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RESEARCH ARTICLE

SUBSONIC AIRCRAFT WING PERFORMANCE AT TRANSONIC SPEEDS AND AT DIFFERENT TURBULENT INTENSITY LEVELS

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ABSTRACT

Aircraft's both for combat and civil purpose work under different environment conditions, especially when it is a combat Aircraft or a jet trainer it will be more often subjected to high level of turbulence due to max manoeuvring and shorter runway takeoff operations. The environment conditions at which civil and combat aircraft operates are unpredictable and will change from one location to other based on weather and climate conditions. In this paper the lift and drag coefficient of a finite 3D wing of a subsonic Aircraft is presented at transonic speed conditions and at different turbulence intensity levels. 3D wing is of NACA 2412 profile. The results obtained by this work on subsonic wing can be used to compare the performance of this subsonic wing with respect to performance of Transonic and supersonic wings. A overview of Transonic aerodynamics and the Turbulence model used is presented in this paper. This subsonic wing is analyzed using K-ω SST Turbulence model and for two different AOA of 0^0 and 4^0. The Mach number of flow over the wing and the pressure plot at the wing symmetric plane at different aircraft speed and at different turbulence intensity levels are discussed in this paper. CFD software Ansys Fluent is used to analyze the aircraft wing at two different AOA and at two different transonic speeds.

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INTRODUCTION

In olden days transonic flow limitations of propeller Aircrafts kept Airplanes from flying fast enough to encounter transonic flow over the rest of the Airplane. Since in propeller Aircraft the propeller will be rotating at higher speeds than the Aircraft, the adverse transonic aerodynamic problems occurred on the propeller first, limiting the speed of Aircraft. During World war II fighters could reach transonic speeds in a dive, and major problems often aroused, one such example was the Lockheed P-38 Lightning, transonic effects prevented the airplane from readily recovering from dives at these speeds and during one flight test, Lockheed test pilot had a fatal accident. The evolution of high speed aircrafts after the propeller Aircrafts was presented in detail by Foss (1978). Later after invention of the jet engine, most of commercial Aircrafts and all combat Aircrafts transports now cruise in the transonic speed range. In this paper a 3D finite wing of a subsonic Aircraft is analyzed using CFD software Fluent at different Turbulence intensity levels ranging from 2% to 15% and at two transonic speeds of 0.7Mach and 0.9Mach. The lift and Drag coefficient of the wing at two different AOA of 0^0 and 4^0 is presented in this paper at above said two transonic speeds and Turbulence intensity levels. Earlier the lift and drag coefficient of the same 3D subsonic Aircraft wing at 0.2mach operating speed and at different AOA was discussed and presented by Balaji and Dash (2016) in their paper. The fluid domain is considered compressible since analysis is carried out at transonic conditions. k-ω SST Turbulence model is used as it was the well proven and widely accepted model for most of external and adverse Turbulence dominated flows. (Hurt, 1965) Douvi C. Eleni and others in their paper discussed about NACA 0012 aerofoil lift and drag prediction using three different turbulence models namely Spalart Allmaras, k ε Realizable and k-ω SST model. They concluded stating k-ω SST model as the most appropriate turbulence model for lift and drag coefficient estimations of a 2d aerofoil. Like this various research conclusions recommended k-ω SST model for Aircraft wing and aerofoil CFD analysis. The Fluid domain of the 3D wing analyzed is composed of Hybrid mesh with Hexahedron, prism, pyramid and Tet elements and care has be taken to ensure the cells are fine near the wing surfaces in the domain to simulate the flow accurately in the boundary layer.

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The results of this 3D subsonic wing analyzed at transonic speeds and discussed in this paper can be used in future for subsequent research work for comparison with the performance of transonic and supersonic wings.

Transonic aerodynamics

For flow over aerofoil and wings if the speed of flow is increased and if it nears transonic speeds at some point over the top surface of wing the local flow becomes sonic where the flow reaches its highest speed locally. This is called critical Mach number, the Mach number at which some point over the wing surface attains sonic condition. As the free stream Mach number increases further, a region of supersonic flow develops. Normally the flow is brought back to subsonic speed by the occurrence of a shock wave in the flow. When the Mach number increases, the shock moves aft and becomes stronger. As the Mach number continues to increase, a supersonic region and shock wave also develops on the lower surface. As the Mach number approaches one, the shocks move all the way to the trailing edge. The flow pattern over an aerofoil with different flow speed is shown in Figure 1. This typical progression of the flow leads to variations in drag, lift and pitching moment with change in Mach number.

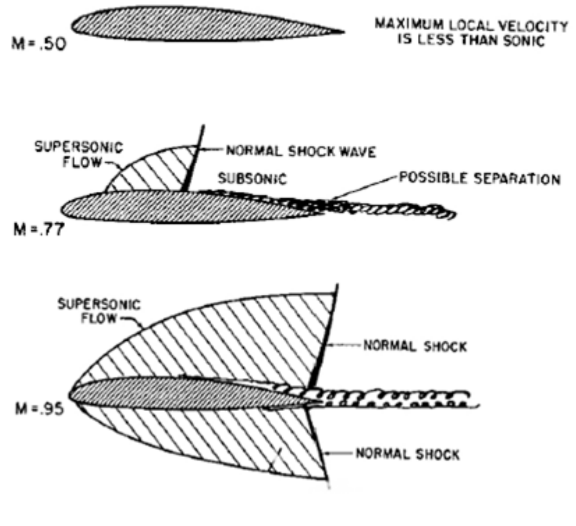

Fig.1. Characteristics of flow over Aerofoil with change in Mach Number

To predict the performance of wings at transonic speed it was so difficult in earlier days since well proven computational methods and Turbulence models were not available. In 1970's Earll Murman and Julian Cole (1970) came up with a scheme using transonic small disturbance theory, that could be used to develop a practical computational method. In Murman and Coles's scheme shocks emerged naturally during the numerical solution of the scheme equation (1).

$$(1 - M_\infty^2 - (\gamma + 1)M_\infty^2 \varphi_{\lambda i})\varphi_r + \varphi_{yy} = 0 \quad \text{- - - (1)}$$

