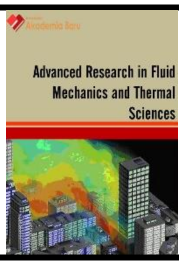
Penerbit Akademia Baru

Journal of Advanced Research in Fluid Mechanics and Thermal Sciences

Journal homepage: www.akademiabaru.com/arfmts.html
ISSN: 2289-7879



High Altitude Transonic Aerodynamics of Supercritical Airfoil at different Turbulence Levels

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ARTICLE INFO

Article history:
Received 22 May 2019
Received in revised form 25 June 2019
Accepted 20 July 2019
Available online 31 August 2019

Keywords:
Transonic Flow; Turbulence Intensity; Supercritical airfoil; Co-efficient of lift; Normal Shock wave

ABSTRACT

Development of the high speed business and commercial aircrafts has become a key research area due to the requirement of fast means of transport. Research on up-gradation of high subsonic flights in-to supersonic flights is in progress. While reaching supersonic velocity, the flow has to pass through the transonic regime. This regime is in-between Mach 0.8 to Mach 1.2. A quantitative research work has already been done in this regime, but still the flow is unpredictable even in 21st century. Hence, these research linchpins on understanding the performance of supercritical airfoil and the normal shock wave behaviour at transonic velocity regime at high altitude (8km above sea level). A supercritical airfoil SC20412 has been chosen for this work. The airfoil is modeled two dimensionally in Ansys Design Modeller and meshed in ICEMCFD. Computational Fluid Dynamics approach was engaged for analysis. The analysis is carried out at 2% and 10% Turbulent Intensity (TI) levels at Mach 0.8 and Mach 0.9 with each at 0^0 and 5^0 Angle of Attack (AoA). Post processing of the results revealed that the Co-Efficient of Lift (C_L) of the airfoil decreases as the Mach number reaches 0.9 form 0.8. Increase in AoA leads to the increase in C_L. The results also showed weak Mach wave and a normal shock wave on the lower and upper surface respectively at Mach 0.8. The normal shock wave starts moving towards leading edge as the AoA increases. Shock wave started moving away from the leading edge as the Mach number increased. The shock wave moves towards the leading edge as the turbulence intensity increases. The variation in upstream boundary conditions of normal shock wave results in the motion of its-self towards or away from leading edge.

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1. Introduction

Humans have always desired to travel long distance in short time. In this development stage came all the transportation vehicles. When the human race reached an era of aircrafts as fast travelling vehicles, there were lot of extensive research conducted during the development face. The desire of humans didn't stop when the subsonic passenger aircraft were successfully built, but they even had

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the desire to travel faster. We have witnessed an extensive research done on transonic velocity regime in past few decades. The flow in this regime is most unpredictable due to the change in phase of flowing fluid from laminar to turbulence [1, 2]. A transonic and supersonic flow analysis was carried out on NACA 2412 airfoil by Balaji and Dash [3, 4]. It led to a conclusion that, the drag of the airfoil increases due to the formation of normal shock wave on the upper surface of the airfoil. C_L of the airfoil decreases as the turbulence intensity increases. This led to a new thought for research of developing an airfoil which delays the formation of shock wave. Large scale research has been conducted by Harris [5] at NASA to develop the airfoils which are known as supercritical airfoils. Supercritical airfoil was designed by reducing the aft camber substantially and the region of the upper surface by using the concept of isentropic recompression and local supersonic flow. An unsteady flow calculation of transonic regime on NACA 64A010 airfoil was done by Chyu *et al.,* [6]. The concluding remarks asserted that the shock wave excrusion and pressures on the surface of the airfoil was well calculated by computational method which falls in an agreement with experimental data's. The effect of ground also plays a vital role which is called as Wing in Ground Effect. A simulation on transonic aerodynamics of RAE2822 airfoil at ground effects was carried out by Boshun Gao and Agarwal [7]. The results were obtained for different AoA, whose post processing reviled that, at transonic flow regime, at high ground clearance shock buffet phenomenon is observed, but at low ground clearance the oscillation of pressure at trailing edge of the airfoil occurs. A CFD analysis work was done on NACA 0012 airfoil at transonic velocity regimes [8, 9]. The results signified that the cause of dynamic instability of the airfoil at this velocity regime was due to the shock waves turbulence characteristic which was signified by effective Prandtl number.

A research on supercritical airfoil's shock location offset was done by Sonia Chalia [10]. The formation of the shock wave is delayed by the flat upper surface of the airfoil, but it results in a reduction of lift. Hence more curvature is added at the trailing edge of the upper surface to regain the lift reduced, states the conclusion. A study on performance characteristics of NACA 0012 airfoil at different turbulence intensities was carried out by Shao-wu LI, *et al.,* [11]. The peroration of the research states that, there is a significant effect of turbulence intensity on drag and lift of airfoil, and higher turbulent intensity leads to stalling of airfoil. A consistent work was done by John B. McDevitt *et al.,* [12] to understand the behaviour of transonic velocity over airfoil with thick circular arc. The importance of turbulence model was determined by this research. The conclusion states that, the employed turbulence model would be adequate if the shock-boundary-layer interaction is weak, if it is strong, then a higher accurate turbulence model should be used.

An unconditional research was carried out numerically in transonic flow regime to actively control the formation of shock wave by Qin *et al.,* [13]. The performance of the airfoil increases as the suction surface is incorporated in the airfoil. It also signified that the consequence of this suction surface has a minimal effect on reduction of wave drag and the strength of the shock wave. Another study was done by William Rose and Arnan Seginer [14] to understand the transonic flow behaviour over NACA64A010 airfoil. They used Navier-Stokes method for solving the equations of flow filed and concluded by saying it is easier to predict the supercritical airfoil's aspects using Navier-Stokes equations.

