# Simulation Study of Supersonic Natural Laminar Flow on Wing with Biconvex Airfoil

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***Abstract*— This Paper is focused on study of Supersonic Natural Laminar Flow (SNLF). With the help of Supersonic Natural Laminar Flow turbulence less flight can be possible at supersonic speed. To achieve that, unique wing designs are required to generate high lift and low drag. As the wings are most important part of the aircraft which uses pressure created by air to its advantage. The paper studies behavior of thin airfoil at supersonic speed with Supersonic Natural Laminar Flow with the thin airfoil used to design wings for Supersonic Business Jet (SBJ).**

**Keywords —** ***Supersonic aircraft, Supersonic Speed, Wing Design, SNLF, Biconvex airfoil, Lift, Drag reduction.***

## I. INTRODUCTION

Few decades ago when Concorde took its first supersonic flight it was dream come true moment for aeronautic industry. The aircraft was cable of traveling at Mach 2.04 (2180km/hr.) speed and was able to cover New York to London distance in less than four hours. But after few years due to expensive flight rates and due to regulations on supersonic flights for safety, manufacturing and research of supersonic flights almost stopped.

Since that supersonic transport is been limited to military aircrafts. But recently there has been research on supersonic flights to be more suitable for civilian aircraft. This paper is based on research on wing design for Supersonic Business Jet (SBJ). Goal is to design wing which can generate more lift force and lower the drag[1].

Whenever supersonic jet is discussed most of the time delta shaped wings are considered for the supersonic cruise. Other times instead of delta back swept wings at high sweet angles are believed to be excellent choice for the supersonic cruise. In this paper instead of delta shape, high sweep angle alternative of low swept wing is considered[1] for with low sweep angle[2].

The paper shows the shock wave analysis on this wings at different sections of the wings. And provides the data for the lift and drag coefficient.

## II. LITERATURE REVIEW

The Supersonic aircraft uses biconvex and double wedge airfoil but any analysis data for these two airfoils is not available easily. Since details of the coordinates and analysis results for pressure distribution for both airfoils are not present in NACA (National Advisory Committee for Aeronautics) [3] or UICC aerospace airfoil database[4].

Data related based on symmetrical airfoil for low Reynolds no. can be predicated with CFD[5, 6] study but for supersonic flow, nature of lift and drag changes at high Reynolds no is not available.

Study available for designing subsonic flight and detail study is available with required formulas on wind design, load and lift drag distribution study[7, 8] . Using data based on lift/drag coefficient is used for the selection of the airfoil and wing design. Supersonic fighter jet F-16 aerodynamic data was available which provides insights on wing optimization and detail drag analysis. Study of distribution of pressure overall wing at transonic speed and lift generation data for Mach 1.414 was presented for 60$^{0}$ swept[9]. At same time even if wings with swept angles are used sometimes it comes with instability due to the mechanisms which was dangerous if aircraft goes above critical speed[10].

For optimization of the wings of the supersonic flight Multiple Objective Genetic Algorithms (MOGAs) are used to get optimizing planform shape by using Euler equations[11].

Linearized supersonic theory and other shockwave related phenomenon was provided to measure specifically for bodies movies at supersonic speed and changes[12]

## III. BACKGROUND

To design wings, it is important to know aerodynamic parameters. To do that earlier days only available option was mathematical models. Based on that airfoil modifications were done to achieve required lift and drag.

Now a day's simulation of the airfoil and wings can be done on different software's. And for simulation of 2D airfoil can be done on simple java applet. Which requires designer to put all the inputs to get required design and required airflow conditions. NASA's Foilsim and Xfoil XFLR5 are great to generate or modify the airfoil and to Simulation results but this software's are not useful for high velocity airflow and high Reynold's no[7].

On the industrial level Software's Cart3D, Rage[13], OpenFOAM, Autodesk simulation, COMSOL, SimScale are used for such simulation but due to High cost of licensing and high system requirement we considered using ANSYS Fluent or Solidworks Flow Simulation.

Hence to get the results for supersonic flight software's like Ansys CFD/Fluent or SolidWorks Flow Simulation can be helpful to understand wing analysis at supersonic flow.

## IV. CASE STUDY

Aerion Corporation is developing Supersonic Business Jet (SSBJ) named Aerion AS2 with CTO Dr. Richard Tracy. The thin & smooth wing development with Supersonic Natural Laminar Flow which can reduce drag [14].


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The AS2 and data presented by Aerion corporation is used as case study. Since this project is still under development not lot of data was not publicly presented and hence for this paper we are considering some assumptions.