They used finite difference approximations for the partial derivatives in the transonic small disturbance equation. The key to making the scheme work was to test the flow at each point to see if the flow was subsonic or supersonic. If it was subsonic, a central difference was used for the second derivative in the x direction. If the flow was supersonic, they used an upwind difference to approximate this derivative. The nonlinear coefficient of the φxx term in Eq (1) is a first derivative in x, φx. A central difference approximation can be used for this term. Since the solution is found by iteration, old values can be used for φx. This approach is known as "mixed differencing" and it was a simple way to capture the physics of the mixed elliptic-hyperbolic type of the partial differential equation. This method is termed a "shock capturing" method and was much simpler than "shock fitting" method at that time. After Earll Murman (1970), Antony Jameson (1979) coded up the method himself and then went on to extend the approach to solve the full potential equation in body fitted coordinates and then came up with extremely efficient flow code FLO36. The next logical development happened was to add viscous effects to the inviscid calculations, and to switch to the Euler equations for the outer inviscid flow. Now CFD software's and codes are widely used to perform this transonic and supersonic analysis. The key to transonic aerofoil design is to control the expansion of the flow to supersonic speed and its subsequent recompression. Researchers have come up with different aerofoil configuration like super critical aerofoils to have a drag rise Mach number much higher than subsonic aerofoils. In this paper the transonic performance of a subsonic wing is presented so that it can be used in later research work to comparing the performance and capabilities of subsonic wings with Transonic speed aerofoils and wing shapes.

Turbulence model

Turbulence is a kind of fluid motion which is unsteady and highly irregular in space and time. The turbulent motion has a wide spectrum of eddy sizes. The largest eddies are associated with low frequency fluctuations and responsible for most of the momentum transport. The smallest eddies are associated with high frequency fluctuations, and are determined by viscous forces. The Turbulent flows are solved using the Navier-Stokes equations together with the continuity equation. The "Direct Numerical Simulation" method in which the Turbulent flow is solved by exact equations with appropriate boundary conditions is computationally time consuming and it is rarely followed nowadays. There are two alternative methods for the prediction of turbulent flows and they are Large-Eddy Simulation (LES) and Turbulence modelling. LES is not used widely for practical flows because of the complexity it involves with increase in Reynolds number and the difficulties of applying boundary conditions. This makes the Turbulence models handy and use full in solving modern day fluid flow problems. The Turbulence models are classified as One Equation Model, Two Equation Model, Reynolds Stress Model and Algebraic Stress Model. In this the widely accepted and well proven Turbulence model is K-ω SST model as described in Introduction section.

In K-ω SST model when compared to standard $k - \omega$ model the turbulent viscosity is modified to account for the transport of the principal turbulent shear stress. This feature gives the SST $k - \omega$ model an advantage in terms of performance over both the standard $k - \omega$ model and the standard $k - \varepsilon$ model. The transport equation for SST $k - \omega$ model is given by

$$\frac{\partial}{\partial t}(\rho k) + \frac{\partial}{\partial x_i}(\rho k u_i) = \frac{\partial}{\partial x_j}\left(\Gamma_k \frac{\partial k}{\partial x_j}\right) + G_k - Y_k + \quad \text{- - - (2)}$$

$$\frac{\partial}{\partial t}(\rho \omega) + \frac{\partial}{\partial x_i}(\rho \omega u_i) = \frac{\partial}{\partial x_j}\left(\Gamma_\omega \frac{\partial \omega}{\partial x_j}\right) + G_\omega - Y_\omega + D_\omega + S_\omega \quad \text{- - - - - (3)}$$

where Γ_k and Γ_ω are the effective diffusivity of k and ω respectively. Y_k and Y_ω represent the dissipation of k and ω due to turbulence and D_ω represents the cross diffusion term. S_k and S_ω are user defined source terms. G_k represents the generation of turbulence kinetic energy due to mean velocity gradients and G_ω represents the generation of ω,

$$G_k = -\rho\overline{u_i u_j}\left(\frac{\partial u_j}{\partial x_i}\right) \quad ----- (4)$$

$$G_\omega = \frac{\alpha}{\nu_t} G_k \quad ----- (5)$$

Coefficient α is $\alpha = \frac{\alpha_\infty}{\alpha^*} * \left(\frac{\alpha_0 + Re_t/R_\omega}{1+Re_t/R_\omega}\right)$ --- (6)

where $\alpha_\infty = F_1\alpha_{\infty,1} + (1-F_1)\alpha_{\infty,2}$ and

$$Re_t = \frac{\rho k}{\mu\omega}$$

$$\alpha^* = \alpha^*_\infty\left(\frac{\alpha^*_0 + Re_t/R_k}{1+Re_t/R_k}\right) \quad ----- (7)$$

Constants $R_k = 6$, $R_w = 2.95$, $\alpha^*_0 = \frac{\beta_i}{3}$ and $\beta_i = 0.072$

Also $\alpha_{\infty,1} = \frac{\beta_{i,1}}{\beta^*_\infty} - \frac{\kappa^2}{\sigma_{\omega,1}\sqrt{\beta^*_\infty}}$

and $\alpha_{\infty,2} = \frac{\beta_{i,2}}{\beta^*_\infty} - \frac{\kappa^2}{\sigma_{\omega,2}\sqrt{\beta^*_\infty}}$

where constant $\kappa = 0.41$ and $\alpha^*_\infty = 1$ and $\beta^*_\infty = 0.09$

Also $\beta_{i,1} = 0.075$ and $\beta_{i,2} = 0.0828$ ----- (8)

The effective diffusivities in SST $k-\omega$ model is given by

$$\Gamma_k = \mu + \frac{\mu_t}{\sigma_k} \text{ and } \Gamma_\omega = \mu + \frac{\mu_t}{\sigma_\omega} \quad ----- (9)$$

where σ_k and σ_ω are the turbulent Prandtl numbers for k and ω respectively.

The turbulent viscosity $\mu_t = \frac{\rho k}{\omega}\frac{1}{\max\left(\frac{1}{\alpha^*}, \frac{\Omega F_2}{a1\omega}\right)}$ ----- (10)

where $\Omega \equiv \sqrt{2\Omega_{ij}\Omega_{ij}}$

$$\sigma_k = \frac{1}{F_1/\sigma_{k,1} + (1-F_1)/\sigma_{k,2}} \text{ and } \sigma_\omega = \frac{1}{F_1/\sigma_{\omega,1} + (1-F_1)/\sigma_{\omega,2}} \quad ----- (11)$$

Ω_{ij} is the mean rate-of-rotation tensor and α^* is defined in (7). where F1 and F2 are blending functions.