A set of researcher namely Spaid and McDonnel Douglas Corp [15] worked on the boundary layer measurements over a super critical airfoil. It was found that the Cebeci-Smith method were reasonably good for predictions made on the airfoil's upper surface, whereas the thicker boundary layer near trailing edge was very well predicted by Nash-Macdonald method.

Yap Tze Chuen *et al.,* [16, 17] carried out an experimental and computational analysis on NACA 0015 and aircraft's 150 airfoil. The results signified that the increase in turbulence intensity delays the stall angle. The concluding remarks states that, the increase in turbulence intensity leads to increase in C_L in experimental methods, whereas the computational method didn't signify a much increase. A computational study on performance of NACA (2)-0714 was done by Ravikumar *et al.,* [18]. The simulation was carried out at different AoA. It was found that, as the AoA is increased, the C_L increased. The flow separation was found to be at 15^0 AoA. A transonic flow study on RAE 2822 supercritical airfoil was carried out by Harish Kumar *et al.,* [19]. Due to raise in drag co-efficient at transonic regimes, the overall drag of the airfoil increases. The maximum lift coefficient is more stable near the stall angle. Ravi Shankar *et al.,* [20] worked on understanding the flow over supercritical airfoil SC (02)-0714 and NACA 4412 airfoil. The conclusions states that the strength of shock wave decreases as the curve line of the airfoil at 70% is made flat.

A study on transonic flow over airfoil at unsteady state was carried out by Chyu *et al.,* [21] which states that, viscous flow computations showed closer agreement with experimental data. A study of high subsonic flow over delta wing was carried out experimentally and computationally by Mustafa Hadidullabi *et al.,* [22, 23]. Relative to upstream point, the sudden drop of suction peak is due to the vortex break down at subsonic flows. Fort *et al.,* [24] worked on understanding the flow behaviour over airfoil at transonic velocity regimes numerically. It stated that the K-omega and Wand model are in good agreement with experimental results. An unconditional research was carried out by Tijdemen *et al.,* [25] on understanding the flow behaviour on an oscillating airfoil at transonic velocities. They have developed the capability to experimentally test an oscillating wing at transonic regime. It is said in concluding remarks that, the prediction for these flows experimentally are within reach.

A study on transonic flow calculation was done by Earll M Murman *et al.,* [26] on thin airfoils. The newly developed finite difference system was solved using elliptic and hyperbolic differential equations. The conclusion states that, the results from the numerical calculations are in conjunction with experimental results. It is observed from experimental and computational methods that as the airfoil or wing approaches the transonic region, there is a sudden increment in drag. Hence a research done by Benedetto Mele *et al.,* [27] carried out a study on decomposition of lift and drag at transonic velocity. The study used RANS method which is based on the volume integration of the Lamb vector field. Finally the model has evidenced to detect spurious drag partially. A group of researchers namely Lin *et al.,* [28] worked on inviscid flow field analysis at transonic velocities. It concludes that the work which has been done has a wide range of applications with good accuracy. Jehad *et al.,* [29] carried out a work on understanding the performance of three turbulence model (SST k-ω, standard k-e, realizable k-e) on a 2-D backward facing step. The result concludes by stating that, with respect to the length of reattachment region, SST k-ω model exhibited very good results than k-e model when compared with experimental data.

The present research linchpin on understanding the shock wave behaviour over the supercritical airfoil at transonic velocity. The objectives of this research are listed below.

i. To analyse the behaviour of C_L at different turbulence intensity levels.
ii. To Study the effect of Mach number and AoA on C_L of airfoil.
iii. To understand the behaviour of shock waves over supercritical airfoil.
iv. To acquire a good insight on the behaviour of shock waves at different boundary conditions.
v. To obtain pressure distribution graph over varies boundary conditions.

2. Methodology

2.1 Pre-Processing/Geometric Modelling

An airfoil numbered SC20412 was selected for research due to its basic appearance in the list of supercritical airfoils. The co-ordinates of the airfoil were obtained from airfoil tools. The 2-D airfoil was modeled in Ansys Design Modeler. Then a control volume was created across the airfoil for analysis as shown in Figure 1.

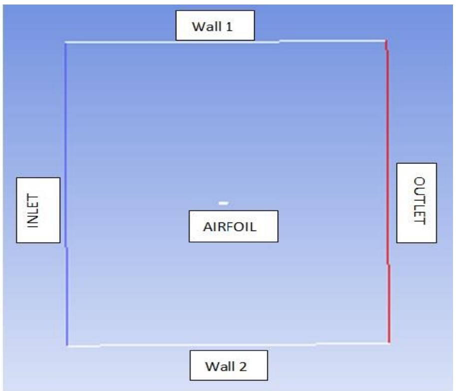

Fig. 1. Geometric Modeling of airfoil

2.2 Meshing

The model has to be meshed before simulation. ICEMCFD was the meshing tool used to build a structured grid around the airfoil as shown in Figure 2. The minimum orthogonal quality was 0.5689, where the values close to 0 was low quality. The minimum ortho skew was 0.451, where the values close to 1 was low quality.

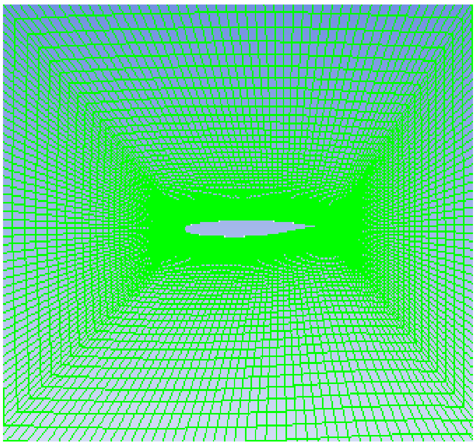

Fig. 2. Structured mesh across the airfoil

2.3 Simulation

Upon the import of mesh file to Ansys Fluent, the model was simulated at different boundary conditions. The airfoil was analyzed at 8 km altitude, 2% and 10% turbulent intensities, 0^0 and 5^0 AoA at Mach 0.8 and Mach 0.9. Boundary Conditions are

i. Airfoil - SC 20412
ii. Mach Number - 0.8 and 0.9
iii. Altitude - 8Km
iv. Angle of Attack (AoA) - 0^0 and 5^0
v. Turbulence Intensities - 2% and 10%
vi. Inlet - Velocity inlet
vii. Outlet - Pressure outlet

