AE3030

ANSYS Bonus Project

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Introduction

The aim of this project was to gain a familiarity with ANSYS and use CFD to generate a transonic flow around a symmetric airfoil. The contour plots generated for the NACA0012 airfoil at a small angle of attack showed that at a freestream velocity of Mach 0.7, velocity increased to Mach 1 over the top of the airfoil but did not show a shock wave. The static pressure contour plot showed a very low pressure in the area of sonic flow. The temperature contour plot showed a low temperature over the area of sonic flow. Stagnation points occurred where predicted and the boundary layer was observed to be attached throughout. When experimental data for Pressure coefficient was compared with the modeling, ANSYS proved to be an accurate modeling tool for transonic flow.

Methodology

I used ANSYS 2021b, Student version to generate the plots below. ANSYS is a CFD program that utilizes governing equations for conservation of mass, energy, momentum, and species to solve for the forces on 3-dimensional fluid control volumes. How the 3-dimenional flow is discretized is user-defined and depends on application. For this mesh, mesh and airfoil data was already included. Since I have limited understanding of ANSYS, I followed a tutorial provided by ANSYS that walked through the steps involved in generating a transonic plot.

Results and Discussion

Pressure Coefficient

Figure 1 shows that the highest pressure for both the lower and upper surface occurs at the leading edge of the airfoil.

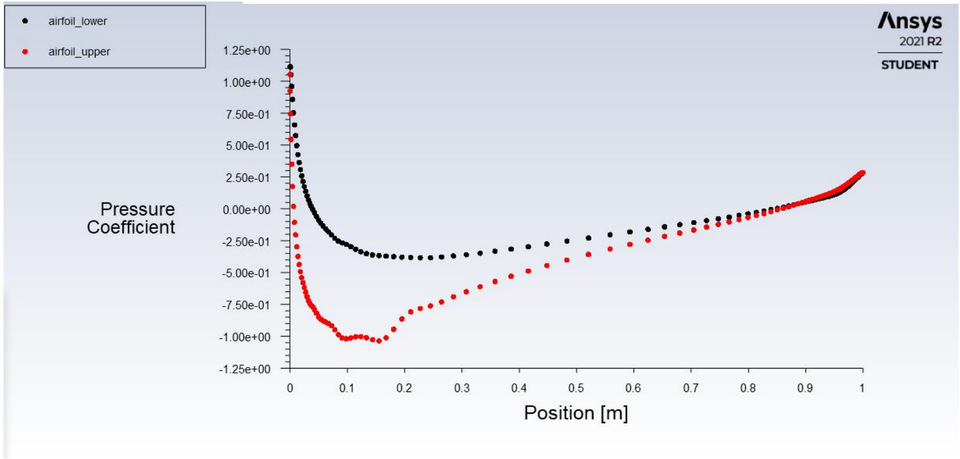

Figure 1: C_p vs. chord over at the surface of the airfoil

This is a stagnation point, where air halts due to the presence of the airfoil itself. The lowest pressure coefficient occurs on the upper surface, at 10% of the airfoil. This known as the “suction peak” and is a result of the air speeding up over the top of the airfoil. The highest pressure-differential also occurs at this mark. Pressure on both surfaces lowers consistently towards the trailing edge, where both upper and lower surfaces have the same pressure. This is consistent with the idea that pressure at the trailing edge must meet or be very close to the pressure that is in the surrounding air, since discontinuities do not occur in nature.

The following plot in Figure 2 shows that the ANSYS model very closely matches with experimental data [2].

