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A Study on Transonic Aerodynamics of Axial Compressor Cascade Blades at Higher Altitudes

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ABSTRACT: Axial compressor plays a vital role in jet engines. As the requirement of an aircraft becomes to fly at higher velocities, the velocity with which the air approaches the compressor blades is also higher. Hence this research focuses on understanding the behaviour of that high speed (transonic) flows through the axial compressor cascade blades at varying Angle of Attacks (AoA) and at different altitudes. Static pressure is the main parameter of the flow which is monitored at different boundary conditions. Computational Fluid dynamics approach was used to simulate the flow over cascade blades. The cascade blades are modelled in Ansys Design Modeller. ICEM CFD tool is used to mesh the modelled cascade. The modelled and meshed cascade blades are analysed at 0°, 5°, 10°, 15°, 20°, and 25° AoA at 4km, 6km, and 8km altitudes. When the results are post-processed, it was found that, as the flow passes through the cascade blades, due to high velocity, shock waves are formed. A weak oblique shock wave is found at the leading edge of each blade, and a normal shock wave inside the cascade. The pressure rise was due to shock diffusion. Increase in AoA resulted in static pressure raise and also the normal shock wave started moving towards the leading edge. Static pressure raise started decreasing as the altitude increased. It was found that to maintain the same pressure rise at higher altitudes, the AoA should be increased. The flow over the 3rd blade started separating as the AoA reached 20° and above. The position of the shock wave and its behaviour is monitored. Rise in static pressure at each AoA and altitude is monitored and presented in the paper.

KEYWORDS: Axial compressor, Transonic flow, Static pressure, Angle of Attack (AoA), Altitude, Normal shock wave

I. INTRODUCTION

An aircraft flies by generating lift from its wings. This lift is generated due to the flow of air over the wings. For the air to flow over the wings, the aircraft should move forward. This forward force is produced by their engines. Turbojet, turboprop, turbo shaft, are the different kinds of engines used in aircrafts. Jet engine consists of 5 major parts namely, intake, compressor, combustion chamber, turbine, and nozzle. The compressor plays a leading role in a jet engine. Jet engine compresses the atmospheric air to higher pressures and then ignites it with adding fuel. While the combusted gas expands, the work is extracted from it. Hence compressing the air to higher pressures is the most important task of jet engine. There are two different kinds of compressors which does this task, they are, Centrifugal compressor and Axial Compressor. Axial compressors are the main focus of this study.

Certainly named as rotating airfoil based compressor, it continuously pressurizes the gasses. The fluid flows parallel to the axis of the rotation of the compressor, due to which it has been named as the axial compressor. The airfoil blades for this kind of compressor are such that the inlet area between the blades is less than the outlet area. This unique shape

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Vol. 8, Issue 3, March 2019

helps the air to diffuse inside the cascade blades. Due to the increase in area from the inlet to the outlet of the cascade, the cascade acts like a diverging duct which helps air to diffuse easily. But in this study, an attempt has been made to understand the flow behaviour at transonic velocity through the cascade blades. The major importance to choose transonic flow is that, at these velocities, the behaviour of flow is different. As the transonic flow approaches the compressor blades, the flow accelerates. This acceleration causes the flow to reach sonic speed inside the cascade blades. At sonic speeds, there are shock waves formed inside the cascade blades. To understand this shock wave behaviour at varying Angle of Attack (AoA) and at different altitudes is the aim of this research. Hence the analysis is performed on the axial compressor cascade blades from 0° AoA to 25° AoA in the difference of 5° AoA at 4 km, 6 km and 8 km altitudes.

Transonic flow is a major regime where the flow behaviour is unpredictable even in the 21st century. Two researchers namely Roberto Biollo and Ernesto Benini worked on understanding the recent trends in axial compressor aerodynamics at transonic velocity [1]. They did a blade profile study to reduce the flow velocity upstream of the passage shock. Conclusion stated that, a deep understanding of losses associated with the relative inherent supersonic flow was possible when computational techniques and optical measurement techniques were progressed. Rotor tip clearance plays an important role in axial compressors. S Satish Kumar, RanjanGanguli, S B Kandagal, Soumendu Jana, has carried out extensive research on a transonic compressor by applying a time-averaged numerical simulation on average rotor tip clearance in axial compressors [2]. It was found that performance characteristics of the transonic axial compressors at 50% and 100% of the designed speed. The results showed that a low adverse pressure gradient is experienced by the tip leakage vortex when the compressor is running at the choked condition. This can lead to a phenomenon which is purely known as surge/compressor stall.

