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Analysis of Shock over NACA 66-206 at Supersonic Regime

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Abstract

Aerofoil is a primary element to be designed during the initial phase of creating any new aircraft. It is the component that forms the cross-section of the wing. The wing is used to produce lift force that balances the weight which is acting downwards. The lift force is created due to pressure difference over the top and bottom surface which is caused due to velocity variation. It took about half a century to break the sound barrier since man started to fly. The difficulties in breaking the sound barrier was the infinite drag experienced by the aircraft in the transonic range and inability to sustain lift by the aircraft wing at higher Mach numbers. A supersonic flight require wing capable of maintaining lift at high Mach numbers, induce oblique shock and bow shock at optimum Mach numbers. Critical speed is the velocity at which there is sudden loss in Mach number and Lift and it is a function of shape of the aerofoil, angle of attack and thickness of the aerofoil. In this paper, a supersonic aerofoil NACA 66-206 is designed and analyzed using CFD to study the shock and aerodynamic properties at different Mach numbers(1.2, 1.5, 2.0, 2.5, 3.0). The objective of this paper is to design a supersonic aerofoil with minimal wave drag at high Mach numbers and sustain lift.

Keywords: Supersonic aerofoil; Shock formation; Viscous flow; CFD.

1. Introduction

An aerofoil is a shape of a wing or blade that causes the aerodynamic forces. Forces acting in a plane are Lift, Drag, Thrust and Weight, of which aerofoil is responsible for Lift and Drag. Lift acts upwards perpendicularly to the direction of forward motion and Drag acts parallel along the direction of flow. The characteristic of an aerofoil is that to attain maximum lift and reduce drag. Even though there is a considerable amount of increase in drag with increase in lift i.e. (0.2L ~ 0.02D). Our aim is to design a supersonic aerofoil with reduced wave drag and capable of sustaining lift at supersonic flying conditions. An supersonic aerofoil (NACA 66-206) was designed using 'GAMBIT' and analyzed at supersonic Mach numbers using 'FLUENT'.

2. Aerofoil Design

The supersonic speed aerofoil is a cross section geometry designed to generate lift atsupersonic speeds. Such aerofoils are explicitly designed for aircraftsthat need to operate consistently in the supersonic regime. Supersonic aerofoil is a thin section of either angled planes or of opposed arcs, having sharp leading and trailing edges so as to prevent formation of detached bow shock. In a typical aerofoil, at some point along the aerofoil, a shock is generated when the local velocity of flow reaches Mach 1 which increases the pressure coefficient to the critical value. The aerofoil chosen for this analysis was a NACA 66-206 aerofoil, a 6% thick aerofoil, as seen fig.1.

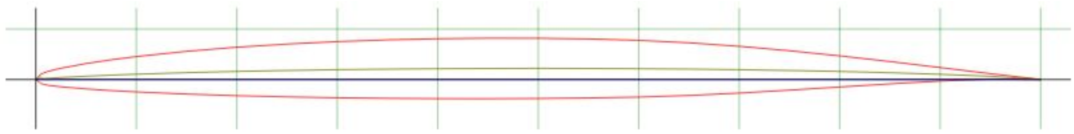

Figure 1: NACA 66-206.

3. Analysis of the Aerofoil

The NACA 66-206 aerofoil was plotted in GAMBIT with its co-ordinates and was Tri-Meshed for improved accuracy and then imported for analysis, the NACA 66-206 was examined at supersonic Mach numbers. Analysis was carried out for different Mach numbers from 1.2 to 3.0 and the resulting pressure and Mach number plotting was done. Fluent is based on iteration and the model is iterated till converging results are obtained. The largest difference was for the pressure drag force which had a relative change of 1.3%. L/D. The following graph plots shows the variation in Mach number along the surface of aerofoil. Two main differences occur with the sharp leading edge compared to roundedleading edge that would change the performance of the aerofoil. With the sharp leadingedge, less of the aerofoil chord length was cut away. The greater chord length provides alarger lifting surface during supersonic flight. However, a rounded leading edge prevents flow separation. From the analysisother performanceslike total pressure, velocity magnitude etc. can also be obtained