3. Results of Test Cases

3.1 Test Case1 - 0^0 and 5^0AoA at 2% Turbulence Intensity at Mach 0.8

This section discusses the computational results obtained from the simulation. Figure 3 and Figure 4 shows the velocity contours at 0^0 and 5^0 AoA. As mentioned earlier, the two parameters which are monitored in this research are the position (location on the chord line of airfoil) of the shock wave and the C_L of the airfoil. Acceleration of the flow can be observed from Figure 3 on the airfoil's upper surface. But the shock wave formation is delayed due to the flat section on the airfoil's upper surface. When the airflow passes over the lower surface, the Mach number tends to reach near supersonic. As the AoA is increases, there is a small separation of flow at the airfoil's trailing edge. The shock wave tends to move towards the leading edge as shown in Figure 4. A steep increase in static pressure in Figure 5 signifies the presence of normal shock wave. The normal shock wave was at 0.3796m away from leading edge at 0^0 and is at 0.3509m away from leading edge at 5^0. This shows that the shock wave is traveling towards the leading edge as the AoA increases. The C_L of the airfoil at 0^0 is 0.3443 and at 5^0AoA is 0.5350. We can clearly observe a substantial increment in C_L as the AoA increases.

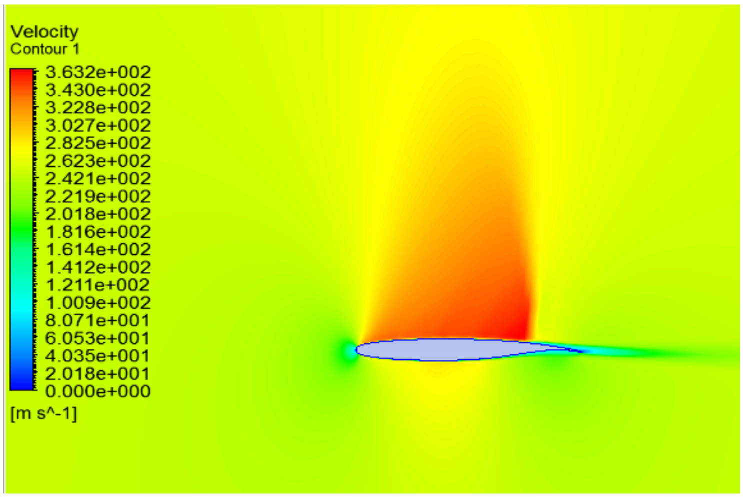

Fig. 3. Velocity contours at 0^0AoA

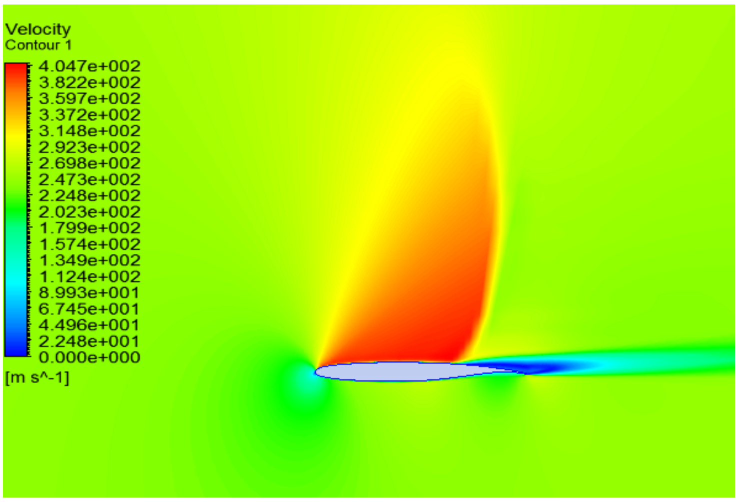

Fig. 4. Velocity contours at 5^0AoA

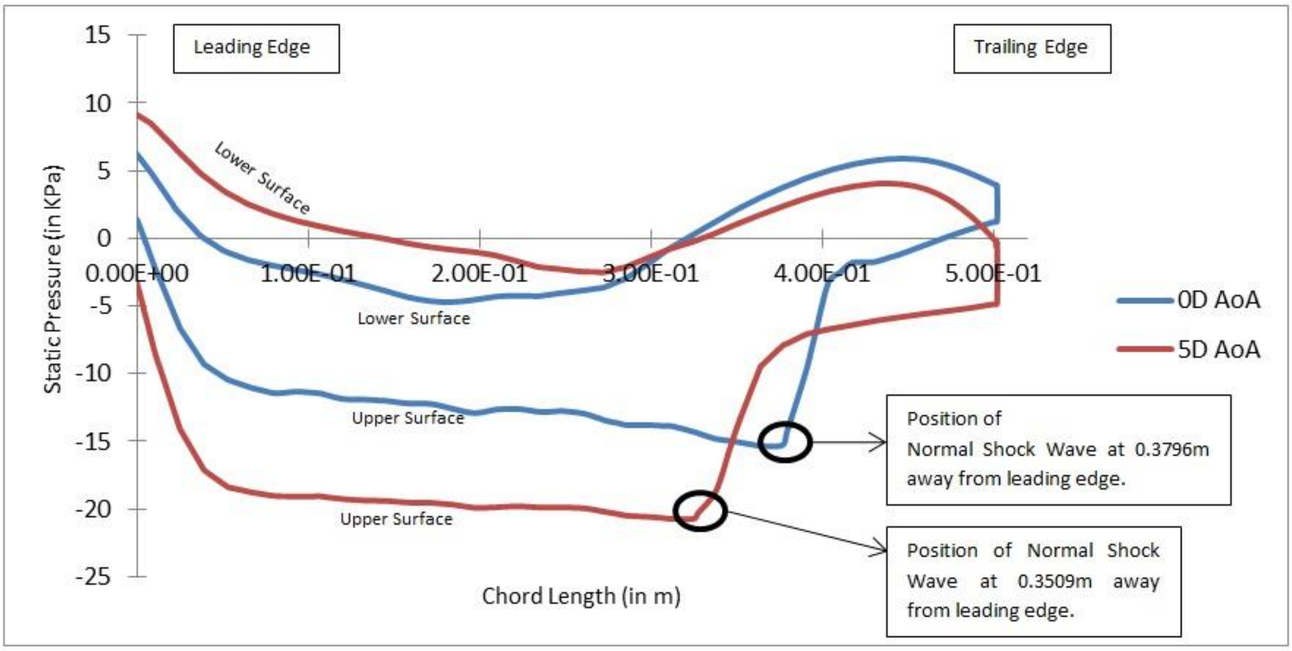

Fig. 5. Chord Length vs. Static Pressure at 0^0 and 5^0AoA at Mach 0.8

3.2 Test Case 2 - 0^0 and 5^0AoA at 2% Turbulence Intensity at Mach 0.9