Katsushi Nagai, Kazuaki Ikesawa, Takao Sugimoto, Toshinao Tanaka, Hiroshiumino, and Takeshi Ryuguji designed a transonic axial compressor with Double Circular Arc Airfoil (DCA), Multi Circular Arc Airfoil (MCA) [3]. To cope with the precise treatment, the design heavily relied on Navier-Stoke code, Euler code, and CFD techniques. The overall test was carried out at 80%, 90%, and 100% of the designed speed on the DCA and MCA compressor blades. The result states that, for DCA compressor, the adiabatic efficiency was 2% lower than MCA compressor. The paper also emphasizes that in DCA compressor, there is a small normal shock wave formed at the inlet of the blades which is the reason for the reduction in efficiency. Their paper concluded by stating that in transonic regimes, the MCA airfoils are more suitable than the DCA airfoil.

Shock waves are the major part of transonic flows. Shockwave induces turbulence and increases the entropy. Hence elimination or controlling of these shockwaves is more important. A research done by Yangang Wang, Arvind G. Rao and Georg Eitelberg give a significant method to control the shock wave generated inside the cascade [4] at transonic flows. The two methods which they have prescribed are; suction type and blowing type. They have analyzed the sequence of blowing and suction at four different points on the cascade blades. When post-processed, it was understood that the separated flow downstream of the normal shock wave can possibly be made not separated by blowing the air from the blades towards the flow. But when the method of suction was incubated at the downstream of the shock wave where the flow is separated, it may decrease the momentum of the wall boundary layer. Due to this, there will be an expansion effect inside the passage and hence resulting in an overall reduction of shock strength.

Using large eddy simulation with free stream disturbances [5], John Leggett, Stephan Priebe, AamirShabbir, Vittorio Michelassi, Richard Sandberg, and Edward Richardson carried out loss prediction model on axial compressor cascade at off-design incidence angles. They used RANS and LES model for their simulation and compared the results. Their result concluded by saying that at off-design conditions a thorough verification is required for LES when compressor profile is optimized using RANS model under various incidence angles. In June 2018 a set of researchers worked on obtaining a design geometry which would permit high-level aerodynamic performance for axial compressors [6]. To increase the performance level, they developed a method based on optimization of adaptive feature which was known as 'surrogate model'. The conclusions were that a successful model is developed to optimize the blade geometry. A numerical simulation at low Reynolds number on transonic axial compressor rotor [7] was carried out by T. Arima, T.

International Journal of Innovative Research in Science, Engineering and Technology

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Vol. 8, Issue 3, March 2019

Sonoda, M. Shiratori, A. Tamura, and K. Kikuchi using K- ϵ model. They did the analysis for Rotor 37 and Rotor 67 and compared the results. The result showed that the Rotor 37 was quantitatively good at design conditions rather than rotor 67.

An experimental investigation was done on double circular arc airfoil [8] by NASA. The peak efficiency was about 0.92 (92%) at lower speeds and even at 60% designed speed. At transonic flow regime, the stall occurred as the weight flow rate decreased. An experimental study was done on aft swept axial transonic compressor stage [9] by Stephan Kablitz, Jörg Bergner, Dietmar K. Henneke, Manfred Beversdorff and Richard Schodl. The result said that the blades experienced for too negative incidence after 60% of chord length.

With shock induced corner stall, Anton Weber, Heinz-Adolf Schreiber, Reinhold Fuchs, and Wolfgang Steinert did an experiment on compressor cascade with the 3D transonic flow [10]. The results showed the formation of passage vortex. The result showed that, by successive code development especially in the corner region, improvements in corner results can be obtained. The interaction of fluid and the compressor blades [11-15] are important in understanding working of principles of compressors. N. M. McDougall, N. A. Cumpsty, T. P. Hynes worked on axial compressor stall inception [16]. The tip stall occurs when the rotor tip clearance is small, and the crucial point of stall is because of boundary layer formed at end walls due to blockage.

M. Pandey, S. Chakraborty, K. Deb has done a flow simulation analysis on compressor cascade using CFD [17]. They have worked on finding the effect of AoA on the static and dynamic pressure behavior of the flow inside the cascade. The conclusion is that, at a slightly higher AoA, the pressure raise was more and advantageous. John R. Erwin, Melvyn Savage, and James Emery J carried out an experiment on the 2-D compressor at low speeds [18]. The result stated that the better high-speed capabilities were obtained by A₆14b blade at high solidities and low inlet angles rather than A₂1B or the A₁₀ blades. A 1-D shock loss model on a supersonic axial compressors cascade [19] was proposed by Hengtao Shi, Baojie Liu and Xianjun Yu in their paper. The concluding remark was that, at a fixed transonic number and at an optimal shock angle, a total pressure re-cover co-efficient can be obtained. As the subsonic flow has a supersonic velocity component in the axial direction, the compressor blades experience supersonic flows. Hence there will be a formation of shock waves at the leading edge and across the cascade. R. E. York and H. S. Woodard worked on the investigation of the effect of varying leading edge radius on the behavior of shock waves using rotational flow computational program [20]. The result stated that the model accounts for the bow shocks which will be formed at the leading edge of the blades.