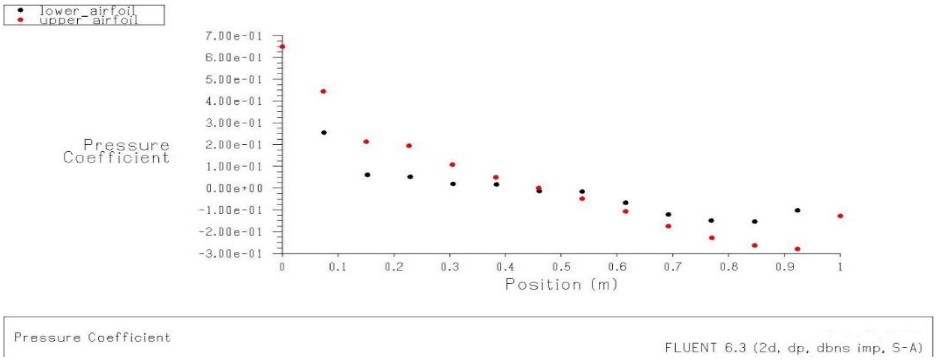

Figure 2: Pressure coefficient of NACA 66-206 aerofoil at M= 1.2.

4. Results and Discussion

The iterations of the solver used laminar flow conditions. Even though the flow can be considered fully turbulent, since the Reynolds number is 9.8 x 10^6, the laminar flow solver was faster than the turbulent equations. The flow was solved for 250 iterations and the results wereconverged. As free stream velocity increases along the aerofoil, at some point the critical Mach number is achieved. After critical Mach number is achieved, shock is produced at the trailing edge with a bow shock detached on the leading edge. To avoid boundary flow separation, speed of the aircraft has to be increased rapidly at Mach 1, thus making the shock wave converging at the trailing edge as the velocity increases. At some time the bow shock is eliminated by the high velocity experienced at the leading edge. These variations are shown in the figures 3,4,5.

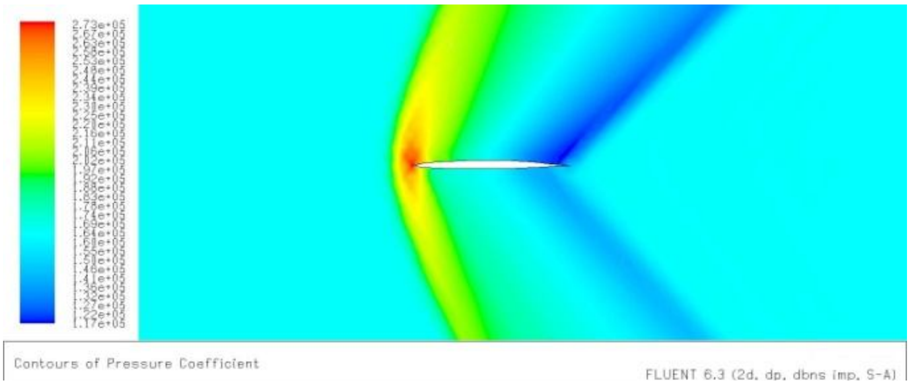

Figure 3.1: Pressure Variation over NACA 66-206 aerofoil at M=1.2.

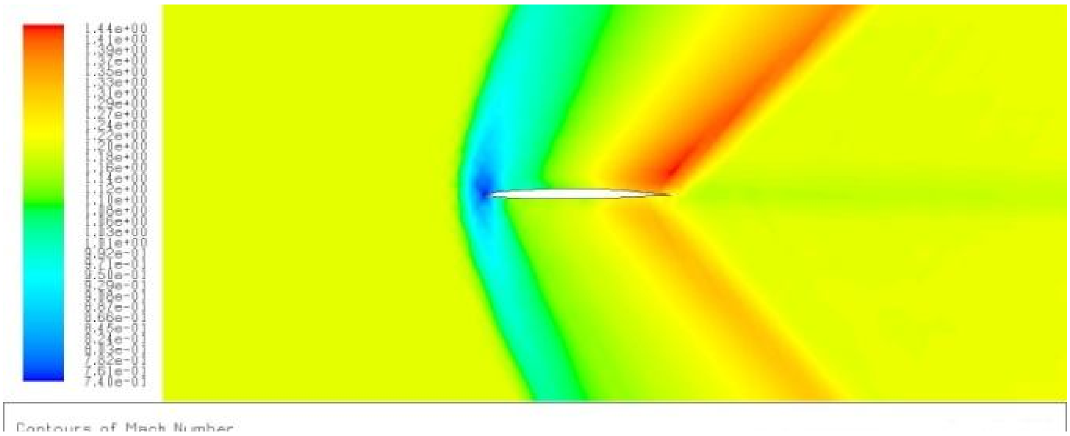

Figure 3.2: Mach number Variation over NACA 66-206 aerofoil at M=1.2.

