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Aerodynamic Analysis of a Supersonic Airfoil at a Fixed Angle of Attack Using Computational Fluid Dynamics

Soumitra M. Dodkey
Dept. of Aerospace Engg,
Hindustan Institute of Technology & Science,
Padur, Chennai,
India.soumitradodke@gmail.com

Manthan K. Patil
Dept. of Aeronautical Engg,
Annasaheb Dange College of Engg. & Technology,
Ashta, Maharashtra, India.
manths777x@gmail.com

Madhumitha M.
Dept. of Research & Development,
Abyom Space Tech and Defense,
Gorakhpur, Uttar Pradesh, India.
m.madhumitha1511@gmail.com

Abstract- Airfoil selection and its design is the most significant step in the aircraft design process. An airfoil is a cross sectional geometry of a wing, which is responsible for the aerodynamic forces. Multiple airfoils when stacked together form a wing. Lift force is generated by the wing which balances the weight force acting downwards. Before selection of any airfoil for supersonic aircraft, detailed analysis of the shock, boundary layer and other flow parameters must be performed. Aerodynamic analysis is done mainly to find the lift coefficient and drag coefficient of the vehicle, predict high- and low-pressure areas and the separation points which affect vehicle dynamics. This analysis has been carried out using CFD, which comes under fluid mechanics. Data structure and Numerical analysis are involved for analyzing and computing problems that include flow of the fluid. The aerodynamic analysis carried out, gives us an idea on various parameters and properties of the airfoil NACA 66- 206. This paper deals with the analysis performed on a NACA 66 series supersonic airfoil. Analysis is done to find out the coefficients of lift and drag at Mach 2 with a fixed angle of attack. Using Computational Fluid Dynamics, pressure, velocity, M, temperature &Reynolds number distribution have been studied over the top and bottom surfaces of the airfoil. This study also includes analysis of the shock pattern at supersonic speed. The subsequent analysis can be further used for determining the drag divergence effect on the lift generated by the NACA 66-206 airfoil.

Keywords: Shock analysis; Supersonic Airfoil; NACA 66- 206; viscous flow; Boundary Layer; CFD.

I. INTRODUCTION

The fundamental forces acting on the aircraft are Lift, Thrust, Weight and Drag. The airfoil generates lift, and drag force acts subsequently on it. The force which acts perpendicularly upwards to the forward motion is the lift and the force which acts along the direction of the flow is the drag experienced.

Lift is generated when there is a difference in pressure over the top and bottom surfaces of an airfoil. When an airfoil moves in a viscous fluid, there is a specific flow regime, where a low thickness, large velocity gradient will be formed around it, known as boundary layer.

The immensity of lift and drag produced by the airfoil are affected by the changes caused by this flow regime. Based on the air velocity, classification of the aircraft can be done. Velocity of an aircraft is usually as Mach number. This is the result of the value of speed of the object divided by the speed of sound. M is the denotation for Mach Number. When the M is less than 0.8, it can be classified as a subsonic aircraft. Transonic aircrafts fly in the range of 0.8 to 1.2M, and beyond 1.2M is identified as supersonic aircraft.

The major purpose of a supersonic airfoil is to generate an adequate amount of lift in the supersonic flow regime. These airfoils are designed in such a way that would not allow formation of a detached bow shock. For this reason, the leading edge and trailing edge of the airfoil are kept sharp.

Airfoils which are used in supersonic aircrafts are usually a thin section consisting of planes which are at an angle, or arcs that are in opposition to each other. When the fluid flow's local velocity attains sonic condition (Mach 1), a shock is generated at a point along the airfoil.

The aim is to perform aerodynamic analysis over a supersonic airfoil at a fixed angle of attack using Computational Fluid Dynamics (CFD). In this analysis, the airfoil chosen is NACA 66-206. Thickness is 6% and the analysis is done keeping the angle of attack fixed at positive 5 degrees. The aerodynamic parameters and shock are examined for Mach 2 using ANSYS FLUENT.

II. METHODOLOGY

This study includes the use of NACA 66-206 airfoil. The design of the airfoil is in two-dimensional (2-D) shape and has a chord length of 0.97 cm and the direction of the unidirectional flow is kept at 5°.

Airfoil NACA 66-206 has been plotted in the design modeller which is shown in Fig. 2 and exported to the commercial computational fluid dynamics software (ANSYS). Analysis was done at Mach 2 and corresponding plots and contours were obtained.