Understanding the Mach number effects on compressor blades is important. A research paper by John F. Klapproth, did a study on the effect of Mach number on compressor blades [21]. It showed that at higher Mach number, the use of inlet guide vanes can be eliminated and operating at higher pressure ratios can minimize the excess tip loading. Aldo Rona, Renato Paciorri and Marco Geron have confirmed the periodicity gain in compressor cascade by building and testing the axial compressor cascade blades model [22]. The pressure measurements were done with and without optimized test tail boards. A study on swept and lean transonic compressors blades (Rotor 35) [24] was carried out by Nevin Francis and J. Bruce Ralphin Rose which gave a radical solution that the performance of the swept and leaned blade was higher than Rotor 25. And the boundary layer growth was also minimized. To understand the flow behavior, visualization is most important. A paper by John A. Slireski and William Roberts B. has visualized the transonic flow using smoke flow visualization [25] to understand the shock losses induced in the compressors due to shock waves. The results showed it as one of the best model to visualize the flow which also concluded by saying that even this technique can be applied to determine induced drag.

II. METHODOLOGY

Before starting the simulation over the cascade blades, the blades have to be rightly modelled. For axial compressor cascade blades, NACA 65210 of NACA 65 [23] series airfoil is chosen. The reason behind choosing this airfoil is that it has a maximum thickness at 10% of chord and 0.2 as its lift coefficient. The coordinates were obtained from airfoil

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Vol. 8, Issue 3, March 2019

tools and imported to geometric modeller of Ansys Workbench. As the blades were modelled, a control volume was built across the blades. To match the 0° AoA of the blades, the control volume is inclined at an angle. The control volume acts like a rectangular duct in which the cascade is placed through which the fluid flows at transonic speed.

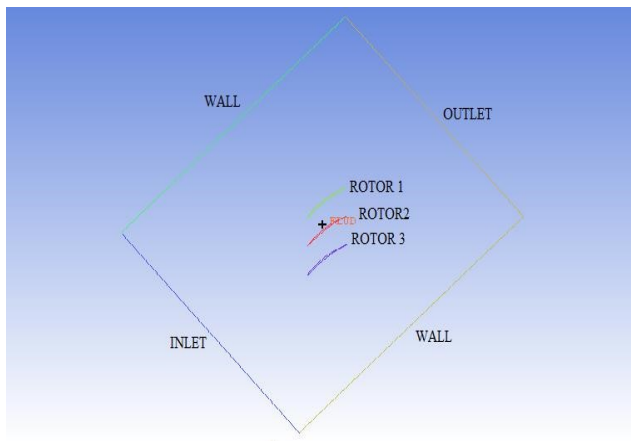


Fig 1 Geometric modelling of blades and control volume

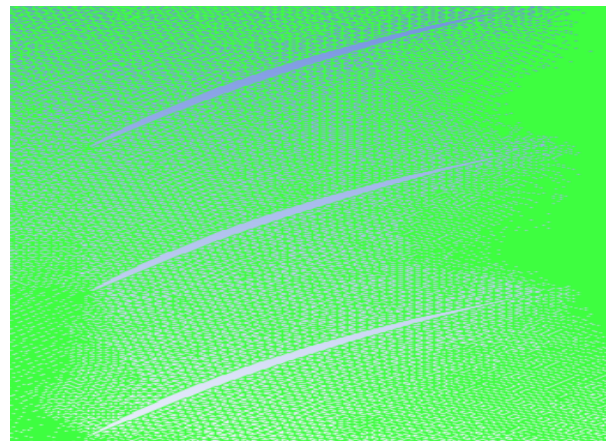


Fig 2 Triangular mesh across the cascade blades

The obtained model from geometric modeller is meshed using ICEM CFD. The model was meshed using an unstructured grid. A triangular unstructured mesh was generated across the blades which were inside the control volume resulting in 627505 elements (cells) and 313747 nodes.

