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Supersonic Aerodynamics of Swept Wings at Different Turbulence Levels

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ABSTRACT: Due to rising demand for fast means of Transportation, high-speed commercial and business Aircrafts development is one of the key research area for researchers and research establishments around the world. Even though quantitative amount of work was done in this Supersonic regime by various scientists and establishments, the country specific law and safety requirements is one of the key barriers in implementing and using this technology for commercial and business aircrafts developments. One such requirement is sonic boom at ground level, which restricts present aircraft speeds to near sonic speeds of 0.85 to 0.9 mach. At present research establishments like NASA is working on different research projects to reduce this boom effect and to enable Aircrafts to travel at supersonic speeds there by facilitating fast means of transportations. Adding to this the other challenge is the Atmospheric Turbulence, which is considered as one of key external factor that leads to Aircraft vibrations and accidents that are reported so far. Understanding the performance of wings at different Turbulence levels is crucial and it will enable Aircraft manufacturers to understand the sensitivity of wings at different turbulence levels during preliminary design phase there by paving way for robust ride control systems and aircraft structure. In this research paper considering the development and feasibility of supersonic Air Transportation, the performance of swept wing at low supersonic speeds is analysed and presented for different Turbulence intensity levels. 3D Swept wing of NACA 2412 profile is considered as it is preferred wing profile for high lift commercial aircrafts. The lift and drag coefficient of this 3D wing at two different sweep angles of 300 and 400 is tabulated. Low Supersonic speed of 1.2 Mach and 1.4 Mach is considered for this analysis. Turbulence intensity levels considered for low supersonic speed are 2%, 5%, 10% and 15% and the results are tabulated. K-ω SST Turbulence model is used with Ansys Fluent as CFD software for the computational analysis

KEYWORDS: Swept wing, Shockwave, Supersonic Aerodynamics,1.2 Mach, 1.4Mach, 3D wing, K-ω SST Turbulence model, lift, drag.

I. INTRODUCTION

Image Supersonic speeds in Air Transportation is one of the research areas that is being explored for past so many years after world war II. The supersonic Aircraft production was stopped due to prevailing ban, after its initial commercialisation during 1969 through Concorde program. For instance this ban which is in effect from 1973 in United States is due to environment effects like sonic boom caused by these supersonic flights. The time has come for the Aviation regulators like FAA to end its ban and create a supersonic noise standard that will allow aviation innovation to once again flourish to commercialize the supersonic transportation. Currently NASA is working on Quiet Supersonic Technology (QueSST) program, which is all about developing supersonic speed Aircraft design and configuration with quiet sonic boom levels. The expected completion of the first flight as per the program milestones is 2021. As the preliminary design review is successful, it is expected that supersonic flights are going to be commercialized once again in near future after completion of this QueSST program across most of countries following

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United States of America. Considering this technological improvement in the supersonic transportation the aerodynamic performance of swept wing is analysed at supersonic speeds and the results are presented in this paper. As the effect of Turbulence on the wings at supersonic speeds is still a grey area, this research work will help understand the performance of wing at different turbulence level. The performance of the swept wing which is of NACA 2412 profile at Transonic speed is discussed and the results are tabulated in [2]. In continuation to the CFD simulation completed by the same authors in [2], the supersonic performance of the same swept wing model is analysed and presented in this paper. Aircraft wings are designed based on the following important application parameters like speed, payload requirement and altitude at which the Aircraft will cruise most of time. By this the fuel consumption can be reduced for economical flight transportation. Due to this reason, even though the preferred wing profile for Supersonic speed is Double wedge or Biconvex profile, the NACA 2412 is considered for the analysis. NACA2412 is one of the preferred wing profile for commercial aircraft due to its high lift performance. Considering this design need NACA 2412 wing profile is considered and the lift and drag coefficient of wing at low supersonic speed of 1.2 Mach and 1.4 Mach is analysed and presented in this paper. Sweep angles of 300 and 400 is considered as the same 3D wing model analysed in previous work [2] is analysed at low supersonic speed in this research work. Turbulence intensity level of 2%, 5%, 10% and 15% is considered for each of flow speed and the results are tabulated.

II. SUPERSONIC AERODYNAMICS

Supersonic flows are the one with flow velocities greater than Mach 1 and lesser than Mach 5. At these speeds the flow compressibility to be taken in to consideration without which the results obtained from flow equations will be erroneous. Compressible flow is realized when there is change in pressure related to change in density. For all the commercial and defence Aircrafts the flow medium is none other than Air and the compressibility of Air to be considered while computing the flows over Aircrafts at speeds above Mach 0.7. The compressible flow equations are required to find the unknown flow quantities which are velocity components, density and pressure changes with respect to spatial co-ordinates X, Y and Z. The compressible flow equations are obtained from the three conservation principles, which are conservation of Mass, conservation of Momentum and conservation of Energy, and along with Equation of state for the flowing fluid. As compressibility is dependent on the Flow speed it is necessary to know the Aerospace term Mach and the Speed of sound. Mach is deined as the ratio between the Flow speed(V) and the speed of sound(a) in the flow at specific flow conditions.

$$Mach\ M = \frac{V}{a} \qquad \text{--------(1)}$$

The speed of sound in compressible flow theory is the speed at which small disturbances propagate and it is defined in terms of other flow parameters as below

$$Speed\ of\ Sound\ a = \sqrt{\frac{dP}{d\rho}} \qquad \text{------(2)}$$

Where P and r are the pressure and density of flow.

The Supersonic flow over a flat plate and circular Arc Aerofoil is shown in Figure 1. As seen in figures the flow is characterised by shock waves.