The behaviour of the flow changes as the Mach number increases from Mach 0.8 to Mach 0.9. Figure 6 indicates that at 0^0 AoA, the flow characteristics over airfoil's upper and lower surface are nearly the same till 30% of the chord length making airfoil inactive at that location. Then there is a difference in pressure at the aft side of the airfoil which leads to a minimal amount of lift generation. The weak Mach wave formed on the airfoil's lower surface in test case 1 has developed in to a Normal shock wave in test case 2. Hence the velocity on both the surface of airfoil is supersonic. As the AoA is increased to 5^0 (Figure 7), the velocity of the flow on the airfoil's upper surface increases, but the lower surface experiences a decrement in velocity. This phenomenon leads to a pressure difference over the airfoil causing more lift generation. The normal shock wave at 0^0AoA has reduced to a Mach wave on the airfoil's lower surface at 5^0AoA.

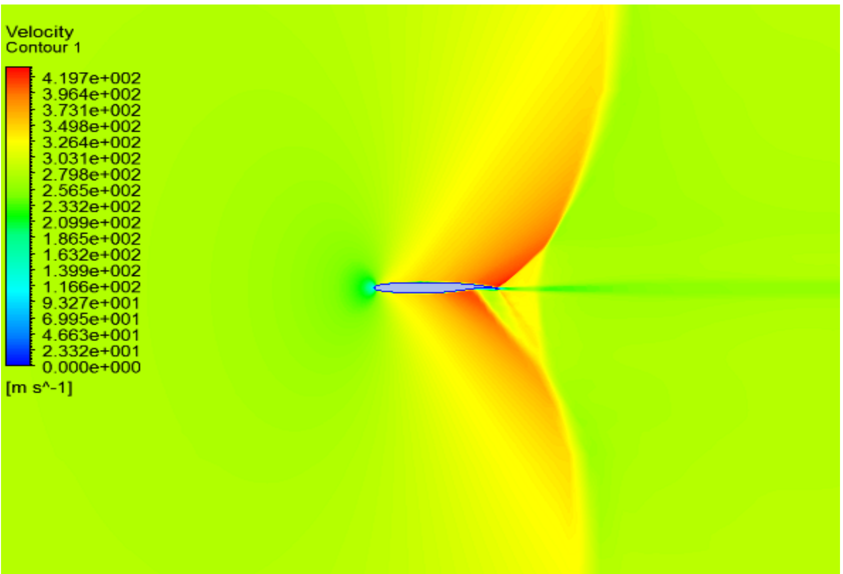

Fig. 6. Velocity contours at 0^0AoA

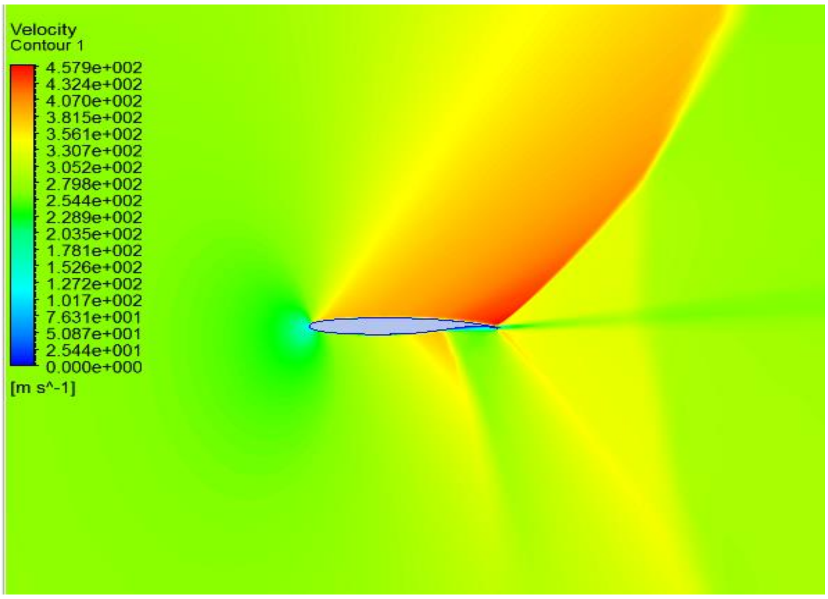

Fig. 7. Velocity contours at 5^0AoA

Comparing the two graphs from Figure 5 and Figure 8 of test case 1 and test case 2, it can be inferred that the airfoil's upper surface experiences a formation of normal shock wave but in case 2 the lower surface experiences the normal shock wave. The occurrence of this phenomenon is because of increment in velocity from Mach 0.8 to Mach 0.9. The upper surface's normal shock wave of the airfoil in test case 1 has moved well far behind the trailing edge, but in test case 2, on the airfoil's lower surface, the normal shock wave has just developed at Mach 0.9. The presence of the shock wave can be obtained from Figure 8. The steep increase in static pressure is now on the lower surface line in graph signifying the position of normal shock wave on it.