Boundary conditions:-

- Inlet \rightarrow Velocity Inlet \rightarrow 295.78m/s
- Outlet \rightarrow pressure Outlet \rightarrow 0 bars
- Walls \rightarrow Wall
- Rotor 1, 2, 3 \rightarrow Walls
- Altitude \rightarrow 4km, 6km, and 8 km
- Angle of Attack (AoA) \rightarrow 0° - 25° AoA

III. EXPERIMENTAL RESULTS

The simulation of the cascade blades was done in ANSYS FLUENT. Computational Fluid dynamics approach was used to understand the behaviour of flow through cascade at transonic velocity. The analysis was carried out at an incremental AoA of 5° from 0° to 25° at 4 km, 6 km, and 8 km altitudes. The obtained results are presented from Fig. 3 to Fig. 15. As the transonic flow approaches the upper surface of the cascade blades, due to the reduction in area, the flow accelerates to supersonic velocity. When the flow reaches the diverging section of the cascade blades, the flow diffuses through a normal shock wave. This is the main reason for the diffusion of air in the cascade. Hence pressure raise at these velocities is purely due to shock diffusion. The static pressure of the air is raised when it passes through the normal shock wave. At all altitudes the behaviour was similar, but the amount of raise in static pressure varied. The flow separation was found at 20° AoA