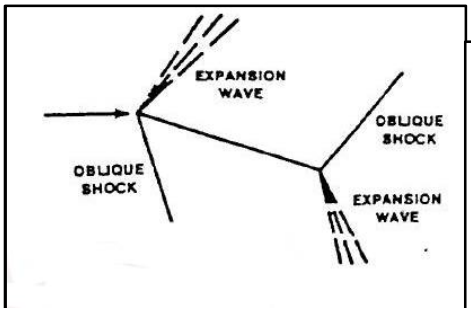

Fig. 1. Shockwave over Flat Plate

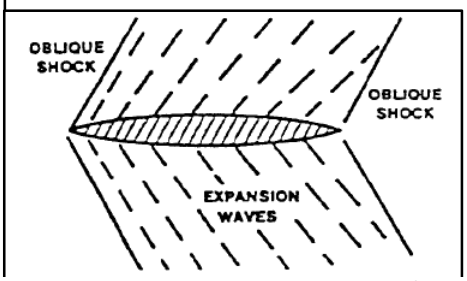

Fig.2. Shockwave over Circular Arc Aerofoil

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The shock waves are further classified broadly in to Normal shock wave and Oblique shock wave and are detailed in following sections.

III. NORMAL SHOCK WAVE

The extremely thin discontinuity, which forms between Supersonic and Subsonic flow, is considered as Shock wave. As the speed of the flow changes suddenly from Supersonic to subsonic speeds in a short distance, the kinetic energy of the supersonic upstream molecules is converted to pressure-volume (PV) and Thermal Energy. Since the changes, which occur due to Normal shock occurs in a shorter distance, the changes are irreversible and the shock wave is not isentropic. While studying Normal shocks the two valid assumptions that are to me made are the flow through a shock is adiabatic and has a constant cross section area between the front and rear face of the shock. With this assumption and applying conservation equations the changes in flow properties caused by shock in terms of initial flow Mach number M_1 is given as

$$\frac{P_2}{P_1} = \frac{1-\gamma+2\gamma M_1^2}{1+\gamma} \quad \text{- - - - - - - -(3)}$$

$$\frac{\rho_2}{\rho_1} = \left[\frac{2+(\gamma-1)M_1^2}{(\gamma+1)ff_1}\right]^1 \quad \text{- - - - - - - -(4)}$$

$$\frac{T_2}{T_1} = \left[\frac{1-\gamma+2\gamma M_1^2}{1+\gamma}\right]\left[\frac{2+(\gamma-1)M_1^2}{(1+\gamma)M_1^2}\right] \text{- - - - - - -(5)}$$

$$\text{Also} \qquad M_2^2 = \frac{M_1^2+\frac{2}{\gamma-1}}{(\frac{2\gamma}{\gamma-1}M_1^2)-1} \text{- - - - - - - -(6)}$$

Where P_1, r_1& T_1 are flow properties (Pressure, Density and Temperature) upstream of Shock wave where the flow is supersonic and P_1, r_1& T_1 are the flow properties downstream of shock wave as shown in Figure 3.

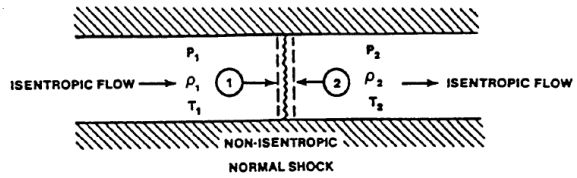

Fig.3. Flow properties near to Normal Shock Wave

IV. OBLIQUE SHOCK WAVE

A normal shock is a special form of a pressure discontinuity in a fluid. Experimental observations of the supersonic flow discontinuities revealed that the discontinuities are inclined to free stream velocity and are called oblique shocks. Oblique shock occur in supersonic flow because continuous compression waves formed by a concave surface in flow tend to merge forming an oblique discontinuity at a finite distance from the surface. Oblique shocks occurs in almost all practical flow situations but the mere existence of supersonic flow does not imply that there must be shock waves somewhere in the flow. Forming the relations between fluid properties on both sides of the oblique shock is not tough as expected since most of Normal shock equations with slight modification are valid for Oblique shock.

Assume a stationary observer sees the flow at Station 1 which suddenly decelerate and compress to the flow at Station 2 because it has traversed a normal shock wave as shown in Figure 4. Then imagine that observer moves along the

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shock wave with a velocity V_t. The moving observer will be seeing a flow situation in which the incoming free stream flow is inclined to shock and the flow undergoes a sudden change in direction when it crosses the shock as seen in Figure 5.

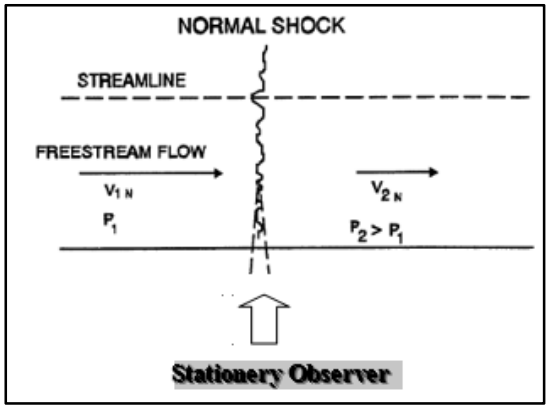

Fig.4. Flow shock wave as seen by stationary observer

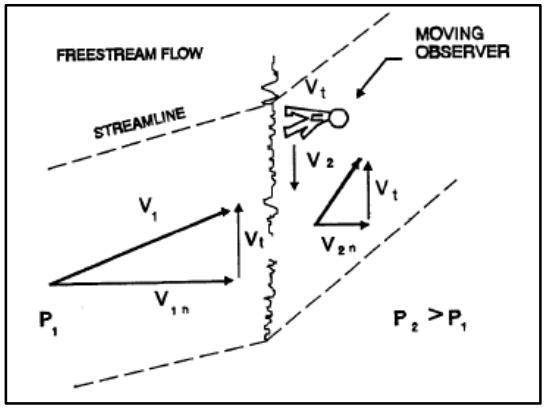

Fig.5. Shock process as seen by moving observer

The oblique flow pattern constructed in this manner has equal tangential velocity componentV_t, on both sides of the shock as shown in figure 5. By considering a solid wall along one of the streamline in Figure 5 and rotating the picture so that incoming velocity V_1, is horizontal, the supersonic flow situation in the neighborhood of a concave corner *is* shown in Figure 6

