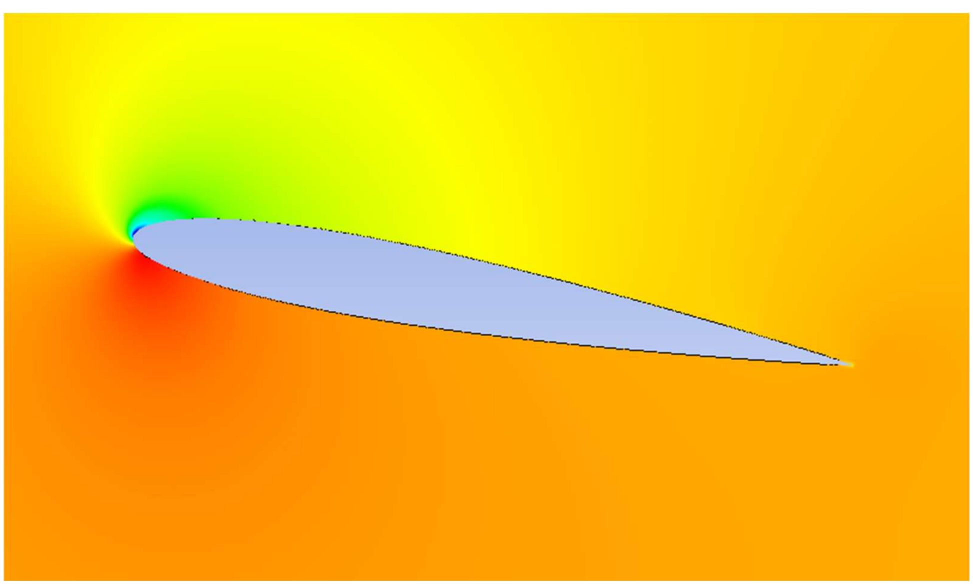
Velocity gets affected the most inside the boundary layer. The flow is slower inside the boundary layer for both cases. For larger α, the boundary layer is more and hence the slower flow is more apparent. Also, near the nose, the flow is different mainly due to the collision with the airfoil. The flow is faster near the nose in both cases.

Pressure Field Contour Plot

❖ $\alpha = 3^o$



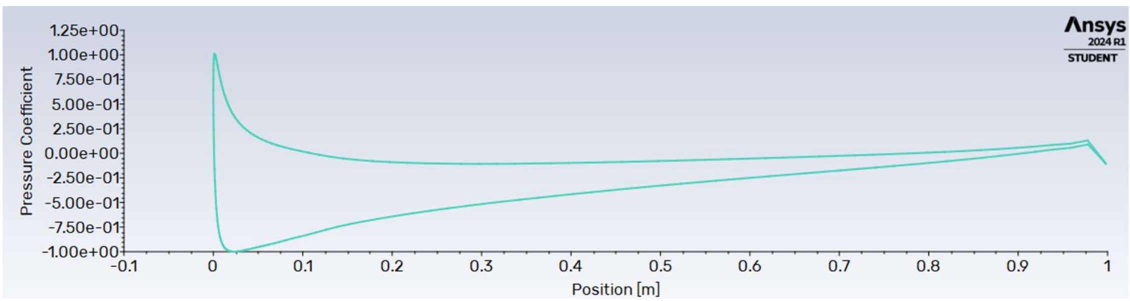
❖ $\alpha = 10^o$



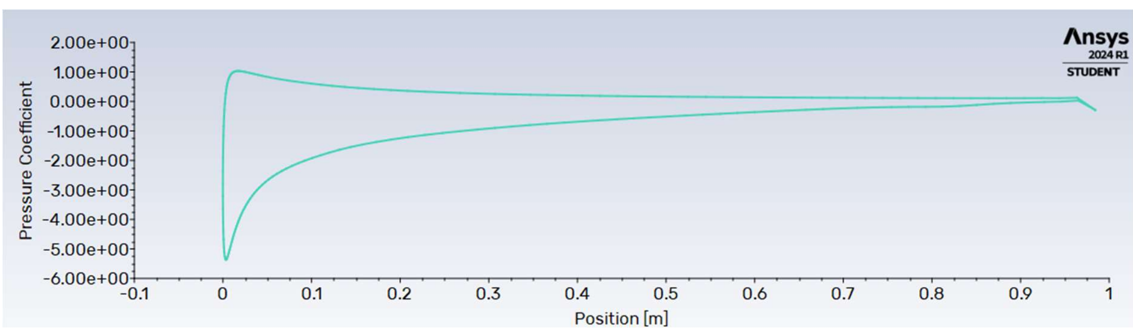
Flow has larger pressure below the airfoil compared to above in both cases. This can be seen from the red region below and yellow above in α = 10º case. Similarly, green below and blue above in α = 3º case. This is due to the fact that for the streamlines to meet at the trailing edge, they should flow past the airfoil in same time. Due to the curvature of the airfoil, the suction surface has longer length than pressure surface. This means velocity should be more above than below. As per Bernoulli's equation, this will imply that the pressure should be larger below the airfoil than ambient pressure while the pressure should be lesser compared to the ambient pressure above the airfoil.

