

# Performance Analysis of Surfaced Modified NACA 2412 Airfoil using CFD at Various Angle of Attacks and Reynolds Number

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**ABSTRACT:** The present research work linchpins on understanding the behaviour of surface modified NACA 2412 airfoil at different Reynolds number. The airfoil's lower surface is introduced with a inward cut of 2.5mm. Three airfoils were built by introducing the cut at different locations. One airfoil has a cut at 25% of chord length and other two at 50% and 75% respectively. The model is developed in Ansys Design Modeler and meshed in ICEMCFD. All three meshed models are simulated at 0<sup>0</sup>, 4<sup>0</sup>, 8<sup>0</sup>, 12<sup>0</sup> and 16<sup>0</sup> Angle of Attacks (AoA) at 60000 and 100000 Reynolds number. The post processing of the result revealed that, the increase in AoA resulted in increase in lift Co-efficient (C<sub>L</sub>) for all three airfoils at both Reynolds number. The lift co-efficient started increasing as the position of cut moved towards the trailing edge. The flow started separating after 12<sup>0</sup> AoA. It was found that, the airfoil with inward cut at 75% of chord length resulted in attaining highest lift co-efficient when compared to other two airfoils.

**KEYWORDS:** NACA 2412 Airfoil, Chord Length, Angle of Attack, Reynolds Number, Lift Co-efficient, Flow Separation.

## I. INTRODUCTION

Airfoils are the main lift producing devices in an aircraft. The airfoil generates lift by creating a pressure difference over its surface. Angle of Attack (AoA) also plays a vital role in generating lift. Change in AoA results in variation of performance of airfoil [1]. Due to the shape of airfoil, the air passes over both the surface at different velocities and pressures. This pressure difference causes the airfoil to lift in its vertical direction [2]. Hence, the lift generated by airfoil purely depends on the shape of the airfoil's surface. As the airfoil's geometrical surface changes, the lift generated by the airfoil also changes. This research focuses on understanding the phenomenon of how an airfoil behaves when its surface is modified. Hence a computational research is carried out on surface modified airfoil at different AoA and at different Reynolds number.

A research done by Alan A [3] on surface modified airfoil resulted in a phenomenal understanding of behaviour of NACA 2412 airfoil when its surfaced is modified to a fish kind model. The researcher carried out a computational study on behaviour of NACA 2412 airfoil when modified to a wahoo fish, sword fish and tarpon fish shape. The results concluded by indicating that, the aerodynamic performance of tarpon fish shape airfoil is higher than NACA 2412 airfoil. A study on development of spoiler on the surface of the NACA 2412 airfoil was done by Scott Douglas Lindsay, Paul Walsh [4] to understand the aerodynamic performance. It is found from post processing of the result that, at high AoA, deployment of spoilers results in increase of lift and drag, but at low AoA, the lift decreases and the drag increases. Er. Shivam Saxena and Mr. Rahul Kumar [5] worked on understanding the aerodynamic performance of NACA 2412 airfoil at different Reynolds number and AoA. It was concluded by saying that, at thick surface of airfoil, static pressure remains constant and at the lower ends of airfoil, the dynamic pressure remains constant. Computational fluid dynamics approach was used to enhance the lift generation process during high lift take off condition for an MAV NACA 2412 wing by Arvind Prabhakar and Ayush Ohiri [6]. The results showed, at high take off condition, double slotted flap extended to 40 degrees is the ideal position. At this configuration of slots, the stall angle raised from 20<sup>0</sup> to 54<sup>0</sup>. A research on effect of air flow over NACA 2412 airfoil at high Reynolds number was carried out by Shivananda Sarkar and Shaheen Beg Mughal [7]. 5<sup>0</sup> AoA was found to be an optimum angle which results in maximum lift to drag ratio and the stall angle was at 15<sup>0</sup>. As the air flows over the airfoil, the temperature of the air also plays a vital role in aerodynamic performance of that airfoil. A research on understanding the characteristic effect of temperature on NACA 2412 airfoil was done by Yogesh Thawrani and Ajith Kumar [8]. Richardson number (R<sub>i</sub>) at 0<sup>0</sup> AoA is directly proportional to the lift co-efficient and the increase in temperature leads to earlier flow separation states the conclusion. An unconditional work was done on surface modification of the airfoil by adding a vortex generator and a dimple on the airfoil to increase the aerodynamic performance by Sonia Chalia and Manish Kumar

Bharati [9]. The concluding remarks states that, the introduction of vortex generator leads to increase in fuel consumption. Dimple over the airfoil surface results in more lift than a vortex generator. Md. Tariqul and Amit M. E. Arefin[10] worked on understanding the performance of airfoil at different Reynolds number. The results said that at lower AoA the Reynolds number had a significant effect on airfoil than at higher AoA. Vinayaka N and Akshaya C[11] worked on flow simulation over cascade blades at different angle of attacks. They concluded by quoting that as the angle of attack was increased, the pressure raised and flow separation happened at more than 15 degree angle of attack. A simulation was carried out on compressor blades at transonic velocity at different AoA by Akshaya C and Vinayaka N[12]. The solution stated that increment in velocity and AoA leads the shock wave to move at different position. A study on transonic airfoil was carried out by Vinayaka N and Akshaya C[13] at transonic regime to understand the behaviour of shock wave at higher altitudes and at different turbulence intensity levels. The paper says that as the turbulence intensity increases with AoA the shock wave position changes.

