AE-244
Assignment 1

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22B0078
2nd Year Aerospace

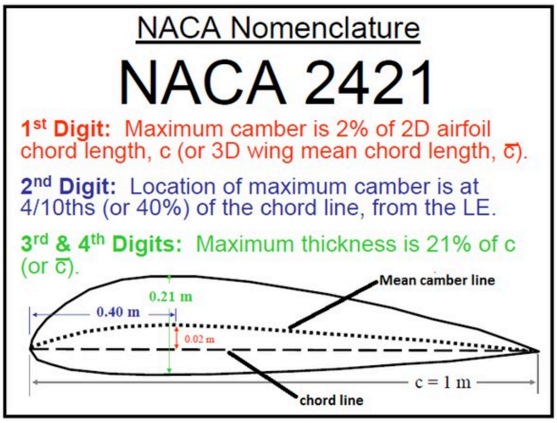
Airfoil Creation and Simulation Mesh Generation

▸ Creating Airfoil →

My roll no. Is 22b0078
And according to assignment,

Maximum camber = 7.8%
Thickness= 22/2 = 11%
Maximum camber position = (22+78)/2 = 50%
Chord Length = 1m

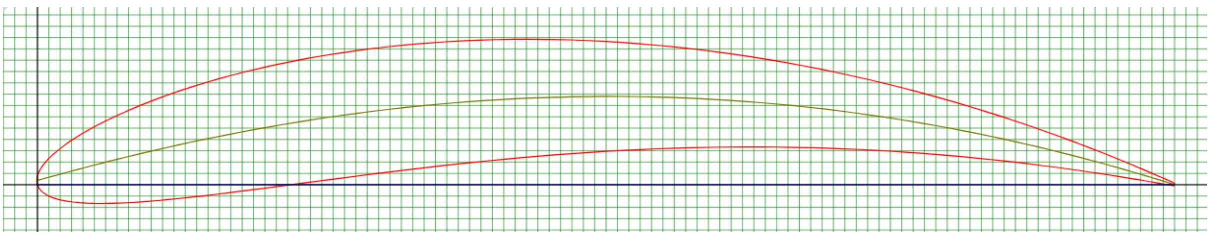
NACA Airfoil Nomenclature



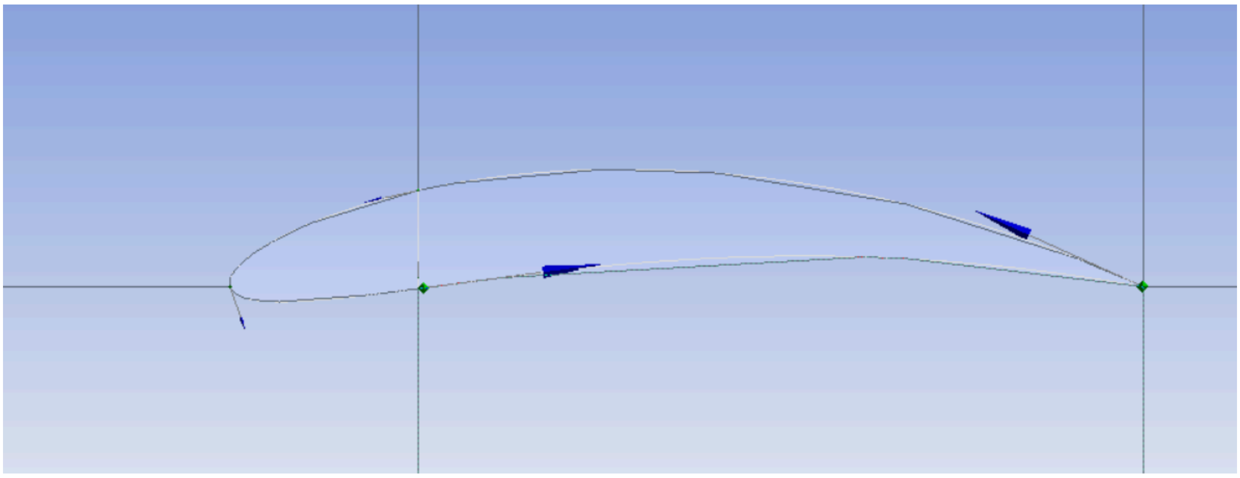
So,
Based on this nomenclature and our airfoil data above,

Our Airfoil is **NACA 7511**

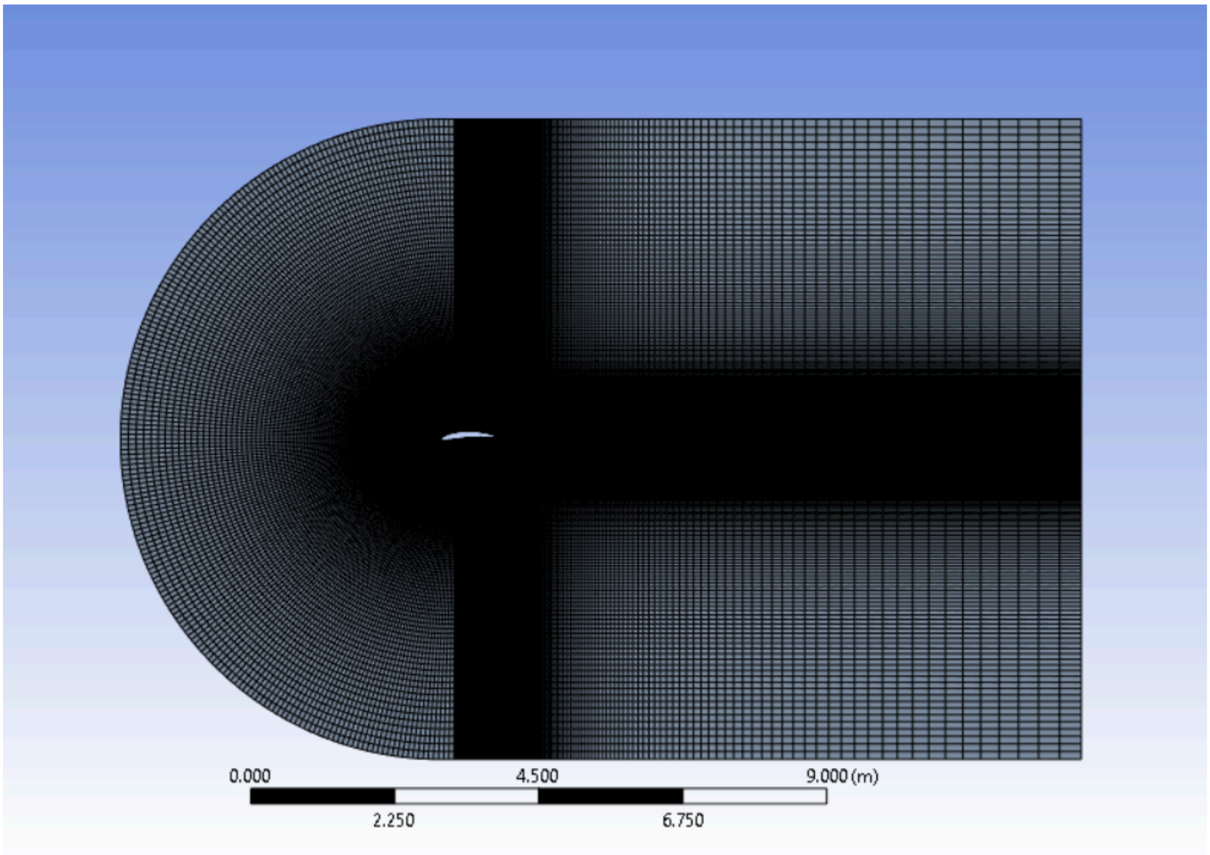
▸ Airfoil shape plot→



- CAD Image of Airfoil→



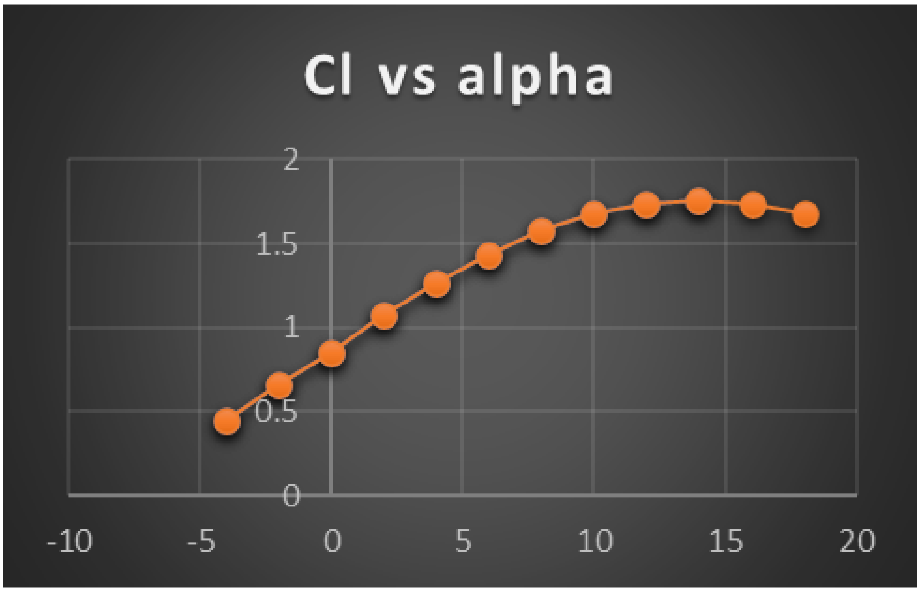
- Image of structured mesh→



Airfoil Simulation and Results

▸ Cl vs α plot

Alpha	Cl
-4	0.4451
-2	0.6641
0	0.8477
2	1.0811
4	1.2696
6	1.4365
8	1.5745
10	1.6777
12	1.7403
14	1.7595
16	1.7364
18	1.6771



Interpretations→

1. Linear Region: Shows the linear relationship between lift coefficient and angle of attack, indicating predictable lift changes at small angles.