Coefficient of Pressure Plot

- α = 3º



- α = 10º



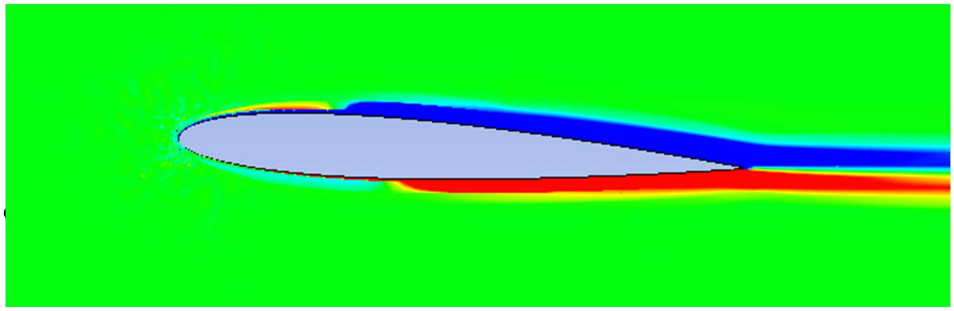
The upper curve represents the C_p values of the lower surface of the airfoil, while the lower curve is that of the upper surface of the airfoil. The C_P curve peaks at the stagnation point. The Pressure slowly increases from the nose till the stagnation point and then starts decreasing.

It can be observed that for lower α, pressure on upper surface is reasonable more and increase steadily. Whereas, for higher α, the pressure on upper surface, is much less and increases rapidly as we move towards the trailing edge.

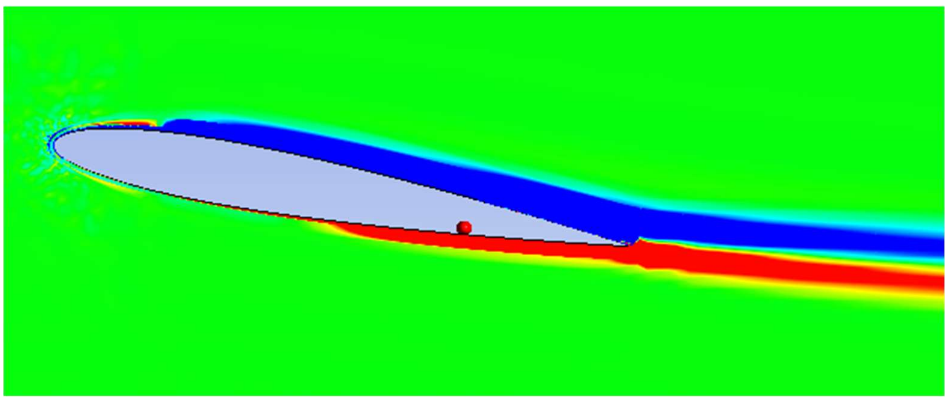
For the lower surface, the situation is reversed. The pressure decreases steadily for higher α, while it decreases more rapidly for lower α

Vorticity Field Contour Plot

- $\alpha = 3^o$



- $\alpha = 10^o$



Vorticity is defined as Curl of the vector field.

$$\text{Vorticity} = \nabla\times\mathbf{V}$$

Since the flow is 2D, only z component exists, vorticity then becomes

$$\text{Vorticity} = \partial v/\partial x - \partial u/\partial y$$

Considering the $\partial u/\partial y$ component on the upper surface of the airfoil, since the x component of velocity (u) increases from 0 to V_∞ with increasing y, hence that quantity is positive. So, vorticity is decreased due to its presence, as shown by the blue region. On the lower surface however, velocity increase as y decreases. Hence, $\partial u/\partial y$ is negative. Hence, vorticity increases, as indicated by the red region in the above screenshots.