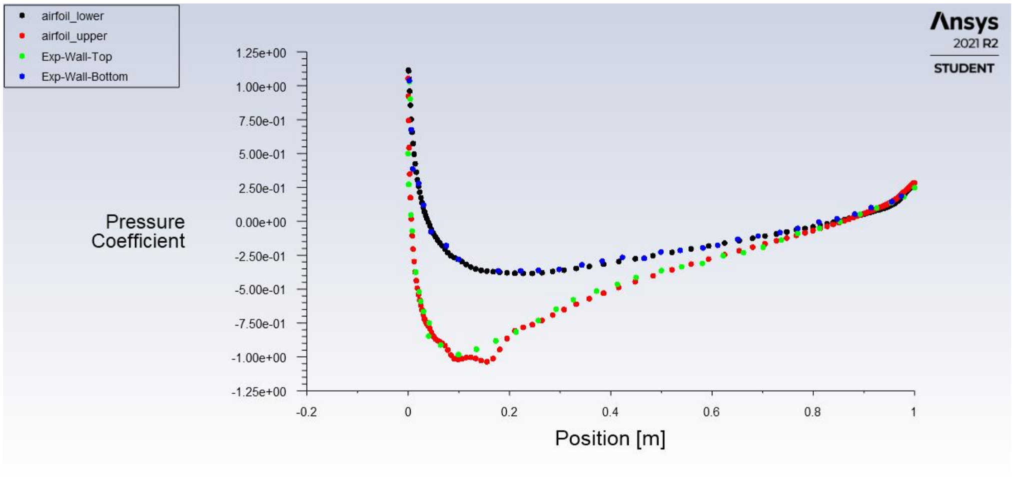

Figure 2: C_p vs. chord position ANSYS simulation (red and blue) and Experimental Data (green and black)

Skin Friction

The highest skin friction coefficient (Figure 3) occurs at the leading edge and decreases along the length of both upper and lower surfaces of the chord.

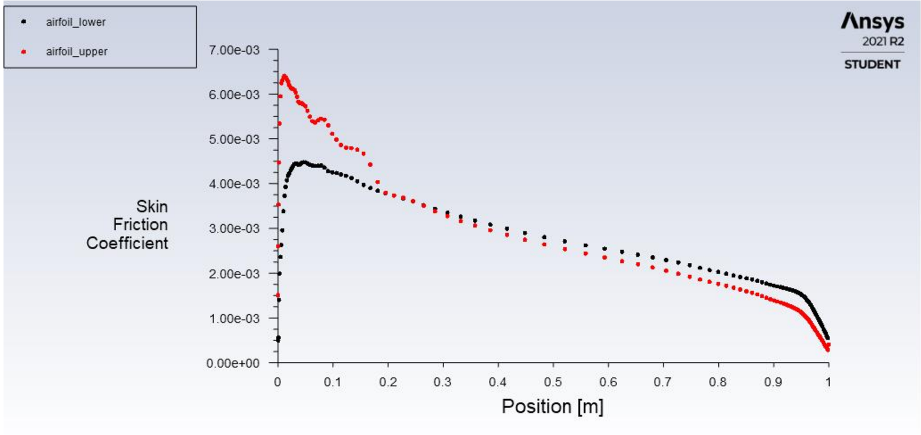

Figure 3: Skin friction coefficient vs. chord position

At and around the leading edge, the air is hitting the airfoil at an angle that is either normal to or close to normal to the airfoil. Because the angle is steeper, there is more kinetic energy on the surface close to the leading edge, which increases the skin friction.

Mach number

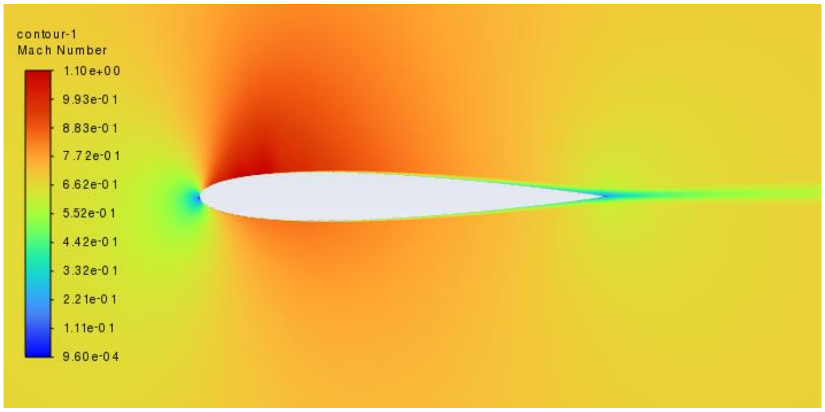

Figure 4: Contour plot of Mach number over the NACA0012 at freestream Mach number of 0.7

Figure 4 shows that there is a very low velocity at the leading edge of the airfoil. This corroborates that there is a stagnation point at that spot. Air must halt due to the presence of the airfoil interrupting flow.