## II. METHODOLOGY

### Geometric Modelling

The coordinates of NACA 2412 airfoil were obtained from airfoil tools. Ansys Design Modeler is used to model the airfoil. There were three surface modified airfoils generated from NACA 2412 airfoil. A 90° inward cut of 2mm is introduced on the lower surface of the airfoil at 25%, 50% and 75% of chord length as shown in Figure 1.

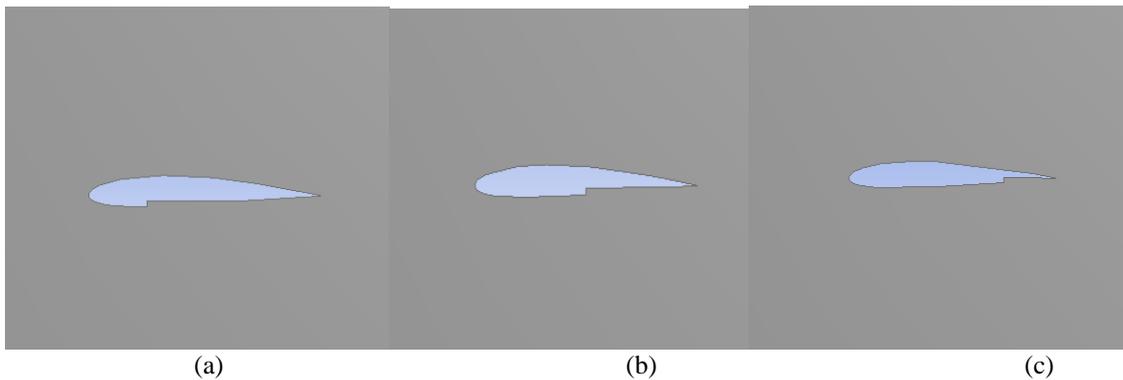


Fig. 1. Text Detection and Inpainting (a) Geometric Modelling of airfoil with inward cut at 25% of chord length (b) Geometric Modelling of airfoil with inward cut at 50% of chord length (c) Geometric Modelling of airfoil with inward cut at 75% of chord length.

### Meshing

The model has to be meshed before it is simulated using Ansys Fluent. ICEMCFD is used to mesh the domain. An unstructured grid is created across the domain. There are 165116 cells created across the airfoil. The orthogonal quality is 0.574, whereas values close to 0 signifies the low quality. The orthogonal skew is 0.343, whereas the value near to 1 is low quality. The meshed model is shown in Figure 2.

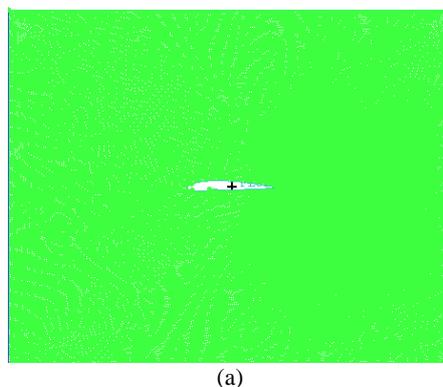


Fig. 2.(a) Unstructured Mesh across the airfoil

**Simulation**

The simulation on the meshed model is done using Ansys Fluent. The simulation was carried out at 60000 and 100000 Reynolds number at 0°, 4°, 8°, 12° and 16° AoA on airfoil which has cut at 25%, 50% and 75% on the lower surface along chord length. The convergence of the solution was decided when  $C_L$  reached a constant value and the continuity equation was satisfied. The graph of converged solutions is given below in Fig 3.

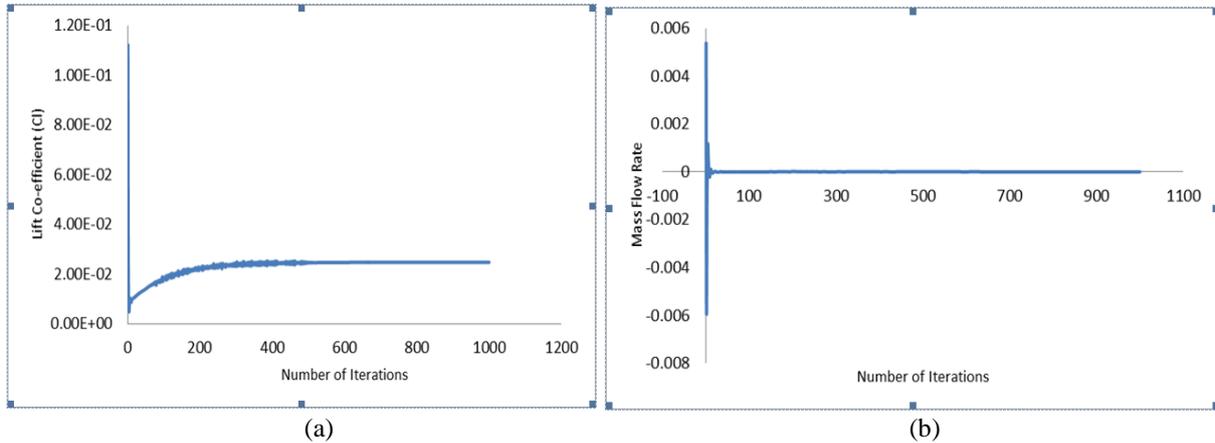


Fig. 3. (a) Convergence criteria w.r.t lift co-efficient (b) Convergence criteria w.r.t mass flow rate