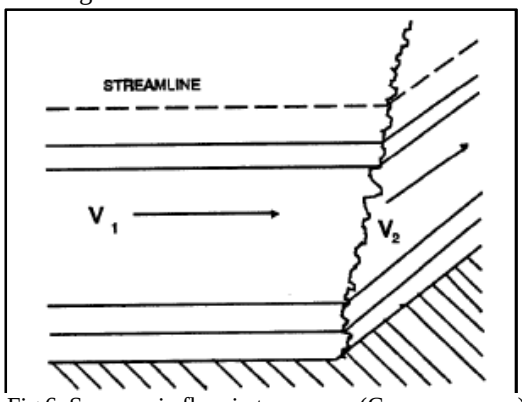

Fig.6. Supersonic flow in to a corner(Concave corner)

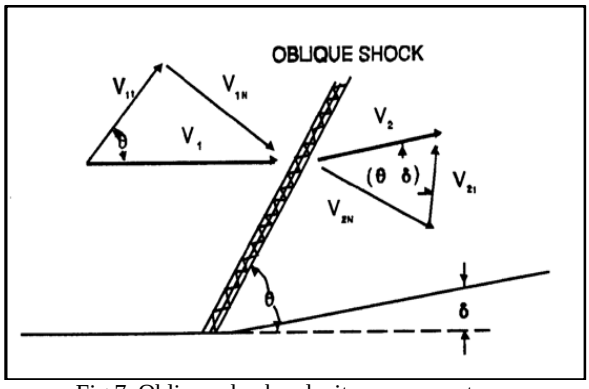

Fig.7. Oblique shock velocity components

By imparting a uniform velocity to the flow field along any shock, a straight segment of an oblique shock can be transformed into a normal shock.The magnitude ofV_t is arbitrary and depends upon the angle the oblique shock makes with the horizontal streamline in front of the shock and the velocity of the approaching flow. This leads to an additional defining parameter in the oblique shock relations, where as in Normal shock equation the only defining parameter required is M1 (Approaching Mach). For Oblique shock equations two independent parameters M1 and Wave angle are required.The shock will be termed as normal or oblique depending upon the relative motion of the observer, and hence the differences between normal and oblique shocks can be explained in geometric terms.The flow notation, and angle descriptions used to express Oblique shock in terms of Normal shock is shown in Figure 7. The angle through which the flow is diverted due to concave corner is called the Turning/wedge angle δ. The angle the oblique shock make with the incoming stream flow is the shock wave angle θ. The upstream flow condition is mentioned with subscript 1 and the downstream is mentioned with subscript 2.

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The flow Mach number perpendicular to Normal shock (M_{1N}) can be expressed as

$$M_{1N} = M_1 \sin\theta \text{ - - - - - - -(7)}$$

Where M_1 is the flow Mach in front of Oblique shock. By using above expression all the Normal shock equations can be expressed in terms of oblique shock, except for those equations that relate to Total properties. The Oblique shock equations are listed below

$$\frac{P_2}{P_1} = \frac{1-\gamma+2\gamma M_1^2 sin^2\theta}{1+\gamma} \text{- - - - - -(8)}$$

$$\frac{\rho_2}{\rho_1} = \left[\frac{2+(\gamma-1)M_1^2 sin^2\theta}{(\gamma+1)M_1^2 sin^2\theta}\right] \text{- - - - - -(9)}$$

$$\frac{T_2}{T_1} = \left[\frac{2\gamma}{\gamma+1} M_1^2 sin^2\theta - \frac{\gamma-1}{\gamma+1}\right]\left[\frac{\gamma-1}{\gamma+1} + \frac{2}{(\gamma+1)M_1^2 sin^2\theta}\right] \text{- - - - - -(10)}$$

$$M_2^2 sin^2(\theta-\delta) = \frac{M_1^2 sin^2\theta + \frac{2}{\gamma-1}}{\frac{2\gamma}{\gamma+1}M_1^2 sin^2\theta - 1} \text{- - - - - - -(11)}$$

Where M_2 is the flow Mach that is concave to front inflow.

V. MIN AND MAX WAVE ANGLE AND δRELATION

From Normal shock analysis, it is know that shock can only occur when free stream Mach is greater than 1. The same is applicable for Oblique shock where the free stream flow component normal to shock must be greater than 1 for the shock to occur. The minimum wave angle can be found from equation (7) and it is given below

$M_{1N} = M_1 \sin\theta = 1$

$$\text{Hence} \qquad \theta_{Min} = \sin^{-1}\frac{1}{M_1} \text{- - - - - -(12)}$$

Notice that the minimum oblique *shocḱ* wave angle θ_{min}, for the given free stream Mach M_1, is the same as the Mach angle μ formed by an isentropic pressure disturbance traveling at Ml> 1.

Where Mach Angle $\mu = \frac{a}{V} = \frac{1}{M}$ - - - - - - -(13)

This shows that an oblique shock wave at minimum wave angle to the free stream flow is a zero strength or isentropic shock.The maximum oblique shock wave angle for a given free stream Mach is 90^o. This is the limiting case and is a normal shock.