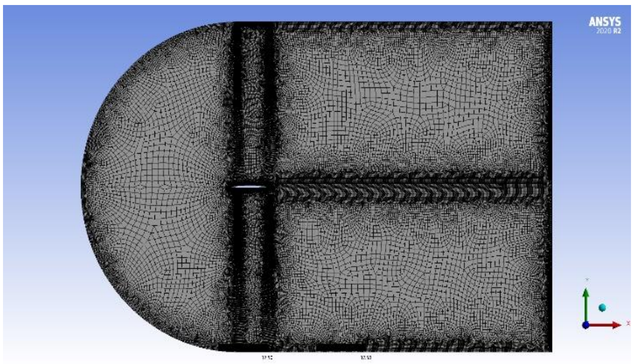

Fig 1. Complete mesh generation of NACA 66- 206 airfoil.

The following section is used in ANSYS to discuss the analysis for the NACA 66-206 airfoil. For the meshing part of the airfoil, various sections are used and depending on the region, thin and dense mesh is implemented. The generated mesh is shown in Fig.1 & Fig.2 respectively.

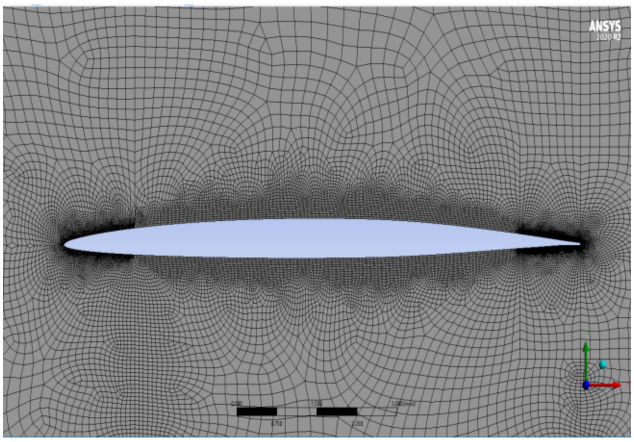

Fig 2. Expanded version of the generated mesh.

In this case, the fluid used is air and the value of Cp is a constant i.e., 1006.43 J/Kg-K. Precision depends on the structure of the meshing part. The nodes were set at 139052 and the elements were at 137579.

Precision again depends on the number of nodes and elements. Had the nodes and elements in the generated mesh be higher, the outcome would have been accurate. But they were fixed around 137000to reduce the complexity of the simulative process. The values for the simulation which needs to be input were fixed are given in Table 1.

Table 1. Parameters set or NACA 66- 206 airfoil for simulation.

Fluid	Air
Density	Ideal gas
Cp	1006.43 J/Kg-K
No. of elements in the mesh to be generated	137579
No. of nodes to be generated in the mesh	139052
No. of iterations of mesh generation	1000
Model	SST K-Omega
Velocity of Laminar Flow	1 ms^{-1}
Density of air	1.225 kgm^{-3}
Viscosity	Sutherland Three Coefficient Method

1. Boundary Conditions:

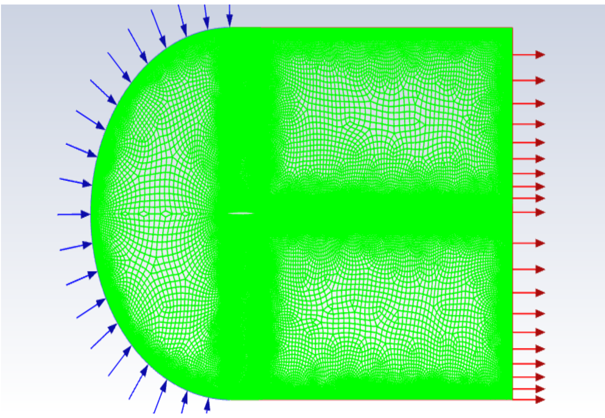

Fig 3. Meshing in the domain with inlet and outlet boundary position.

Table 2. Boundary conditions for NACA 66- 206 airfoil.

S. No.	Position	Boundary condition	Value
1	Inlet	Velocity inlet	X-component: 686 cos (5°) m/s Y-component: 686 sin(5°) m/s
2	Outlet	Pressure outlet	Gauge Pressure= 0 Pa
3	Airfoil	No-slip	-
4	Far field	Pressure Far Field	Mach=2

2. Graphs:

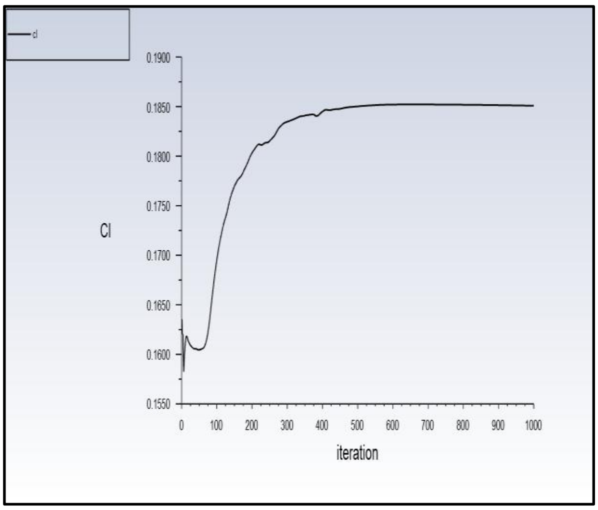

Fig 4. Coefficient of lift.

