AE 244 - Glider Design Assignment

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1. Airfoil Creation and Simulation Mesh Generation:

    1.1 NACA airfoil

    Airfoil – NACA 7411

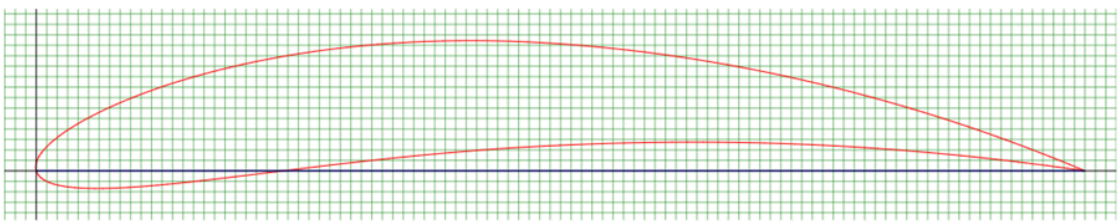
    Max Chamber = 7.3 %

    Thickness = 11 %

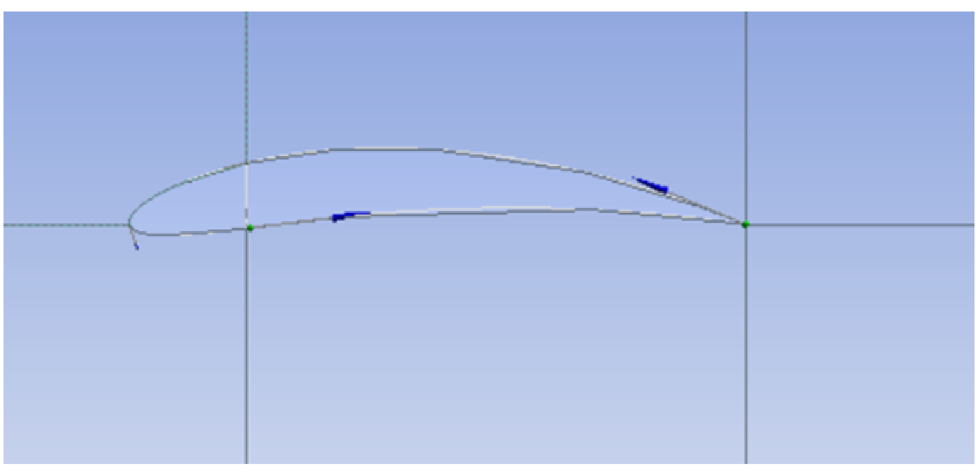
    Max Chamber Position = 47.5 %

    Chord Length = 1 metre

    1.2 Airfoil shape plot and image of airfoil CAD



Name = NACA 7411 Airfoil M=7.3% P=47.5% T=11.0%
Chord = 1000mm Radius = 0mm Thickness = 100% Origin = 0% Pitch = 0°



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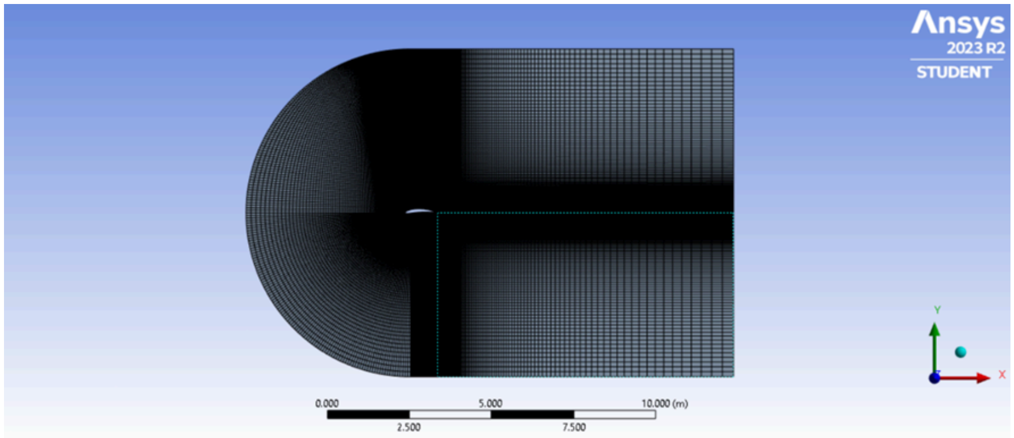
1.3 Image of structured mesh