**III. EXPERIMENTAL RESULTS**

**Test Case 1 – Position of cut at 25% of chord length. Re=60000**

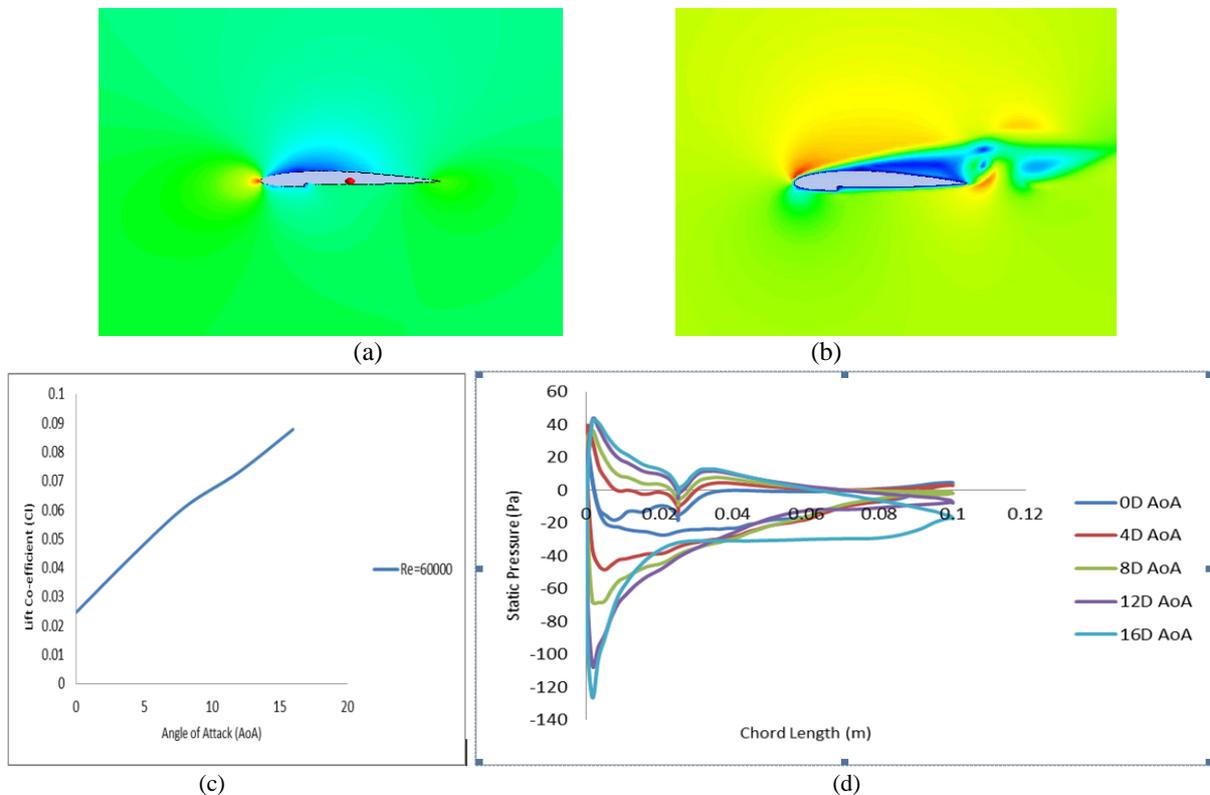


Fig. 4. (a) Static Pressure Contours at 0° AoA (b) Velocity Contours at 16° AoA (c) CL vs. AoA graph at 60000 Reynolds Number (d) Static Pressure vs. Chord Length graph at varies AoA.

The results obtained from the computational analysis are presented in this section. The simulation was carried out at  $0^\circ$ ,  $4^\circ$ ,  $8^\circ$ ,  $12^\circ$  and  $16^\circ$  AoA at 60000 Reynolds number. Fig 4 (a) indicates the static pressure over the airfoil at  $0^\circ$  AoA and Fig 4 (b). shows the velocity contours at  $16^\circ$  AoA. The lift co-efficient of the airfoil increases as the AoA increases. Lift co-efficient at  $0^\circ$  AoA is 0.0246 whereas the max  $C_L$  is obtained at  $16^\circ$  AoA which is 0.0878. The results indicates a flow separation over the airfoil at  $16^\circ$  AoA as shown in Fig 4 (b). Hence, the airfoil stall angle is in-between  $12^\circ$  to  $16^\circ$  AoA. The dip of static pressure on the upper line (lower surface of airfoil) in the graph at 25% of chord length indicates the cut on airfoil. The dip indicates the rise in velocity due to decrease in pressure. This phenomenon occurs due to the presence of cut at 25% of chord length where the flow will be accelerating. The steep rise in pressure at  $16^\circ$  AoA in Fig 4 (d) signifies the stall of airfoil.