2. Stall Region: Indicates a sharp drop in lift coefficient beyond a critical angle, signifying airflow separation and reduced lift efficiency.

3. Maximum Lift Coefficient: Represents the peak lift capability of the airfoil before stall, crucial for determining optimal performance and manoeuvrability.

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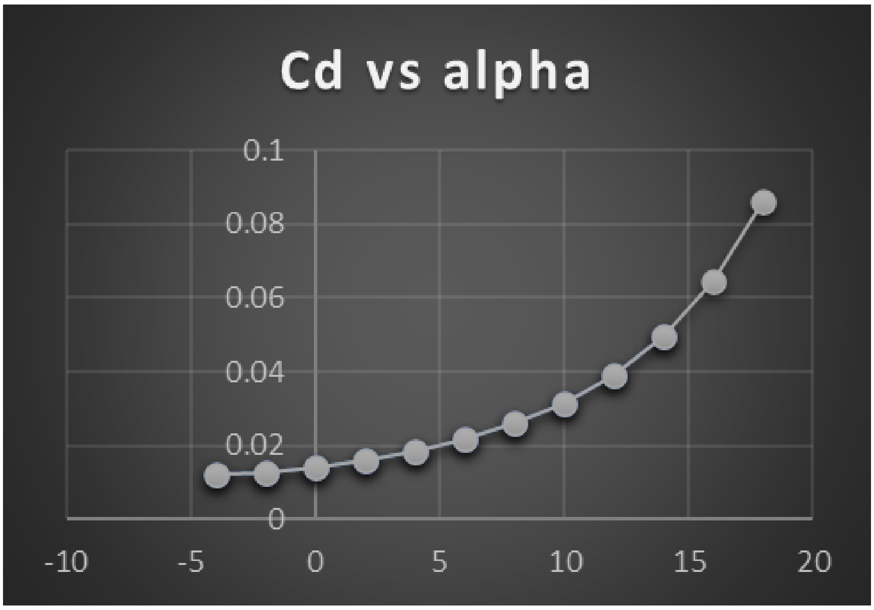
▸ Cd vs α plot

Alpha	Cd
-4	0.012
-2	0.0126
0	0.0139
2	0.0159
4	0.0184
6	0.0217
8	0.0259
10	0.0314
12	0.0388
14	0.0494
16	0.0646
18	0.086



Interpretations→

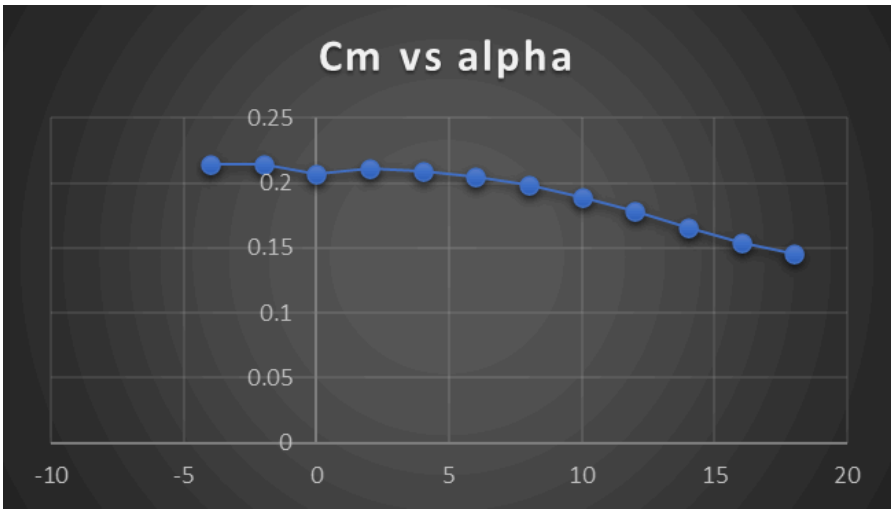
1. It visually represents how the aerodynamic efficiency of an airfoil varies as its angle of attack changes, aiding in the optimization of performance for different flight conditions.

2. The plot offers insights into the airfoil's stall behaviour, highlighting the angle at which drag sharply increases due to flow separation, leading to loss of lift.

3. It helps engineers understand the trade-off between lift and drag at different angles of attack, crucial for designing efficient aircraft wings and propellers.

▸ Cm vs α plot

Alpha	Cm
-4	0.2139
-2	0.214
0	0.207
2	0.2117
4	0.209
6	0.2048
8	0.1983
10	0.1895
12	0.1784
14	0.1661
16	0.1544
18	0.1456



Interpretations→

1. The Cm vs alpha plot for an airfoil depicts the variation of pitching moment coefficient (Cm) with angle of attack (alpha), indicating the airfoil's tendency to rotate nose-up or nose-down.

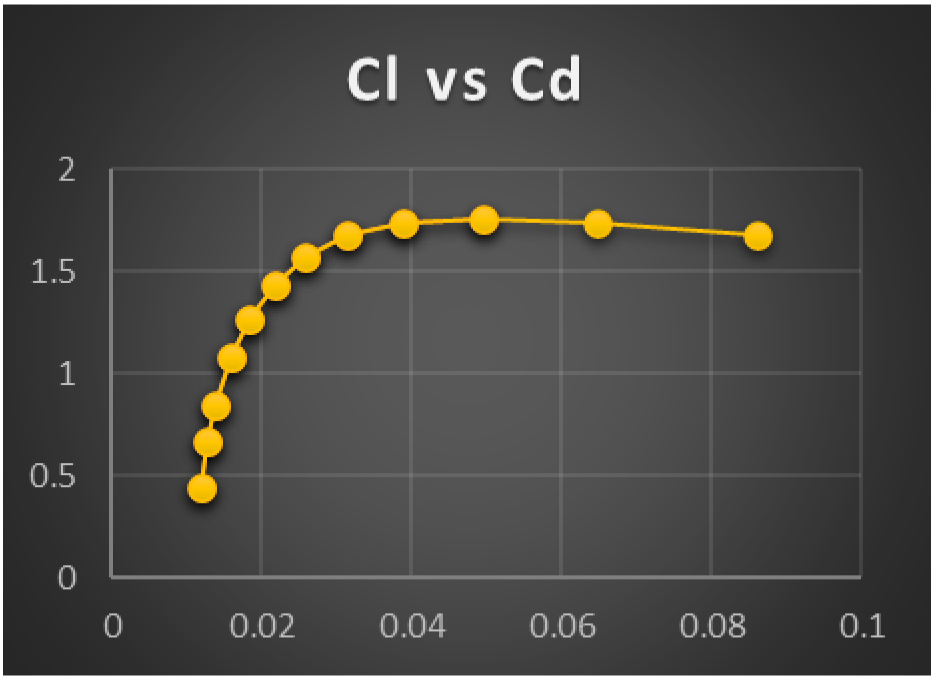
2. It provides critical information about the airfoil's stability characteristics, showcasing how its pitching moment changes with changes in angle of attack, essential for aircraft stability and control analysis.

3. This plot aids in understanding the aerodynamic balance of an airfoil, revealing regions of positive or negative pitching moments relative to the centre of gravity, influencing the aircraft's trim and manoeuvrability.

4. Engineers utilise this plot to assess the airfoil's behaviour near stall conditions, where abrupt changes in pitching moment can impact the aircraft's controllability and handling qualities.

▸ Cl vs Cd plot

Cl	Cd
0.4451	0.012
0.6641	0.0126
0.8477	0.0139
1.0811	0.0159
1.2696	0.0184
1.4365	0.0217
1.5745	0.0259
1.6777	0.0314
1.7403	0.0388
1.7595	0.0494
1.7364	0.0646
1.6771	0.086



Interpretations→

1. The Cl vs Cd plot for an airfoil illustrates the trade-off between lift coefficient (Cl) and drag coefficient (Cd), indicating the airfoil's efficiency in generating lift relative to the drag it produces.

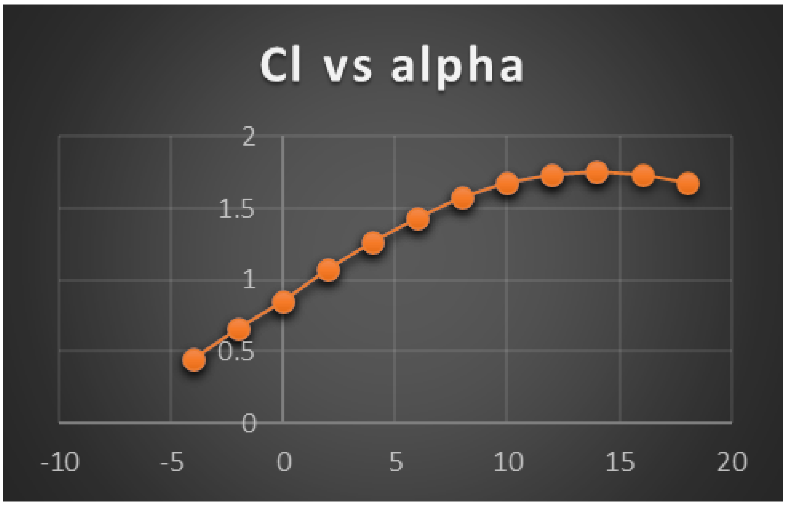
2. It helps in determining the airfoil's optimal angle of attack for maximum lift-to-drag ratio, crucial for achieving the best performance and fuel efficiency in aircraft and wind turbine design.