Airfoil Simulation and Results

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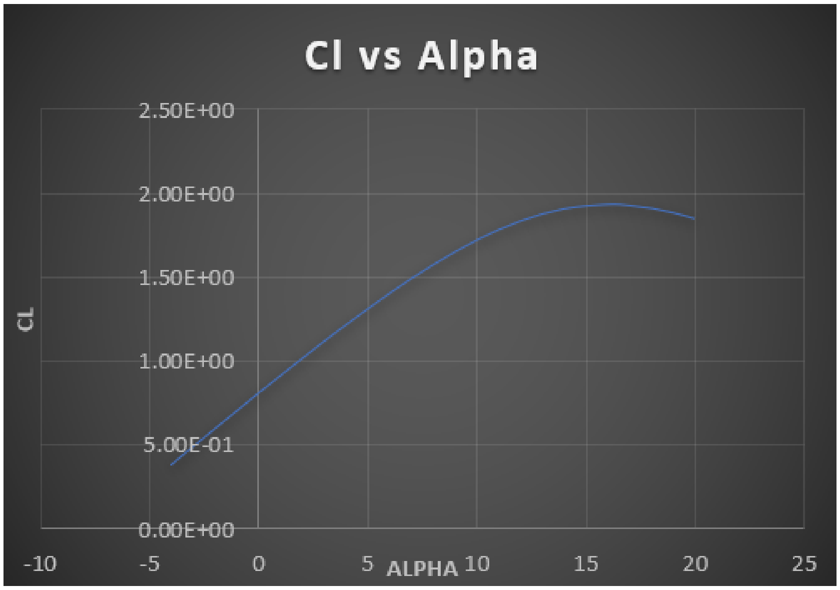
2. Airfoil Simulation and Results:

2.1 Cl vs α plot

Cl	alpha
3.81E-01	-4
5.97E-01	-2
8.09E-01	0
1.02E+00	2
1.12E+00	3
1.22E+00	4
1.40E+00	6
1.57E+00	8
1.72E+00	10
1.84E+00	12
1.91E+00	14
1.93E+00	16
1.93E+00	17
1.91E+00	18
1.88E+00	19
1.85E+00	20

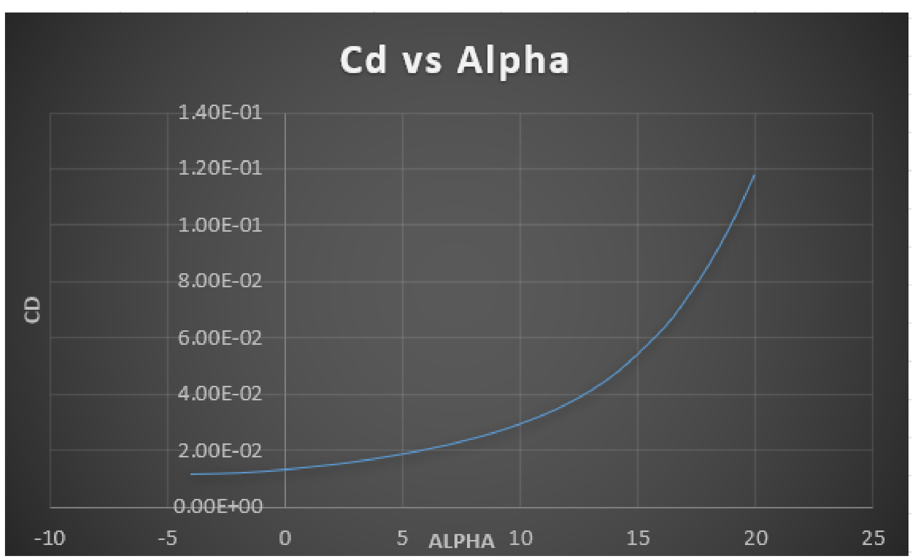


Interpretations:

1. At small angles of attack, the lift coefficient typically increases linearly with the angle of attack. This linear region represents the region of "linear lift behaviour" where the lift coefficient increases proportionally with the angle of attack.
2. As the angle of attack continues to increase beyond a certain point, the lift coefficient reaches its maximum value, after which it starts to decrease rapidly.
3. The stall occurs when the airflow over the upper surface of the airfoil separates from the surface, leading to a disruption in the generation of lift and an increase in drag.

2.2 Cd vs α plot:

Cd	alpha
1.15E-02	-4
1.19E-02	-2
1.31E-02	0
1.49E-02	2
1.59E-02	3
1.72E-02	4
2.02E-02	6
2.41E-02	8
2.93E-02	10
3.65E-02	12
4.69E-02	14
6.24E-02	16
7.30E-02	17
8.55E-02	18
1.01E-01	19
1.18E-01	20



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Interpretations:

1. At very low angles of attack, the CD vs. alpha curve is typically relatively flat.This is because the airfoil experiences minimal drag in these conditions.
2. As the angle of attack continues to increase, the CD vs. alpha curve exhibit a sudden increase, indicating the onset of stall.
3. In the stall region, the airfoil experiences a significant increase in drag as flow separation occurs, leading to a decrease in lift and an increase in drag.