## V. PROBLEM STATEMENT

The Supersonic Business Jet with maximum speed of 1.4 Mach needs wings which can generate enough lift. For landing and take-off, the flight speed will be reduced to 0.95 Mach due to supersonic flight regulations. The wings will have thin airfoil. The maximum allowable wingspan is of 23.5m and wing area will be 140 sq. m. Maximum aircraft weight is around 60,000kg and fuel weight is to be considered around 26,800 kg.

The aircraft minimum speed is in subsonic regime but for the research we will be only analyzing wing in supersonic regime. Also, we will not be considering the flaps and aileron since at the time supersonic cruise it will flap & aileron will be closed.

## VI. METHODOLOGY

To design wing for supersonic wing airfoil selection and design parameters need to be choose carefully since selection criteria does not work for supersonic aircraft like it works for subsonic aircraft. For most of the subsonic flight Prandtl-Glauert's linearized compressibility correction is used giving Cp = $Cp,0/(\sqrt{1-(M\infty)^\wedge 2})$. While foe supersonic flights Cp =$Cp,0/(\sqrt{(M\infty)^2-1})$ as for supersonic speed M∞ > 1. Designing wing for supersonic aircraft requires modified parameters than regular wing parameters.

Based on the requirements airfoil selection is done but for most of the supersonic aircraft use extremely thin airfoil. It is necessary to see that airfoil satisfy the required forces. For that we calculated the lift and drag coefficients for the airfoil and based on the calculated parameter the design was checked with Prandtl's lift lining theory to obtained elliptical lift distribution and coefficient of lift from the MATLAB programming.

Rest of the calculations and CAD geometry was done by using Linearized supersonic theory for flat plate.

### A. Speed of Sound

Speed of the sound in perfect gas is always depended on the temperature of the gas. As altitude changes the air temperature changes giving the different speed to the sound at different altitude. The high-speed gas flow at higher altitude is calculated in the terms of Mach number. The Mach number above 1 is to be considered as supersonic regime.

When object moves in air with some velocity it transfers some amount of the energy in neighboring gas molecules giving them kinetic energy.

### B. Shock Wave and Effects

When body travels with more speed than the speed of the local sound wave, molecules in surrounding starts moving in every direction and comes together and create different physical properties around body.[15]

In supersonic flow shock waves are thin flow at which disrupt changes in pressure and other properties can be seen.

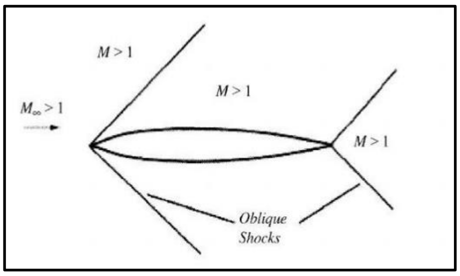


Fig. 1. Biconvex Airfoil at Shock wave

### C. Supersonic Natural Laminar Flow.

The streamlines in this flow are smooth and regular. The airflow on the wing will move smoothly along streamline.[16]

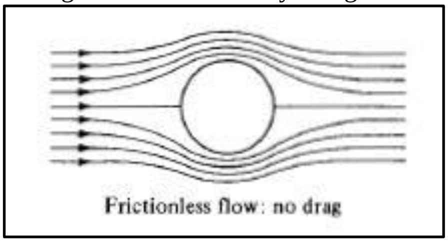


Fig. 2. Laminar Flow around the Sphere

Shear stress in laminar flow is less than turbulent flow and will give low skin friction drag on the surface of the wing body. LFC is applied at the leading edge to provide smooth transition of the boundary layer.

### D. Angle of Attack

Angle of attack is the angle between wing/ aircraft or wind. For most of the jet reference line is between fuselage centerline. For supersonic flight requires AOA to be in between 0-1º.

### E. Prandtl's Classical Lifting-Line Theory

Prandtl's lifting line theory gives practical lift distribution on the 3D aircraft wing. The mathematical model is used to distribute lift in symmetrical shape overall wing. The mathematical model will be useful to modify the dimensions[8].

### F. Linearized Supersonic Theory

Linearized supersonic theory is based on the perturbation velocity potential equation which holds for both supersonic and subsonic flow. Since we need design which can hold itself in the both flow this theory will be useful to predict lift and drag forces[12].

## VII. WING DESIGN

The wing design is based on the many parameters and required to be compatible with other body parts of the aircraft body. The primary purpose of the wing is to generate enough lift such that it can provide sufficient force for aircraft to fly.

### A. Parameters

*a) No of Wings:* one single wing is used as conventional modern aircraft.

*b) Wing Location*: the wings vertical location is to be considered as high wing which gives more lift than mid/low wings and will reduce stall. Also, it provides nose-up pitching moment. Correspondingly, for safety considerations high wings are used.