The dissipation of k and ω in SST $k-\omega$ model is given by

$$Y_k = \rho\beta^* k\omega \quad ----- (12)$$

$$Y_\omega = \rho\beta\omega^2 \quad ----- (13)$$

where $\beta = \beta_i\left(1 - \frac{\beta^*_i}{\beta_i}\zeta^* F(M_t)\right)$ and

$$\beta^* = \beta^*_i(1 + \zeta^* F(M_t)) \quad ----- (14)$$

and $\beta_i = F_1\beta_{i,1} + (1-F_1)\beta_{i,2}$

$$\beta^*_i = \beta^*_\infty\left(\frac{4/15 + (Re_t/R_\beta)^4}{1+(Re_t/R_\beta)^4}\right)$$

Compressibility function, $F(M_t)$ in (14) is

$$F(M_t) = \begin{cases} 0 & M_t \le M_{t0} \\ M_t^2 - M_{t0}^2 & M_t > M_{t0} \end{cases}$$

where $M_t^2 \equiv \frac{2k}{a^2}$, $M_{t0} = 0.25$ and $a = \sqrt{\gamma RT}$,

The constants $\sigma_{k,1} = 1.176$, $\sigma_{\omega,1} = 2.0$, $\sigma_{k,2} = 1.0$, $\sigma_{\omega,2} = 1.168$, $\zeta^* = 1.5$, $R_\beta = 8$, $\beta^*_\infty = 0.09$
$a_1 = 0.31$, $\beta_{i,1} = 0.075$, $\beta_{i,2} = 0.0828$, $\alpha_0 = \frac{1}{9}$

3D wing profile and domain

The 3D wing analyzed is shown in Figure 2.

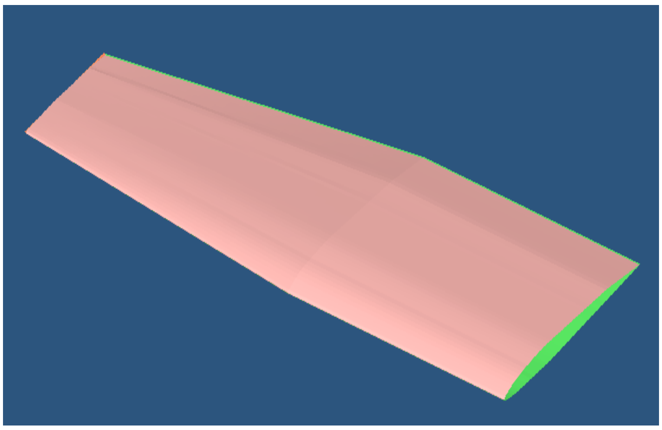

Fig.2. 3D wing of Transonic speed Aircraft

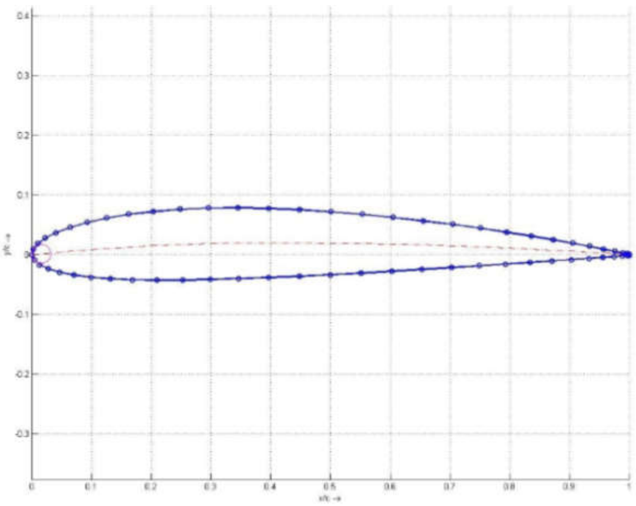

Fig.3. NACA2412 Aerofoil Profile

As seen in figure the 3d wing configuration is, for a portion of span it is straight and it is swept for the later span till wing tip. The profile of wing is NACA2412 and it is shown in Figure 3. The fluid domain is meshed with 1877281 cells to simulate the flow similar to wind tunnel. Initially convergence study of lift coefficient at transonic speed of 0.7Mach is done with varying mesh density and size near to wing surface in the domain and the domain with 1877281 cells was found as optimum for converged

Solution at Transonic speeds. The meshed domain of model is shown in Figure 4. The domain is meshed with fine elements near to the wing surface to capture flow separation, shocks and vorticity accurately.

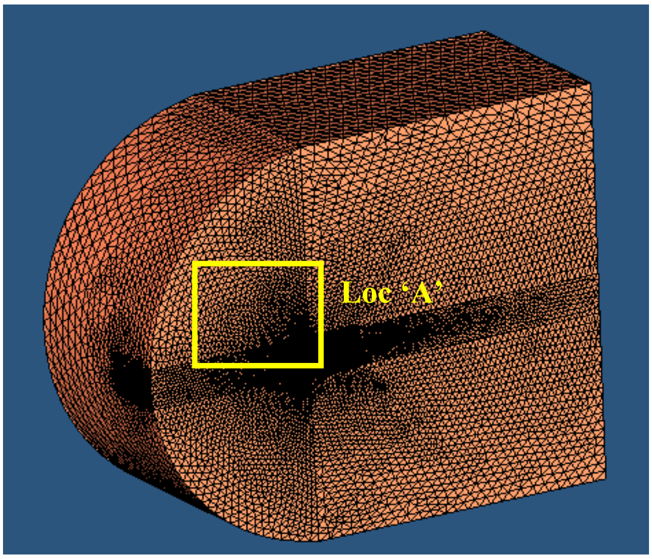

Fig.4. Fluid Domain with 3d wing

At Location 'A' as shown in figure 5 the mesh size is kept very fine near to wing surface throughout the wing span.