2.3

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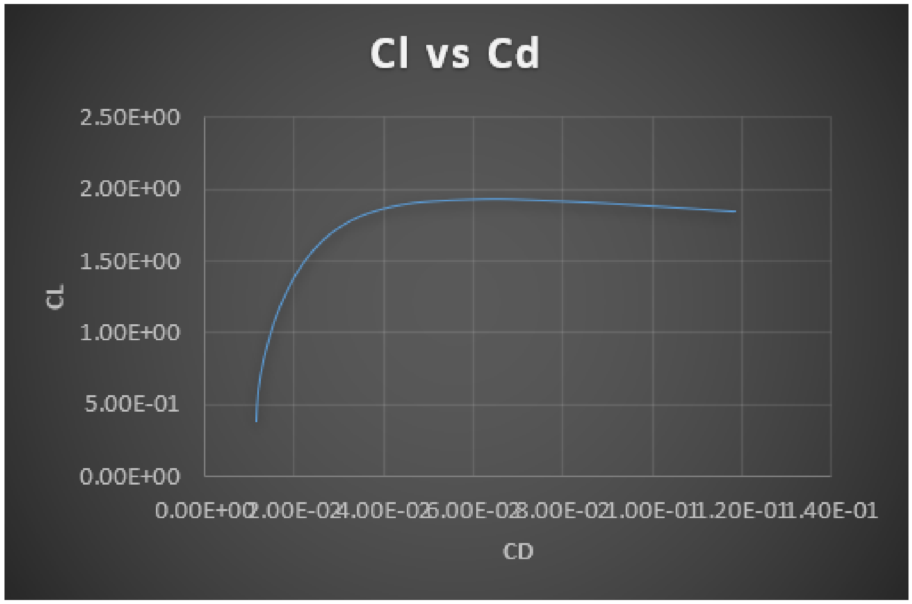
2.4 Cl vs Cd plot:

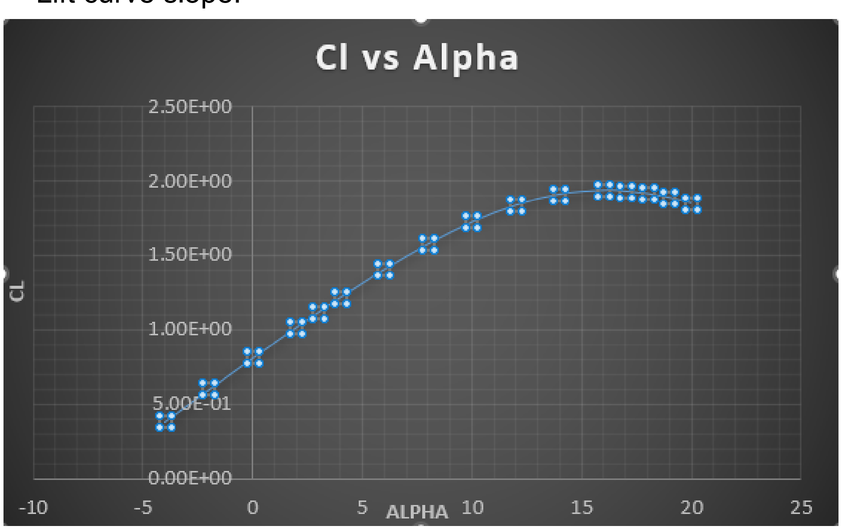
Cl	Cd
3.81E-01	1.15E-02
5.97E-01	1.19E-02
8.09E-01	1.31E-02
1.02E+00	1.49E-02
1.12E+00	1.59E-02
1.22E+00	1.72E-02
1.40E+00	2.02E-02
1.57E+00	2.41E-02
1.72E+00	2.93E-02
1.84E+00	3.65E-02
1.91E+00	4.69E-02
1.93E+00	6.24E-02
1.93E+00	7.30E-02
1.91E+00	8.55E-02
1.88E+00	1.01E-01
1.85E+00	1.18E-01



Interpretations:

1. At lower angles of attack, both CL and CD typically increase linearly with increasing angle of attack.
2. There is usually a point on the graph where the lift-to-drag ratio (CL/CD) is maximised. This point corresponds to the optimum angle of attack for the airfoil.
3. Beyond the optimum angle of attack, both CL and CD may deviate from their linear trends, with CL potentially reaching a peak and then decreasing while CD increases rapidly.
4. After stall occurs, the lift-to-drag ratio decreases significantly as both CL and CD increase, indicating a less efficient operating condition

2.5 Lift curve slope:



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Considering two points to calculate the slope of Cl VS Alpha graph

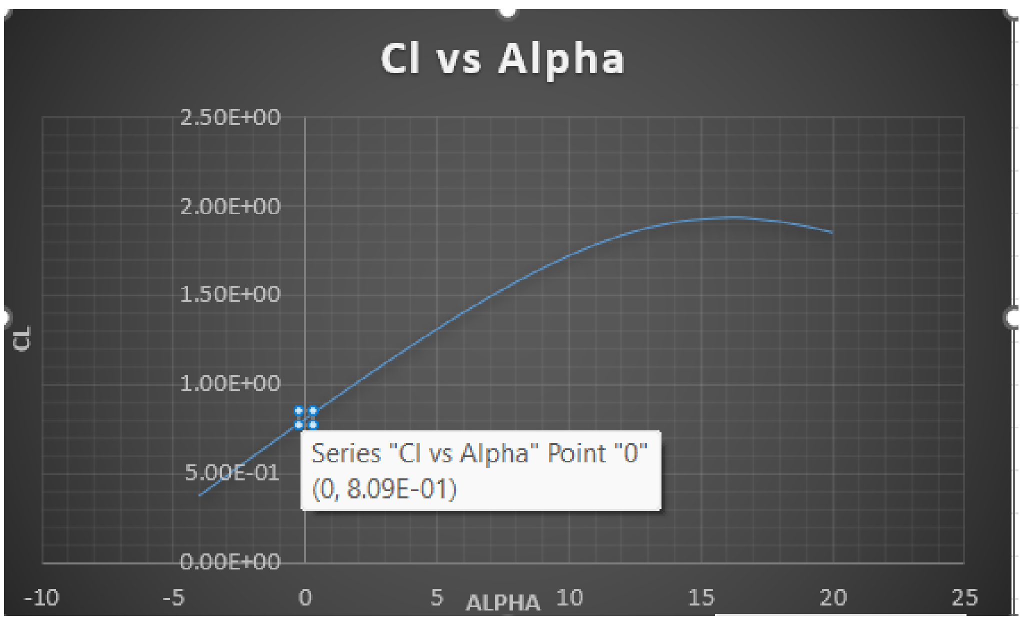
X1 = -4 Y1 = 0.38129
X2 = 10 Y2 = 1.7215