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*c) Aspect Ratio (AR)*: As for supersonic flight short wings are necessary and ideal aspect ratio for supersonic aircraft is in between 2-4, we assume it to be 3.5.

*d) Taper Ratio*: taper ratio for trapezoidal wings are in between 0-1. For our purpose we selected $\lambda$=0.6

*e) Tip & Root Chord & Mean Aerodynamic Chord(MAC)*

$$AR = b2\,/S \quad (1)$$

Therefore, $b = \sqrt{S} * \sqrt{AR} = \sqrt{140 * 3.5} = 22.13$ m (2)

Also, AR = b/MAC (3)

MAC = b/AR

= 22.13/3.5 = 6.32 m

Based on MAC, we can calculate Cr and Ct

Since, $MAC = \frac{2*Cr*(1+0.6+0.6\text{^}2)}{3*(1+0.6)}$ (4)

Cr = 7.714 m & Ct = 4.6 m

*f) Twist angle*: For supersonic aircraft adding twist angle can reduce the lift hence no twist angle is desired.

*g) Sweep angle*: to improve the wing aerodynamic drag, lift at supersonic speed sweep angle is required.

$$\mu = Sin^{-1}\frac{1}{M} = Sin^{-1}\frac{1}{1.2} = 56.44^o \quad (5)$$

$$\Delta = 1.2 * (90 - \mu) = 1.2*(90-56.44) = 40.27^o$$

But since we want to develop wing with low sweep angle, the final selected angle was $13^0$.

*h) Dihedral angle*: for Supersonic aircraft with high wing ideal dihedral angle is between 0 to -5º. For our purpose the selected angle is 0º.

*i) Incidence ($i_w$)*: also known as wing setting angle for ideal supersonic aircraft is in between 0 to 1 which is required to generate minimize drag and maximize lift.

*j) Flaps & aileron*

At the time of supersonic cruise flaps and aileron will be closed hence we will not consider them for our analysis.[17, 18] For this design Droop nose type slat is recommended for leading edge.

*B. Airfoil Selection*

To achieve Supersonic flight two types of airfoils are considered.

1. Biconvex Airfoil
2. Double Wedge Airfoil

the airfoil used for this purpose are biconvex airfoil with 4% and 3% thickness to chord ratio. The supersonic aircraft requires airfoil with minimum (t/c) max.

The thin airfoils are used for root and tip of the wing. The thin airfoil is used to control laminar flow at leading edge and trailing edge. The biconvex airfoil used for root is 4% thickness and tip biconvex airfoil 3%.

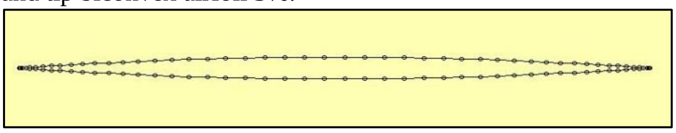

Fig. 3. Biconvex 4% airfoil

Above fig shows biconvex airfoil with 99 plotting points which was obtained using JAVAFoil applet. The thickness to chord ration is 4% and maximum thickness location is at 50%. This airfoil will be used at the root of the wing.

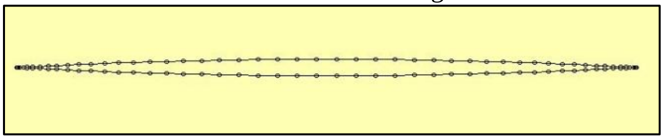

Fig. 4. Biconvex 3% airfoil

Above fig shows biconvex airfoil with 99 plotting points which was obtained using JAVAFoil applet. The thickness to chord ration is 3% and maximum thickness location is at 50%. This airfoil will be used at the tip of the wing.

*C. Lift and Drag calculations*

Considering aircraft is traveling at supersonic Mach 1.4 speed at altitude of 15km we will be calculating lift and drag parameters.

At altitude 15km air properties are as follows

- $p_\infty$ = 1.2112*10^4 N/$m^2$
- $T_\infty$ = 216.66 K
- $б\infty$ = 0.19475 m/s

Hence, Speed of sound at 15km will be

$a_\infty = \sqrt{YRT\infty} = \sqrt{1.4 * 287 * 216.66}$ =295.04 m/s (6)

$V_\infty = M_\infty * a_\infty$ = 1.4*295.04 = 413.056 m/$s^2$ (7)

Also, dynamic pressure

$q_\infty = (0.5) *(б_\infty) *(V_\infty)$ = (0.5*0.19475*413.056^2) (8)

$q_\infty$= 16613.66 N/$m^2$

For supersonic aircraft coefficient of lift is calculated by using[19],

$$C_l = \sqrt{\frac{Wavg}{q\infty * S}} = \sqrt{\frac{48750*9.81}{16613.66*140}} = 0.4518 \quad (9)$$