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Vol. 8, Issue 3, March 2019

Test Case 1 → 4 km Altitude

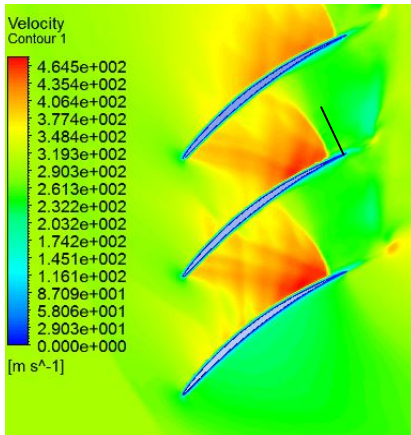


Fig 3 Velocity contours at 0° AoA

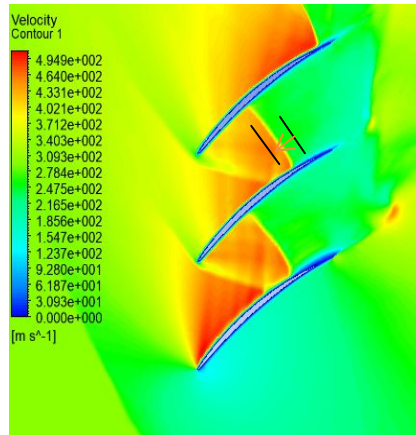


Fig 4 Velocity contours at 10° AoA

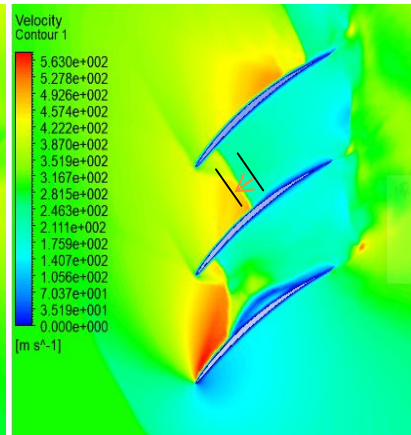


Fig 5 Velocity contours at 20° AoA

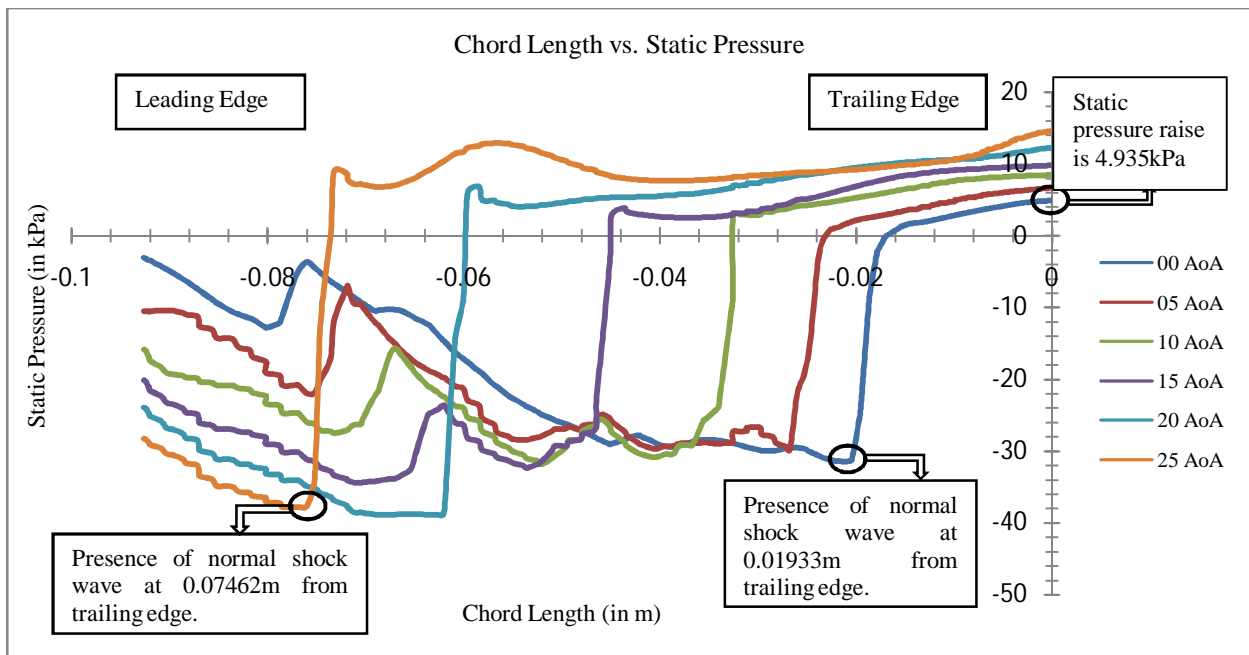


Fig 6 Chord Length vs. Static Pressure graph at 4 km altitude and at different Angle of Attacks (AoA)

The above velocity contours show the behaviour of flow at 0°, 10° and 20° AoA. Fig. 6 shows the variation of static pressure along the chord length at transonic speed through the cascade from 0°-25° AoA at 4km altitude. We can observe that there is a formation of a normal shock wave inside the cascade in all 3 figures. In Fig. 6, it is represented by a steep increase in static pressure at all AoA. By analysing the velocity contours and graph we can notice that as the AoA increased, the normal shock wave starts moving towards the leading edge of the blade. At 0° AoA the normal shock wave is at a distance of 0.01933m from trailing edge. But at 25° AoA, the same shock wave is at a distance of nearly 0.07462m from the trailing edge. The static pressure raise at 0° AoA is 4.935 kPa. The flow separation started at

International Journal of Innovative Research in Science, Engineering and Technology

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Visit: www.ijirset.com

Vol. 8, Issue 3, March 2019

20° AoA on rotor blade 3 as shown in Fig. 5. Hence the optimum point of operation at this altitude is in between 15°-18° AoA at which the pressure raise is approximately 9kPa to 10kPa.