Slope = $(Y2 \ - Y1/X2 - X1)$ = 9.57 * 10^-2

Interpretations:

1. For smaller angles of attack the Cl is directly proportional to alpha and that is why its a linear for smaller angles.
2. As the angle increases the slope changes and it starts reducing and becoming plat and then suddenly it becomes negative due to separation of flow occur at surface of airfoil.

2.6 Y-Intercept



Y-intercept = 0.809

Interpretations:

1. Here the y-intercept on this curve represents the coefficient of lift at zero angle of attack, it is an unsymmetric airfoil and its giving lift at zero angle of attack.
2. If we have symmetric airfoil it may give zero lift at zero angle of attack.
3. Another thing to be noted is that it also gives lift at a negative angle of attack which increases the lift efficiency of aircraft.

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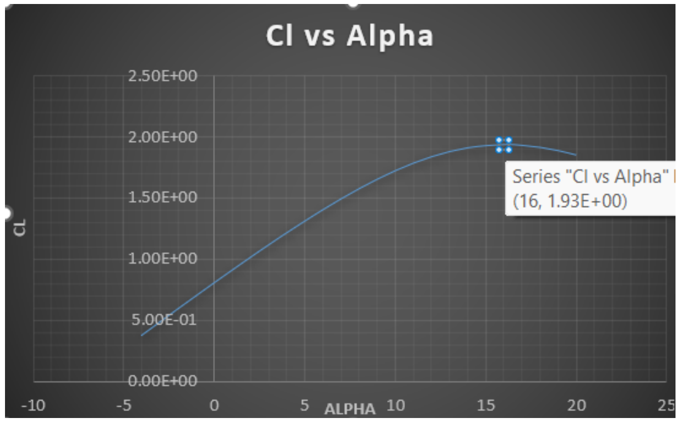
2.7 Stall angle

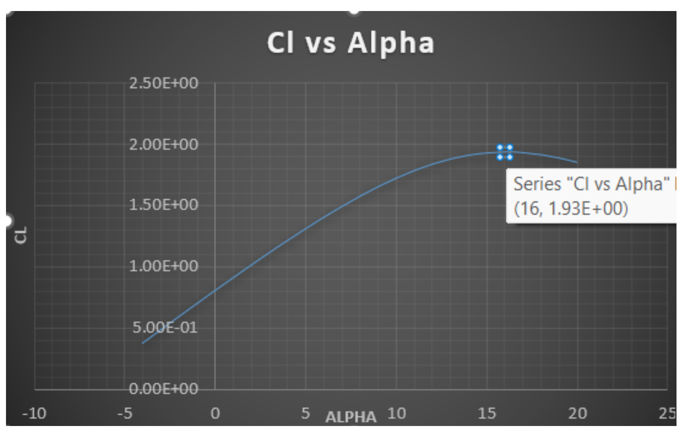


Max lift is obtained at alpha = 16 degrees. So the stall angle is 16 degrees.

Interpretations:

1. The stall angle of attack marks the point at which the airflow over the airfoil or wing separates from its surface.
2. This separation leads to a disruption in the generation of lift and an increase in drag.
3. This increase in drag can lead to a decrease in airspeed, which may exacerbate the stall condition.

2.8 Maximum Cl



Cl max = 1.9329

Interpretations:

1. The max CL represents the peak lift coefficient that the airfoil or wing can achieve before reaching the stall angle of attack.
2. The max CL is a key factor in determining the aircraft's takeoff and landing performance. Higher max CL values allow for shorter takeoff and landing distances.

