<u>Computational Aerodynamics Initiative #3</u>

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The purpose of this project was to numerically study the effects of subsonic freestream Mach number on the coefficient of lift (C_l) and coefficient of drag (C_d) of an airfoil (NACA 2412), especially as the air is locally accelerated beyond the speed of sound. These local Mach numbers and stagnation pressures were then used to generate a contour map around the airfoil to analyze how the flow behaves as the Mach numbers varied. This was achieved through the use of analytical methods in the form of an Euler Solver.

The first observation was that as Mach number increases, the coefficient of lift, C_l, increases. This is expected because the effect of compressibility is associated with an increase in the C_l. This effect is primarily due to a higher suction peak resulting from the energy stored in density being used to accelerate the flow.

Another observation was the increase in the coefficient of drag, C_d. This was expected as the air creates a normal shock to transition from supersonic to subsonic smoothly, which creates wave drag. The normal shocks can be seen on the top of the airfoil in Figure 3a, and on both the top and bottom of the airfoil in Figure 2a. The C_d plot in Figure 1b shows the exponential increase in drag as the airfoil developed this additional normal shock on its underside.

The formation of these normal shocks can be seen in the progressions from Figure 2a to Figure 5a, which shows how the shocks decrease in magnitude before disappearing as the free-stream Mach number decreases. Figures 2b and 3b confirm that these are, in fact, normal shocks because the stagnation pressure decreased. More importantly is the observation that as Mach number increases, the normal shock begins further along the airfoil, which is expected since the air will have more energy and can resist forming a shock until much later.

It's also worth mentioning the limitations of this numerical model. First the fact that the C_d plot in Figure 1b goes to zero or even negative numbers shows that this model lacks viscous profile drag, which normally provide a baseline drag coefficient.

These limitations necessitated the need to validate the accuracy of this numerical solver. This was done by comparing the computational data of the pressure, stagnation pressure, and density ratios to those found in the normal shock tables. Table 1 and Table 2 compare these values, and we see that the post-shock Mach number as well as the pressure, stagnation pressure, and density ratios are all very close to what we expect.

In conclusion, we saw how compressibility affects the properties of an airfoil such as its lift and drag coefficients. We also saw how the properties of air, such as stagnation pressure is constant in isentropic flow and how it changes across the shock.

Table 1: Mach 0.82 Data Comparison

	Euler Solver	Normal Shock Tables
Ma_1 (Before Shock)	1.3410	
Ma_2 (After Shock)	0.766	0.766*
P_2/P_1	1.908	1.931*
ρ_2/ρ_1	1.595	1.587*
$P_{0,2}/P_{0,1}$	0.964	0.972*

Table 2: Mach 0.75 Data Comparison

	Euler Solver	Normal Shock Tables
Ma_1 (Before Shock)	1.264	
Ma_2 (After Shock)	0.825	0.868*
P_2/P_1	1.652	1.698*
ρ_2/ρ_1	1.445	1.453*
$P_{0,2}/P_{0,1}$	0.992	0.985*

* Note that these values were approximated using linear interpolation since the normal shock tables don't contain the data for the given incoming Mach numbers