From Figure 7. $\tan\theta = \frac{V_{1N}}{V_{1t}}$

and $\tan(\theta-\delta) = \frac{V_{2N}}{V_{2t}}$

Eliminating V_{1t} and V_{2t} as both are same for oblique shock and considering continuity equation, the relation between θ&δ is expressed in terms of M_1 as

$$\frac{\tan(\theta-\delta)}{\tan\delta} = \frac{(\gamma-1)M_1^2 sin^2\theta+2}{(\gamma+1)M_1^2 sin^2\theta} \text{- - - - - -(14)}$$

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Where $M_1$is inlet flow Mach number. To show the dependency of δ on θ& M_1, the equation (14) is manipulated with trigonometric laws to show the explicit relation of δ as

$$\tan\delta = 2\cot\theta \frac{M_1^2 sin^2\theta - 1}{M_1^2(\gamma + \cos 2\theta) + 2} \text{- - - - - -(15)}$$

VI. SUPERSONIC WING MODEL AND DOMAIN

Swept wing of NACA 2412 profile is considered for this analysis and only one wing is considered. By applying symmetric boundary condition the effect of flow over aircraft with two wings shall be simulated. For NACA 2412 wing profile two wings are modelled, one with 30^0 sweep and the other wing with 40^0 sweep. For both this wing sweep the wing area, root chord and wing span is kept constant. The Swept wing which is of 30^0sweep and of NACA2412 profile is shown in Fig 8. The 3D wing area is 5.74m^2 and the wing span is 5.5m with root chord length of 1.4m. To simulate the flow over the wing the 3D model along with fluid domain is meshed in Ansys Workbench software module.

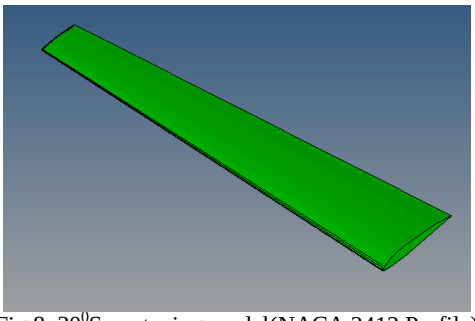

Fig.8 30^0Swept wing model(NACA 2412 Profile)

The size of fluid domain is selected accordingly to simulate the flow similar to wind Tunnel Tests. The domain size used is 50m x 40m x 20m. Hybrid Mesh methodology is used in meshing software which constitutes of both Tet and Prism elements. The size of mesh elements near to 3d wing model in the domain is kept fine to capture the boundary layer effects, flow separation and shock wave formations. SST K-ω Turbulence model which is the used Turbulence model in Ansys Fluent for this research work uses wall function principle of flow calculation, which allows the use of a relatively coarse mesh in thenear-wall region. Irrespective of this the mesh is kept fine near to wing surface. Mesh convergence study is done for both sweep angles to see the variation of lift coefficient by refining the element size near to wing surface and the element size that resulted in converged lift coefficient value is selected and used for subsequent analysis for different speeds and Turbulence level. The total mesh element count is 1757578 elements for 30^0 swept wing model with domain and it is 1961250 elements for 40^0 swept wing model with domain.

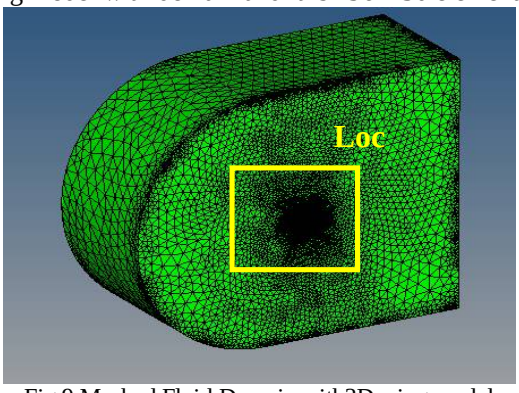

Fig.9 Meshed Fluid Domain with3D wing model

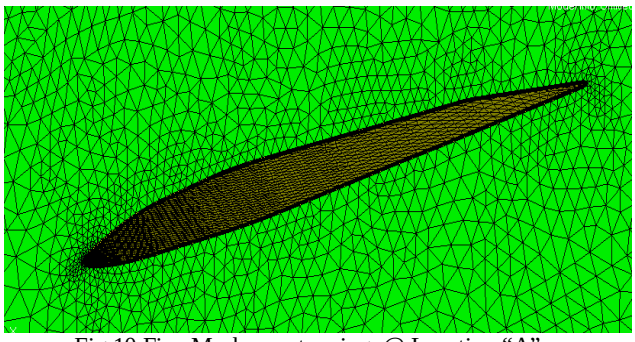

Fig.10 Fine Mesh near to wing @ Location "A"

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The Mesh model of 30^0 swept wing with domain is shown in figure 9. As shown in figure 10 the mesh size is kept fine near to wing surface to capture the viscous boundary effects perfectly. Aircraft Aero dynamists will always try to keep the swept wing envelope within this Mach cone angle to minimize the shock wave losses and drag at supersonic speeds. For supersonic speed of 1.2 Mach the Mach cone angle μ is 56.40 and hence keeping the sweep angle lesser than this will lead to reduced drag and loss. For subsonic and Transonic speeds where there will be no Mach cone effect having higher sweep angles leads to reduced lift/weight ratio of aircraft which is not economical for commercial aircraft. Hence based on the speed at which the aircraft will cruise most of time an optimum sweep angle to be selected which will lead to higher lift coefficient and reduced drag over a wide range of flight speed.

VII. TURBULENCE MODEL, PROPERTIES AND BOUNDARY CONDITIONS

CFD software Ansys Fluent is used to analyse the swept wing models at supersonic speeds of 1.2Mach and 1.4 mach. SST K-ω model is used as the Turbulence model for this analysis as it is the well proven and widely accepted Turbulence model. One of the research paper by [3]Douvi C. Eleni and others concluded with stating that k-ω SST model as the most appropriate turbulence model for lift and drag coefficient estimations of a 2d aerofoil. The compressibility effect is included as the flow is supersonic and the flow medium is Air. The properties of fluid at 35000ft is applied and used, as most of Aircraft Turbulence occurrence were reported at this altitude range of 30000ft to 40000ft [4]. Only one wing is considered and symmetric boundary condition is applied to simulate the flow similar to flow around the Aircraft where each of wing will be symmetric to other wing with respect to longitudinal axis of Aircraft.