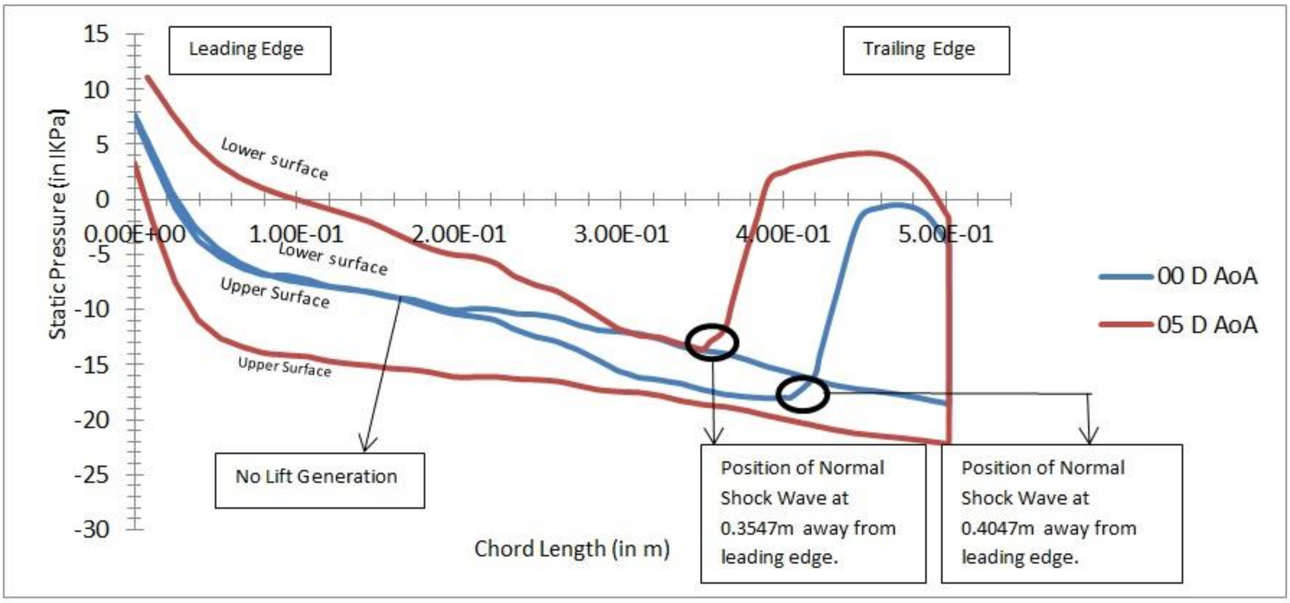

Fig. 8. Chord Length vs. Static Pressure at 0^0 and 5^0AoA at Mach 0.9

3.3 Test Case 3 - 0^0 and 5^0AoA at 10% Turbulence Intensity at Mach 0.8

Aerodynamic Performance of an airfoil will also depend on turbulence intensity of the flow. Figure 9 and Figure 10 conveys the velocity contours at 0^0 and 5^0 at Mach 0.8 and at 10% turbulence intensity level. The shock wave is formed on the airfoil's and a weak Mach wave on the airfoil's lower surface. At 0^0AoA, from leading edge of airfoil, the normal shock wave is at 0.3901m in this case, whereas the normal shock wave was at 0.3796m away from leading edge in case 1. These figures depicts that the shock wave has further moved from leading edge towards tailing edge. The C_L of the airfoil is found to be 0.3375 in case3, but in case 1 it was 0.3443, signifying that, as the turbulence intensity increases, the C_L decreases. But as the AoA increase, the shock waves position also changes. The position of the shock wave was 0.3509m away from leading edge in case 1, but in case 3 at 5^0AoA,it is 0.3250m. C_L at 5^0AoA is 0.5307 which is higher than the previous AoA. Figure 11 depicts the graph of chord length vs. static pressure