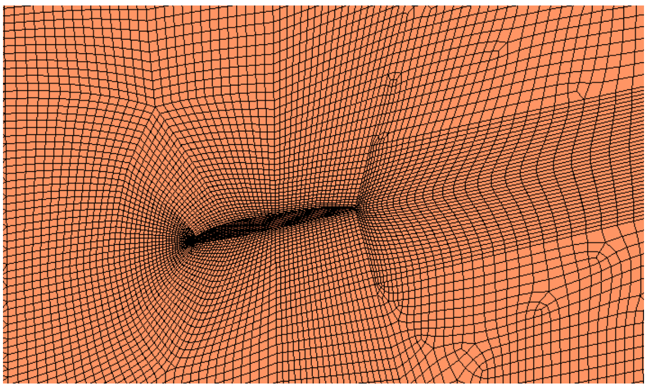

Fig.5. Location 'A'. Fine mesh near to wing surface

Turbulence model, properties and boundary conditions

For flows above 0.3 Mach, compressibility of fluid has to be considered and hence Air which is the fluid medium of the domain is considered compressible. Software "Ansys Fluent" is used for Analysis of the fluid domain. The domain temperature is assumed as 0^0c(273k) considering high altitude operations. On the inlet and outlet face shown in figure 6 "Pressure Far Field" boundary condition is applied. Symmetric boundary condition is applied on one of the face of domain to which the wing is attached, this is because only one of the Aircraft wing is considered and the effects of fuselage is not included in this analysis. Wall boundary condition is applied to all other faces of domain. On the wing surface "No Slip Wall" boundary condition is applied to take in to effect of boundary layer, separation and shocks.

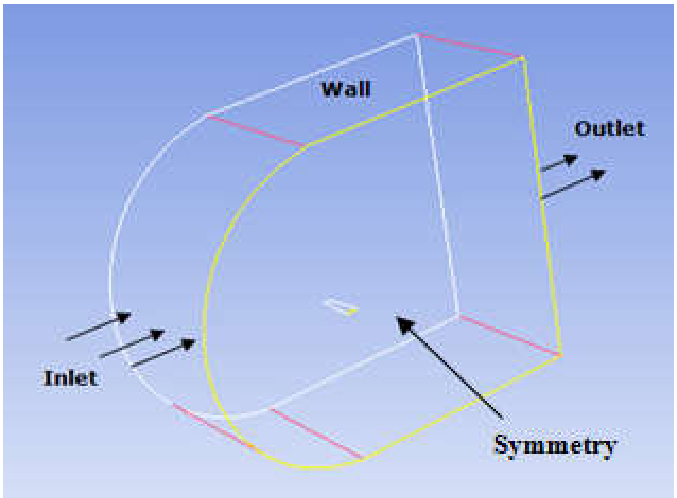

Fig.6. Fluid Domain Boundary conditions

k-ω sst Turbulence model is the Turbulence model used for this analysis in Fluent software since it is the well proven and widely accepted Turbulence model for external flows as explained in Introduction. Analysis is carried out for four different Turbulent intensity conditions of 2%, 5%, 10% and 15% for each of Transonic flow speed 0.7Mach and 0.9 Mach. The AOA of wing is kept as 0^0 degree for 1 set of load case and it is kept as 4^0 degree for second set of load case.

RESULTS AND DISCUSSION

3D wing of a subsonic Aircraft which is of NACA2412 profile is analyzed at two different AOA of 0 degree and 4 degree. For each of this AOA CFD simulations were carried out for two different Aircraft speed of 0.7 Mach and 0.9 Mach and results were discussed below. The results were obtained using Ansys Fluent software with k-ω SST turbulence model.

Results at 0^0 AOA

In Table1 Lift and Drag coefficient of the wing at 0^0 AOA is listed for Turbulence intensity levels of 2%, 5%, 10% and 15%. The results are presented at two different Transonic speeds of 0.7Mach and 0.9 Mach.

Table I. Lift and Drag coefficient of the 3D wing at 0^0 AOA and at different Turbulence Intensity level

S.No.	Mach	Turbulence Intensity %	Lift Coefficient (Cl)	Drag Coefficient (Cd)
1	0.7	2	0.139	0.019
2	0.7	5	0.133	0.020
3	0.7	10	0.126	0.022
4	0.7	15	0.124	0.024
1	0.9	2	0.131	0.039
2	0.9	5	0.113	0.041
3	0.9	10	0.104	0.046
4	0.9	15	0.103	0.049

From Figure 7 we see the lift coefficient decreases with increase in Turbulence Intensity level from 2% to 15%, also it is seen that with increase of speed of Aircraft from 0.7Mach to 0.9Mach the lift coefficient decreases this is because of the isentropic deceleration and compression waves due to sonic shock waves discussed in Transonic dynamics section of this paper. Also it is seen that the delta decrease in lift coefficient is high at 0.9Mach when compared to 0.7 Mach.

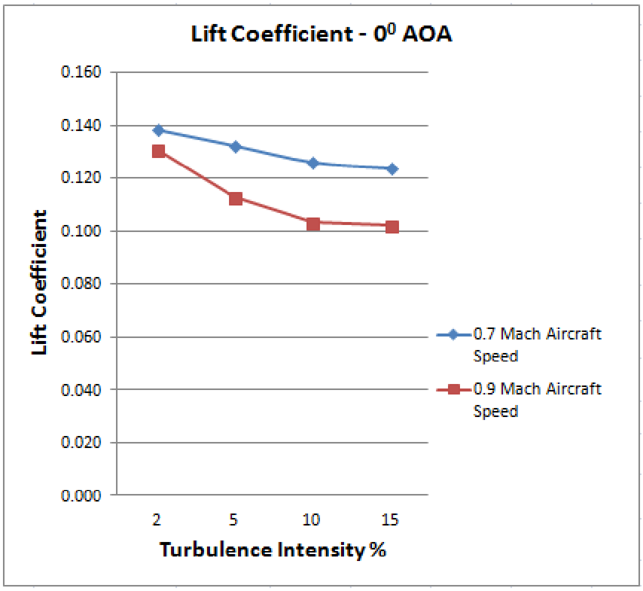

Fig .7. Lift coefficient of 3D wing at 0^0 AOA and at Transonic speed of 0.7 Mach and 0.9 Mach

From figure8 we see the drag coefficient increases with increase in Turbulence intensity level from 2% to 15%, also it is seen that with increase of speed of Aircraft from 0.7Mach to 0.9Mach for a particular Turbulence intensity level the drag coefficient will be higher at 0.9Mach speed when compared to 0.7Mach speed. The Static Pressure at symmetric plane of the domain near to wing surface is shown in Figure 9 for 0.7Mach flow speed and at 2%Turbulence intensity level.

The Mach number of flow over the wing surface at symmetric plane at 2% Turbulence intensity level and at 0.7Mach flow speed is shown in Figure 10. In the figure we see the flow has not reached sonic speed over the wing surface with the aircraft speed of 0.7 Mach for this wing configuration.