Test Case 2 → 6 km Altitude

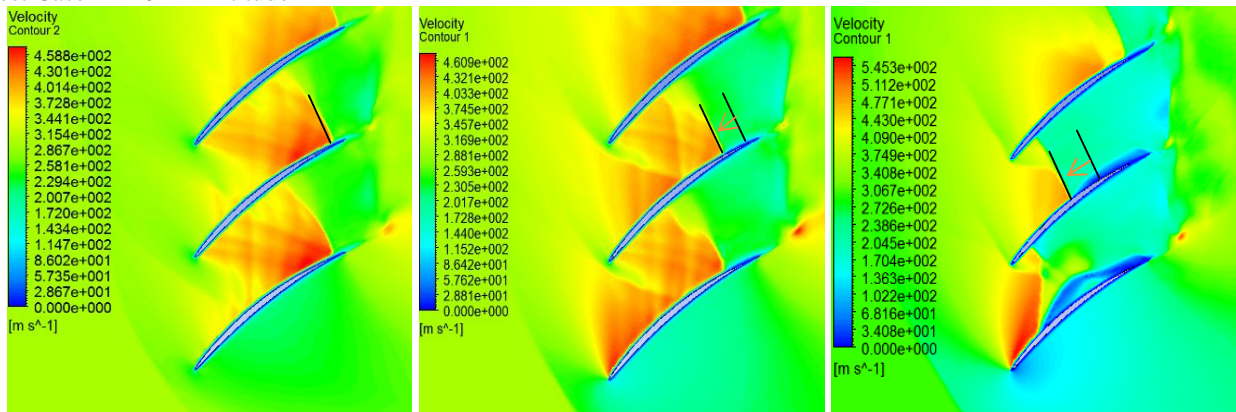


Fig 7 Velocity contours at 0° AoA

Fig 8 Velocity contours at 10° AoA

Fig 9 Velocity contours at 20° AoA

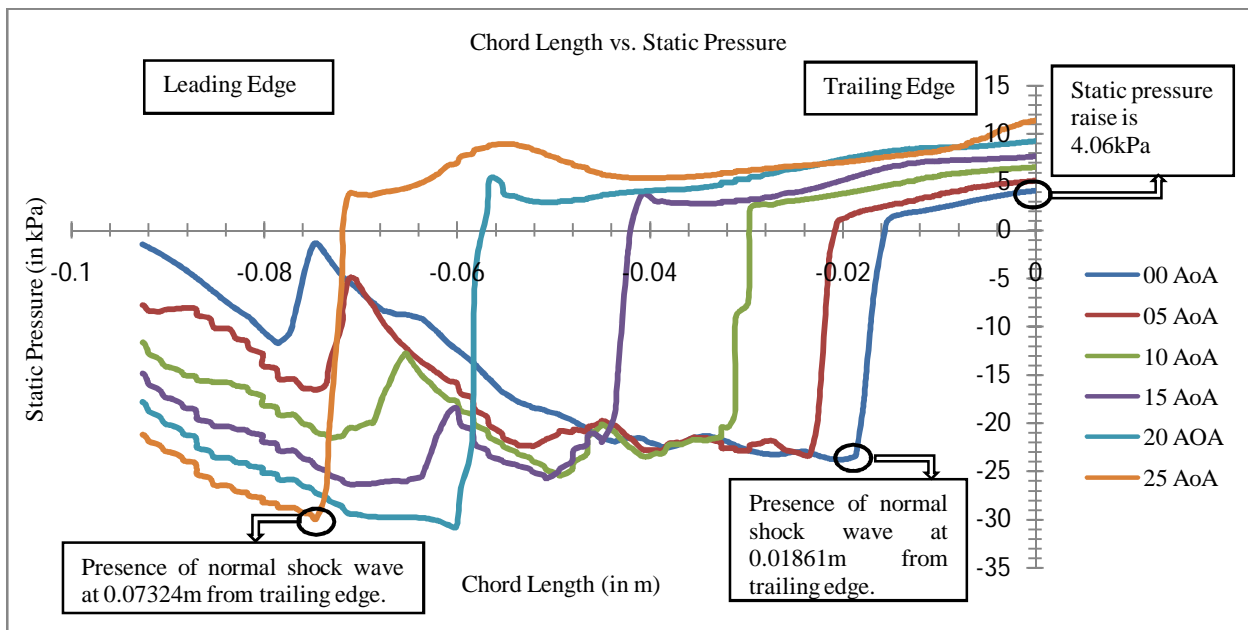


Fig 10 Chord Length vs. Static Pressure graph at 6 km altitude and at different Angle of Attacks (AoA)

As the altitude increases, behaviour of the flow also changes. Above figures from Fig. 6 to Fig. 8 represent the velocity contours at 6km altitude. At higher altitudes, due to the decrease in atmospheric pressure and density, the amount of pressure raise is low. At 4 km 0° AoA the amount of pressure raise and the position of the shock wave were 4.935kPa and 0.01930m from trailing edge, but in this test case, the pressure raise and the position of the shock wave is 4.06kPa and 0.01861m from the trailing edge respectively. The position of the normal shock wave has moved towards trailing edge as compared to the Test Case 1. The occurrence of this phenomenon is due to the reduction of pressure at higher altitudes. The normal shock wave travelstowards the leading edge. But the pressure raise obtained at 4 km altitude

International Journal of Innovative Research in Science, Engineering and Technology

(A High Impact Factor, Monthly, Peer Reviewed Journal)

Visit: www.ijirset.com

Vol. 8, Issue 3, March 2019

cannot be obtained at this altitude at a particular AoA. The flow separation was found to be at around 20° AoA. The steep increase in static pressure in Fig. 10 represents the presence of normal shock wave. A high fluctuation in static pressure is observed at 20° and 25° AoA in Fig. 10 after the flow pass through the shock wave, which is due to the turbulence induced by flow separation.