3. This plot facilitates the selection of airfoils based on specific performance requirements, such as maximising lift for takeoff and minimising drag for cruising, aiding in the design of efficient aerodynamic systems.

4. Engineers use this plot to analyse the aerodynamic performance of airfoils across a range of operating conditions, enabling informed decisions during aircraft design and optimization processes.

▸ Lift Curve Slope

We take two coordinates on the straight line in Cl vs alpha plot to know its straight line slope.



x	y
2	1.0811
0	0.8477

Lift curve slope = 0.1167

Interpretations→

1. The lift curve slope of an airfoil quantifies its ability to generate lift with changes in angle of attack, indicating how efficiently lift increases as the angle changes.

2. It represents the rate at which lift coefficient (Cl) changes with angle of attack (alpha), essential for predicting an airfoil's lift performance and stability characteristics across varying flight conditions.

▸ Y-Intercept-

Y intercept of Cl vs alpha plot = 0.8477

Interpretations→

1. The Y-intercept of an airfoil's lift coefficient curve represents the lift coefficient when the angle of attack is zero, providing a baseline value for lift generation at a neutral angle.

2. It signifies the lift produced by the airfoil when it is aligned with the freestream airflow, serving as a reference point for understanding its aerodynamic behaviour and performance at low angles of attack.

▸ Stall Angle-

According to the Cl vs alpha plot our stall angle is **14 degrees** as we get maximum lift coefficient on 14 degrees of angle of attack.

Interpretations→

1. The stall angle of an airfoil is the critical angle of attack beyond which flow separation occurs, leading to a sudden loss of lift and increased drag.

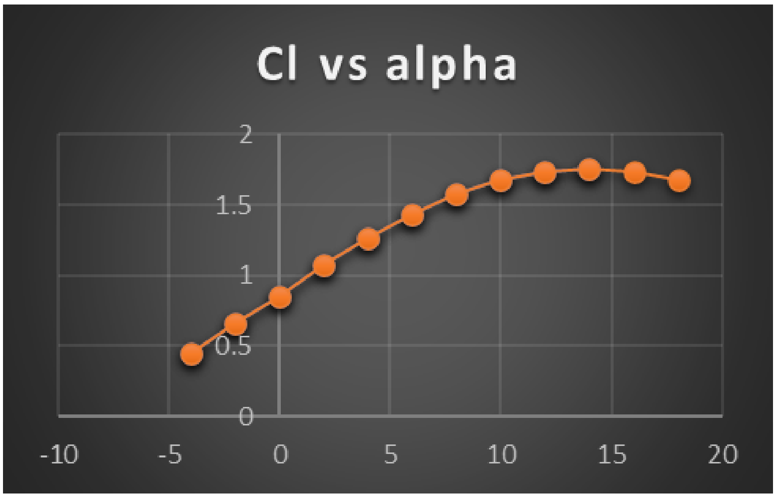
2. It marks the boundary between efficient lift production and aerodynamic stall, impacting aircraft performance and control during manoeuvres, emphasising the importance of avoiding this angle during flight operations.

3. The stall angle of an airfoil is a key parameter indicating the maximum angle of attack before turbulent airflow detachment, crucial for pilots to maintain safe flight and prevent loss of control.

4. It represents the limit of the airfoil's aerodynamic capabilities, influencing aircraft design considerations and operational procedures to ensure safe and efficient flight performance.

▸ Maximum Cl-

The maximum lift coefficient in my case is ***1.7595*** which we got at the angle of attack of 14 degrees.



Interpretations→

1. An airfoil's maximum lift coefficient indicates the maximum lift force per unit area that may be achieved, giving information on the airfoil's capacity to produce lift under particular aerodynamic conditions.

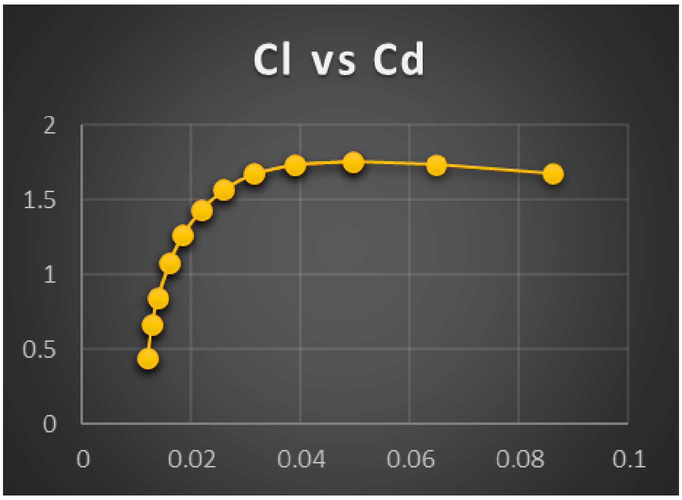
2. It symbolises the airfoil's maximum lift-generating efficiency, which helps engineers optimise wing designs for improved lift performance during the flight's takeoff, landing, and manoeuvring phases.

3. The maximum Cl is a crucial factor in figuring out the stall characteristics of the airfoil since it acts as a guide to prevent achieving extreme angles of attack that might cause a loss of lift and control.

4. It affects factors related to aircraft performance that are necessary for safe and effective operation in a range of flight conditions, including payload capacity, manoeuvrability, and stall speed.

▸ Maximum Cl/Cd-

The maximum value of Cl/Cd in my case is **69.00** which I got at an angle of attack of **4 degrees**.



Interpretations→

1. An airfoil's maximum Cl/Cd ratio is the best design for producing lift while reducing drag, showing the ideal lift-to-drag ratio.

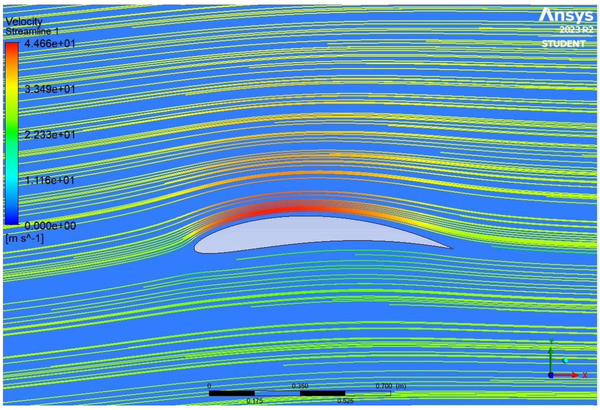
2. It is an important performance metric for aircraft designers, assisting in the identification of airfoils that provide the best balance between aerodynamic efficiency and lift generation–a critical step in getting the highest possible fuel economy and range.

3. The airfoil's greatest lift-to-drag efficiency is shown by the Cl/Cd ratio, which gives pilots important information for maximising glide ratios and endurance while in flight.

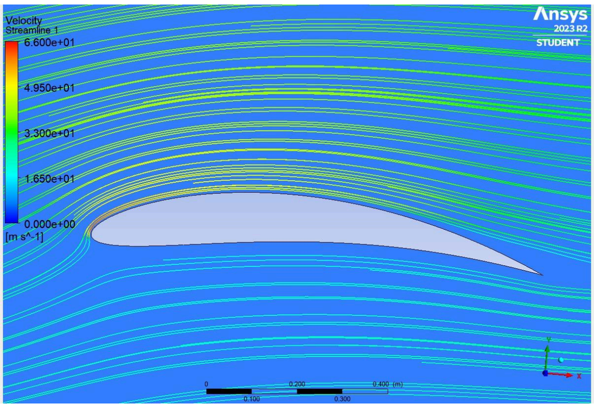
4. It acts as a foundational standard for choosing airfoils for a range of uses, from wind turbine blades to aeroplane wings, guaranteeing best performance and energy conversion efficiency in a variety of aerodynamic systems.