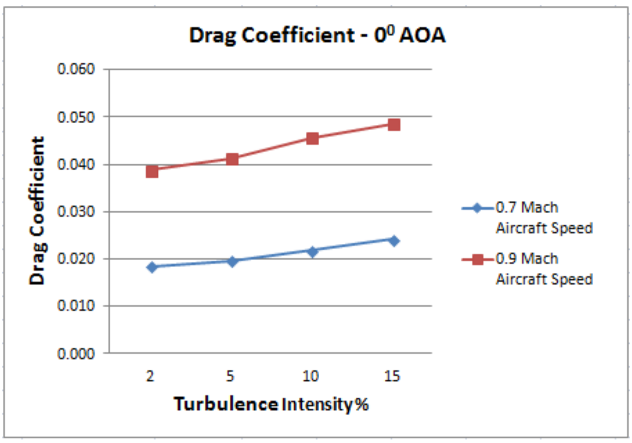

Fig.8. Drag coefficient of 3D wing at 0^0 AOA and at Transonic speed of 0.7 Mach and 0.9 Mach

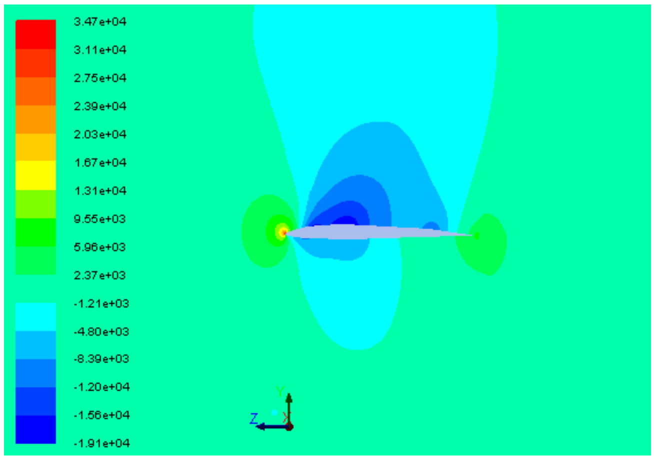

Fig.9. Static Pressure plot at symmetric plane at 0^0 AOA, Turbulence intensity of 2% and at 0.7 Mach

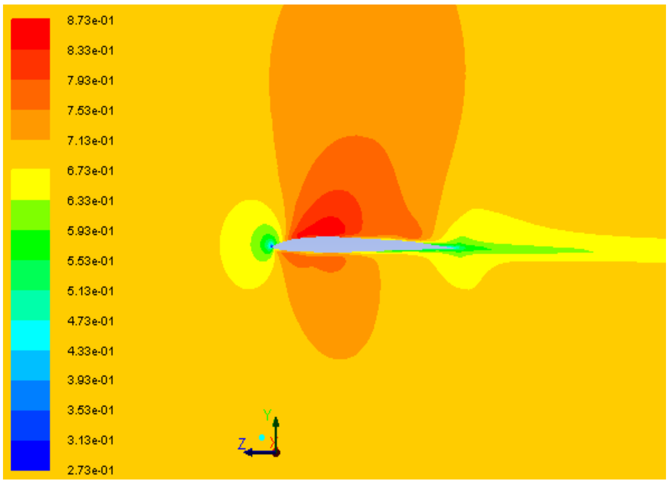

Fig.10. Flow Mach number at symmetric plane for 0^0 AOA, 0.7Mach inlet condition and at 2% Turbulence intensity

The Static Pressure at symmetric plane of the domain near to wing surface is shown in Figure 11 for 0.7Mach flow speed and at 2%Turbulence intensity level.

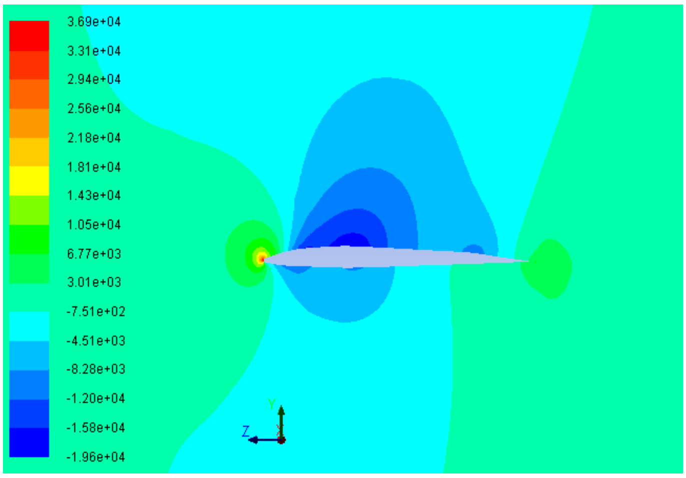

Fig.11. Static Pressure plot at symmetric plane at 0^0 AOA, Turbulence intensity of 15% and at 0.7 Mach

The Mach number of flow over the wing surface at symmetric plane at 15% Turbulence intensity level and at 0.7Mach flow speed is shown in Figure 12.

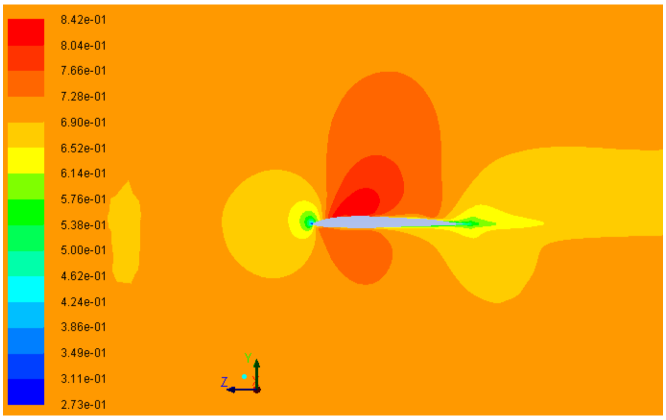

Fig.12. Flow Mach number at symmetric plane for 0^0 AOA, 0.7Mach inlet condition and at 15% Turbulence intensity

The Static Pressure at symmetric plane of the domain near to wing surface is shown in Figure 13 for 0.9Mach flow speed and at 2%Turbulence intensity level. We can see there is sudden change in pressure near to wing trailing edge due to shock and recompression effects.