**Test Case 2 – Position of cut at 25% of chord length. Re=100000**

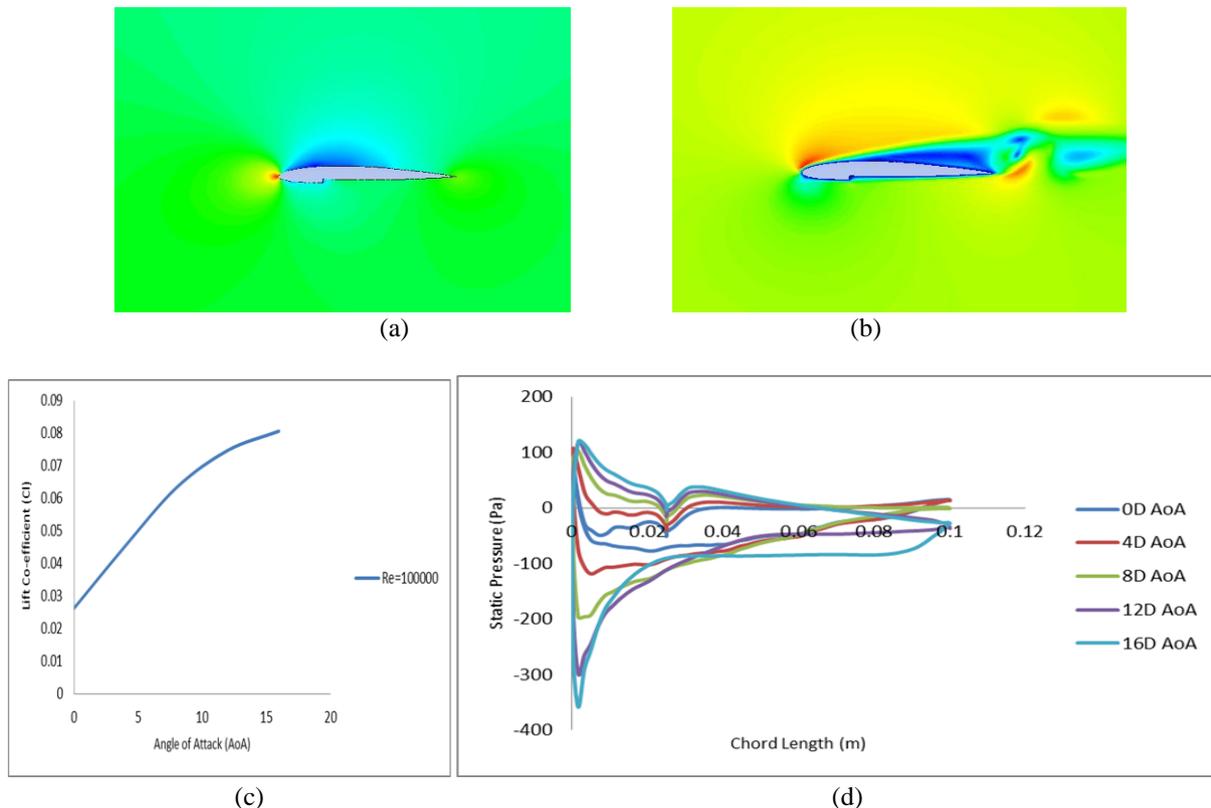


Fig. 5. (a) Static Pressure Contours at  $0^\circ$  AoA (b) Velocity Contours at  $16^\circ$  AoA (c) CL vs. AoA graph at 60000 Reynolds Number (d) Static Pressure vs. Chord Length graph at varies AoA.

Static pressure contours and velocity contours are shown in Fig 5 (a) & (b). This section discusses the results of cut at 25% of chord length at 100000 Reynolds number. In this test case, it is found that, as the AoA increases, the lift co-efficient increases. In test case 1, the lift coefficient was 0.0246 at  $0^\circ$  AoA but in this case, the lift co-efficient is 0.0263, which signifies that the increase in Reynolds number leads to increase in lift. Flow separation occurs at  $16^\circ$  AoA as shown in Figure 5 (b). The airfoil stalls earlier than compared to test case 1. The lift co-efficient was 0.0878 at  $16^\circ$  AoA in test case 1 whereas the lift co-efficient is 0.0806 in this case, signifying that the airfoil stalls early. Fig 5 (d) represents the static pressure distribution over the airfoil at 100000 Reynolds number. It can be seen that, the area inside the curve in graph has increased when compared to test case 1. This increase in area signifies the increase in lift. Both test case 1 and test case 2 shows a dip in pressure on upper line of the curve (lower surface of the airfoil) in Fig 4 (d) and Fig 5 (d) indicating the loss of lift.