Mach number increases to Mach 1 very quickly over the upper surface of the airfoil. The velocity of the airflow increases around the upper surface even though the airfoil is symmetric because there is a non-zero angle of attack. The theory behind this increase in velocity is that the air surrounding the airfoil needs to meet the Kutta condition at the trailing edge: that is, the flow exiting the upper and lower surfaces of the airfoil must be at the same velocity (magnitude and direction) at the end of the airfoil. Since there is a non-zero angle of attack, the air at the upper surface has further to travel, and so travels faster along the upper surface.

The Kutta condition also explains the low velocity at the trailing edge of the airfoil. Because the air is traveling a different direction at the upper surface from the lower surface, in order for the velocities of both airflows to agree at a single point (physically necessary, as the Kutta Condition describes), the velocity must be zero.

Since there is a stagnation point at the trailing edge, it can also be surmised that there has been no separation of boundary layer. This may be a result of a high level of kinetic energy from the high velocity in the freestream causing the boundary layer to change to a turbulent flow, allowing the boundary layer to stay on the airfoil throughout.

Though the airflow along the top of the airfoil reaches Mach 1, there does not appear to be any shockwave.

Pressure

There is a very low pressure zone along the top surface of the airfoil (Figure 5) that corresponds to the high velocity zone (Figure 4):

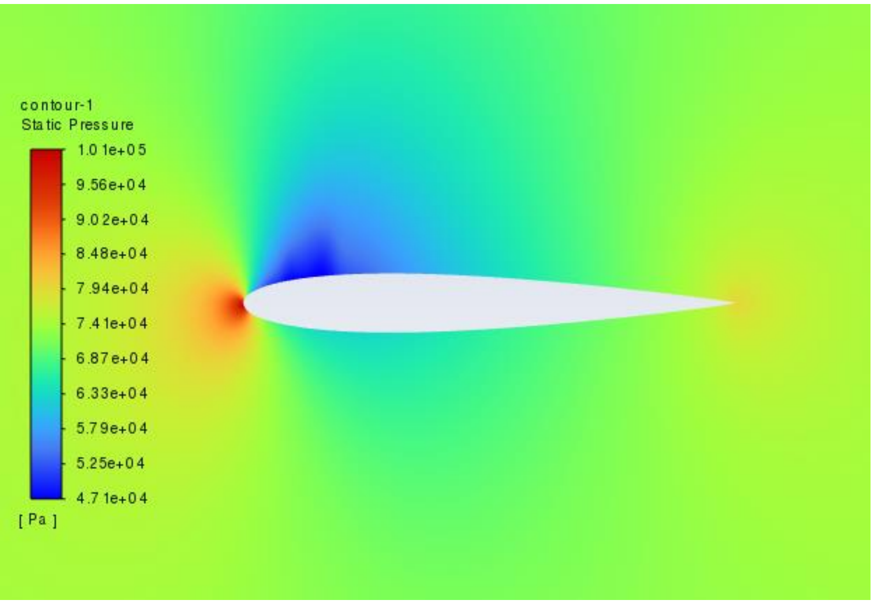

Figure 5: Contour plot of Static Pressure over the NACA0012 at freestream Mach number of 0.7

Normally, one would associate a low-pressure zone with high velocity: faster traveling air decreases the pressure according to Bernoulli’s equation. However, Bernoulli’s equation assumes an incompressible situation, and that is not the case with transonic flow. We can analyze this using compressible flow equations:

$$\frac{P}{P_0} = \left(1 + \frac{k-1}{2}\mathrm{Ma}^2\right)^{-k/(k-1)} \qquad (1)$$

From this equation (most likely used by ANSYS), Pressure at Mach 1 is about 52% of the pressure of the freestream.