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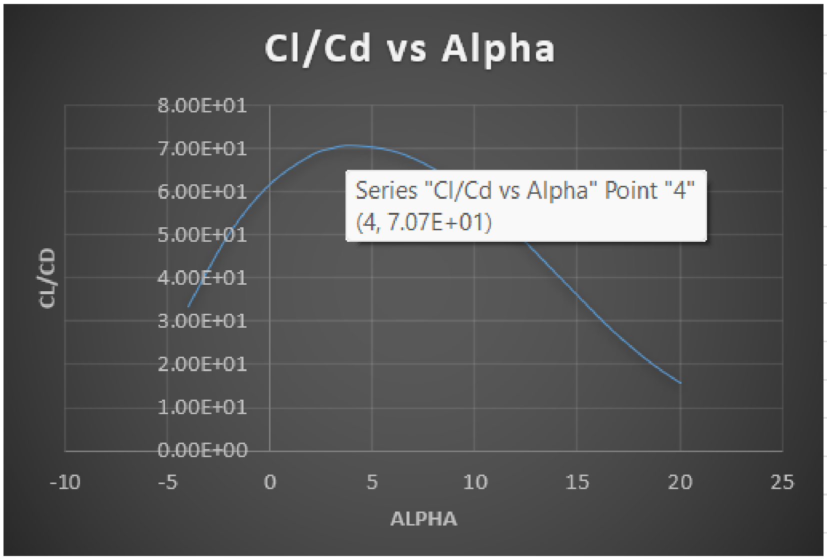
3. The max CL also influences the aircraft's manoeuvrability, particularly during low-speed flight regimes such as during approach and landing.

2.9 Maximum Cl/Cd:



Max Cl/Cd = 70.71
At Angle Alpha = 4 degree

Interpretation:

1. The maximum lift-to-drag ratio (CL/CD) is a key parameter in aerodynamics that represents the efficiency of an airfoil or wing in generating lift relative to the drag it produces.
2. At a specific angle of attack where the lift-to-drag ratio is maximised. This angle of attack represents the most favourable operating condition for the airfoil or wing in terms of aerodynamic efficiency.
3. The max CL/CD ratio is particularly important for aircraft operating over long distances.Aircraft with higher max CL/CD ratios can achieve greater range and endurance by minimising drag and maximising lift.

Flow Study at 3 degree and 10 degree angle of attack

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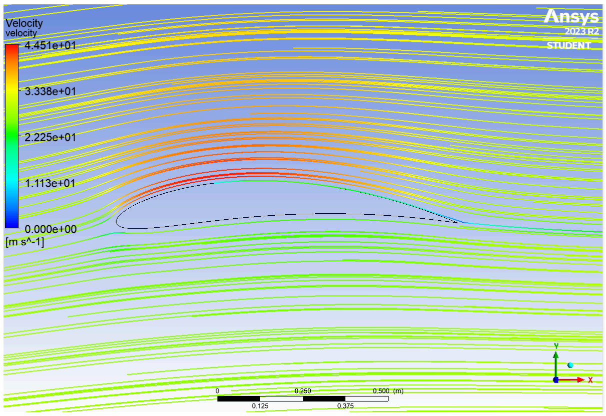
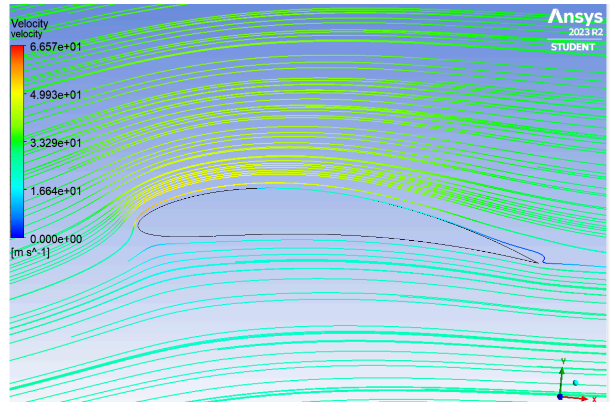
3. Flow Study (at α = 3 and 10)

3.1.1 Streamline plots



α = 3

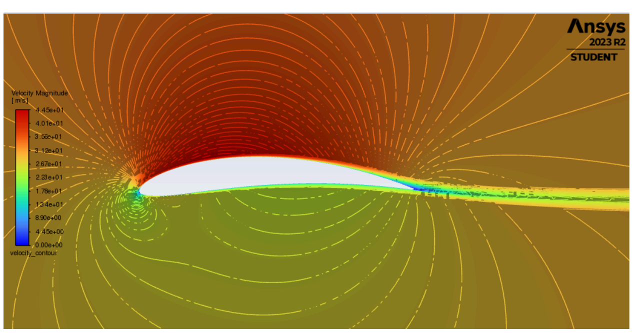


α = 10

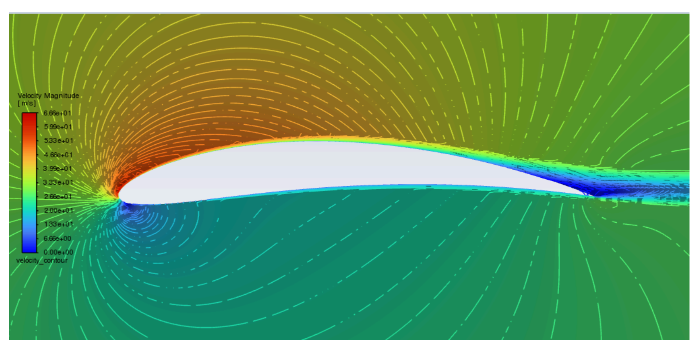
Interpretations:

1. Streamlines indicate the direction in which fluid elements are moving at each point in a flow field.
2. Closer streamlines represent regions of higher velocity gradient, while more widely spaced streamlines indicate lower velocity gradient
3. As it is visible streamline is near on the top of the upper surface which means that they have high velocity as compared to lower surface streamlines are less dense which means they have less velocity.
4. The Separation point at alpha 10 degree has reached earlier as compared to 3 degree.