**Test Case 3 – Position of cut at 50% of chord length. Re=60000**

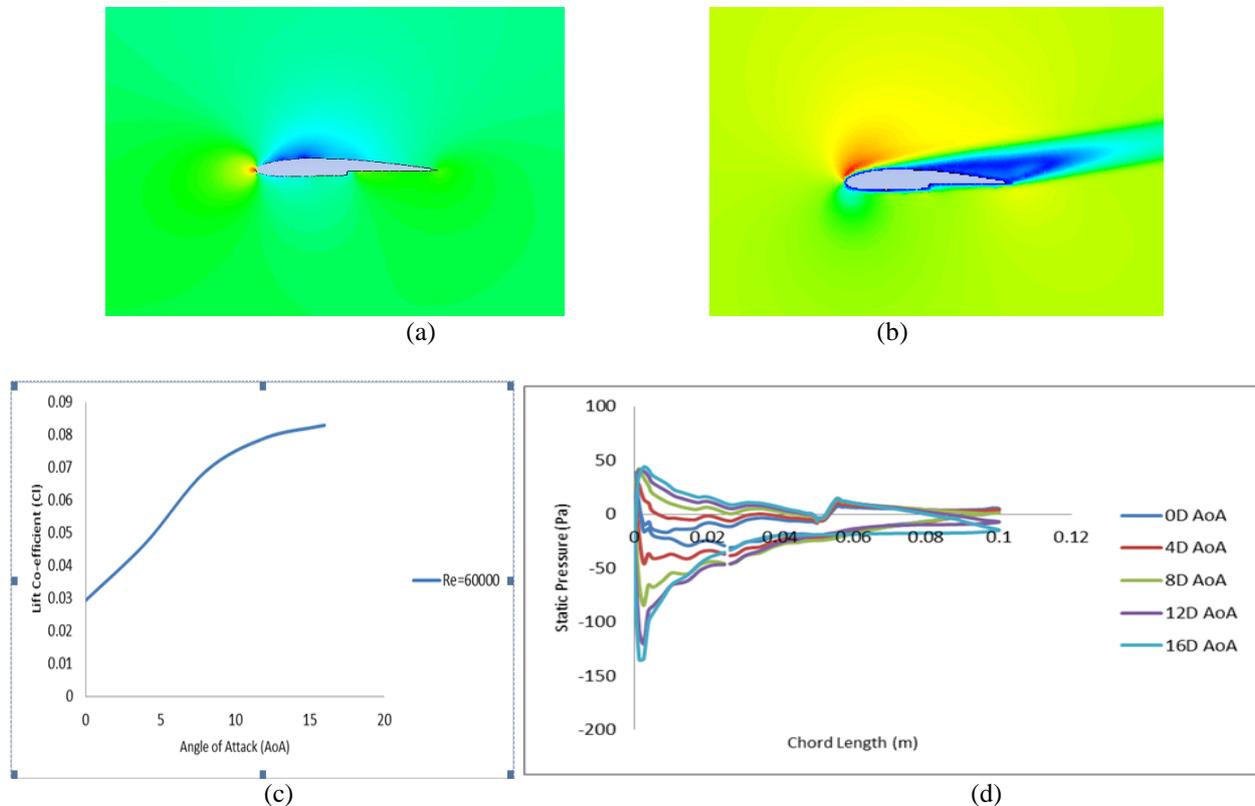
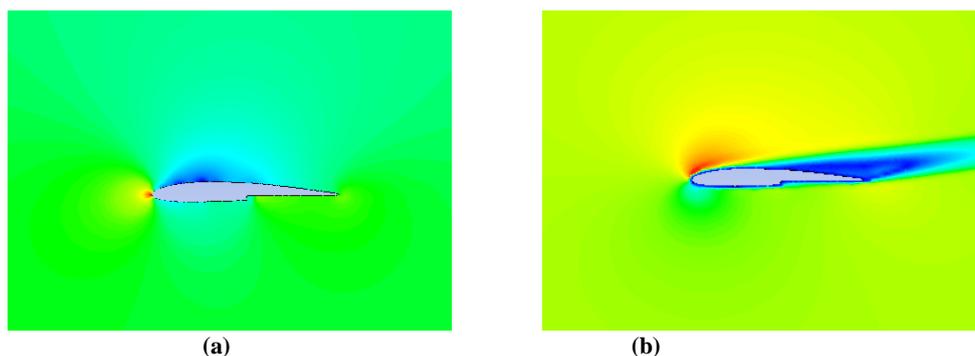


Fig. 6. (a) Static Pressure Contours at 0°AoA (b) Velocity Contours at 16°AoA (c) CL vs. AoA graph at 60000 Reynolds Number (d) Static Pressure vs. Chord Length graph at varies AoA.