Flow Study (at α = 3° and 10° AOA)

▸ **Streamline plots-**



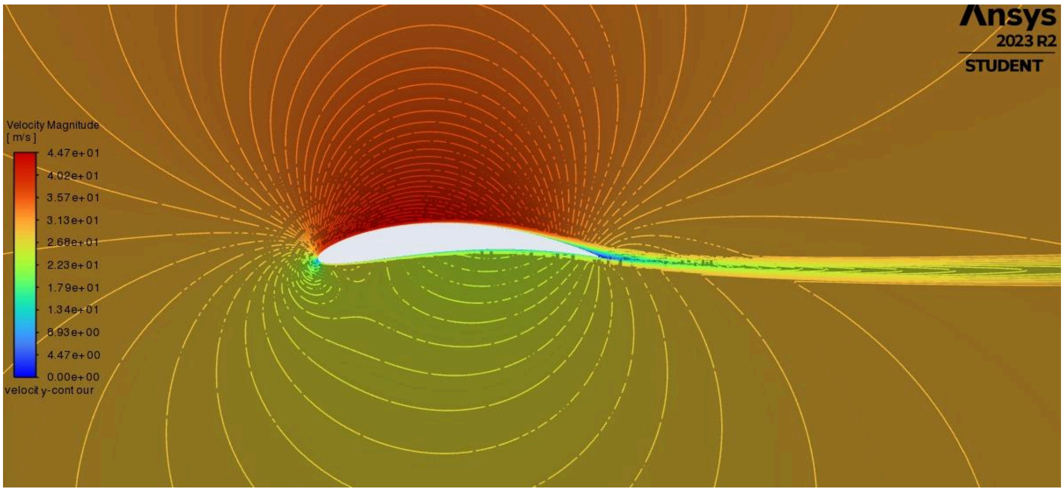
(a) 3°



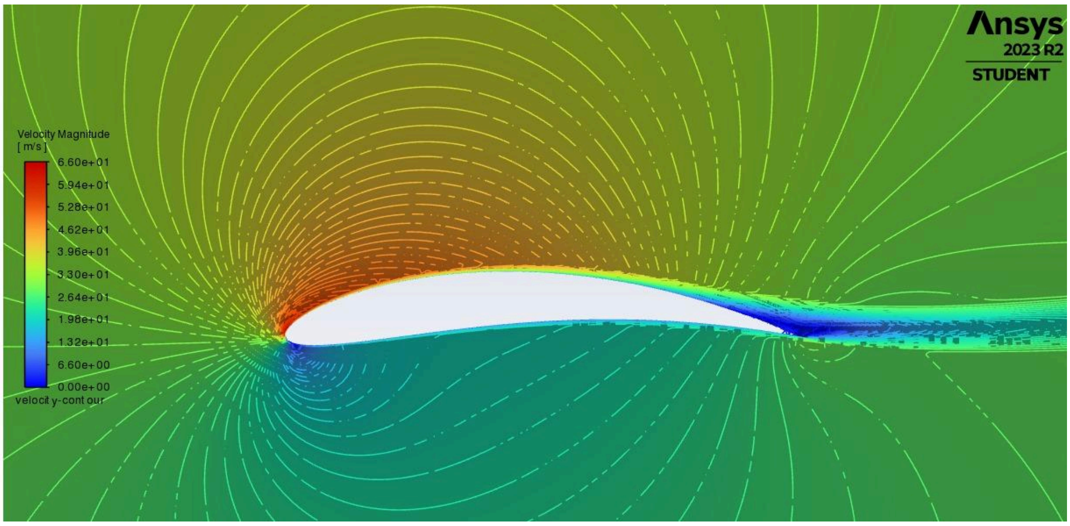
(b) 10°

At 3 degrees angle of attack, the streamlines typically exhibit smooth and attached flow over the airfoil surface, with minimal separation. This configuration suggests efficient lift generation with relatively low drag, indicating favourable aerodynamic performance. In contrast, at 10 degrees angle of attack, the streamlines show signs of flow separation, particularly near the trailing edge of the airfoil. This separation indicates a higher likelihood of turbulent flow and increased drag, potentially leading to reduced lift and stability.

▸ Velocity field magnitude contour plots



(a) 3°

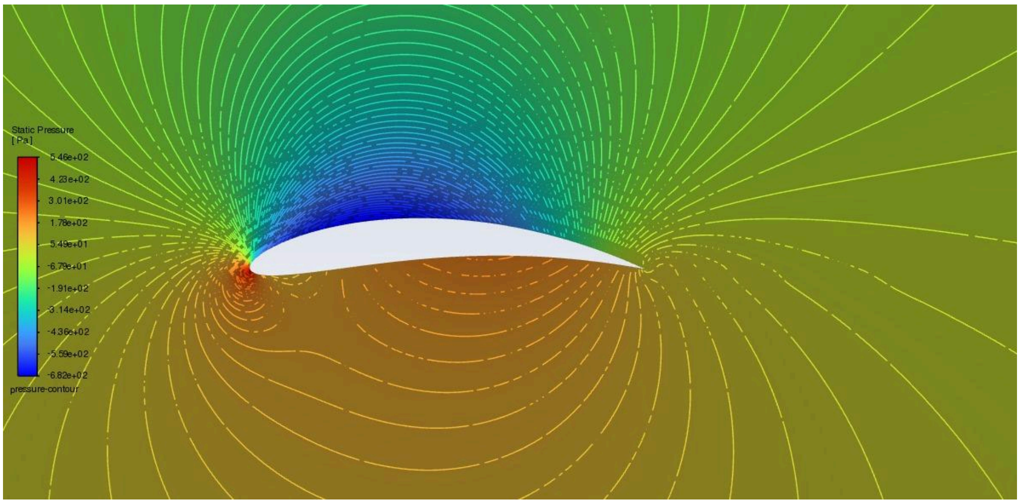


(b) 10°

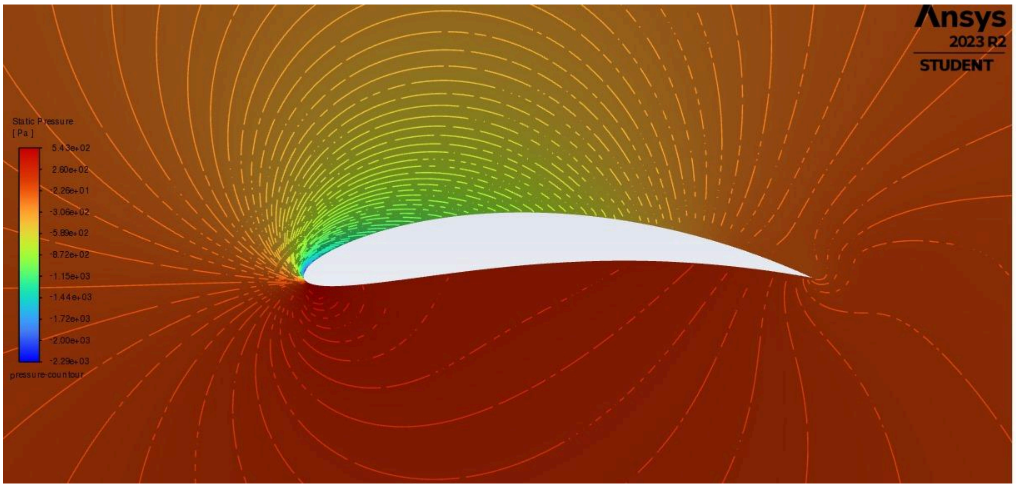
At 3 degrees angle of attack, the contour plots typically reveal smoothly varying velocities along the airfoil surface, with higher velocities observed near the leading edge and lower velocities towards the trailing edge. This pattern indicates relatively undisturbed airflow and efficient lift generation.

However, at 10 degrees angle of attack, the contour plots show regions of higher velocity near the leading edge, followed by areas of lower velocity and even stagnation points near the trailing edge. This distribution suggests flow separation and increased turbulence, resulting in higher drag and potentially reduced lift.

▸ Pressure field contour plots



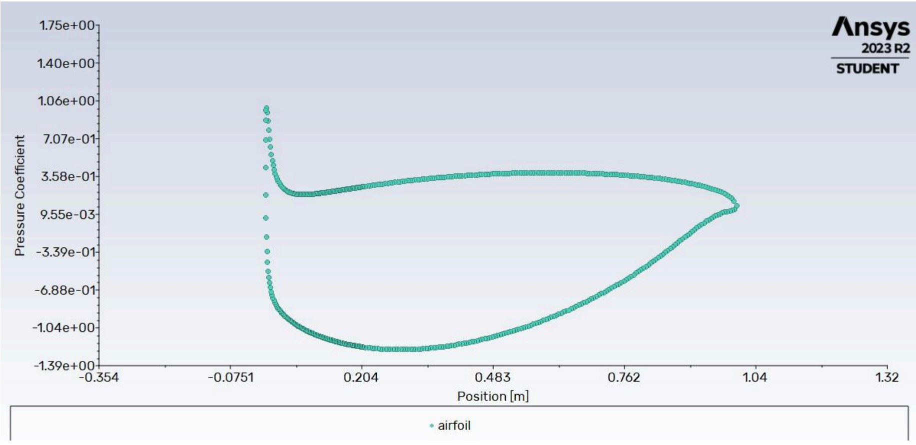
(a) 3°



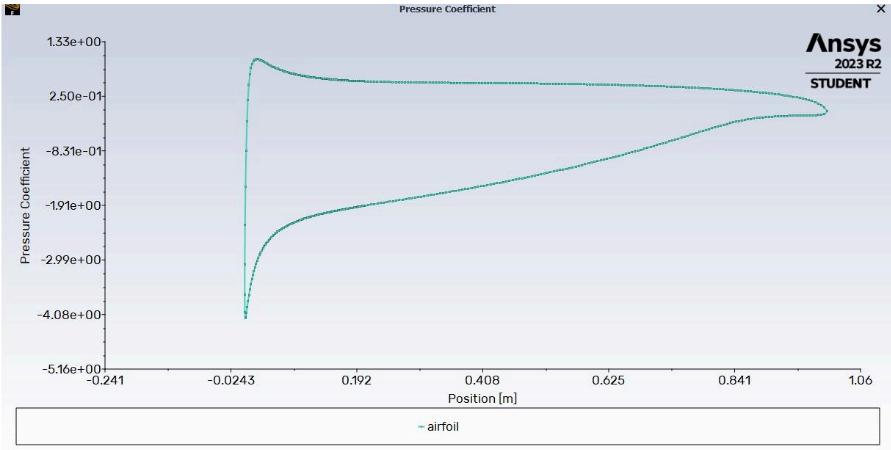
(b) 10°

At 3 degrees angle of attack, the contour plots typically depict higher pressure regions near the leading edge, gradually decreasing along the upper surface and increasing slightly along the lower surface. This distribution indicates favourable lift generation with minimal separation and relatively low drag.
However, at 10 degrees angle of attack, the contour plots reveal regions of lower pressure near the leading edge, followed by sudden drops in pressure and even separation bubbles along the upper surface. This pattern signifies flow separation and increased drag, resulting in reduced lift efficiency.