3.1.2 Velocity field magnitude contour plots



α = 3

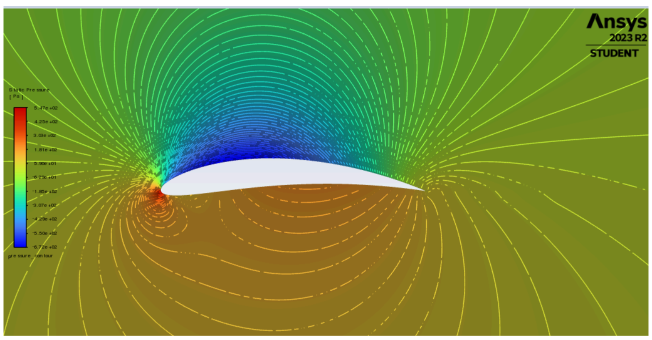


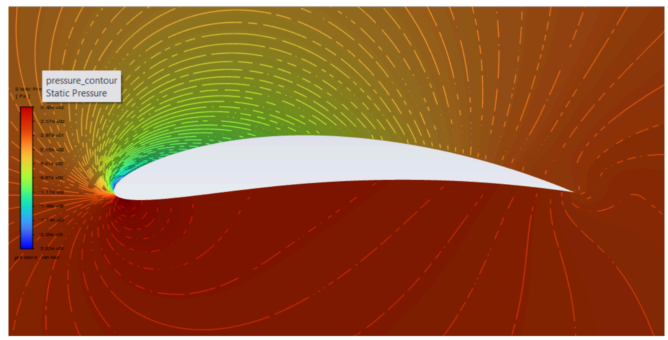
α = 10

Interpretations:

1. The stagnation point, where the flow velocity drops to zero, is likely to shift closer to the leading edge of the airfoil at 10 degrees compared to 3 degrees.
2. At angle of attack 10 degree the velocity above the upper surface has less velocity at the tip of the airfoil as compared to 3 degree this is due to reverse flow to air.
3. At 3 degrees of angle of attack, the velocity contours are likely to show a smoother and more uniform velocity distribution over the airfoil surface compared to 10 degrees.
4. This is because the lower angle of attack allows for a more attached flow, resulting in less separation and a more evenly distributed velocity profile.

3.1.3 Pressure field contour plots





α = 3 α = 10

Interpretations:

1. This is the Pressure contour for angle of attack 3 degree and 10 degree. The stagnation point in this contour is shifted away from the leading tip of the airfoil for 10 degree as compared to 3 degrees.
2. For the 10 degree contour plot the pressure below the airfoil is large as compared to the pressure below the airfoil for 3 degrees.
3. At 10 degrees of angle of attack, the pressure contours may exhibit steeper pressure gradients, particularly near the leading edge of the airfoil. This is indicative of higher-pressure gradients and potentially stronger suction forces on the upper surface due to the higher angle of attack.

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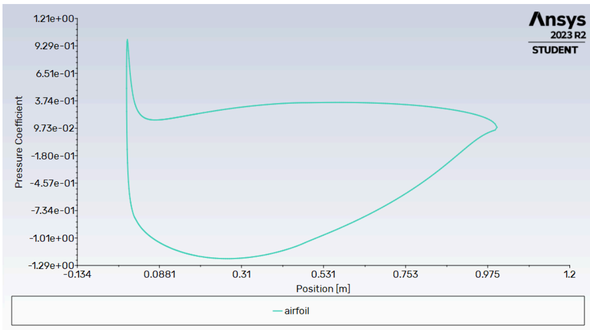
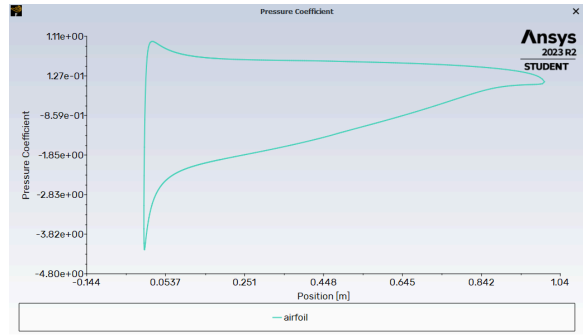
3.4 Coefficient of Pressure plots along airfoil surface