Now to calculate $C_{lw}$ for wing

Ideal wing lift coefficient

$$C_{lw} = C_l/0.95 = 0.4756 \quad (10)$$

Ideal airfoil lift coefficient

$$C_{li} = C_{lw}/0.90 = 0.5284 \quad (11)$$

Considering HLD with $\Delta C_{flap}$ = 0.45 are used and for supersonic conditions Flap will be up net Clw will be

Net $C_{lw} = C_{li} - \Delta C_{flap}$ = 0.5284 – 0.45 = 0.0784 (12)

But since we are considering thin airfoil and by linearized supersonic theory

$$C_{li} = \frac{4\alpha}{\sqrt{M\text{^}2-1}} \quad (13)$$

Therefore α = 1.09º which is in the range of supersonic flight. Also, for both our airfoil maximum lift is around 1.09º.


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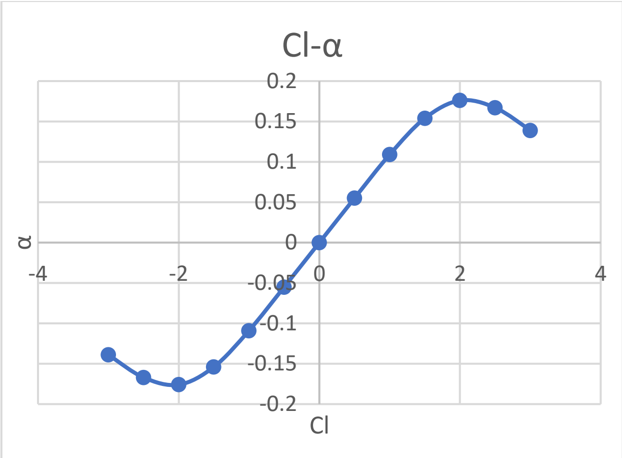


Fig. 5. Cl-α Graph for biconvex 4% airfoil.

But after observing the curve for smooth laminar flow and considering max usable angle of attack we choose angle of attack 0.7º for our wing with $C_{li}$ = 0.077.
Henceforth for the Biconvex 4% airfoil we calculated the lift slope curve

$$C_l\,\alpha = 1.8 \prod (1 + 0.8\tfrac{t}{c}) = 5.83 \qquad (14)$$

Also, by using Prandtl's Lifting line theory based on our parameters for wing and lift curve slope for our airfoil the lift distribution and ideal lift coefficient obtained from MATLAB programming for our wing was not ideal.

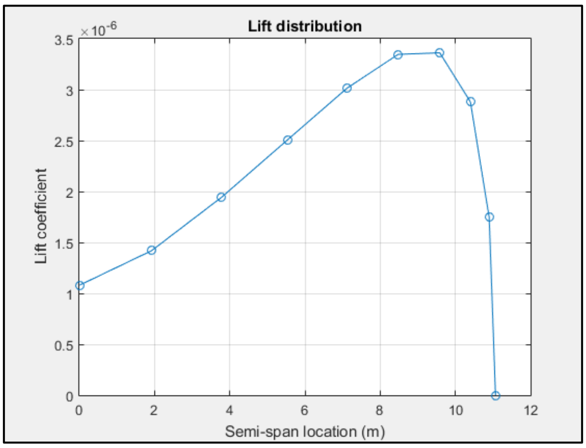


Fig. 6. Lift distribution based on the calculated parameters

Hence after trial and error method we got following new parameters with better lift distribution in desired elliptical form to reduce induced drag.

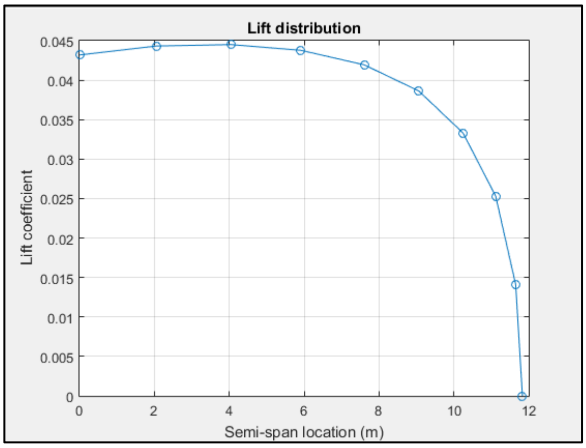


Fig. 7. Lift distribution based on the MATLAB parameters

Hence new parameters and new Lift coefficient for wing can be given as following

TABLE I. DESIGN PARAMETERS

| Design Parameters | Calculated | Optimized |
|---|---|---|
| AR | 3.5 | 4 |
| $\lambda$ | 0.6 | 0.75 |
| α | 1.99º | 0.7º |
| αt | 0º | 0º |
| $C_{lw}$ | 0.0784 | 0.2007 |