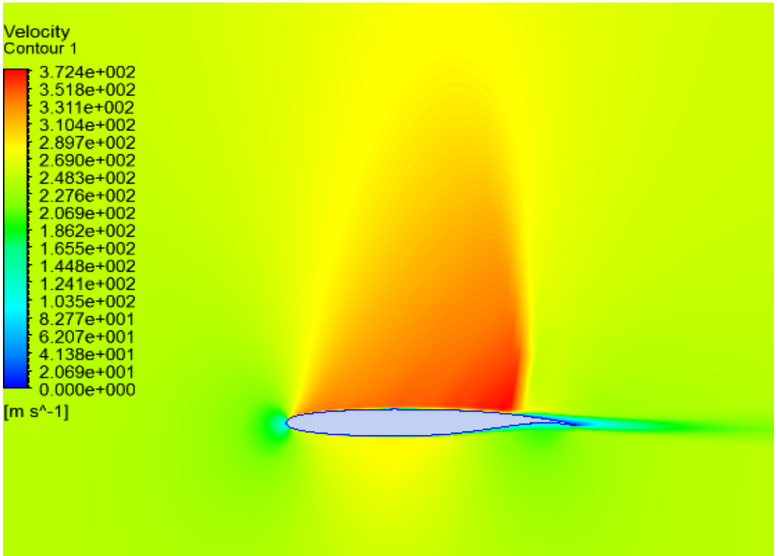

Fig. 9. Velocity contours at 0^0AoA

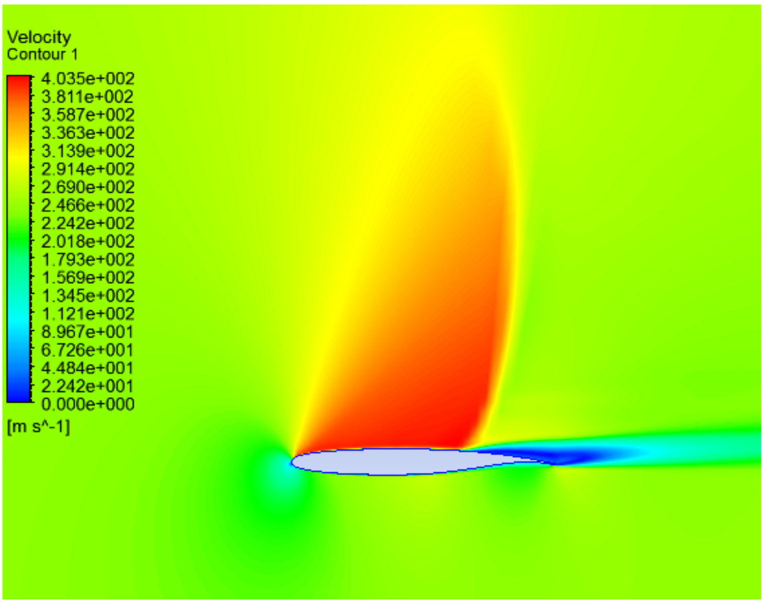

Fig. 10. Velocity contours at 5^0AoA

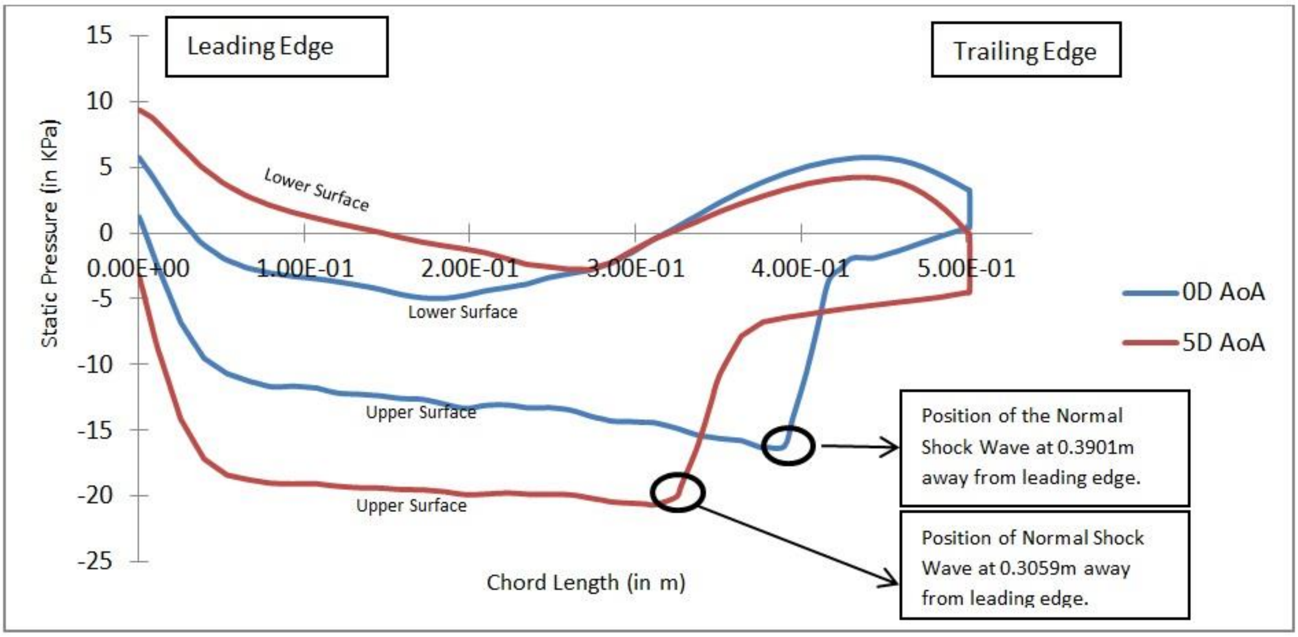

Fig. 11. Chord Length vs. Static Pressure at 0^0 and 5^0AoA at Mach 0.9

3.4 Test Case 4 - 0^0 and 5^0AoA at 10% Turbulence Intensity at Mach 0.9

The Figure 12 and Figure 13 shows the velocity contours at 0^0 and 5^0AoA whereas Figure 14 depicts the graph of chord length vs. static pressure. Unlike in the previous cases, in this case the computational results are obtained for the modelled supercritical airfoil at 0.9 Mach number for 0^0 and 5^0AoA. The shock wave generated on the airfoil's upper surface has moved far behind trailing edge. The shock wave on the lower surface is at the distance of 0.4047m from leading edge. As the AoA is increased from 0^0 to $5^0$the shock wave started moving towards from leading edge. Compared with Case 1 at 2% turbulent intensity, the position of normal shock wave at Mach 0.9 doesn't defer a long distance. But at 5^0AoA at 10% turbulence intensity the position of shock wave is 0.3547m away from leading edge where as in case 2 it was 0.3545m. C_L started decreasing as the Mach number reached to 0.9. C_L was 0.0504 at 0^0 where as it increased to 0.3178 when the AoA was increased.