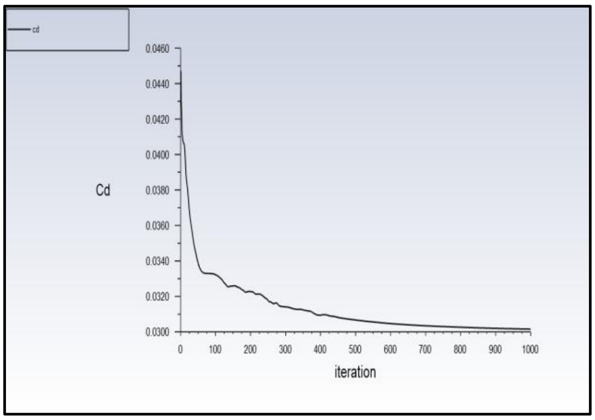

Fig 5. Coefficient of drag for NACA 66- 206.

At Mach 2, the coefficient of lift for NACA 66-206 gives a value between 0.15 to 0.19, when free-stream flow is kept at a slope of 5°. When examining the lift coefficients, after 400 iterations it was found that the values were around 0.1850 and remained constant after that.

For coefficient of drag, the NACA 66-206 gave a continuous declining curve. The lift coefficients were around 0.03, which were very close to zero.

III. RESULTS AND DISCUSSIONS

1. Velocity:

The above contour shows velocity distribution at Mach 2. After analyzing the velocity distribution over the airfoil, it was found that surface velocity over the entire airfoil ranges from 3.01E+02 m/s to 5.26E+02 m/s and velocity increases to the far field in the range of 6.02E+02 m/s to 7.52E+02 m/s.

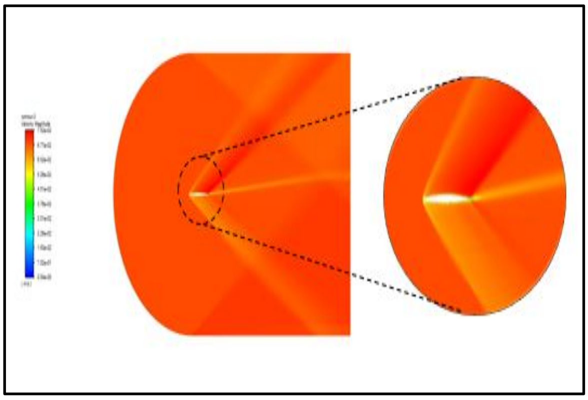

Fig 6. Velocity distribution over NACA 66- 206 airfoil at Mach 2.

2. Dynamic and Static Pressure:

The contour in Fig.7 and Fig. 8 shows that the static pressure is 9.27E+04 Pa at upstream and due to shock, it is decreased to 4.49E+04 Pa. Dynamic pressure found upstream is 3.18E+05 Pa and downstream is 1.59E+05 Pa.

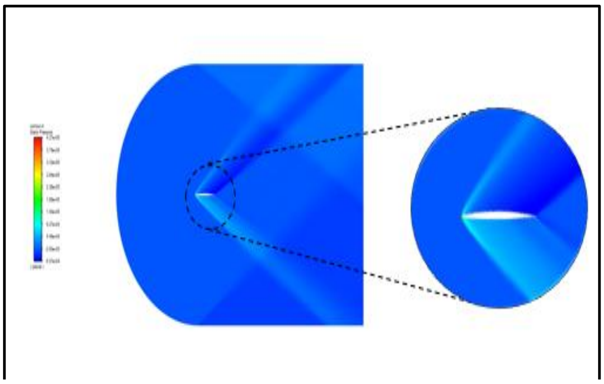

Fig 7. Static Pressure distribution at Mach 2.