Based on the new parameters we calculated remaining parameters.

$$AR = b^2/S$$

Therefore, $b = \sqrt{S} * \sqrt{AR} = \sqrt{140 * 4} = 23.66$ m

$$AR = b/MAC$$

$$MAC = b/AR = 22/4 = 5.5 \text{ m}$$

Based on MAC, we can calculate Cr and Ct

Since, $MAC = \dfrac{2*Cr*(1+0.75+0..75\char94 2)}{3*(1+0.75)}$

Cr = 6.25 m & Ct = 4.7 m

Also, Lift force for supersonic aircraft can be calculated as

$$L = (q\infty) * (S) * (Cl) = 46{,}867.13 \text{ N} \qquad (15)$$

Similarly, for Drag force we calculate different drag coefficient as drag force is combination of all of them.

1. Wave drag: As we have deceived to use linearized supersonic theory

$$C_{d,w} = \frac{4\alpha\char94 2}{\sqrt{M\char94 2-1}} = 0.45*10^{-3} \qquad (16)$$

2. Skin friction drag:

$$a_\infty = \sqrt{\Upsilon RT\infty} = 295 \text{ m/s}^2$$

$$V_\infty = M_\infty * a_\infty = 1.4*295.04 = 413.056 \text{ m/s}^2$$

By Sutherland's Law

$$\frac{\mu}{\mu 0} = \left(\frac{T}{T0}\right)^{\frac{3}{2}} \left(\frac{T0+110}{T+110}\right) \qquad (17)$$

By using value at T=216.66K for μ0 & T0

$$\mu = (1.7894*10^{-5})*(\frac{216.66}{288})\char94 (3/2)\ (\frac{288.16+110}{216.78+110})$$

$$= 1.421*10^{-3} \text{ Kg/ms}$$


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Reynold's No: $\frac{(\sigma\infty)*(V\infty)*C}{\mu} = 3.112*10^7$ (18)

By using graph $C_f$ at Mach 1.4 for biconvex airfoil.

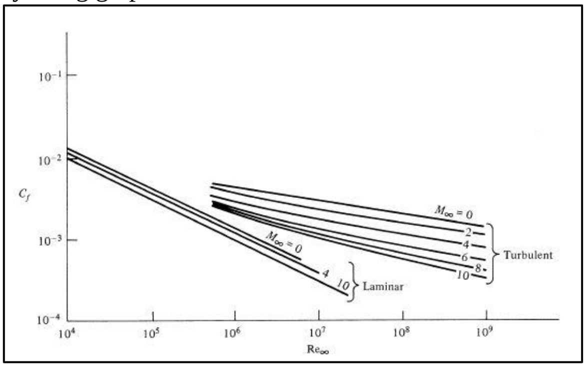


Fig. 8. Frictional drag for flat plate shown (Data based on van Driest calculations)

Based on the careful observation of graph we can say laminar boundary with Re= 3.11 X $10^7$, we can get $C_f$ = 3.25 X $10^{-3}$. Also, for biconvex profile, drag will be twice as two surfaces are involved.

Net $C_f$ = 6.50 X $10^{-3}$

3. Profile drag: Since we are using thin profile leading edge $C_p$ will be very small, almost negligible.

4. Induced drag: Induced drag is generated due to high lift generation.

$$C_{di} = \frac{Cl^{\wedge}2}{\prod eAR} = 3.2 \text{ X } 10^{-3} \quad (19)$$

Total Drag: $C_{d,w} + C_f + C_{di} + C_{dp} = 0.01015$

Total Drag force will be

$$D = (q_\infty) * (S) * (C_d) = 11745.85 \text{ N} \quad (20)$$

*D. Pressure and Temperature Calculations:*

Pressure and temperature at altitude 15Km considered on the surface at supersonic speed[20] can be calculated by:

$$\frac{P0}{P} = \left(\frac{(\Upsilon+1)^{\wedge}2M^{\wedge}2}{4\Upsilon M^{\wedge}2-2*(\Upsilon-1)}\right)^{\wedge}\left(\frac{\Upsilon}{\Upsilon-1}\right)\left(\frac{1-\Upsilon+2*\Upsilon M^{\wedge}2}{\Upsilon+1}\right) \quad (21)$$

$$P_0 = 36928.81 \text{ N/m}^2$$

For the maximum temperature[10]

$$\frac{T0}{T} = 1 + \left(\frac{(1-\Upsilon)M^{\wedge}2}{2}\right) = 301.59 \text{ K.} \quad (22)$$