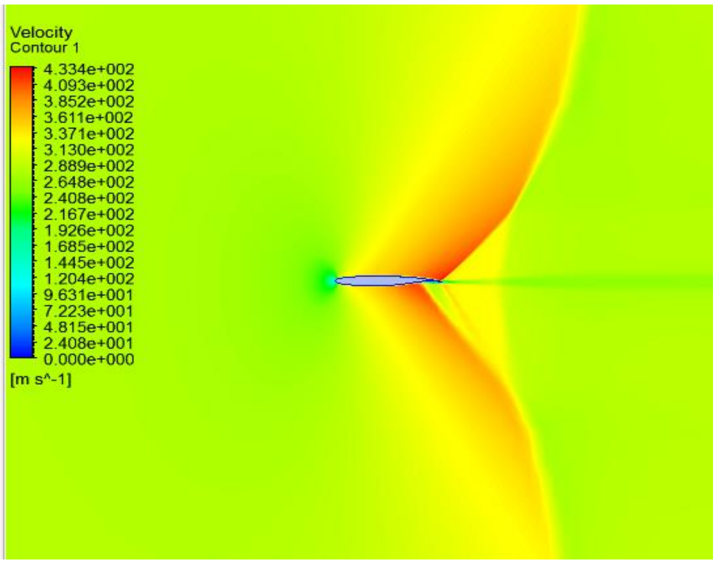

Fig. 12. Velocity contours at 0^0AoA

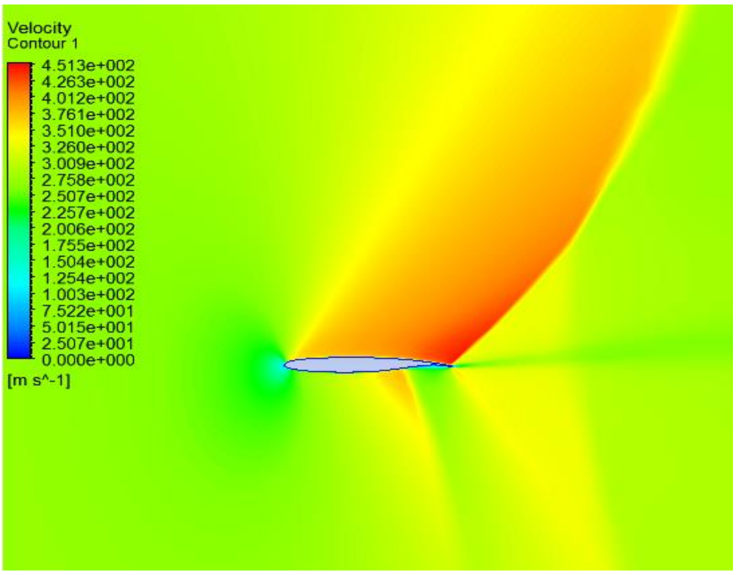

Fig. 13. Velocity contours at 5^0AoA

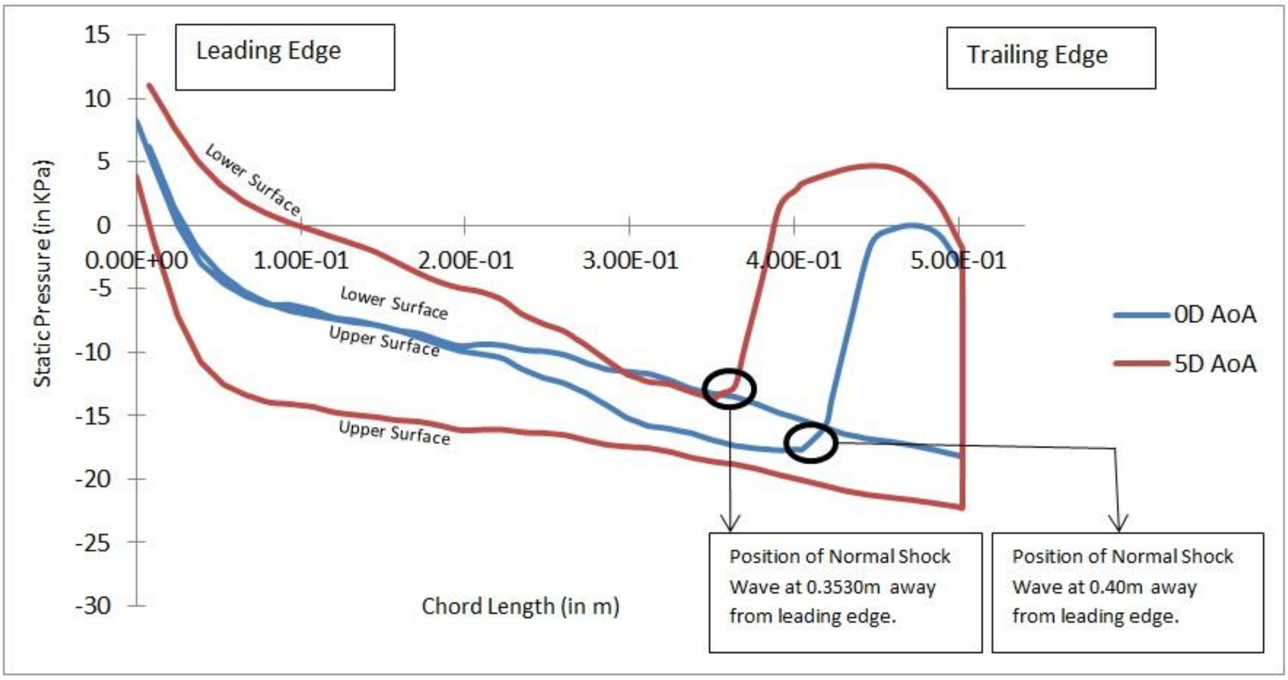

Fig. 14. Chord Length vs. Static Pressure at 0^0 and 5^0AoA at Mach 0.9

3.5 Results of Lift Co-Efficient (C_L) and Position of Normal Shock Wave for Varying AoA

Figure 15 gives an insight about the behaviour of the C_L as the AoA of the airfoil is changed at different Mach number and Turbulence Intensity (TI). The graph signifies that, there will be a reduction in C_L as the turbulence intensity increases. Increase in AoA leads to increase in C_L. It can be seen through the above graph that C_L and Mach number are inversely proportional at Mach 0.8 and Mach 0.9at both turbulence intensity levels.