Test Case 3 → 8 km Altitude

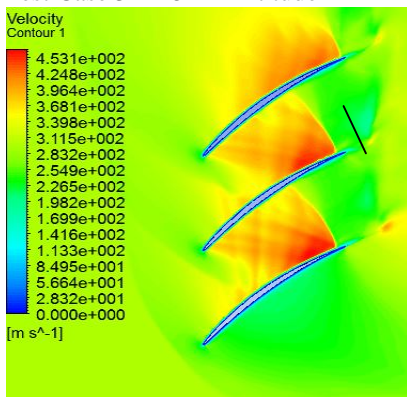


Fig 11 Velocity contours at 0° AoA

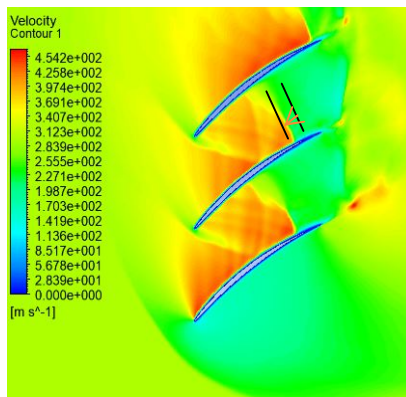


Fig 12 Velocity contours at 10° AoA

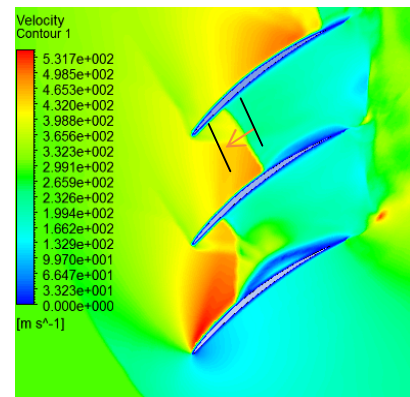


Fig13 Velocity contours at 20° AoA

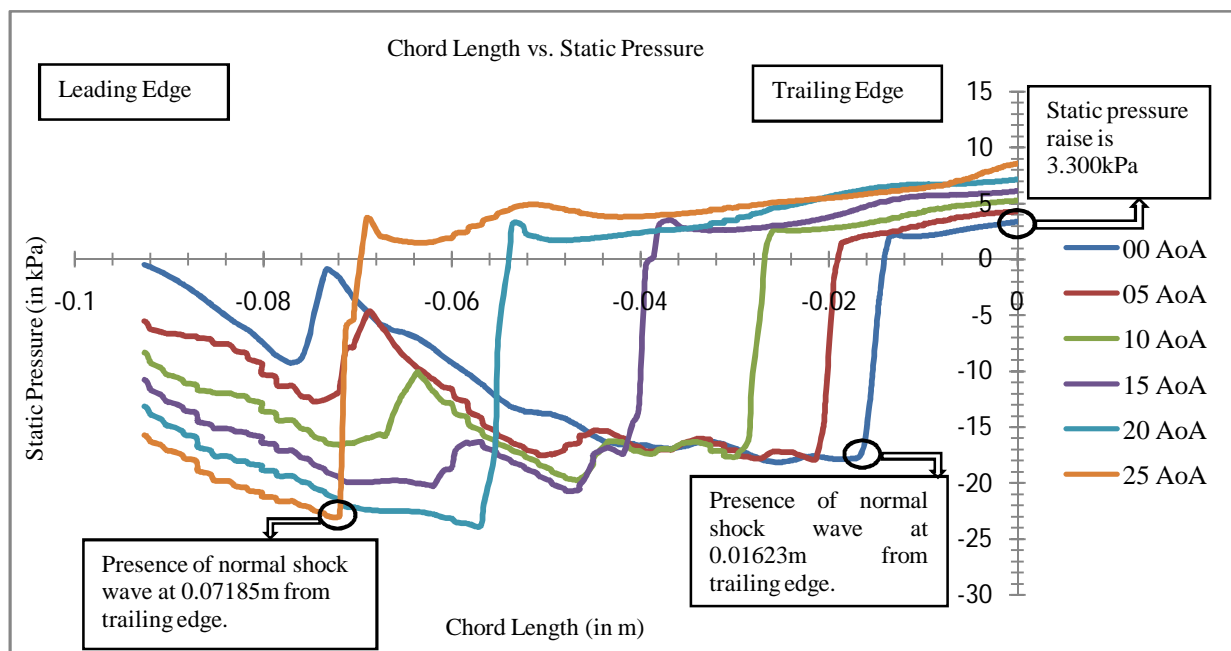


Fig 14 Chord Length vs. Static Pressure graph at 8 km altitude and at different Angle of Attacks (AoA)

The behaviour of normal shock wave purely depends on the upstream flow properties. This is clearly represented in above Fig. 14. At this AoA and altitude, the normal shock has been moved towards the trailing edge when compared to the previous altitudes. At 0° AoA, the amount of pressure raise and the position of the normal shock wave are nearly 3.3kPa and 0.01623m from trailing edge. Fig. 14 shows that, there is an increase in pressure even after the flow passing through the normal shock wave; this is due to the diverging section of the compressor blades. The normal shock wave

International Journal of Innovative Research in Science, Engineering and Technology

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Vol. 8, Issue 3, March 2019

travelled towards the leading edge as the AoA increased. Due to the further reduction in atmospheric pressure and density at 8 km altitude compared to 4 km and 6 km, the amount of maximum pressure raise obtained at this altitude is around 6.5kPa to 7kPa. We can clearly observe from the previous test case that 6kPa to 7kPa was achieved at 10°-12° AoA at 4 km Altitude. But in this case, it is obtained at around 16°-18° AoA. The flow separation was started at 20° AoA. Figure 9 shows the flow separation on the rotor blade 3 which results in turbulence.