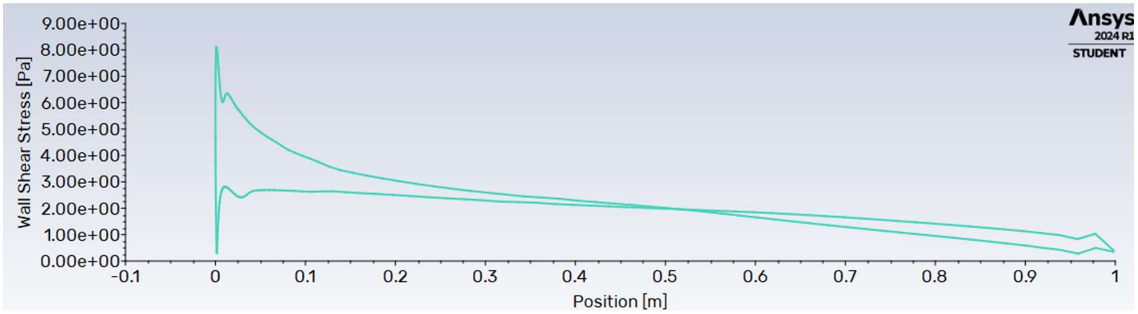
Stagnation Point (x/c)

Points on the airfoil where the previously joint flow splits to go from upper and lower surfaces of the airfoil respectively is called stagnation point. This point is generally located on the lower surface of the airfoil.

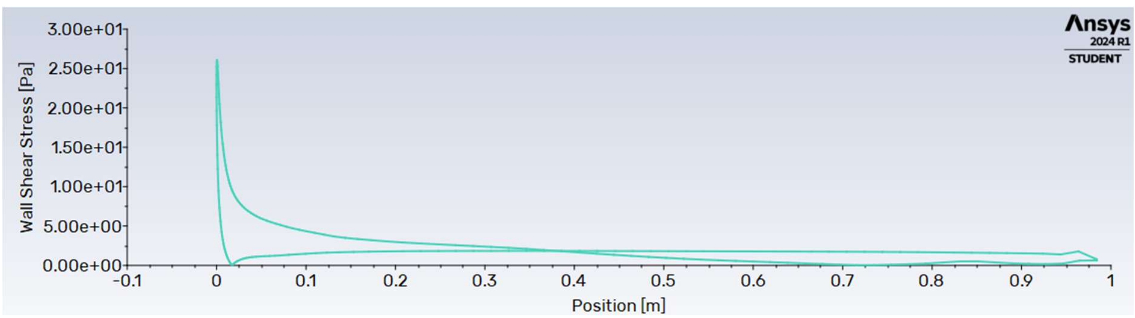
- $\alpha = 3^o$ 0.00174
- $\alpha = 10^o$ 0.0174

Flow Separation Point

❖ $\alpha = 3^{o}$



❖ $\alpha = 10^{o}$

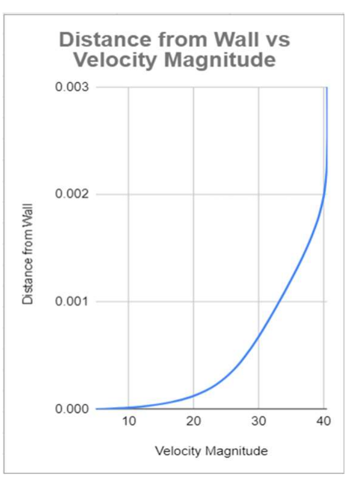


Wall shear stress determines the shear force exerted by the adjoint airflow on the airfoil surfaces. When this force is zero, this implies, the flow is not attached to the airfoil and hence we say flow separation has occurred.

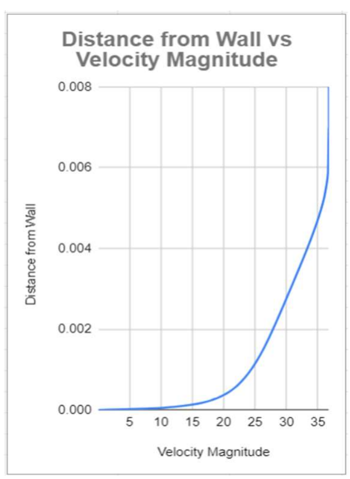
Wall Shear is not zero at any point on the airfoil for both $\alpha = 3^{o}$ & $\alpha = 10^{o}$. Hence, there is no flow separation in both cases.