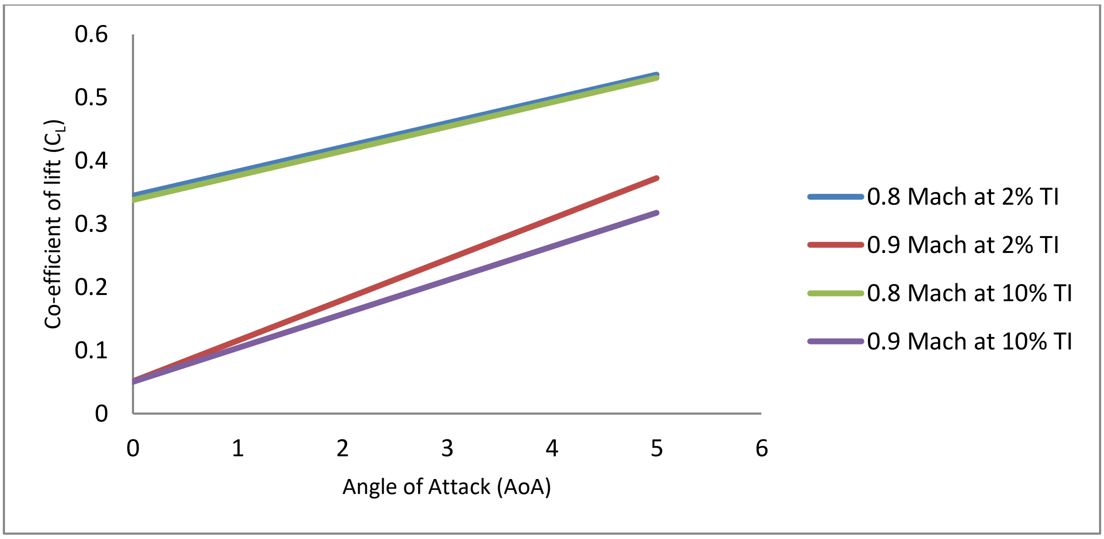

Fig. 15. C_L vs. AoA w.r.t Mach Number and TI

The graph in Figure 16 shows the position of normal shock wave on the chord length of the airfoil. We have observed from previous cases that the normal shock wave appears on the airfoil's upper surface at 0.8 Mach number where as a weak Mach wave on the lower surface. But at 0.9 Mach number the shock wave formed on the airfoil's upper surface, propagates far away from the trailing edge and the weak normal shock wave on the airfoil's lower surface will develop in to a strong normal shock wave. Hence at Mach 0.8, the shock wave is on the airfoil's upper surface and at Mach 0.9 it is on the lower surface. From the above graph, it can be inferred that as the AoA and turbulence intensity increases keeping Mach number constant, the shock wave moves towards leading edge. The increase in Mach number leads to the movement of shock wave away from leading edge.

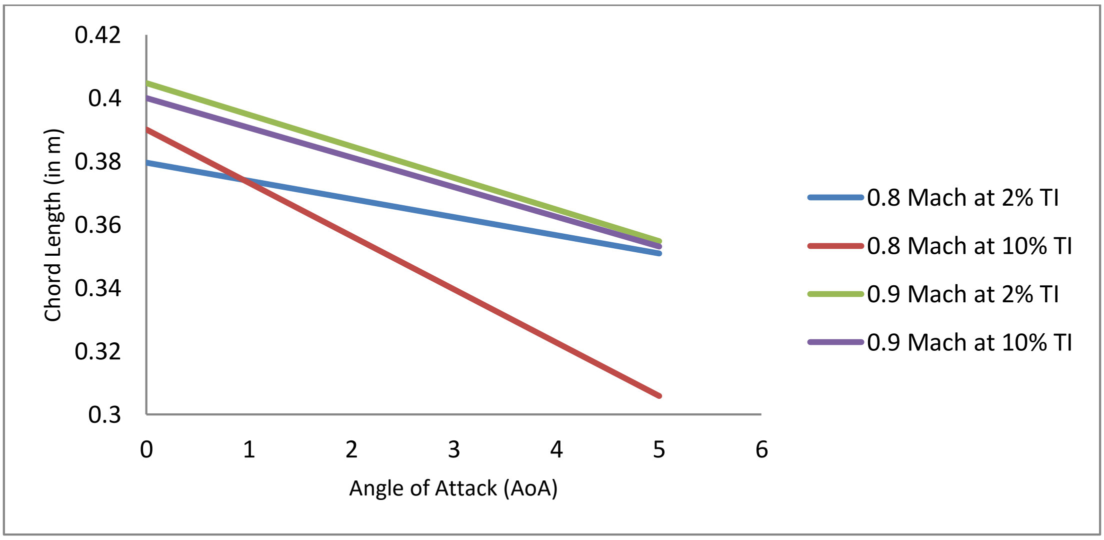

Fig. 16. Position of shock wave on the chord length vs. AoA. B

4. Conclusions

The computational fluid dynamics analysis was carried out at different turbulence intensity levels on a supercritical airfoil SC20412 at transonic velocity regimes. The analysis was also carried out at different AoA. The conclusions inferred from the results are presented below.

i. It was found that the increment in turbulence intensity leads to the decrement in C_L at both AoA and Mach numbers.
ii. As the velocity increases from Mach 0.8 to 0.9, C_L decreases due to the formation of supersonic flow over both the surface of the airfoil. But as the AoA increases from 0^0 to 5^0 at Mach 0.9, the airfoil experiences a decrement in velocity on the lower surface, which leads to the increase in C_L.
iii. The normal shock wave is formed as the accelerated flow reaches the supersonic velocity on the airfoil's upper surface. The shock wave starts propagating towards the leading edge as the AoA increases.
iv. As the AoA and Mach number increases, due to the low pressure formed on the upstream of the normal shock wave, the shock wave starts moving towards the leading edge of the airfoil.
v. The lift is generated due to the pressure difference over airfoil. The pressure distribution over the airfoil changes as the AoA, velocity and turbulence intensity changes.

Hence, the conclusions obtained from the results gives an insight about the behaviour of C_L and normal shock wave at two major transonic regimes and at different turbulence intensity levels.

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