*E. Flow simulation results for designed wing.*

1. Wing Structure

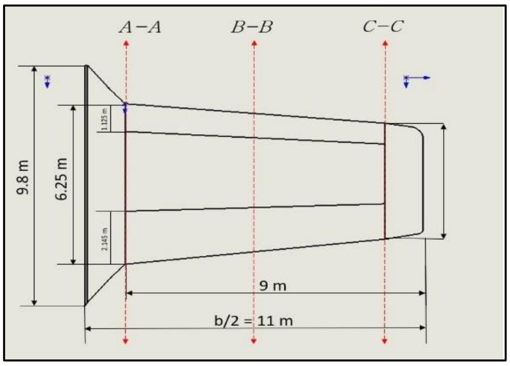


Fig. 9. Wing Dimensions

As show above we are considering three sections of the wing, section A-A, Section B-B, Section C-C.

2. 3D CAD geometry

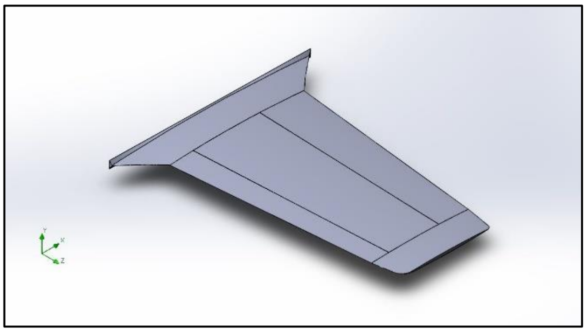

Fig. 10. CAD model

The Wing was designed in SolidWorks and provided all the required features necessary for this study and tried to get reliable model.

3. Flow simulation Domain

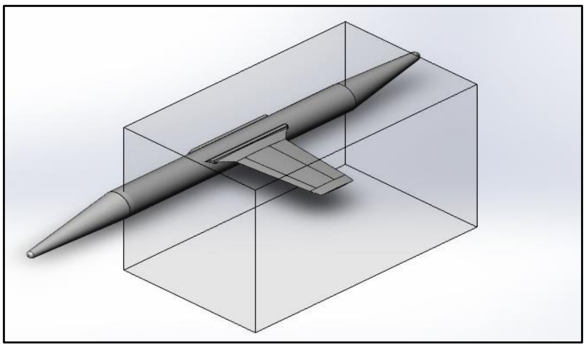

Fig. 11. CAD model

Flow domain was constructed around the wing only and excluded other geometry (i.e. aircraft body) since our focus is to study the effect of shock wave at supersonic speed around wing.

4. Flow Distribution


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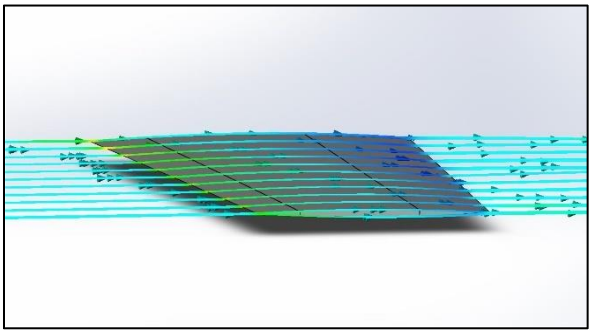

Fig. 11 a) Smooth transition of flow on wing surface

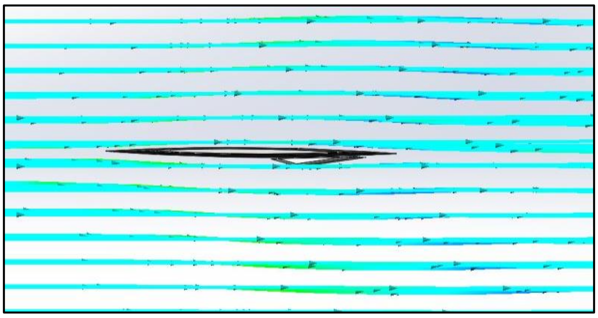

Fig. 12. b) Smooth Laminar Flow distribution

Above Figure 8 a) shows flow around the wing where it can be clearly seen that no turbulence is generated after the transition and flow remerge with the original flow due to thin leading edge.

Figure 8 b) also confirm that flow is not disturbing flow above and below the wing.