Test case 3 discusses results about the airfoil which has cut at 50% of chord length and at 60000 Reynolds Number. Static pressure contours are shown in Fig 6 (a). The lift co-efficient at 0°AoA is 0.0294 in this case, but it was 0.0246 in test case 1. As the cut on the lower surface of the airfoil is shifted from 25% to 50% of chord length, the co-efficient of lift increased. The lift coefficient was 0.0878 in test case 1 whereas in this case the lift coefficient is 0.0829 at 16°AoA. This shows that, the airfoil which has cut at its 50% of chord length stall earlier than the airfoil which has cut at its 25% of chord length. Flow separation on the airfoil is shown in Fig 6 (b). Hence, the stalling angle is in-between 12° and 16°. From Fig 6 (d) it can be signified that the area inside the curve is higher in this case than in test case 1. This is due to the increase in lift. The rise in pressure at 50% of chord length shown in Fig 6 (d) indicates the pressure on the lower surface is increased due to the cut, leading to the excess amount of lift. In test case 1 there is a dip in pressure which resulted in lift loss, but in this case the raise in pressure results in gain of lift.

**Test Case 4 – Position of cut at 50% of chord length. Re=100000**



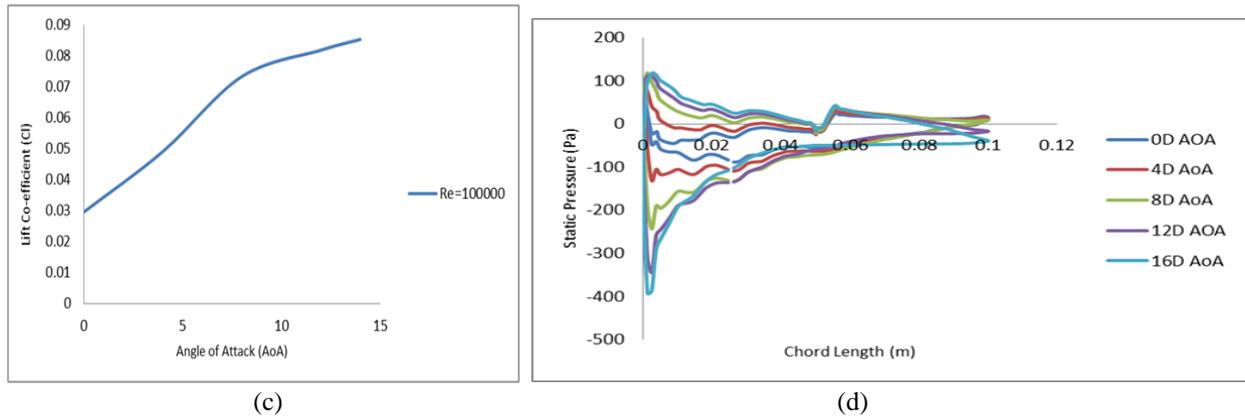


Fig. 7. (a) Static Pressure Contours at 0°AoA (b) Velocity Contours at 16°AoA (c) CL vs. AoA graph at 60000 Reynolds Number (d) Static Pressure vs. Chord Length graph at varies AoA.

The behaviour of the airfoil changes as the flow characteristics changes. Test case 4 is carried out for the same boundary condition as of test case 3 at 50% cut on its chord length but, the Reynolds number is increased from 60000 to 100000. The post processing of results signified that, the lift co-efficient increased as the Reynolds number and AoA is increased as shown in Fig 6 (c). But the difference in increment of lift coefficient from test case 1 to test case 2 at 0°AoA was 6.46% whereas test case 3 to test case 4 at same degree AoA is 11.14%. This signifies that there is a higher increment as the Reynolds number increases at 50% cut at chord length than 25% cut at chord length. The flow separation was found at 16°AoA as signified in Fig 6 (b). The airfoil stalls in this test case earlier when compared with other test cases. The increase in lift is signified by increase in area of static pressure curve as shown in Fig 6 (d). The slight raise in pressure in Fig 6 (d) signifies the cut on the airfoil. The steep increase in pressure on the lower line (airfoil's upper surface) of graph indicates the flow separation on the airfoil's upper surface.