α = 3

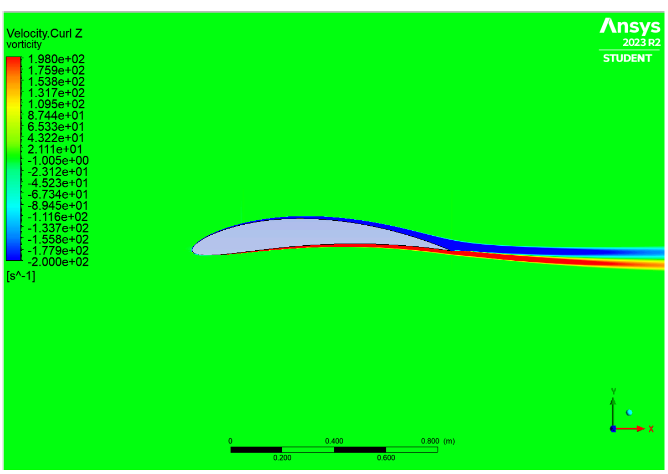


α = 10

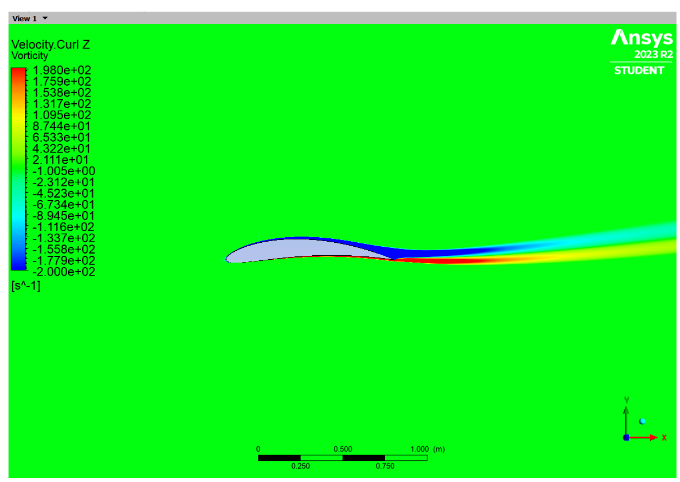
Interpretations:

1. At 10 degrees of angle of attack, the Cp plot may exhibit steeper gradients and larger variations in pressure compared to 3 degrees.
2. The Cp values near the leading edge are more negative (indicating lower pressures) on the upper surface, reflecting stronger suction forces due to increased angle of attack, and decrease more rapidly along the upper surface before reaching regions of flow separation and higher pressures.
3. At the tail of the airfoil the coefficient is more in 10 degree as compared to 3 degree.
4. This is happening because the lift at 10 degree is more as compared to 3 degree because the pressure difference between the upper layer and the lower layer is more in 10 degree as compared to 3 degree.

3.5 Vorticity field contour plots



α = 3



α = 10

Interpretations:

1. At 3 degrees of angle of attack, the magnitude of vorticity may be relatively low and concentrated in specific regions of the flow field, such as near the wingtips or regions of flow separation. The vorticity may be generated primarily by shear and flow curvature effects.

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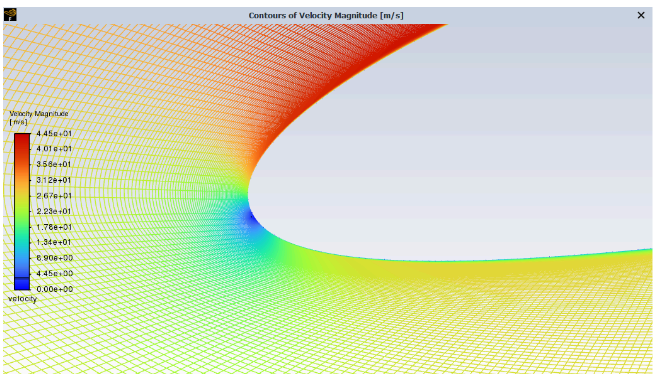
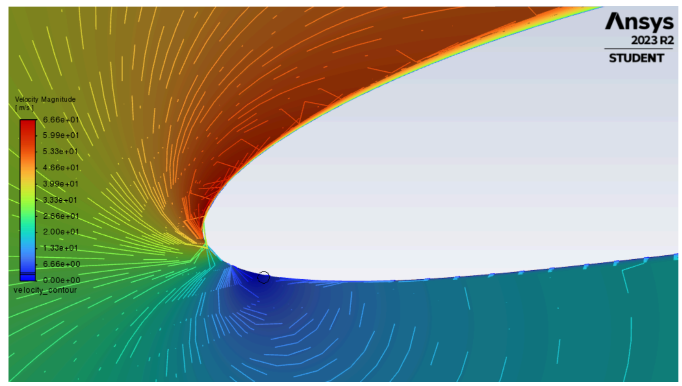
2. At 10 degrees of angle of attack, the magnitude of vorticity is likely to be higher and more widespread throughout the flow field compared to 3 degrees. The higher angle of attack results in stronger flow curvature, shear, and potentially more significant separation, leading to increased vorticity generation.
3. The distribution and magnitude of vorticity at different angles of attack directly influence the aerodynamic performance of the airfoil or wing. Higher levels of vorticity, as typically observed at higher angles of attack like 10 degrees, can lead to increased lift-induced drag and affect overall aerodynamic efficiency.