Test cases/Altitude	Angle of Attack AoA	Position of normal shock wave from trailing edge (in m)
Test case 1 4km	0°	0.01933
	5°	0.02677
	10°	0.03669
	15°	0.04548
	20°	0.06196
	25°	0.07462
Test case 2 6km	0°	0.01861
	5°	0.02234
	10°	0.03113
	15°	0.04255
	20°	0.05845
	25°	0.07324
Test case 3 8km	0°	0.01623
	5°	0.02043
	10°	0.02819
	15°	0.04020
	20°	0.05543
	25°	0.07185

Table 1 The above table shows the normal shock wave's position at 4km, 6km, and 8km altitude from 0°-25° AoA, obtained from the simulation of transonic flow over the axial compressor cascade blades.

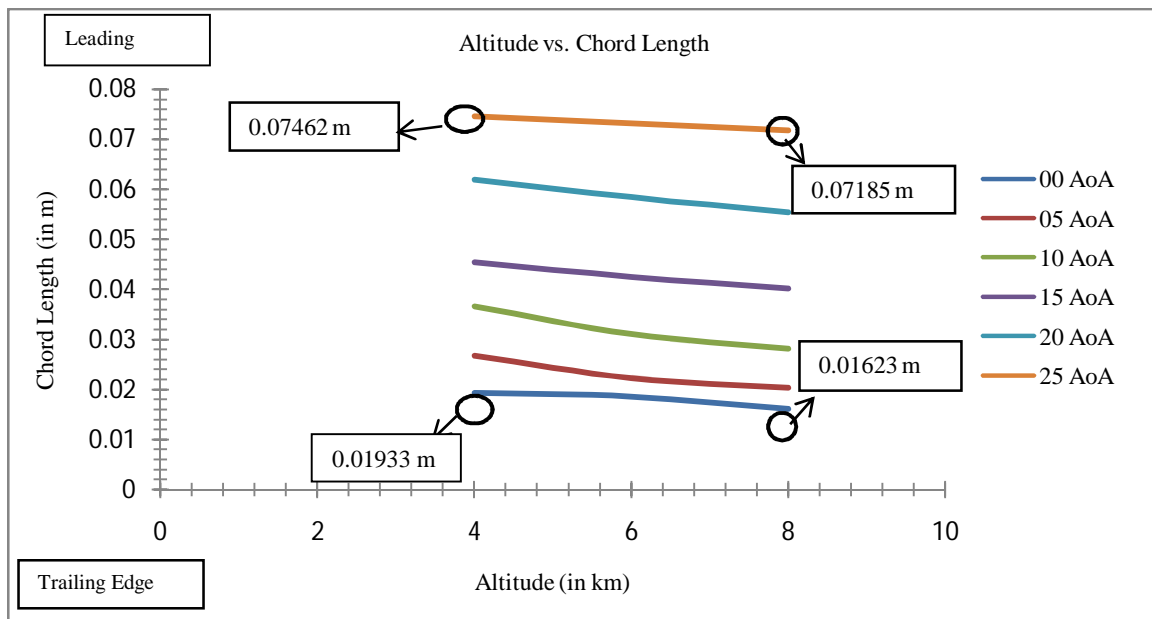


Fig. 15 Altitudes vs. Chord Length graph at different AoA (all the distance mentioned on the graph are from trailing edge)

International Journal of Innovative Research in Science, Engineering and Technology

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Vol. 8, Issue 3, March 2019

The above graph shows that, at a particular AoA, if the altitude is varied, the normal shock wave starts moving towards the trailing edge. It can be observed in the Fig. 15 that at 0° AoA and at 4km altitude, the normal shock wave is at 0.01933m from trailing edge, but at 6km, it is at 0.01861m, which indicates that the normal shock wave is moving towards the trailing edge.

IV. CONCLUSION

Analysis of axial compressor cascade blades at transonic velocity (295.77m/s) at different AoA and altitudes was carried out. When the flow passes through the compressor cascade, the static pressure inside the cascade raises above atmospheric pressure. As the AoA is increased, the amount of static pressure raise started increasing. This phenomenon is the same at all three altitudes. The properties of air is different at different altitudes, hence as the altitude is increased, the pressure raise per angle of attack started decreasing. The decrease in pressure raise per AoA at different altitudes is due to the decrease in pressure and density of the air at higher altitudes. As the altitude increases, to maintain the same amount of pressure raise, the angle of attack has to be increased. When the air flows over the cascade blade, the flow accelerates and reaches supersonic velocities; hence there will be the formation of a normal shock wave inside the cascade. The increase in AoA at a particular altitude resulted in normal shock wave moving towards the leading edge. At particular AoA, as the altitude increases, the shock wave starts moving away from the leading edge. This means, if the position of the shock wave at 0° AoA at 4km and at 8km is analysed, the shock wave is farther from the leading edge at 8km when compared with 4km altitude. Flow separation was started at 20° AoA.

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Vol. 8, Issue 3, March 2019

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