The pressure differential between the upper and lower surfaces generates the lifting force. Since the pressure differential is quite large, there must be a large amount of lift generated at this Mach number.

The contour plot also clearly shows the stagnation areas at the leading edge and trailing edge in red. As the velocity of the freestream slows or stops, pressure is higher.

Temperature

Figure 6 shows a plot of temperature: there us a higher temperature area where stagnation points occur, and a lower temperature area where the velocity increases and pressure decreases.

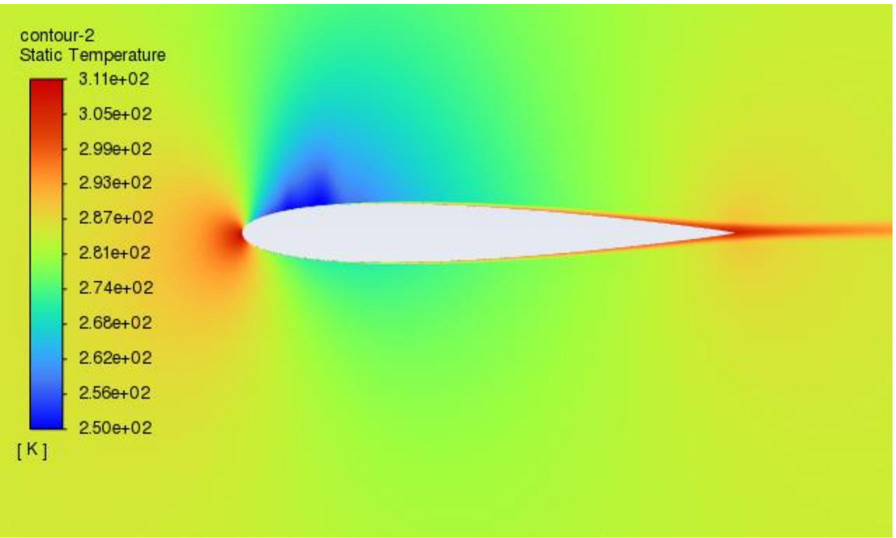

Figure 6: Contour plot of Static Temperature over the NACA0012 at freestream Mach number of 0.7

The temperature spike at the leading edge corresponds to the skin friction coefficient spike in Figure 3. This may be for the same reasons: air is bombarding the airfoil and running into itself, causing particles to collide and heat up.

The higher temperature that is close to the airfoil near the trailing edge, however, so it is not related to skin friction. I believe that the higher temperature may be related to the boundary layer. The boundary layer around the high velocity zone does not have a high temperature, but then there is a higher temperature than the freestream around the last 60% of the airfoil. This may be due to the adverse pressure gradient that we can see in Figure 5 (a slightly orange area at the trailing edge) causing a situation where the boundary layer is disturbed and heated up as a result.

Conclusions

Commercial airplanes often cruise at transonic speeds. Judging by the favorable pressure conditions in Figure 6, it seems like transonic speeds can generate a lot of lift. There also may not be much wave drag associated with this Mach number and this airfoil, because there does not appear to be a shock wave.

The coefficient of pressure matched very closely with experimental data, proving that the equations ANSYS was using were correct models for transonic conditions over this type of airfoil.

References

[1] T.J. Coakley, "Numerical Simulation of Viscous Transonic Airfoil Flows," NASA Ames Research Center, AIAA-87-0416, 1987

[2] C.D. Harris, "Two-Dimensional Aerodynamic Characteristics of the NACA 0012 Airfoil in the Langley 8-foot Transonic Pressure Tunnel," NASA Ames Research Center, NASA TM 81927, 1981