▸ Coefficient of Pressure plots along airfoil surface



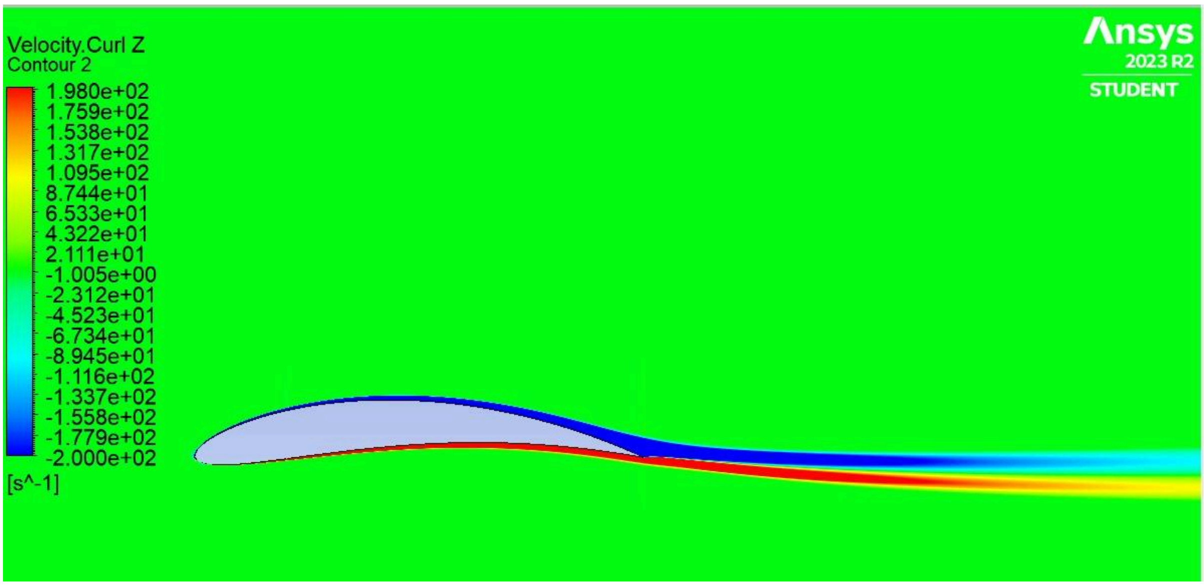
(a) 3°



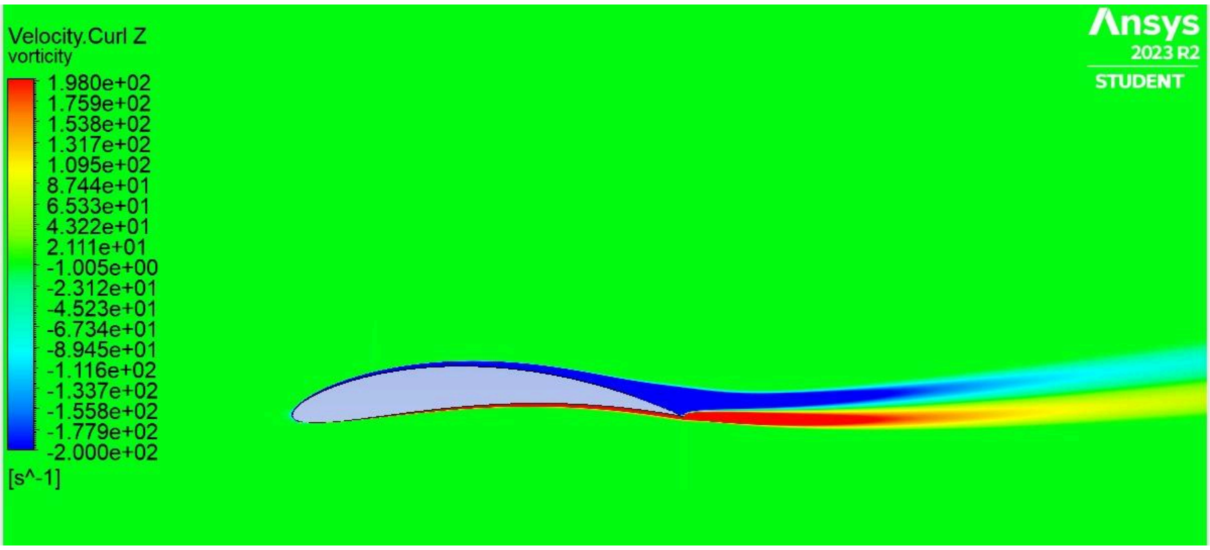
(b) 10°

At 3 degrees angle of attack, the Cp plots typically exhibit a relatively smooth distribution of pressure coefficients along the airfoil surface, with higher values near the leading edge and gradually decreasing values towards the trailing edge. This indicates favourable lift generation with minimal flow separation and efficient airflow attachment.
However, at 10 degrees angle of attack, the Cp plots reveal more significant variations, with lower pressure coefficients near the leading edge and abrupt drops indicating separation bubbles along the upper surface. This pattern signifies flow separation and increased drag, leading to reduced lift efficiency.

▸ Vorticity field contour plots



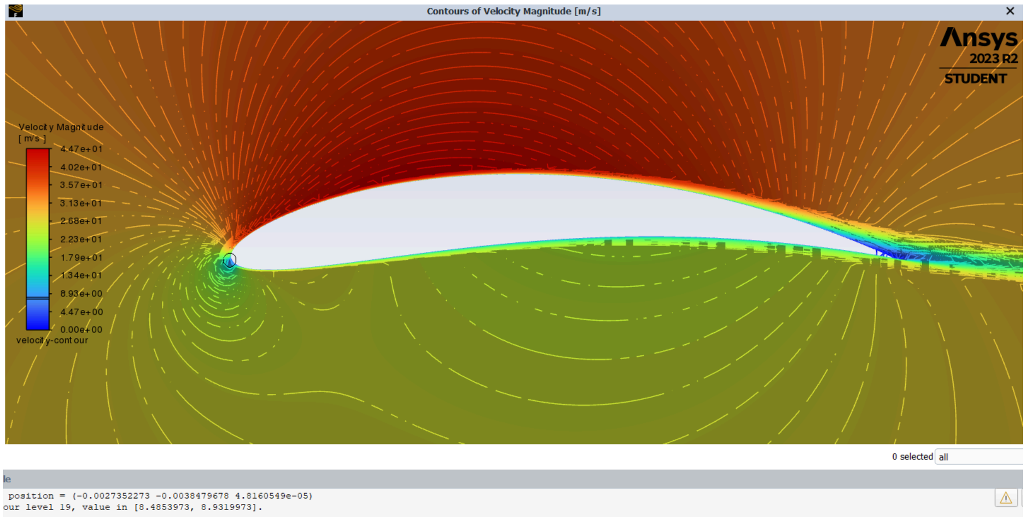
(a) 3°



(b) 10°

At 3 degrees angle of attack, the contour plots typically show relatively weak vorticity concentrations along the airfoil surface, indicating smooth and attached airflow with minimal separation. However, at 10 degrees angle of attack, the contour plots reveal stronger and more pronounced vorticity patterns, particularly near regions of flow separation and turbulence, such as the trailing edge.

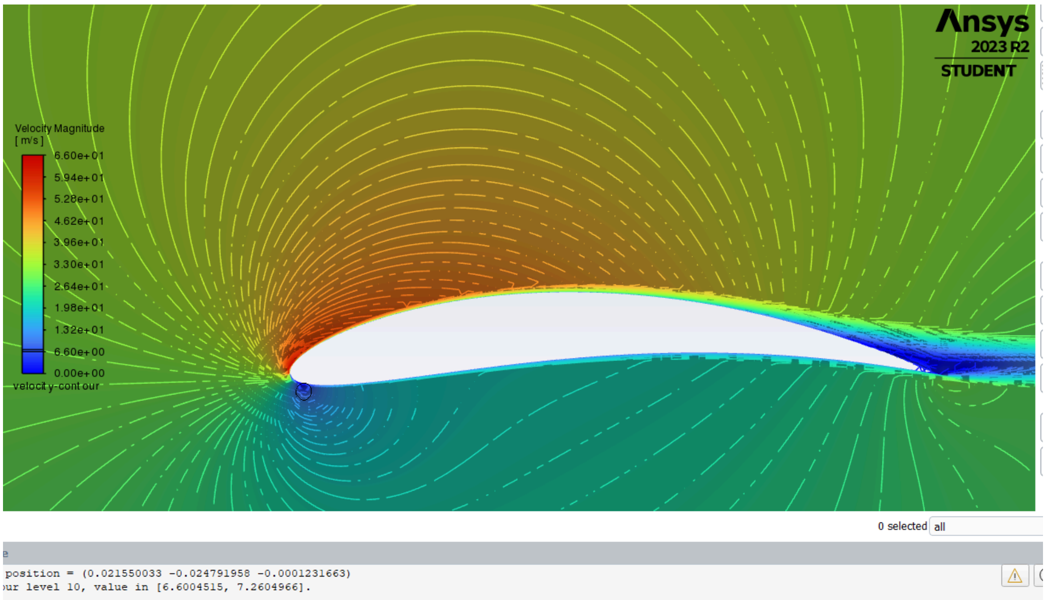
▸ **Approximate location of stagnation and Flow separation in airfoil surface**