**Test Case 5 – Position of cut at 75% of chord length. Re=60000**

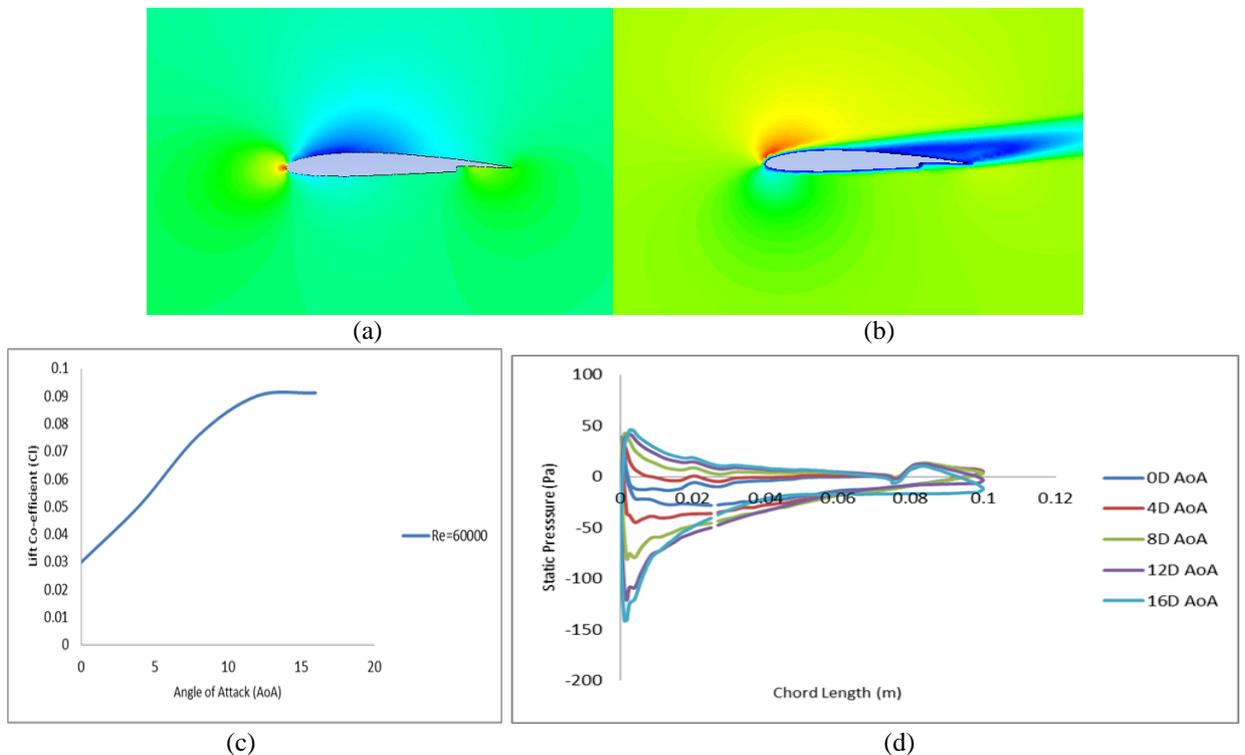
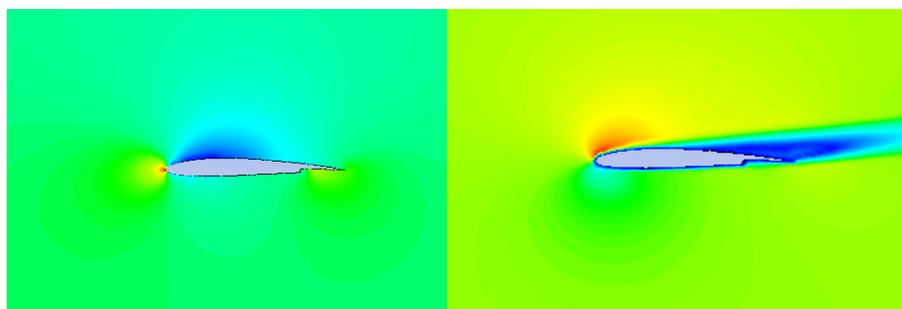


Fig. 8. (a) Static Pressure Contours at 0°AoA (b) Velocity Contours at 16°AoA (c) CL vs. AoA graph at 60000 Reynolds Number (d) Static Pressure vs. Chord Length graph at varies AoA.

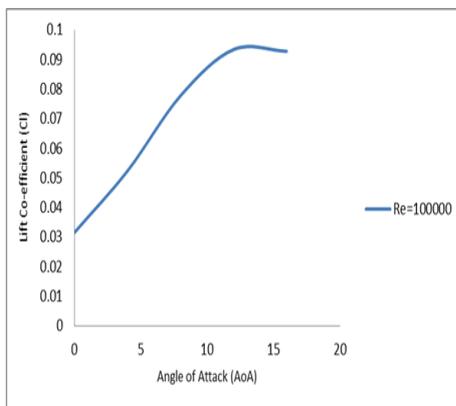
Fig 8 (a) & (b) shows the static pressure and velocity contours on airfoil. In this test case, the cut on the airfoil surface is made at 75% of the chord length. As the AoA is increased, the lift co-efficient also increased. The  $C_L$  of the airfoil is 0.0299 in this test case, whereas in test case 3 it was 0.0294. But the increment in lift co-efficient from  $12^\circ$  AoA to  $16^\circ$  AoA is minimum in this test case. The difference in amount of increment in lift co-efficient from test case 1 to test case 3 was 16.32% at  $0^\circ$  AoA, but the increment from test case 3 to test case 4 is just 1%. The flow separation is shown in Fig 8 (b). The value of lift coefficient is 0.0829 in test case 3 whereas it is 0.0911 in this test case at  $16^\circ$  AoA. Hence, before the airfoil stalls, it produces higher lift when compared to airfoil which has 50% cut on its chord length. The change in lift co-efficient with respect to AoA is shown in Fig 8 (c). The lift generated is higher when compared to previous cases. This rise in pressure at 75% chord length in Fig 8 (d) indicates the presence of cut on the lower surface of the airfoil. This raise in pressure on the lower surface creates an extra pressure difference due to which an excess amount of lift is generated.