As shown in figure 11 the "Pressure Far Field" boundary condition is applied at both inlet and outlet face of the domain. At inlet face the pressure far field boundary condition is applied with 1.2Mach and 1.4 Mach flow speed for which the domain is simulated with wing model. Symmetric boundary condition is applied to the face to which the wing is attached, this is because the flow is considered to be symmetric about this face as the fuselage is not modelled. Wall boundary condition is applied to remaining three faces of the domain and "No slip condition" is considered for these faces. Analysis is done for Supersonic flow speeds of 1.2Mach and 1.4Mach for the wings analysed and presented in this paper.

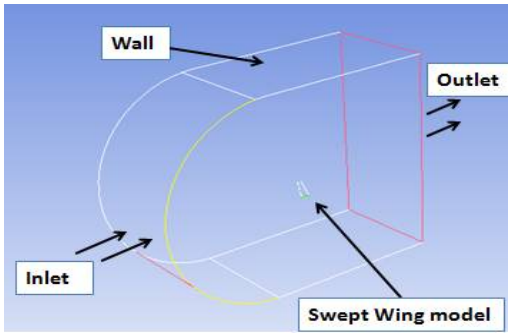

Fig .11 Analysis Domain and Boundary conditions

VIII. RESULT'S AND DISCUSSION

The results of 3d swept wing model of NACA 2412 profile, analysed using Ansys Fluent software is tabulated in this paper. Wing models of 30^0 and 40^0 sweep and 0^0AOA is analysed and the lift and drag coefficients are tabulated for different turbulence intensity levels of 2%, 5%, 10% and 15 %.

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Results of 30^0 Swept NACA 2412 Wing at 0^0 AOA:

In Table 1 Lift and Drag coefficient of the 300 Swept wing of NACA2412 profile at 00 AOA is listed for Supersonic speed of 1.2Mach and 1.4 Mach and at different Turbulence intensity levels of 2%, 5%, 10% and 15%.From figure12 we see that for 30^0 Sweep wing at 1.2 Mach the lift coefficient increases with increase of Turbulence Intensity level from 2% to 15%, but at 1.4 Mach the lift coefficient decrease with increase of Turbulence level. This is because the effect of turbulence leads to larger drag loss at 1.4 Mach compared to 1.2 Mach. This shows that higher Turbulence intensity level at lower supersonic speeds aids in lift increase, where as it decreases the lift at higher supersonic speeds.

Si No	Mach	Turbulence Intensity %	Lift Coefficient (Cl)	Drag Coefficient (Cd)
1	1.2	2	**0.034**	**0.0384**
2	1.2	5	**0.034**	**0.0391**
3	1.2	10	**0.038**	**0.0397**
4	1.2	15	**0.039**	**0.0402**
1	1.4	2	**0.016**	**0.0384**
2	1.4	5	**0.015**	**0.0390**
3	1.4	10	**0.015**	**0.0398**
4	1.4	15	**0.014**	**0.0404**

Table I. Lift and Drag coefficient of 30^0 SweepNACA 2412 wingat 0^0 AOA and at different Turbulence Intensity level

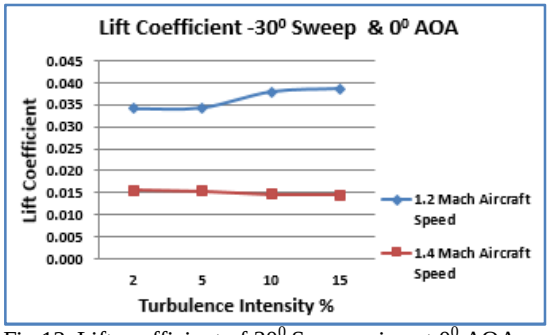

Fig.12 Lift coefficient of 30^0 Sweep wing at 0^0 AOA and at Supersonic speed of 1.2 Mach and 1.4 Mach

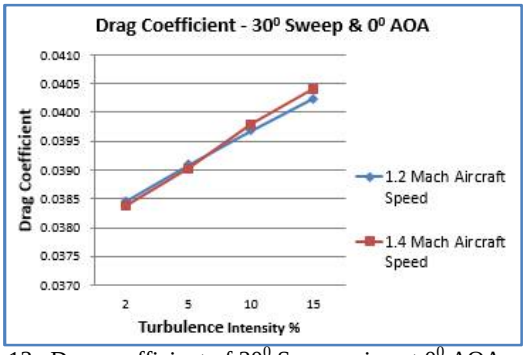

Fig.13 Drag coefficient of 30^0 Sweep wing at 0^0 AOA and at Supersonic speed of 1.2 Mach and 1.4 Mach

Figure13 shows the drag coefficient of 30^0 Swept wing at 0^0 AOA for different Turbulence intensity levels. For both the supersonic speed of 1.2 Mach and 1.4 Mach the drag increases with increase of Turbulence level. At 2% Turbulence intensity the drag at 1.2 Mach and 1.4 Mach are the same. With increase of Turbulence intensity the drag coefficient of wing at 1.4Mach is higher than the drag coefficient of wing at 1.2Mach

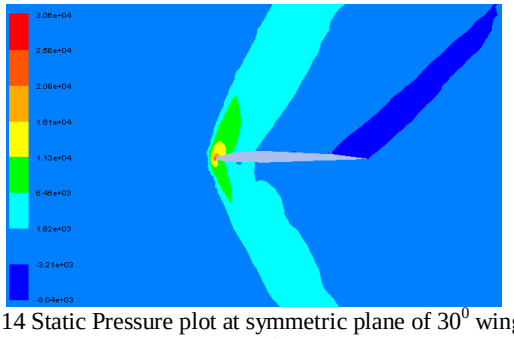

Fig.14 Static Pressure plot at symmetric plane of 30^0 wing 0^0 AOA, Turbulence intensity of 2% and at 1.2 Mach