(a) 3°

Location of stagnation (-0.0027352273,-0.0038479678)

Location of Flow separation (0.86088985 0.057613365)



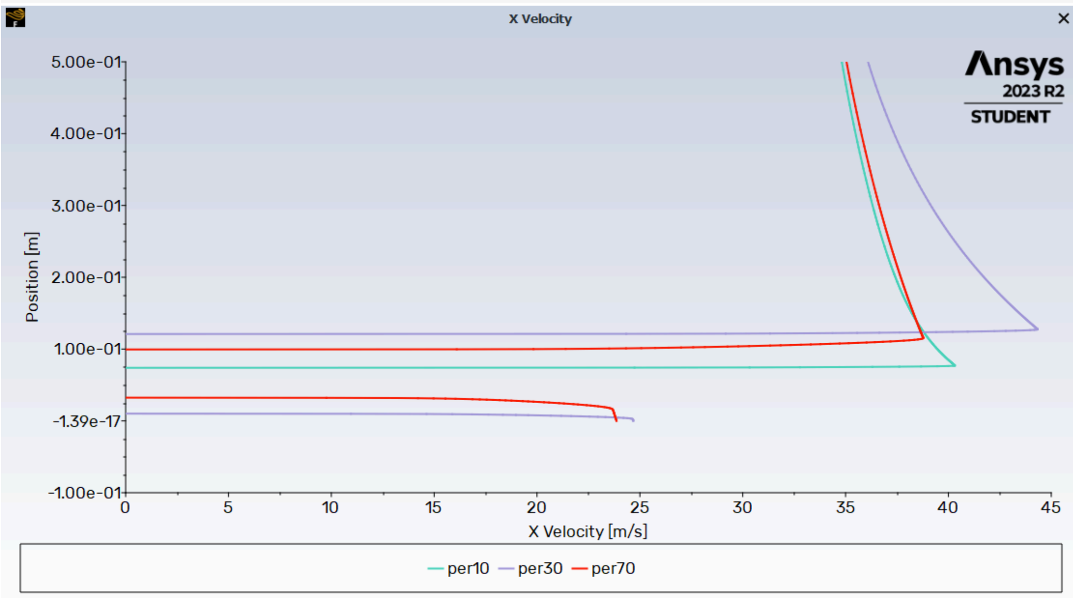
(b) 10°

Location of stagnation (0.021550033,-0.024791958)

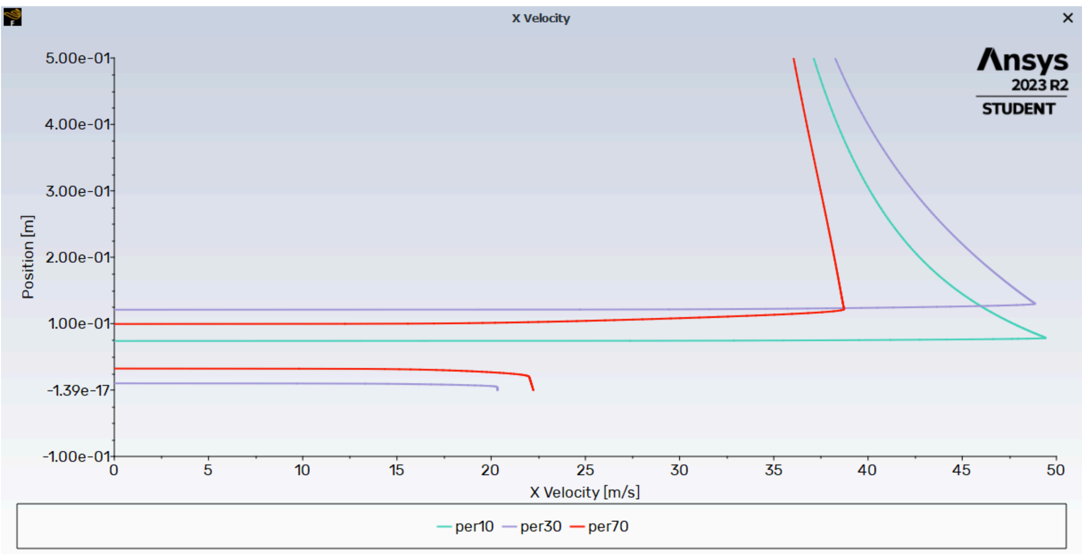
Location of Flow separation (0.68640053 0.10891915)

Velocity contour plots were utilised to find approximate position of the stagnation flow

▸ **Upper surface boundary layer velocity profile At x/c = 10%, 30% and 70%**



(a) 3°



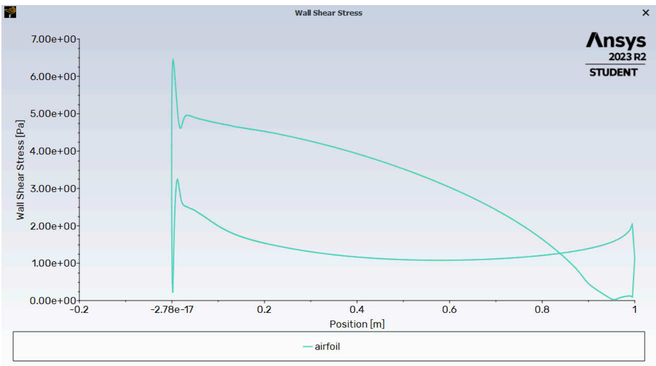
(a) 10°

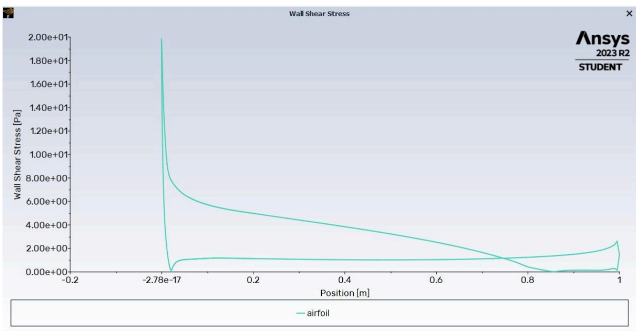
The 99% of the velocity close to the airfoil—that is, the point at which the velocity begins to drop due to slower moving free stream interactions—is used to compute the boundary layers.

It is evident that viscous factors cause the profile to steepen from 0.1 to 0.3 profile. Regarding the grey profile (0.7), it indicates the point where flow separates into turbulent and laminar flow in the 3◦ instance, with a split occurring at 0.8 m. Since the transition for the 10◦ has already happened, a sharp boundary layer is evident.

▸ Estimation of boundary layer thickness

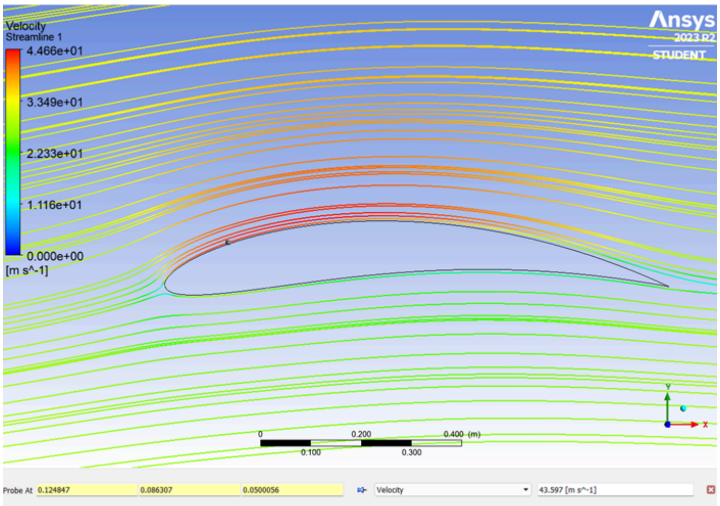
Degrees	x/c = 10%	x/c = 30%	x/c = 70%
3°	*0.121*	*0.007490*	*0.125*
10°	*0.122*	*0.007900*	*0.130*

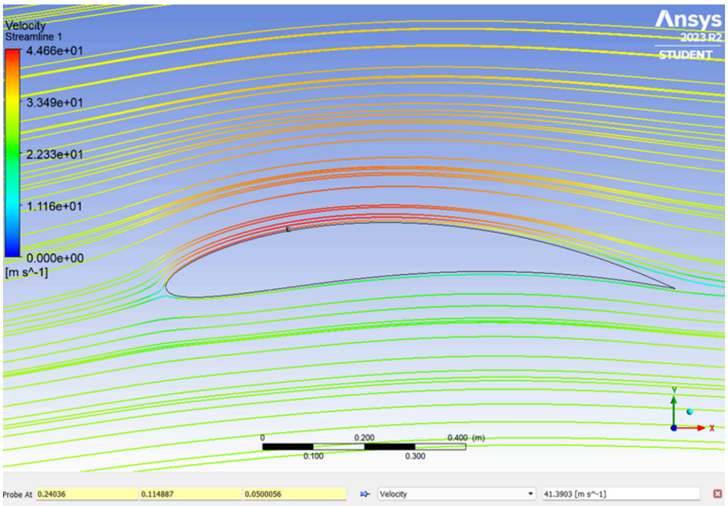




Tangential velocity of flow becomes zero when wall stress becomes zero.