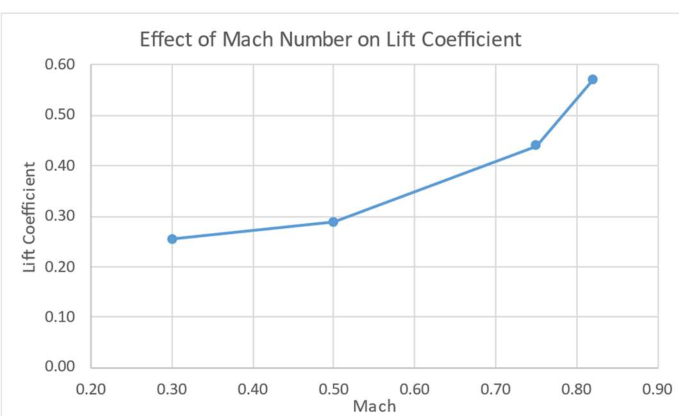

Figure 2a - Lift Coefficient vs freestream Mach number

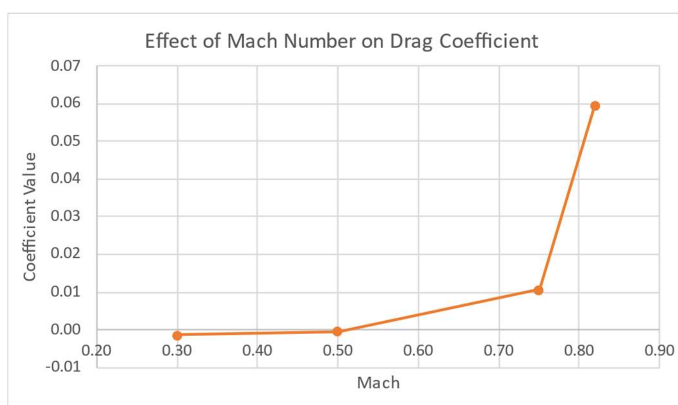

Figure 2b - Drag Coefficient vs freestream Mach number

Contour Plot Data

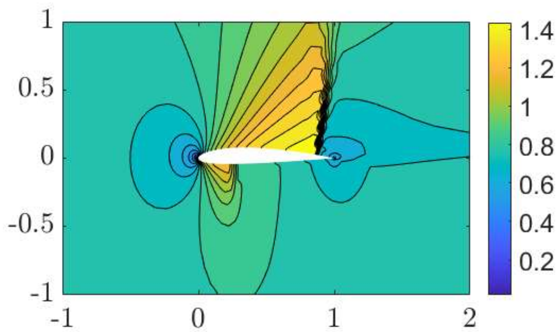

Figure 2a - Mach Contour at Ma = 0.82

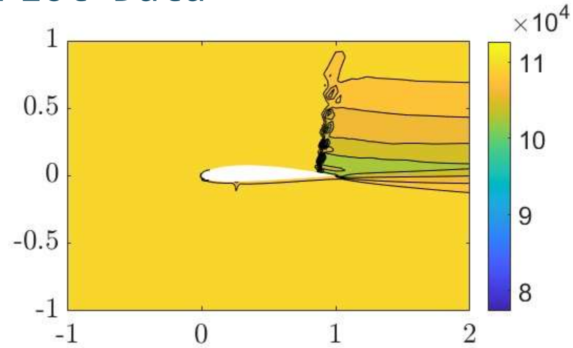

Figure 2b - Total Pressure Contour at Ma = 0.82

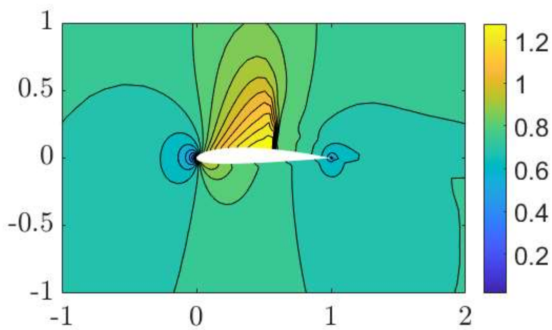

Figure 3a - Mach Contour at Ma = 0.75

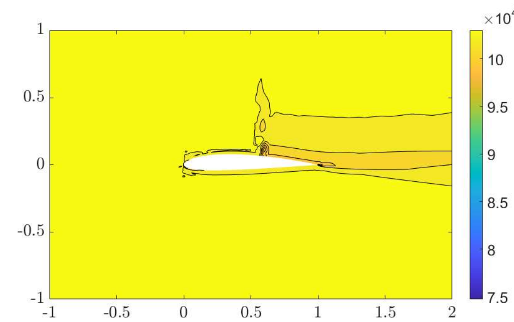

Figure 3b - Total Pressure Contour at Ma = 0.75

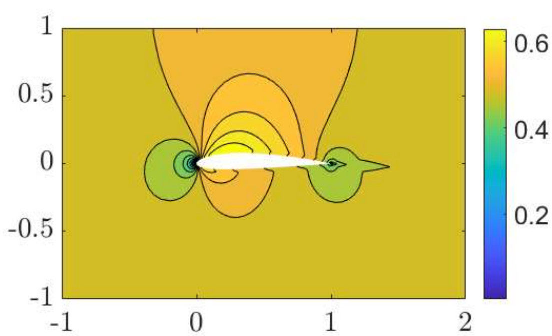

Figure 4a - Mach Contour at Ma = 0.50

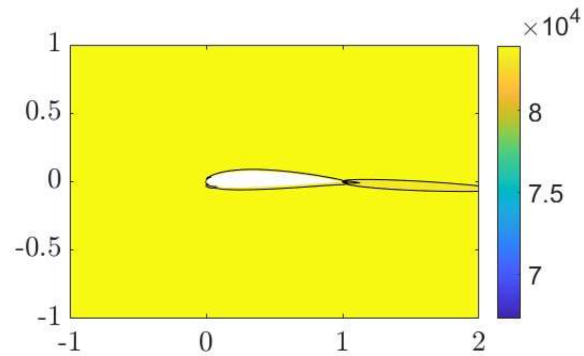

Figure 4b - Total Pressure Contour at Ma = 0.50

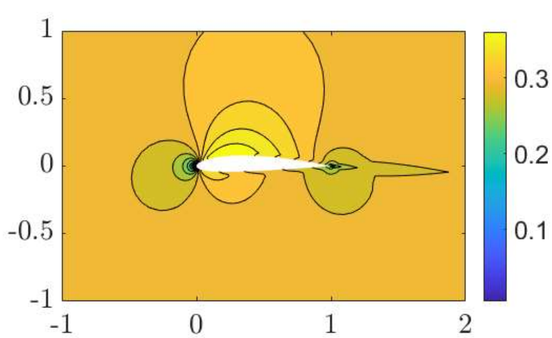

Figure 5a - Mach Contour at Ma = 0.30

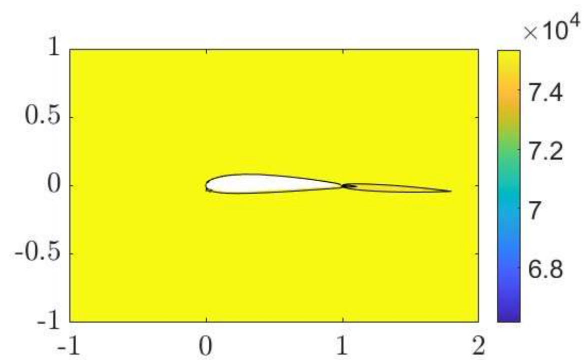

Figure 5b - Total Pressure Contour at Ma = 0.30