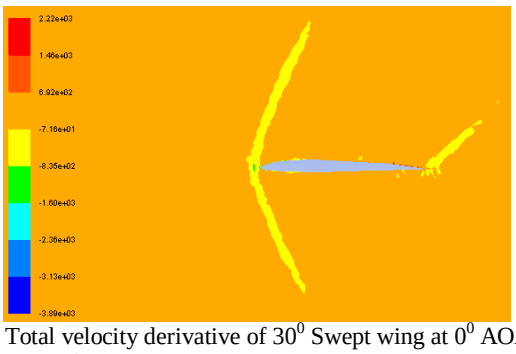

Fig .15 Total velocity derivative of 30^0 Swept wing at 0^0 AOA, and at at 1.2Mach flow speed and 2% Turbulence intensity

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The Static Pressure plot ofthe 30^0 Swept wing at 1.2Machand at 2% Turbulence is shown in Figure 14. The plot near to wing surface in the fluid domain is shown. The pressure decreases drastically near to trailing edge of wing which is one of the indication of presence of expansion wave. The total derivative of velocity along flow direction at 1.2 Mach is shown in Figure 15 for 30^0Sweep wing. It is visible from figure that bow shock is visible affront of wing and normal shock is seen at wing trailing edge. For 30^0 swept wing the Static Pressure plot at 1.2 Mach and at 15% Turbulence intensity level and at 0^0 AOA is shown in Figure 16.It is seen that larger area of wing surface near to trailing edge is experiencing negative pressure due to higher Turbulence level which aids in increasing the lift coefficient.

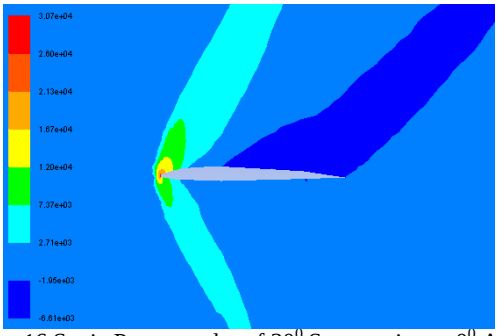

Fig .16 Static Pressure plot of 30^0 Sweep wing at 0^0 AOA, 1.2Mach speed and at 15% Turbulence intensity

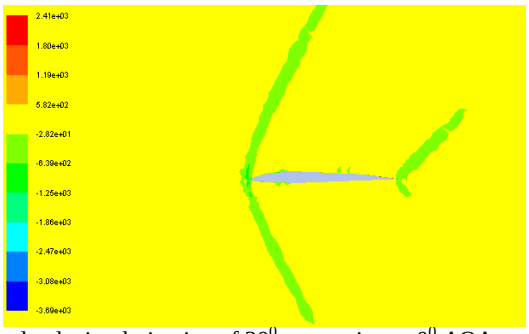

Fig.17 Total velocity derivative of 30^0swept wing at 0^0 AOA, Turbulence intensity of 15% and at 1.2 Mach

The total velocity derivative of flow over the 30^0 swept wing at 0^0 AOA and at 1.2 Mach and 15% Turbulence intensity is shown in Figure 17. It is noticed from figure that increase of turbulence intensity leads to reduced velocity derivate change when compared to figure 15 at bow shock area. The Static Pressure plot ofthe 30^0 Swept wing at 1.4 Mach and at 15% Turbulence is shown in Figure 18. The negative pressure at trailing edge of wing is lesser at 1.4 Mach as compared to negative pressure seen at 1.2 Mach flow speed.The velocity derivative of 30^0 Sweep NACA 2412 wing at 1.4 Mach and at 15% Turbulence intensity level is shown in Figure 19. The shock is still bow shock as the turning angle is larger than the maximum turning angle for the given flow Mach value of 1.4.

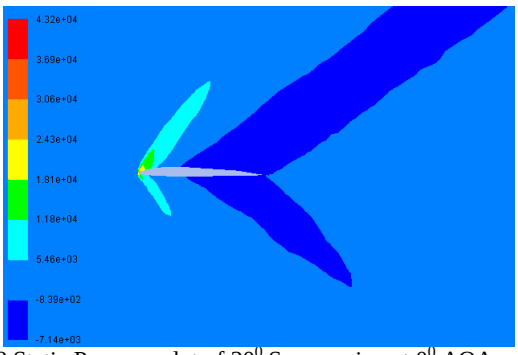

Fig .18 Static Pressure plot of 30^0 Sweep wing at 0^0 AOA, 1.4 Mach velocity and 15% Turbulence intensity

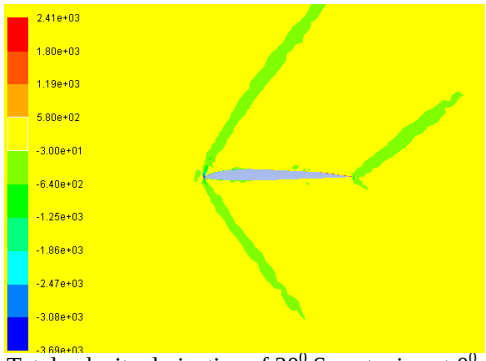

Fig .19 Total velocity derivative of 30^0 Swept wing at 0^0 AOA, at Turbulence intensity of 15% and at 1.4 Mach

Results of 40^0 Sweep NACA 2412 Wing at 0^0 AOA:

The lift and Drag coefficient of 40^0 Swept NACA 2412 wing at 0^0 AOA is tabulated in Table II below for Turbulence intensity levels of 2%, 5%, 10% and 15%. The Static pressure plot and velocity derivative plot is shown at different Turbulence levels. By comparing, the lift coefficient of 30^0 Swept wing and 40^0 Swept wing in Table I and Table II we see the lift coefficient for 40^0 Swept wing is higher when compared to 30^0 Swept wing for all Turbulence intensity

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levels as the related drag is lesser in 40^0 Swept wing.The lift coefficient rise with increase in Turbulence level of 40^0 Swept wing at 1.2 Mach show in figure 20 shows similar characteristics of 30^0 swept wing as shown in figure 12. At 1.4 Mach the lift coefficient decrease is similar to 30^0 swept wing.The Drag coefficient at 1.4 Mach is far separated from 1.2 Mach speed drag coefficient for 40^0 swept wing. As compared to Figure 13 the drag coefficient of 40^0 swept wing is less than drag coefficient of 30^0 swept wing for all Turbulence levels. The other difference noticed here is at 2% Turbulence the drag coefficient at 1.2 Mach is lesser than drag coefficient at 1.4 Mach which is not true for 30^0 swept wing.

Si No	Mach	Turbulence Intensity %	Lift Coefficient (Cl)	Drag Coefficient (Cd)
1	1.2	2	**0.053**	**0.0355**
2	1.2	5	**0.058**	**0.0361**
3	1.2	10	**0.064**	**0.0366**
4	1.2	15	**0.067**	**0.0371**
1	1.4	2	**0.0240**	**0.0365**
2	1.4	5	**0.0239**	**0.0370**
3	1.4	10	**0.0235**	**0.0376**
4	1.4	15	**0.0239**	**0.0397**

Table II. Lift and Drag coefficient of the 40^0 Swept wing at 0^0 AOA and at different Turbulence Intensity level

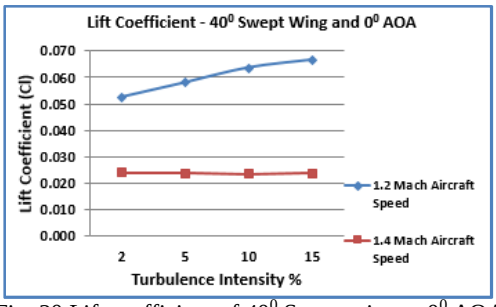

Fig. 20 Lift coefficient of 40^0 Swept wing at 0^0 AOA and at 1.2 Mach and 1.4 Mach

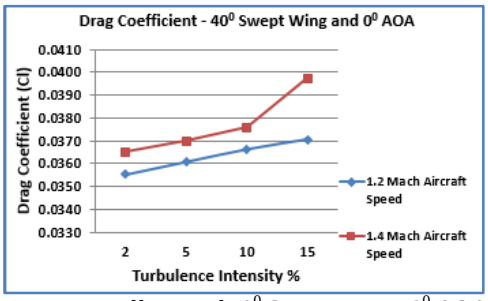

Fig. 21 Drag coefficient of 40^0 Swept wing at 0^0 AOA and at 1.2 Mach and 1.4 Mach

As compared to 30^0 swept wing the pressure observed in trailing edge of 40^0 swept wing is higher as shown in figure 22. The velocity derivative of 40^0 swept wing at 1.2Mach and 0^0 AOA is shown in figure 23. It is noticed the shock is still bow shock as the turning angle of flow is not changed at the leading edge.

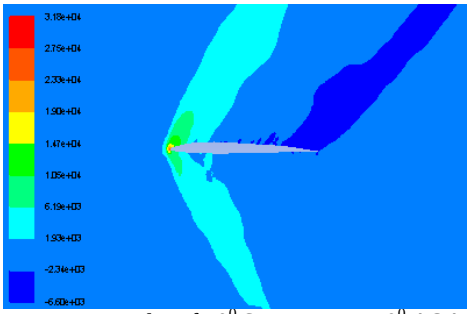

Fig. 22 Static Pressure plot of 40^0 Swept wing at 0^0 AOA, 1.2Mach velocity and 2% Turbulence intensity

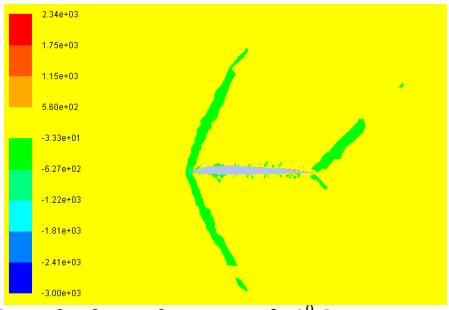

Fig. 23 Total velocity derivative of 40^0 Swept wing at 0^0 AOA, Turbulence intensity of 2% and at 1.2 Mach

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As compared to figure 15 the change in velocity of 40^0 swept wing observed in the shock is -627 1/s as seen in figure 23, whereas it is -835 1/s for 30^0 swept wing which is higher and implicates the shock is strong in 30^0 swept wing. For 40^0 Swept wing at 0^0AOA the static pressure plot is shown in figure 24 for 1.2 Mach flow speed and at 15% Turbulence intensity level. From figure 24, it is noticed that the Maximum positive pressure is less than the Maximum pressure for 30^0 Swept wing shown in figure 16 for same supersonic speed. In addition, the pressure observed at bottom portion of wing is less than the pressure observed in the 30^0 swept wing for same flow speed and Turbulence level.The velocity derivative plot of 40^0 Swept wing at 0^0 AOA and at 1.2 Mach inlet condition and at 15% Turbulence intensity level shown in figure 25. The maximum change in velocity observed is -600 1/s whereas it is -639 1/s for 30^0 Swept wing as shown in figure 18 for same 15% turbulence level. This shows increase in sweep angle reduces the effect of shock.