Upper Surface Boundary Layer Velocity Profile

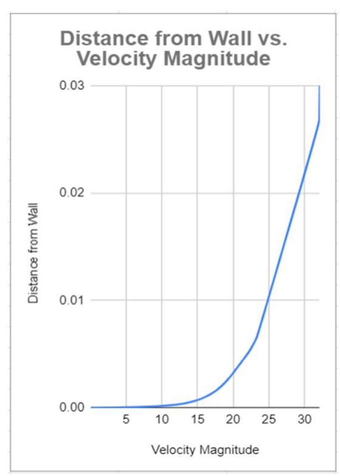
➢ $\alpha = 3^{o}$



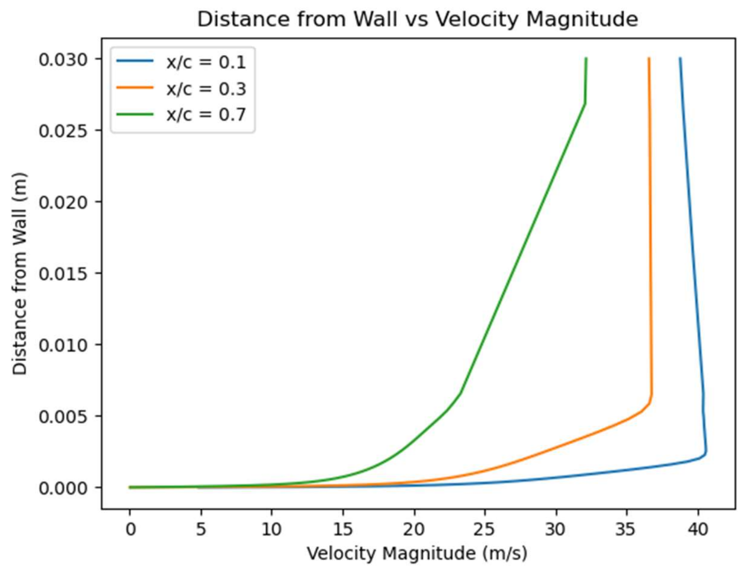
x/c = 0.1



x/c = 0.3



x/c = 0.7



Estimated thickness of Boundary layer

- x/c = 0.1 2.10 mm
- x/c = 0.3 5.86 mm
- x/c = 0.7 26.84 mm

- $\alpha = 10^{o}$

Thickness of the boundary layer can be found by observing the above curves. As the curve approaches asymptotic behaviour, the boundary layer approach its end. The distance from the surface at which the velocity stops changing is the required thickness. It can be observed that the thickness of the boundary layer at x/c = 0.1 is the least, greater for x/c = 0.3, and largest for x/c = 0.7. Hence, we can conclude that the boundary layer grows thicker as we start moving from leading edge to the trailing edge.

Verification of Bernoulli's Theorem along a streamline far from Boundary layer

- $\alpha = 3^{o}$

  Free stream values

  ρ = 1.225 kg/m^3

  P_∞ = 1 atm

  V_∞ = 30 m/s

  q_∞ = 551.25 Pa

  $P_{Stagnation, \infty}$ = 1 atm + 551.25 Pa

  At Some point P far from boundary layer

  P_{Static} = 1 atm – 132.10 Pa

V = 33.47 m/s
q = 684.15 Pa
$P_{Stagnation, P}$ = 1 atm + 552.04 Pa

So

$P_{Stagnation, \infty} \approx P_{Stagnation, P}$

- $\alpha = 10^o$

Free stream values

ρ = 1.225 kg/m^3
P_∞ = 1 atm
V_∞ = 30 m/s
q_∞ = 551.25 Pa
$P_{Stagnation, \infty}$ = 1 atm + 551.25 Pa

At some point P far from boundary layer

P_{Static} = 1 atm – 250 Pa
V = 36.2 ms^{-1}
q = 801.6 Pa
$P_{Stagnation, P}$ = 1 atm + 551.6 Pa

So

$P_{Stagnation, \infty} \approx P_{Stagnation, P}$

Verification of Bernoulli's Theorem along a streamline inside Boundary layer

- $\alpha = 3^o$

For point 1 inside boundary layer

ρ = 1.225 kg/m^3
$P_{Static, 1}$ = 1 atm – 72.50 Pa
V_1 = 29.71 m/s
q_1 = 540.64 Pa
$P_{Stagnation, 1}$ = 1 atm + 613.14 Pa

For point 2 inside boundary layer

$P_{Static, 2}$ = 1 atm – 253.97 Pa
V_2 = 36.04 m/s
q_2 = 795.56 Pa
$P_{Stagnation, 2}$ = 1 atm + 541.59 Pa

So

$P_{Stagnation, 1} \neq P_{Stagnation, 2}$

- $\alpha = 10^o$

For point 1 inside boundary layer

ρ	$= 1.225\ kg/m^3$
$P_{Static,\ 1}$	$= 1\ atm - 137.72\ Pa$
V_1	$= 30.68\ m/s$
q_1	$= 576.52\ Pa$
$P_{Stagnation,\ 1}$	$= 1\ atm + 438.8\ Pa$

For point 2 inside boundary layer

$P_{Static,\ 2}$	$= 1\ atm - 491.47\ Pa$
V_2	$= 40.43\ m/s$
q_2	$= 1001.18\ Pa$
$P_{Stagnation,\ 2}$	$= 1\ atm + 509.71\ Pa$

So

$$P_{Stagnation,\ 1} \neq P_{Stagnation,\ 2}$$

Acknowledgement

- Videos provided by sir
- My friends Rohan Chowdhury and Binay Kumar Shaw
- Special thanks to Rohan. He’s been a great help

References

- Ansys Innovation courses website
- Videos sir provided