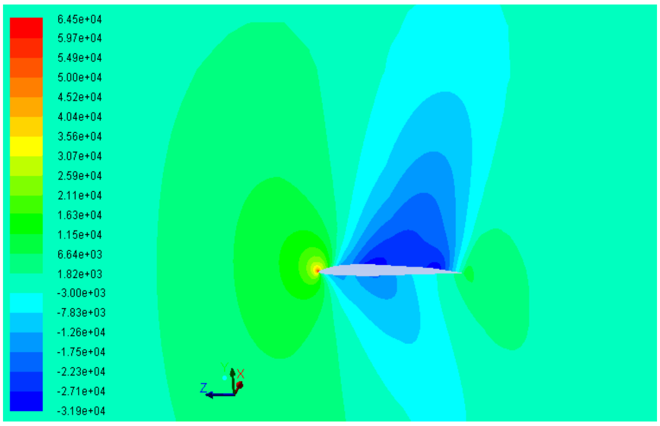

Fig.13. Static Pressure plot at symmetric plane at 0^0 AOA, Turbulence intensity of 2% and at 0.9Mach

The Mach number of flow over the wing surface at symmetric plane at 2% Turbulence intensity level and at 0.9Mach flow speed is shown in Figure 14. As seen in figure 14 the flow will reach sonic conditions soon after the leading edge even at 0.9Mach inlet condition due to decrease in pressure on top surface of the wing.

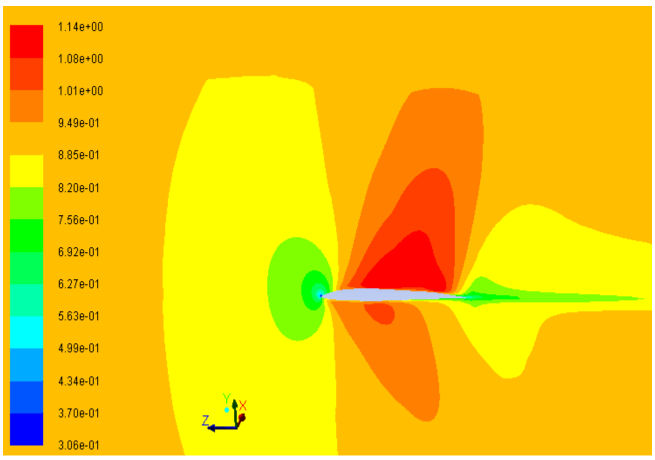

Fig.14. Flow Mach number at symmetric plane for 0^0 AOA, 0.9Mach inlet condition and at 2% Turbulence intensity

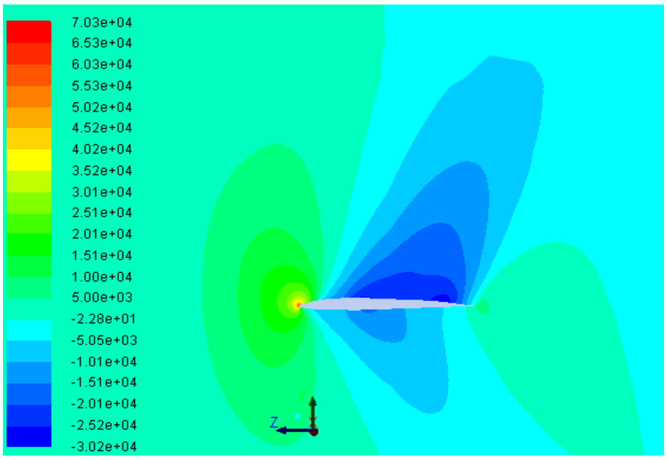

Fig.15. Static Pressure plot at symmetric plane at 0^0 AOA, Turbulence intensity of 10% and at 0.9Mach

The Static Pressure plot and flow Mach number is shown in Figure 15 and 16 for 10% Turbulence intensity level and at a flow speed of 0.9 Mach.

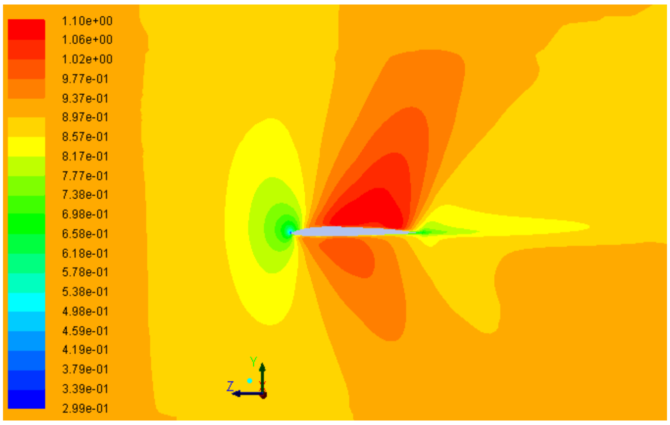

Fig.16. Flow Mach number at symmetric plane for 0^0 AOA, 0.9Mach inlet condition and at 10% Turbulence intensity

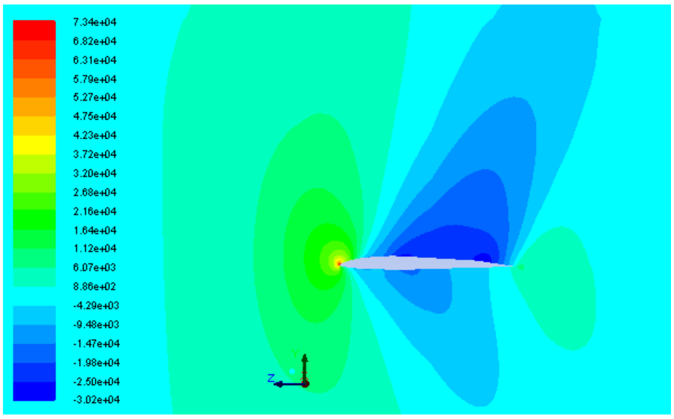

Fig.17. Static Pressure plot at symmetric plane at 0^0 AOA, Turbulence intensity of 15% and at 0.9Mach

The Static Pressure plot and flow Mach number is shown in Figure 17 and 18 for 15% Turbulence intensity level and at a flow speed of 0.9 Mach.

Results at 4^0 AOA: For 4^0 AOA the lift and drag coefficient of the subsonic Aircraft wing at 0.7 Mach and 0.9 Mach is tabulated below at different Turbulence intensity levels of 2%, 5%, 10% and 15%.