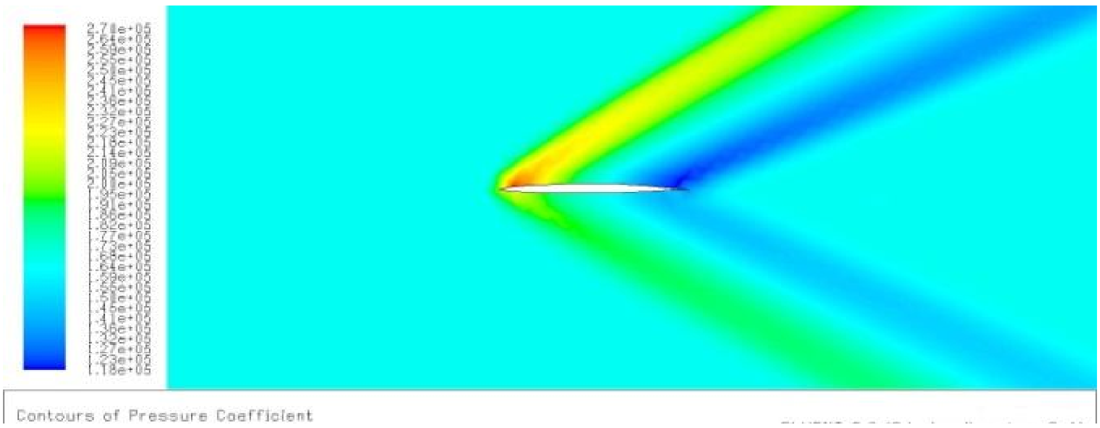

Figure 4.1 Pressure Variation over NACA 66-206 aerofoil at M=2.0

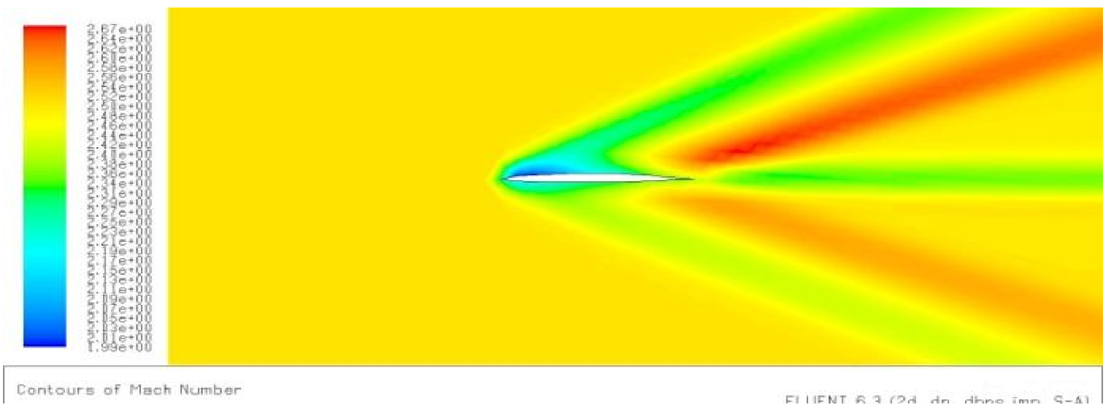

Figure 4.2: Mach number Variation over NACA 66-206 aerofoil at M=2.0.