The thin airfoil is helpful to produces favorable condition with flow stream in order generate significant pressure gradient. This significant pressure gradient

5. Pressure contours at Mach 1.4

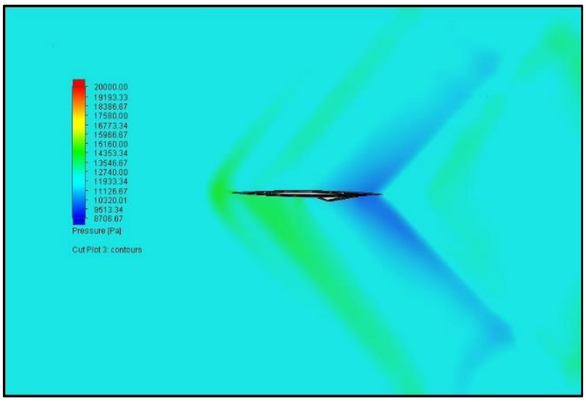


Fig. 13. a) Pressure distribution at wing section A-A

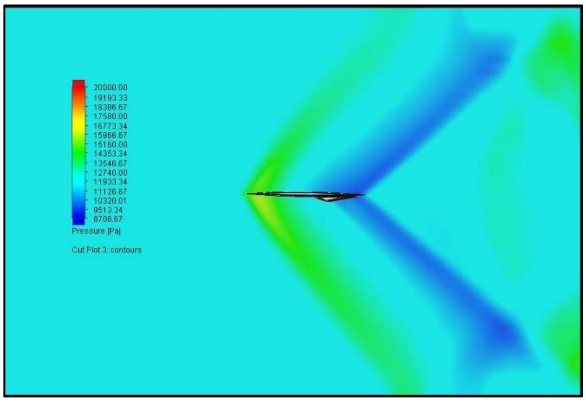


Fig. 13. b) Pressure distribution at wing section A-A

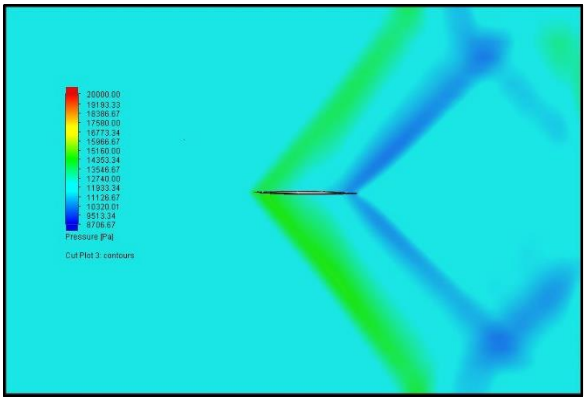


Fig. 13. c) Pressure distribution at wing section A-A

Fig a), b), c) show the pressure distribution across the wing at different section. From above figure it is obvious that the pressure at the leading edge is higher than the trailing edge. The pressure is maximum at section B-B.

In Fig a) lower surface the area at leading edge shows high pressure region than any other surface. If looked carefully it is safe to say that upper surface of the leading edge at that section is not under very large pressure difference. while at the tailing edge upper and lower both surfaces are at low pressure.

In Fig b) at leading edge thin section of the wing seems to be creating more pressure at time of shock wave generation. Also, at this region tailing low pressure region is more effective from tailing edge side.

In Fig c) lower surface of the leading edge since to be under more pressure than upper surface. On the other hand, for tailing edge pressure on the lower surface is decreasing rapidly comparing to the upper surface.


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6. Velocity contours at Mach 1.4