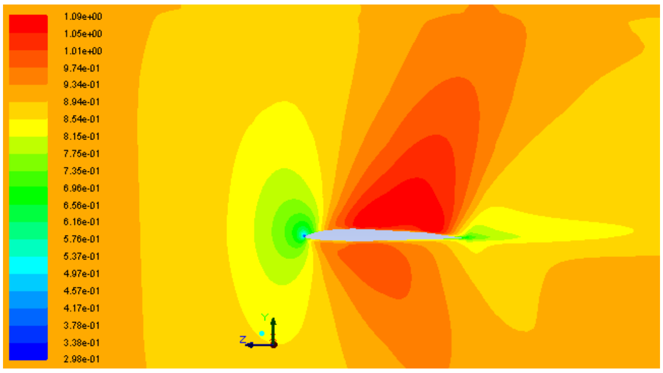

Fig.18. Flow Mach number at symmetric plane for 0^0 AOA, 0.9Mach inlet condition and at 15% Turbulence intensity

Table II. Lift and Drag coefficient of the 3D wing at 4^0 AOA and at different Turbulence Intensity level

S.No.	Mach	Turbulence Intensity %	Lift Coefficient (Cl)	Drag Coefficient (Cd)
1	0.7	2	0.493	0.041
2	0.7	5	0.465	0.041
3	0.7	10	0.448	0.044
4	0.7	15	0.441	0.047
1	0.9	2	0.527	0.075
2	0.9	5	0.495	0.075
3	0.9	10	0.469	0.076
4	0.9	15	0.464	0.079

Comparing Table II with Table I we see lift coefficient Cl is higher for 4^0 degree AOA than 0^0 AOA for the same Aircraft speed and Turbulence intensity level.

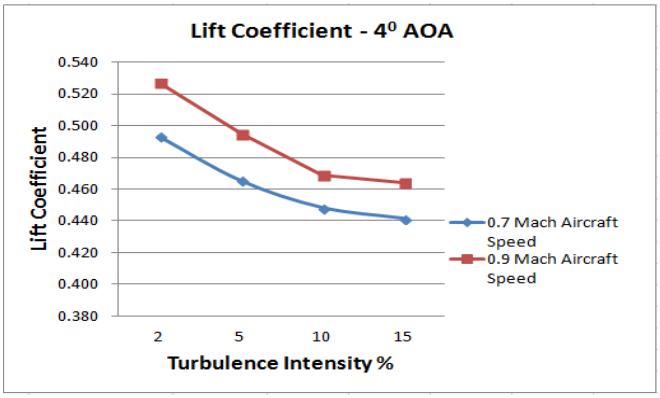

Fig 19. Lift coefficient of 3D wing at 4^0 AOA and at Transonic speed of 0.7 Mach and 0.9 Mach

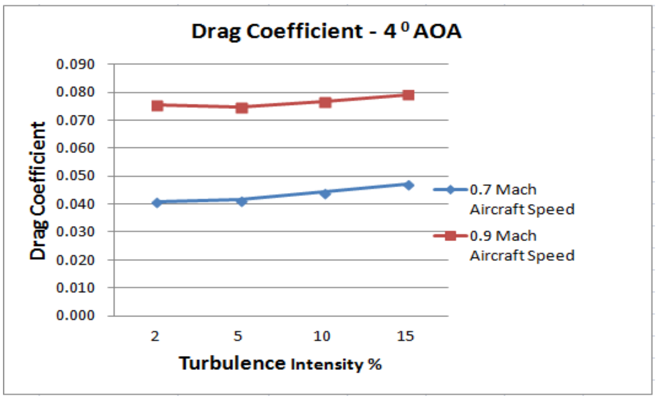

Fig.20. Drag coefficient of 3D wing at 4^0 AOA and at Transonic speed of 0.7 Mach and 0.9 Mach

From Figure19 it is seen that at 4^0 degree AOA lift coefficient of wing at 0.9 Mach number is higher than 0.7 Mach number at all the Turbulence intensity level analyzed. This is different from the simulation results observed at 0^0 degree AOA where Cl of 0.7 Mach is higher than 0.9Mach as shown in Figure 7. This is because at 4^0 degree AOA since the lift coefficient is high, the delta decrease in lift coefficient due to shock and recompression effects is not a large fraction of Cl and hence lift coefficient of wing at 0.9 Mach is higher than lift coefficient of wing at 0.7 Mach. The drag coefficient of wing at 4^0 degree AOA and for Aircraft speed of 0.7 Mach and 0.9 Mach is shown in figure 20 for different Turbulence intensity levels.

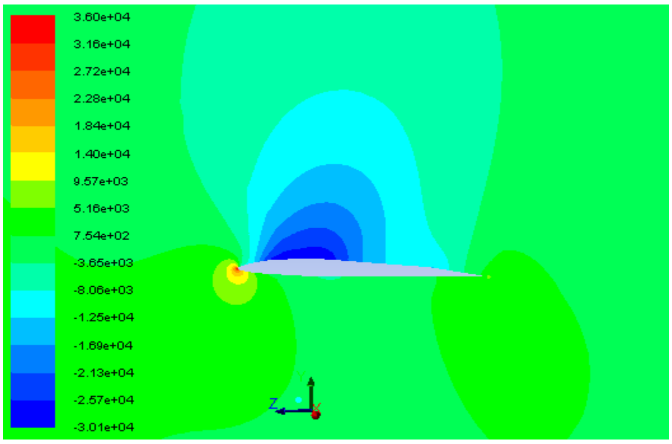

Fig.21. Static Pressure plot at symmetric plane at 4^0 AOA, Turbulence intensity of 15% and at 0.7Mach

The Static Pressure at symmetric plane of the domain is shown in Figure 21 for 0.7Mach flow speed and at 15%Turbulence intensity level. When comparing this static pressure plot with figure 11 we see the negative pressure seen at bottom face of wing with 0^0 AOA is not observed in 4^0 AOA. The Mach number of flow over the wing surface at symmetric plane at 15% Turbulence intensity level and at 0.7Mach flow speed is shown in Figure 22. When comparing this flow Mach number with figure 12 which is for 0^0 AOA, we see the Mach number of flow is 0.924 for 4^0 AOA which is higher than 0.842 Mach observed in 0^0 AOA.