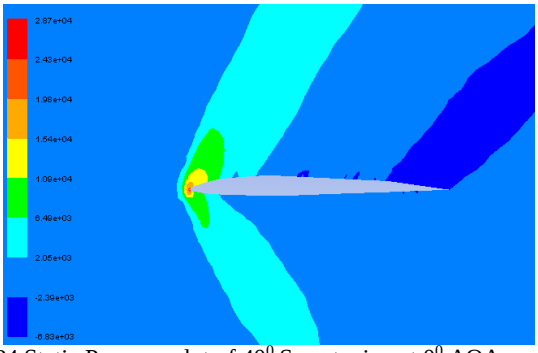

Fig. 24 Static Pressure plot of 40^0 Swept wing at 0^0 AOA, 1.2 Mach inlet condition and at 15% Turbulence intensity

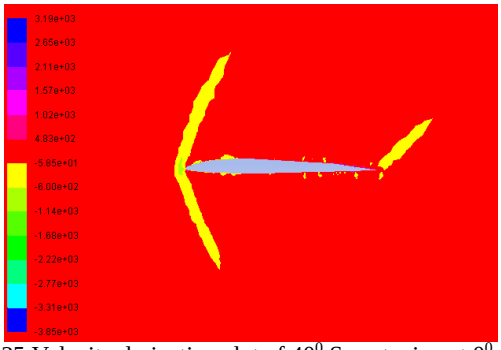

Fig. 25 Velocity derivative plot of 40^0 Swept wing at 0^0 AOA Turbulence intensity of 15% and at 1.2Mach

IX. CONCLUSION

3D Swept wing model of NACA 2412 profile and of two different sweep angles of 30^0 and 40^0 was simulated in this research work. The lift and drag coefficient of both the wings are obtained after simulating the fluid domain with wing model by using Fluent software and the results are tabulated in Table I and Table II. From the table it is noticed that lift coefficient of both the wing models of 30^0 and 40^0 sweep and at 0^0 Angle of Attack increases with increase in Turbulence from 2% to 15% and at 1.2 Mach. This shows that at low supersonic speed increase in Turbulence intensity causes the lift to increase. At 1.4 Mach there is no significant change in lift observed with change in Turbulence for both the wings.

The drag coefficient increases with increase in Turbulence level for both the wing sweeps. For 30^0 Swept wing the increase in drag is approximately 5% due to increased Turbulence level from 2% to 15%. The drag coefficient of 40^0 Swept wing is less than the drag coefficient of 30^0 Swept wing. The static pressure plot and velocity divergence plot is shown for 1.2Mach and 1.4 Mach for few of Turbulence intensity level. The maximum change in velocity calculated in the shock region in 40^0 Swept wing is less than that observed in 30^0 Swept wing, which shows the effect of shock on 40^0 Swept wing is less as compared to 30^0 Swept wing. From the figures shown for Velocity divergence it is evident that the increase in Turbulence intensity level leads to reduced velocity divergence for both the sweep angles. The results discussed above clearly depicts the advantage of wing sweep and shows the effect of Turbulence on shock waves at low supersonic speeds.

REFERENCES

[1] DR. Hermann Schlichting .“Boundary Layer Theory ” McGraw Hill Book company – seventh edition
[2] Balaji. R, Dr.PK Dash, “Swept Wing Aerodynamics at Transonic Speeds and at Different Turbulence Levels” International Journal of Current Research, Vol. 9, Issue, 06, pp.52513-52521, June, 2017.
[3] Douvi C. Eleni, Tsavalos I. Athanasios and Margaris P. Dionissios “Evaluationof the turbulence models for the simulation of the flow over a National Advisory Committee for Aeronautics (NACA) 0012 airfoil”, Journal of Mechanical Engineering Researchvol. 4(3), March 2012.

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International Journal of Innovative Research in Science, Engineering and Technology

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

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[4] Suparnamaaya Prasad, Marie L. Ivanco, Thomas G. Ivanco and Ersin Ancel "Characterization of Atmospheric Turbulence as a Function of Altitude", 17th AIAA Aviation Technology, Integration, and Operations Conference (AVIATION 2017); 5-9 Jun. 2017

[5] Mostafa Hadidoolabi, Hossein Ansarian, "Supersonic flow over a pitching delta wing using surface pressure measurements and numerical Simulations" Chinese Journal of Aeronautics, 31(1), pp. 65-78 (2018).

[6] Hoffmann, J.A. "Effects of Free stream Turbulence on the Performance Characteristics of an Aerofoil", AIAA Journal, 29(9), pp. 1353-1354 (1991).

[7] Abdulkareem Sh. Mahdi Al-Obaidi, Eric Tan Nan Kui "The effect of wing geometry on lift at supersonic speeds", Journal of Engineering Science and Technology EURECA 2013 Special Issue August (2014) 16 – 27

[8] A. Séraudie, J. Perraud, F. Moens, "Transition measurement and analysis on a swept wing in high lift configuration", ICAS 2002 congress 391.1-391.9

[9] Goldstein, M. E. "Scattering of Acoustic Waves into Tollmien-Schlichting Waves by small Stream wise variations in Surface Geometry. J. Fluid Mech. 154: 509–529 (1985).

[10] Saric, W. S., Reed, H. L. & White, E. B. "Stability and Transition of Three- Dimensional Boundary Layers" . Annu. Review of Fluid Mechanics. Vol 35: 413–440. (2003)

[11] Jorge E. Bardina "Turbulence modelling for hypersonic flight". MCAT Institute Progress Report 92-010. NASA CR 190313.

[12] Schlichting H "On the formation of the turbulence of the platelet flow" G¨ottingen, Math. Phys. Class, pp. 171-174 (1933)

[13] Gray, W. E. "The effect of wing sweep on laminar flow". Tech. Rep RAE TM Aero 255. British Royal Aircraft Establishment (1952).