**Test Case 6 – Position of cut at 75% of chord length.  $Re=100000$**

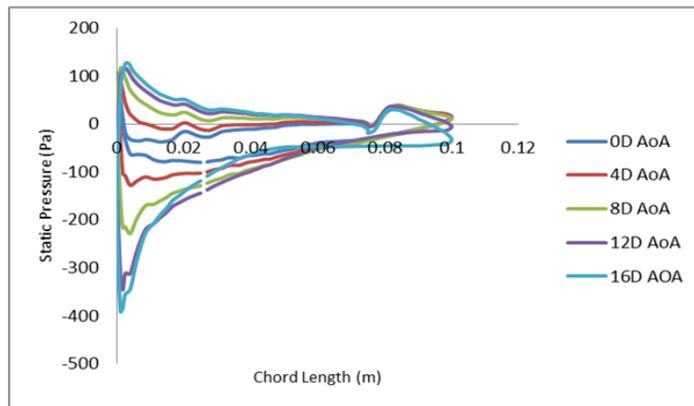


(a)

(b)



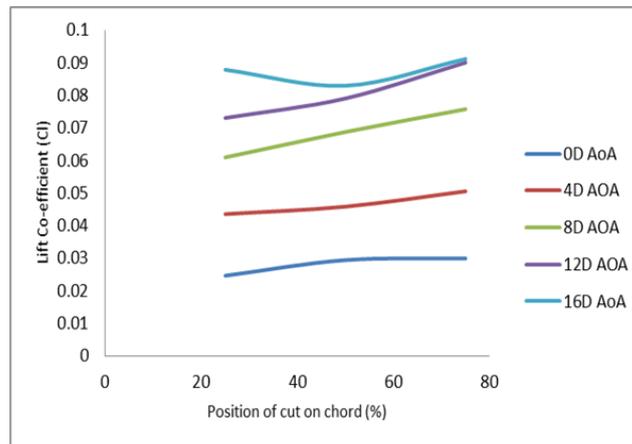
(a)



(b)

Fig. 8. (a) Static Pressure Contours at  $0^\circ$  AoA (b) Velocity Contours at  $16^\circ$  AoA (c)  $C_L$  vs. AoA graph at 60000 Reynolds Number (d) Static Pressure vs. Chord Length graph at varies AoA.

The simulation is done at 100000 Reynolds number in this test case for 75% cut on the chord length. The lift co-efficient is highest when compared to all the previous test cases. The lift co-efficient is 0.0315 which is highest of all. The difference in amount of increment in lift co-efficient in test case 2 and test case 4 at  $0^\circ$  AoA was 11.14%, but the increment in test case 4 and test case 6 is just 6.03% at  $0^\circ$  AoA. The highest  $C_L$  is 0.0933 at  $12^\circ$  AoA in test case 6. The lift co-efficient decreased as the AoA is raised to  $16^\circ$ . The air flow gets separated from the airfoil at  $16^\circ$  AoA as shown in Fig 8 (b). Fig 8 (a) shows the distribution of static pressure across airfoil whereas Fig 8 (b) indicates the separation of air flow over the airfoil's upper surface. The increase in area of the static pressure curve signifies the increment of lift co-efficient. Due to the increase of Reynolds number from 60000 to 100000, the raise in static pressure at the position of the cut is higher, which leads to the higher lift generation. But the airfoil stalls earlier at 100000 Reynolds number than 60000 Reynolds number. The airfoil in this test case has achieved the maximum lift coefficient.

**Results of lift co-efficient at various AoA and position of cut on chord line**

(a)

Fig. 9. (a) Position of cut on chord vs. Lift co-efficient at various AoA.

The graph in Fig 9 (a) shows the variation of lift co-efficient at 60000 Reynolds number with change in position of cut at different AoA. Fig 8 (d) reveals a significant results obtained from all test cases. The graph indicates that, the lift co-efficient increases as the position of cut is changed from 25% to 75% through 50% at various AoA. Lift co-efficient increases as the AoA increases. It can be seen from graph that, the amount of increment in lift from 12<sup>o</sup>AoA to 16<sup>o</sup>AoA is decreased due to the separation of flow. The same phenomenon occurs as the Reynolds number is increased from 60000 to 100000.

**IV. CONCLUSION**

The surface modified NACA 2412 airfoil was analysed using computational method. The simulation was done on three airfoils which have an inward cut on its lower surface at 25%, 50% and 75% of chord length. All three airfoils were analysed at 60000 and 100000 Reynolds number at 0<sup>o</sup>, 4<sup>o</sup>, 8<sup>o</sup>, 12<sup>o</sup> and 16<sup>o</sup>AoA. The post processing of the results revealed that, increase in AoA will result in increment of lift coefficient. The change in cut from 25% to 75% through 50% resulted in increment of lift co-efficient. The difference in increment of lift co-efficient at two Reynolds number is higher for the airfoil which has cut at its 50% of chord length. Due to the presence of cut at 25% of chord length which is near the leading edge of the airfoil, the pressure decreases due to the acceleration of flow and resulting in loss of lift. Airfoil which has cut at 75% of chord length has resulted in maximum lift co-efficient when compared with other airfoils. The flow separation is found in between 12<sup>o</sup> and 16<sup>o</sup>AoA for all three airfoils. Airfoil with cut at 75% on its chord length stalls earlier than any other airfoils. Hence, it can be concluded from the results that, the surface modified NACA 2412 airfoil which has cut at 75% of its chord length yields highest lift co-efficient at all the AoA operating at 100000 Reynolds number.

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