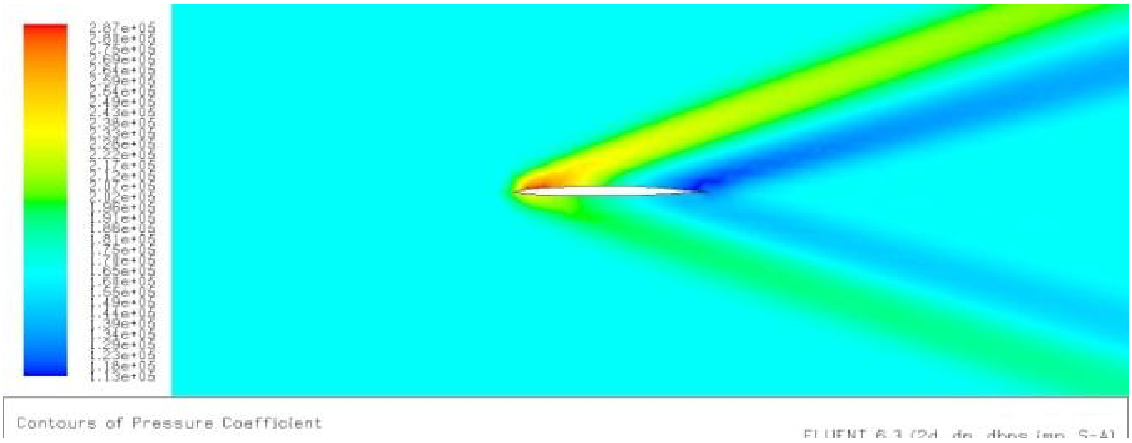

Figure 5.1: Pressure Variation over NACA 66-206 aerofoil at M=3.0.

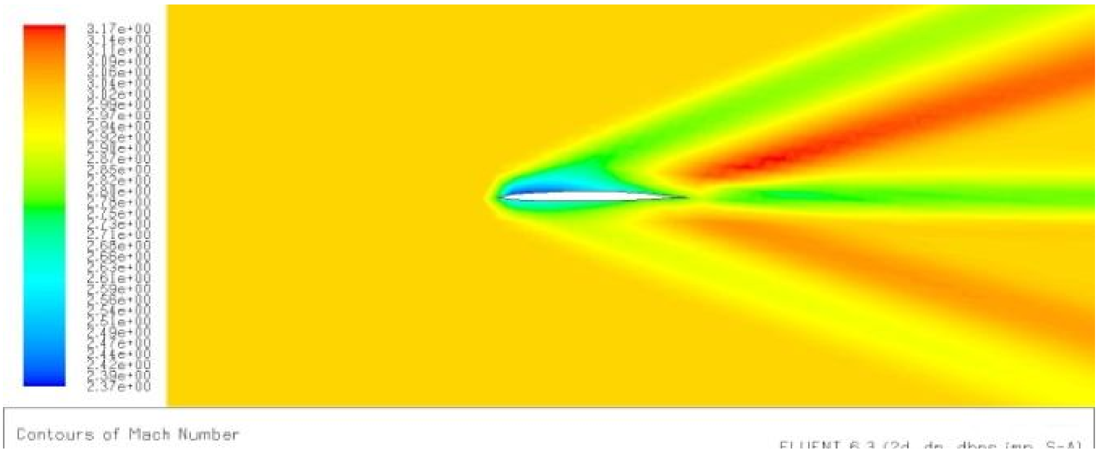

Figure 5.2: Mach number Variation over NACA 66-206aerofoil at M=3.0.

The Pressure Coefficient is a relative pressures present at each point in a flow field. At subsonic speedsthe flow around the aerofoil is not disturbed as that of the supersonic speeds. In subsonic flow, the pressure coefficient in the flow field will be near to the atmospheric pressure. Pressure increases drastically after the shock.

5. Conclusion

At supersonic speeds there is a rapid fall in Lift C_L coefficient and rapid rise in drag coefficient C_D which makes supersonic flight complex. Here a supersonic areofoil is analyzed at various mach numbers and the propagation of shock waves with Mach number is studied. The leading edge of *NACA 66-206* is at subsonic velocity due to oblique shock. The sharp leading edge has significantly reduced the drag and high pressure at the leading edge by attaching oblique shock. The trailing edge has expansion fans that has accelerated the flow to higher supersonic velocities. The same aerofoil shows poor performance like high stall velocity and handling problems in subsonic speeds.

Reference

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