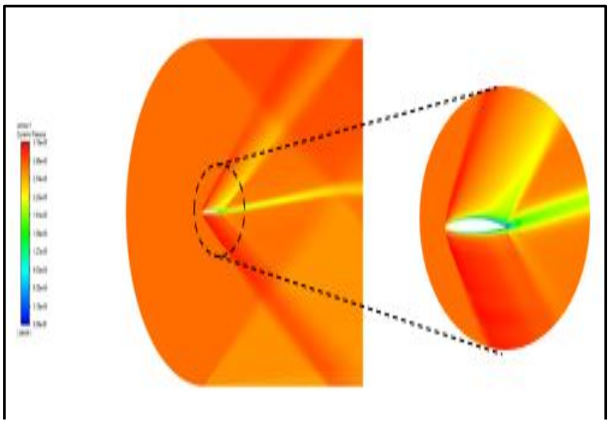

Fig 8. Dynamic Pressure distribution over NACA 66-206 airfoil at Mach 2.

3. Mach number:

The contour in Fig. 9 shows variation in Mach number. After analysing Mach number distribution over the airfoil it was found that surface Mach number over the entire airfoil is ranging from 0.9 to 1.42 and Mach number increases to the far field in the range of 1.89 to 2.36.

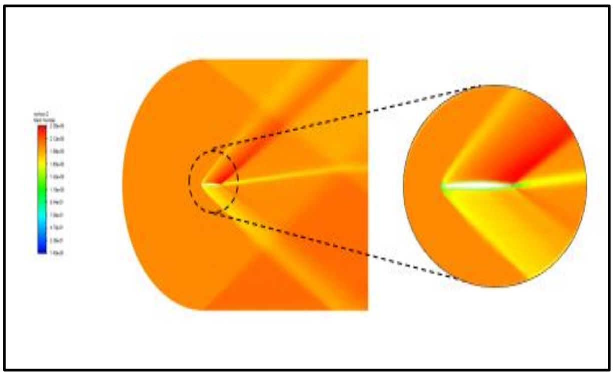

Fig 9. Variation in M on NACA 66-206 airfoil at Mach2.

4. Static Temperature:

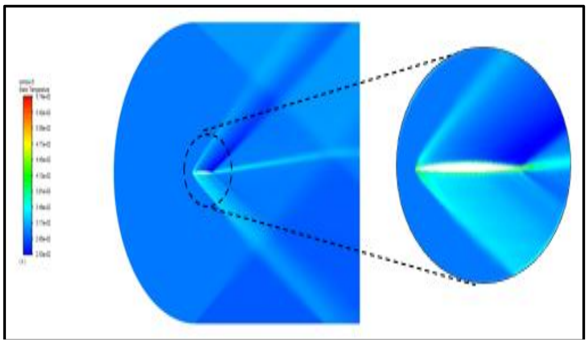

Fig 10. Dynamic Pressure distribution at Mach 2.

The contour below shows that surface temperature is varying between 318 K to 477 K and far field temperature is 285 K.

5. Reynolds Number:

The contour in Fig. 11 shows the variation in Reynolds number over NACA 66-206 airfoil it was found that, in the far field it ranges from 3450 to 7770 which is turbulent flow and at the surface it is found to be ranging between 863 and 1730.

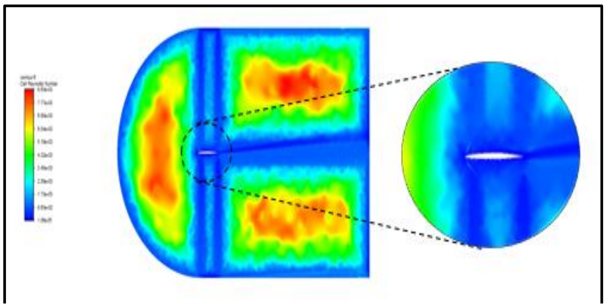

Fig 11. Dynamic Pressure distribution over NACA 66-206 airfoil at Mach 2.

Table 3. Findings from the flow analysis.

S. No.	Parameter	Value
1	Lift Coefficient (Cl)	0.185
2	Drag Coefficient (Cd)	0.03
3	Lift Force	512.43 N
4	Drag Force	83.49 N

IV. CONCLUSION

This paper has mainly focused on finding the lift and drag coefficients obtained for a NACA 66- 206 airfoil at a supersonic speed of Mach 2 when the flow angle was fixed at 5°. Propagation of shock waves was studied. It is analyzed that the flow velocity greatly influences the characteristics of the boundary layer formed.

Parameters like lift and drag forces, coefficients of lift and drag after analyzing are summarized in Table 3. As the free stream velocity increases along the surface of the airfoil, critical M is reached at some point after which shock was formed along the trailing edge.

A detached bow shock is formed at the leading edge. It is also examined that the sharp leading edge on the supersonic airfoil has efficiently reduced high pressure and drag along the leading edge by formation of an oblique shock. When looking at the trailing edge, it's been found that it has expansion fans which have resulted in accelerating the flow to high supersonic speeds.

V. ACKNOWLEDGEMENT

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