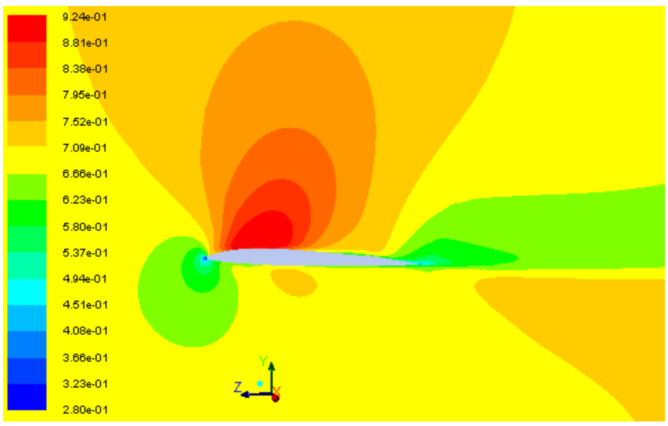

Fig.22. Flow Mach number at symmetric plane for 4^0 AOA, 0.7 Mach inlet condition and at 15% Turbulence intensity

The Static Pressure plot and flow Mach number at symmetric plane of the domain is shown in Figure 23 and figure 24 for Aircraft speed of 0.9Mach and at 15% Turbulence intensity level.

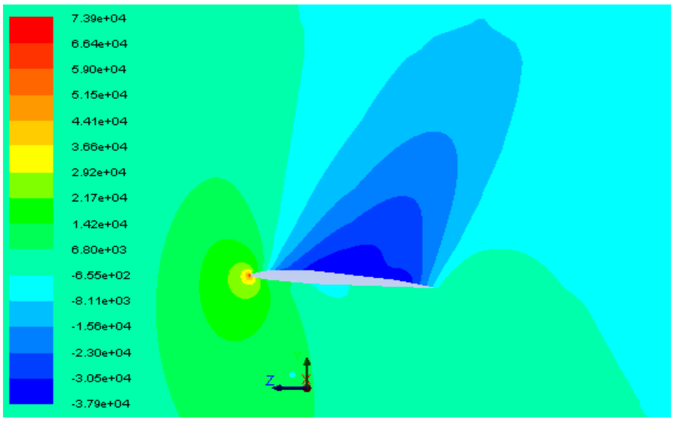

Fig.23. Static Pressure plot at symmetric plane at 4^0 AOA, Turbulence intensity of 15% and at 0.9Mach

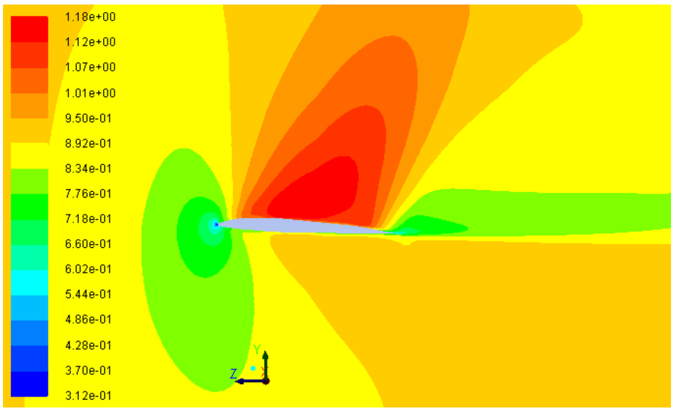

Fig.24. Flow Mach number at symmetric plane for 4^0 AOA, 0.9 Mach inlet condition and at 15% Turbulence intensity

Comparing figure 24 with figure 15 we see at 4^0 AOA high negative pressure over the wing surface is spread across over 40 to 50% of chord length whereas at 0^0 AOA the negative pressure is not high as 4^0 AOA and it is observed over only 10 to 20% of chord length. The max flow Mach number observed over the wing is 1.18 Mach for 4^0 AOA as shown in figure 24 and it 1.1 Mach for 0^0 AOA as shown in figure 16.

Conclusion

The results of a 3D subsonic wing at two different transonic speed of 0.7 Mach and 0.9 Mach is presented in this paper for different Turbulence intensity levels of 2%, 5%, 10% and 15% and the lift and drag coefficients are tabulated in Table I and Table II. CFD software Fluent is used and K-ω SST Turbulence model is the computational model used to arrive at precise results. Convergence study of the domain with varying mesh density is done at 0.7 Mach speed and a optimum mesh size was selected and used for all the analysis which was discussed in this paper. From Table I we see for 0^0 AOA the lift coefficient decreases with increase of Turbulence intensity level and with change in speed of Aircraft/flow from 0.7 Mach to 0.9 Mach the lift coefficient decreases because of the shock and recompression effects. For 4^0 AOA we see in Table II the lift coefficient decreases with increase of Turbulence intensity level, which is showing same trend as at 0^0 AOA. With the change in speed of aircraft from 0.7 Mach to 0.9 Mach with 4^0 AOA, we see the Lift coefficient increases, which is different from the characteristics observed at 0^0 AOA. This is because the delta change in lift coefficient due to shock and recompression effects is not very high at 4^0 AOA due to which the net lift coefficient is higher at 0.9 Mach when compared to lift coefficient at 0.7 Mach. The static pressure plot and flow Mach number of the wing at symmetric plane for two different AOA and different Turbulence intensity level is shown in this paper.

Comparing Figure 9 and Figure 11 it is observed that increase of Turbulence intensity level from 2% to 15% at 0^0 AOA and at 0.7 Mach there is some recompression over the wing top surface due to which the flow mach number has decreased from 8.73 to 8.4 as shown in Figure 10 and Figure 12. With change in Mach number from 0.7 to 0.9 we see from figure 11 and figure 17 that large recirculation and shock region is spread across almost $3/4^{th}$ of the wing top surface at 0.9 mach speed when compared to 0.7 Mach speed due to which flow Mach number is above 1 for most of wing top surface region as shown in figure 18. This leads to decreased lift coefficient with increased speed from 0.7 Mach to 0.9 Mach for 0^0 AOA. Comparing figure 18 and figure 24 we see the flow Mach number increases from 1 to 1.18 due to change in AOA from 0^0 to 4^0. This high Mach number of flow over the top wing surface will be having its associated high shock levels and recompression effects which will have higher loss in lift coefficient compared to 0^0 AOA. This results of Subsonic wing at transonic conditions and at different Turbulence intensity levels can be used in future research work to compare the performance of transonic and supersonic wings with this subsonic wing to determine the advantages and disadvantages of a particular wing configuration.

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