3.6 Approximate location of stagnation and flow separation points in airfoil surface

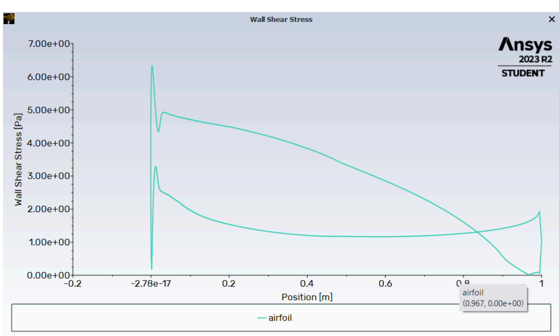




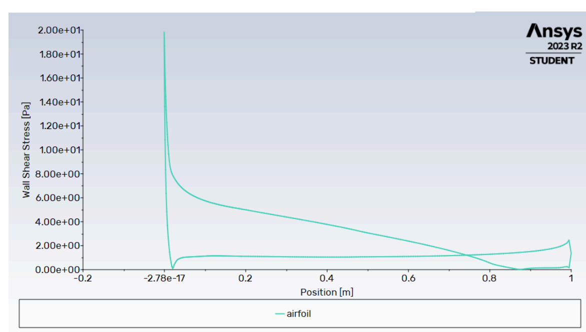
Location of stagnation at α = 3 degree
(3.1154628,3.5605288)

Location of stagnation at α = 10 degree
[2.6631896, 3.3289871]

Wall shear stress vs Position plot



α = 3 degree



α = 10 degree

Interpretation:

1. We can find the point of stagnation and the point of separation on the airfoil by plotting the wall shear stress vs chord length plot. Point where the wall shear stress is max is the point of stagnation and the point where wall shear stress is zero is the point of flow separation..
   For 10 degree separation point = 0.876 , For 3 degree separation = 0.969.

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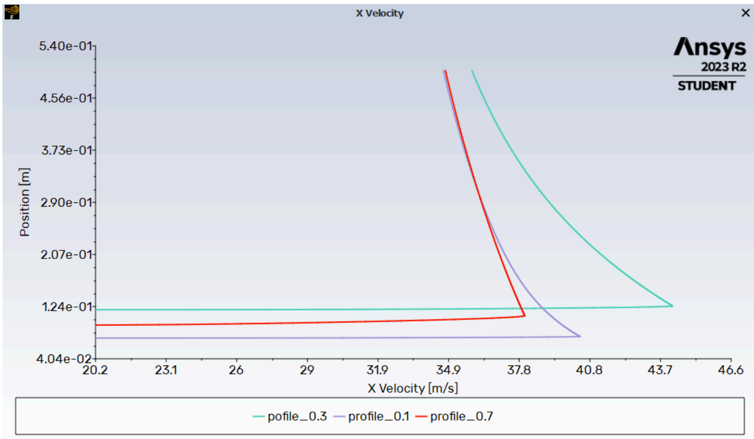
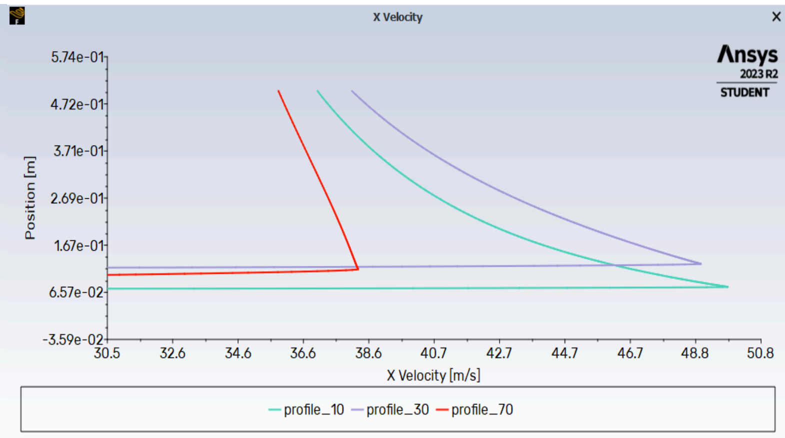
3.7 Upper surface boundary layer velocity profile at x/c = 10%, 30% and 70%





α = 3 degree α = 10 degree

Interpretation:

1. Velocity overshoot results from the need for the boundary layer flow to accelerate to balance pressure forces and maintain mass conservation. Flow separation creates reversed or stagnant flow near the surface, leading to localised regions of increased velocity, known as velocity overshoot, before the flow reattaches or transitions to the free stream.

3.8 Estimate of boundary layer thickness at the three locations:

	Boundary Layer thickness	
	3 degree	10 degree
10%	0.0056	0.0038
30%	0.006	0.06
70%	0.0133	0.0187

4. Conservation laws (at α = 30)

4.1 Net mass flow rate into the control volume.

I have created rectangular control volume having coordinates (-2,-2),(-2,2),(2,2) and (2,-2)

```
                Mass Flow Rate                [kg/s]
-------------------------------- --------------------
                       line-10            -146.66687

                Mass Flow Rate                [kg/s]
-------------------------------- --------------------
                       line-11             7.9904999

                Mass Flow Rate                [kg/s]
-------------------------------- --------------------
                       line-12             -146.4414

                Mass Flow Rate                [kg/s]
-------------------------------- --------------------
                       line-13             7.7618909
```

Mass flow rate = 146.66687 -146.4414 - 7.9904999 + 7.7618909 = 0.003139