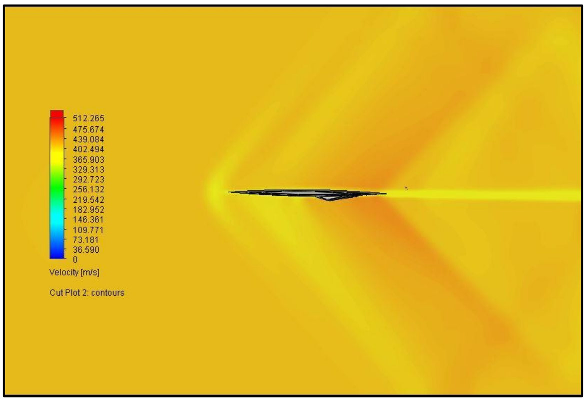


Fig.14. a) Velocity Contours at wing section A-A

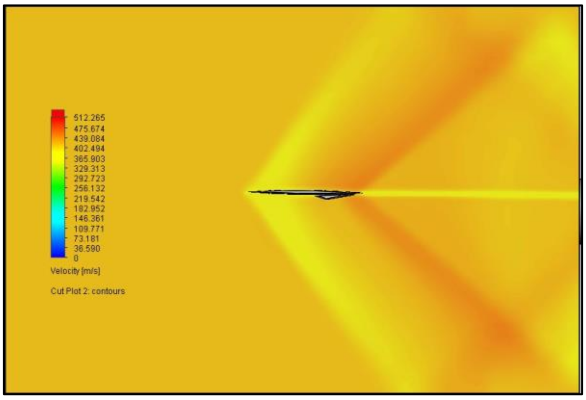


Fig.14. b) Velocity Contours at wing section B-B

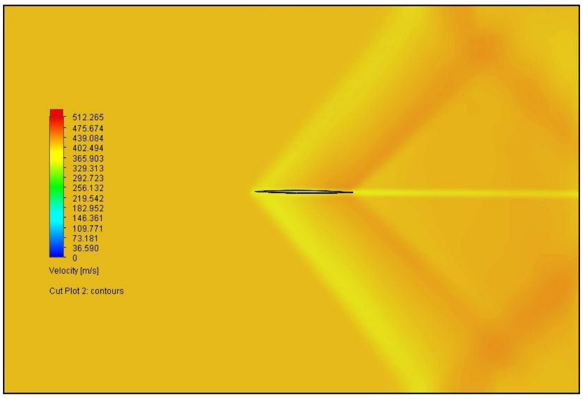


Fig.14. c) Velocity Contours at wing section C-C

Fig a), b), c) show the behavior of velocity across the wing at different section.

From above figure we can say that, shock wave at tailing edge of section A-A shows high velocity region which covers half of the wing surfaces from the tailing edge and it keeps sharpening till the section B-B. Similarly, velocity at shock wave generated at leading edge of section A-A keep increasing gradually to section B-B.

Similarly, from Section B-B to Section C-C velocity at shock wave from leading edge covers more than half of the surface of upper and lower wing and almost start reducing its effect. While from Section B-B to Section C-C velocity at tailing edge gradually decreases.

7. Mach No

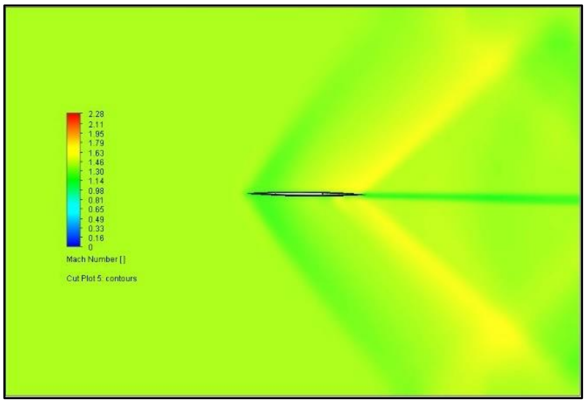


Fig. 15. Mach Number

From observing Fig shows section B-B and shows same effect overall the wing. we can safely say that Mach number changes at the leading edge and tailing edge.[21]

8. Temperature

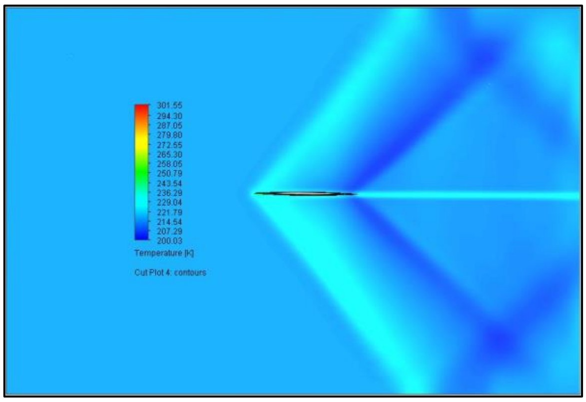


Fig. 12. Temperature around wing

The temperature profile across the wing is similar. The leading edge is at higher temperature than tailing edge. The highest temperature is at the section where wing intersect with fuselage.

### *F. Results*

TABLE II. COMPARISON RESULTS

| Parameters | Calculated | Flow Simulation |
|---|---|---|
| Cl | 0.2007 | 0.1953 |
| Cd | 0.0101 | 0.0111 |
| Lift Force | 46,867.13 N | 44396.70 N |
| Drag Force | 11745.85 N | 12975.74 N |
| Highest Pressure ($N/m^2$) | 36928.81 ($N/m^2$) | 22786.15 ($N/m^2$) |
| Highest Temperature (K) | 301.59 K | 301.55 K |


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## VIII. CONCULSION

The flow simulation results and calculated results are almost similar. Due to iterative natural of flow simulation it can be said that, lift and drag coefficients and the lift and drag forces based on them are almost Close by and satisfactory.

The method used to study behavior of wing at supersonic speed based on the supersonic linearized theory hold good for thin airfoil and the pressure and temperature range are in the satisfactory level.

Lot of scope is available for the future study can be done based on the results. Results can be used to Modify the dimensions and by using high lift devices and other LFC techniques to get higher amount of lift force.

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