- *Validity of Bernoulli equation along a streamline*

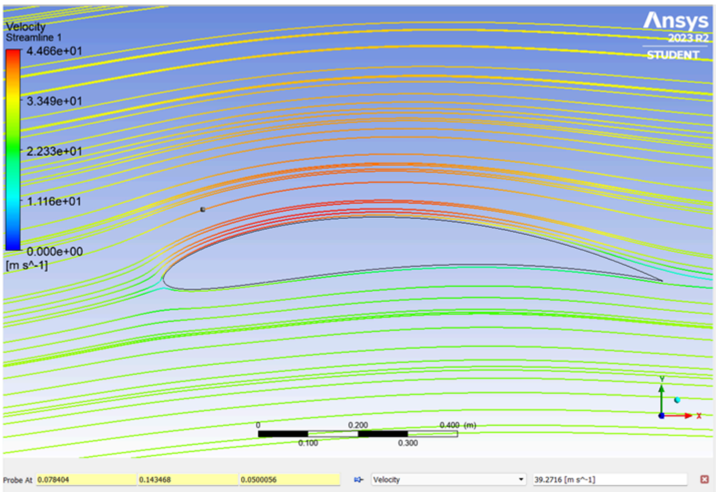


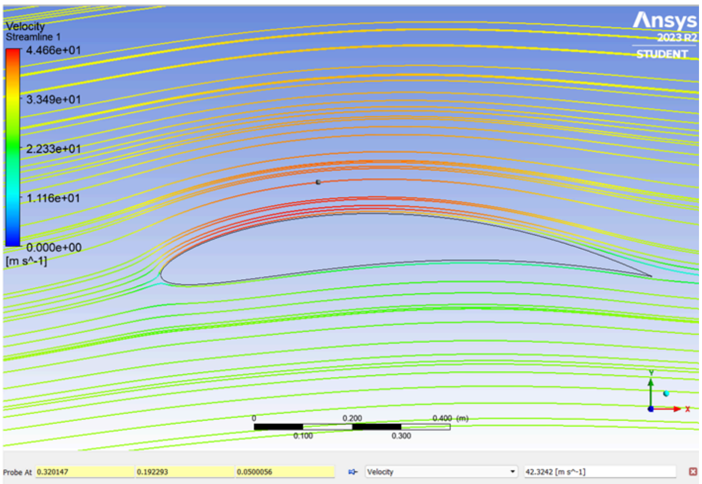


Close to Boundary layer 3 Degrees

Velocity	***43.597***	***41.3903***
Static Pressure	***-616.065***	***-673.46***
Bernoulli	***548.1128***	***375.83***

Bernoulli equations Not Satisfied

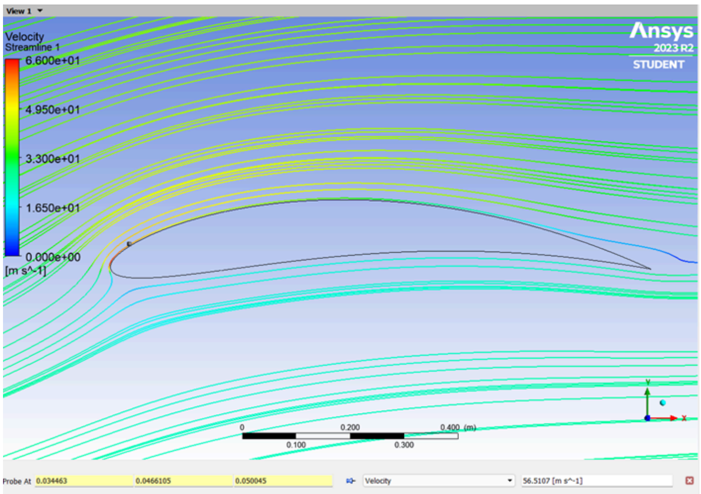


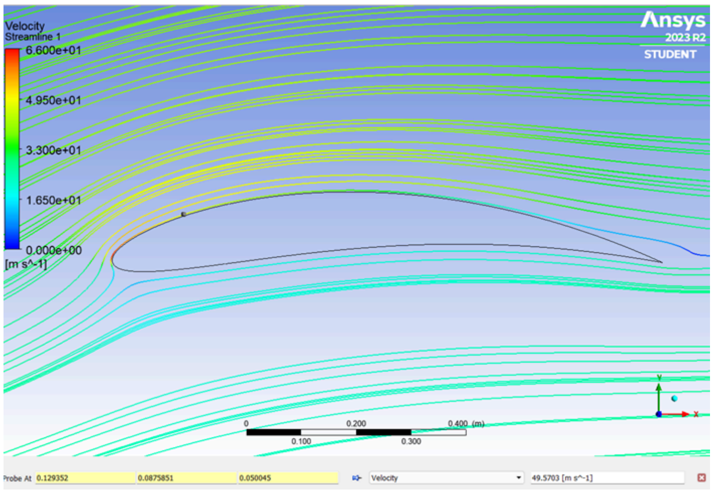


Far from Boundary layer 3 Degrees

Velocity	***39.2716***	***42.3242***
Static Pressure	***-393.493***	***-546.028***
Bernoulli	***551.066***	***550.956***

Bernoulli equations Satisfied

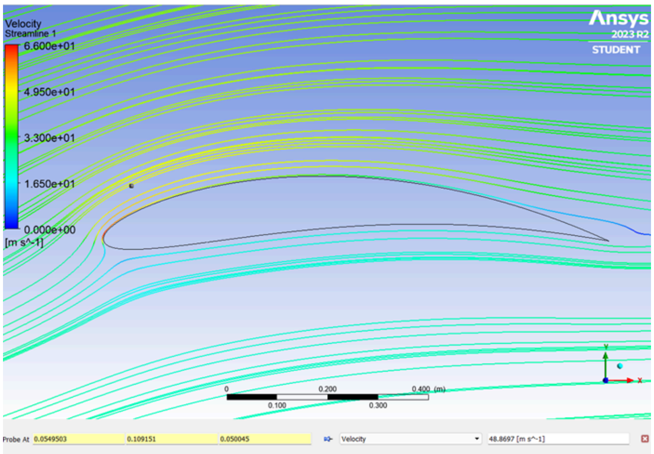


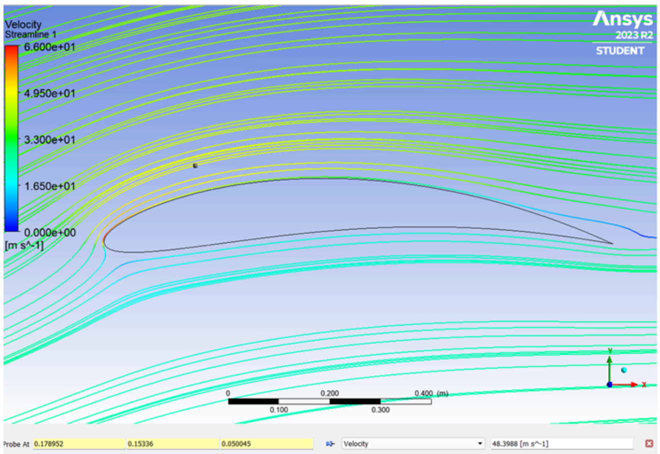


Close to Boundary Layer for 10 Degrees

Velocity	56.5107	49.5703
Static Pressure	-1411.25	-1138.19
Bernoulli	544.69	366.83

Bernoulli equations Not Satisfied





Far from Boundary Layer for 10 Degrees

Velocity	***48.8697***	***48.3988***
Static	***-919.451***	***-891.355***
Bernoulli	***542.77***	***542.87***

Bernoulli equations Satisfied

Conservation laws (at α = 30)

- **Net mass flow rate into the control volume**

I created a rectangular control volume having coordinates (-2,-2) , (-2,2), (2,2) and (2,-2). And then I select mass flow rate and I got

```
                   Mass Flow Rate               [kg/s]
-------------------------------- --------------------
                          line-10           -146.42398

                   Mass Flow Rate               [kg/s]
-------------------------------- --------------------
                          line-11            7.7771261

                   Mass Flow Rate               [kg/s]
-------------------------------- --------------------
                          line-12           -146.66246

                   Mass Flow Rate               [kg/s]
-------------------------------- --------------------
                          line-13            8.0187136
```

Mass flow rate = 146.42398-7.7771261-146.66246+8.0187136

Mass Flow Rate = 0.0031075 Kg/s

▶ **Net rate of momentum change through the control volume**

Along X-axis

Flow Rate X Velocity	[(m/s)(kg/s)]
line-10	-4378.2814
line-11	227.17953
line-12	-4390.2126
line-13	247.8682

Momentum change = 4378.28-4390.21-227.17+247.86

Momentum change = 8.76 m/s kg/s

Along Y-axis

Flow Rate Y Velocity	[(m/s)(kg/s)]
line-10	-47.472344
line-11	12.957469
line-12	-334.85855
line-13	14.279061

Momentum Change = -47.47+12.95+334.85-14.27

Momentum change = 286.06 m/s kg/s

There is lift along y axis so momentum change will not approach zero.

Now we calculate Lift Force using Report Definitions

Lift	[N]
airfoil	651.35

Now we calculate Drag Force using Report Definitions

Drag	[N]
airfoil	-24.702512

Now we calculate Weighted Pressure from surface integrals

Area-Weighted Average Static Pressure	[Pa]
line-10	1.896661
line-11	39.895923
line-12	-2.133776
line-13	-51.272462
Net	-2.903413

Pressure along X axis is 11.38 pa
Pressure along Y axis is 0.24 pa

Area = 16 mm^2

Force along x axis = 182.08 N
Force along y axis = 3.84 N

Net force along x axis = 182.08-24.70 = 157.38 N

Net force along y axis = 641.35-3.84 = 637.51 N

Net Momentum change along x axis = 11.93 m/s kg/s

Net Momentum change along y axis = 20.59 m/s kg/s

So we can compare them and get the results.

Airfoil Design

▸ 5.1 &5.2

My current airfoil is NACA-7511.
Firstly I want to decrease its camber and thickness both. Its Max camber position length is fine (50%) according to my perspective.

So, after doing various simulations on different airfoils. I concluded that I want to design new airfoil
NACA 5508.

It has,

Maximum camber = 5.0%
Thickness = 11%
Maximum camber position = 50%
Chord Length = 1m

Reason why I decrease Maximum camber % and Thickness for improving its performance.

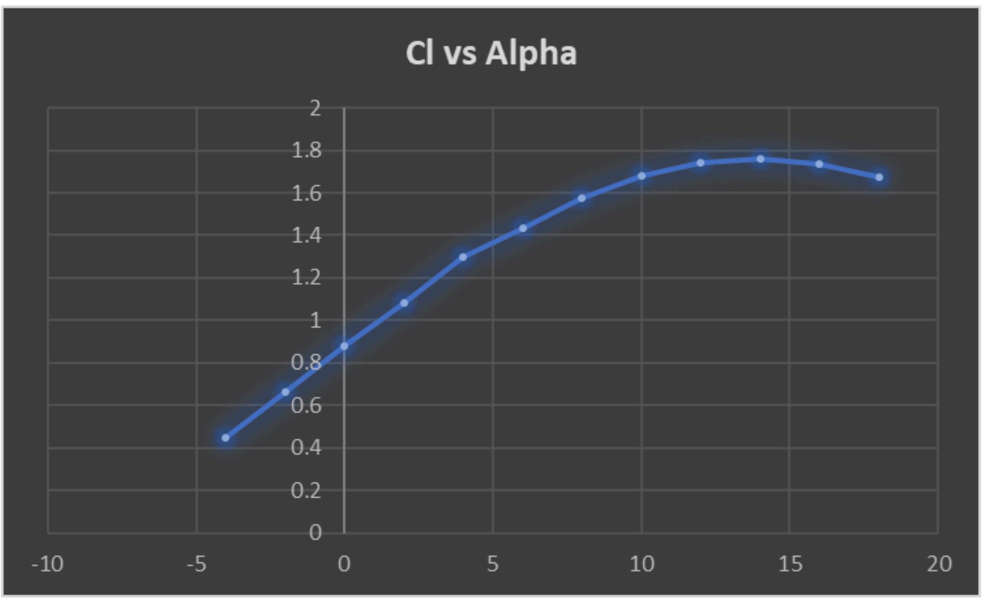
1. **Airfoils with lower camber and thickness tend to have lower drag coefficients. This means they encounter less resistance as they move through the air, leading to improved aerodynamic efficiency.**

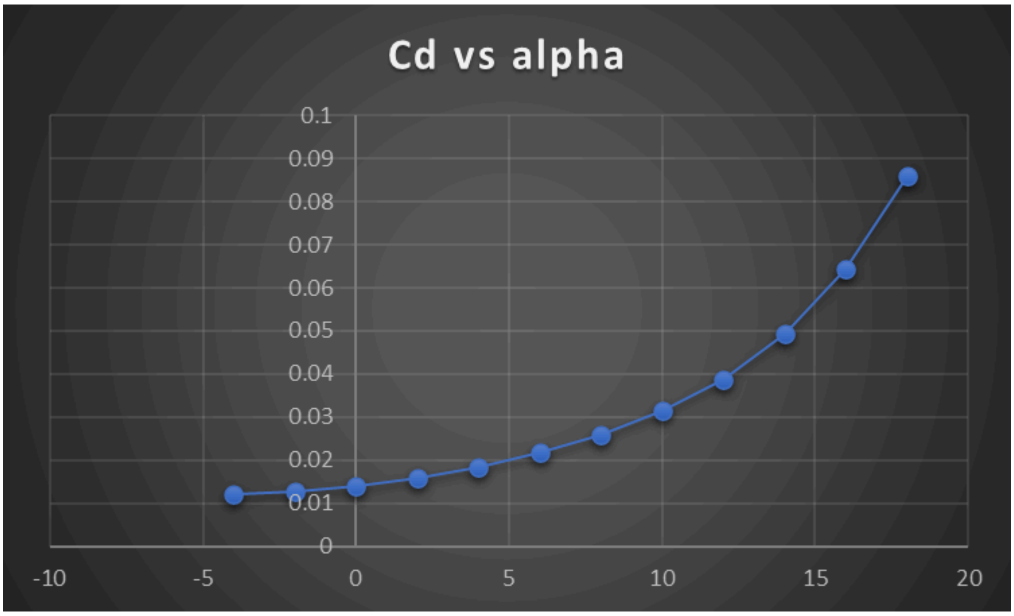
2. **By reducing drag while maintaining or even increasing lift, a thinner airfoil can achieve a higher lift-to-drag ratio. This is beneficial for aircraft as it allows for better fuel efficiency and longer range.**

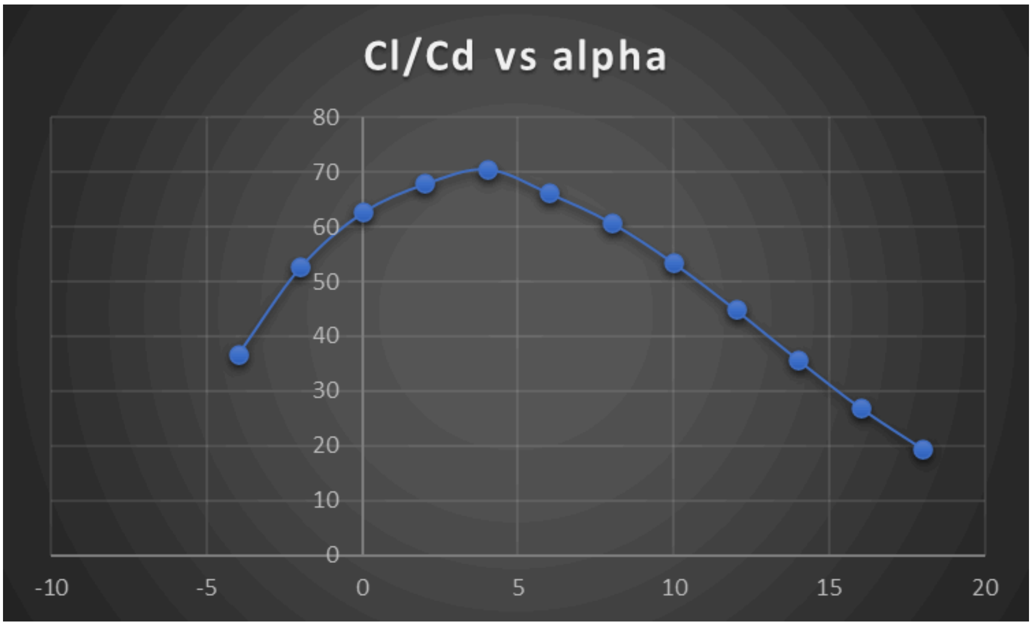
3. **Thinner airfoils often have more gradual stall characteristics, meaning they can operate at higher angles of attack before experiencing a stall. This improves the safety and manoeuvrability of the aircraft.**

For Airfoil NACA5508→

Alpha	Cl	Cd	Cm	Cl/Cd
-4	0.4452	0.0121	0.2139	36.79339
-2	0.6642	0.0126	0.214	52.71429
0	0.8774	0.014	0.2133	62.67143
2	1.0798	0.0159	0.2118	67.91195
4	1.2995	0.0184	0.2091	70.625
6	1.4366	0.0217	0.2048	66.20276
8	1.5745	0.0259	0.1983	60.79151
10	1.6776	0.0314	0.1895	53.42675
12	1.7407	0.0388	0.1785	44.8634
14	1.761	0.0493	0.1663	35.72008
16	1.7381	0.0644	0.1545	26.98913
18	1.6762	0.0861	0.1455	19.46806







Max Cl/Cd = 70.625
Max Cl =1.761
Stall angle = 14 Degrees
Y- Intercept = 0.8774
Slope = 0.1012

- **Performance of new airfoil-**

1. Here we get a new value of Cl/Cd as 70.625 which is higher than the previous one.

2. Second one possesses a higher lift-to-drag ratio (CL/CD) than first, it signifies superior aerodynamic efficiency and performance.

3. A higher CL/CD value indicates that the airfoil can generate more lift relative to the drag it produces, resulting in better overall efficiency during flight.

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