Task 5 - Compressor Airfoil CFD

Meisel, Carlos Juarez, Albert Quintero, Osvaldo

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1 Overview

This report contains the results of the CFD analysis performed on the compressor airfoil our team has chosen from Task 3.

2 Flow Conditions

As defined in the *T.L. User's Manual*, the flow through a cascade is defined by two parameters:

- The inlet flow Mach number M_{in}
- inlet flow angle α_{in}

We have two non-dimensional parameters that define the flow through the cascade:

$$\rho_{in} = 1 \tag{1}$$

$$V_{in} = 1 \tag{2}$$

Where, ρ_{in} is the static density at the inlet and V_{in} is the velocity at the inlet. From here we can calculate the Mach number at the inlet:

$$M_{in} = \frac{V_{in}}{c_{in}} \tag{3}$$

Since V_{in} is a non-dimensional parameter, we need to non-dimensionalize the speed of sound c_{in}. We can do this by taking a look at how V_{in} is defined:

$$V_{in} = \frac{157.5}{157.5} \tag{4}$$

Applying the same logic to the speed of sound we get:

$$c_{in} = \frac{\sqrt{\gamma RT}}{157.5} \tag{5}$$

Where γ is the ratio of specific heats, R is the gas constant and T is the temperature. Since this all takes place in the compressor, the working fluid is air, meaning that $\gamma = 1.4$ and $R = 287.058 \frac{J}{kgK}$.

Resulting in our Mach number to be:

$$M_{in} = 0.462823 \tag{6}$$

We are then able to calculate the static pressure at the inlet:

$$p_{in} = \frac{1}{\gamma M_{in}^2} \tag{7}$$

Readers may recall the standard isentropic relations for pressure and density from AERO 201:

$$p_0 = p\left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}} \tag{8}$$

$$\rho_0 = \rho\left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{1}{\gamma - 1}} \tag{9}$$

We can makew use of these relations to calculate the total pressure and density at the inlet:

$$p_{0_{in}} = p_{in}\left(1 + \frac{\gamma - 1}{2} M_{in}^2\right)^{\frac{\gamma}{\gamma - 1}} \tag{10}$$

$$\rho_{0_{in}} = \rho_{in}\left(1 + \frac{\gamma - 1}{2} M_{in}^2\right)^{\frac{1}{\gamma - 1}} \tag{11}$$

We calculated our total parameters to be:

$$p_{0_{in}} = 2.87924668 \tag{12}$$

$$\rho_{0_{in}} = 0.9004399 \tag{13}$$

We induced a flow angle of $\alpha_{in} = 55$ degrees. Alowing for us to calculate the:

$$\texttt{FLUX} = V_{in} \cos \alpha_{in} \tag{14}$$

$$\texttt{VTAN} = V_{in} \sin \alpha_{in} \tag{15}$$

Resulting in:

$$\begin{aligned}
\texttt{FLUX} &= 0.573576436 \\
\texttt{VTAN} &= 0.819152044 \\
\texttt{UINIT} &= 0.573576436 \\
\texttt{VINIT} &= 0.819152044
\end{aligned}$$

3 Computational Fluid Dynamics Figures

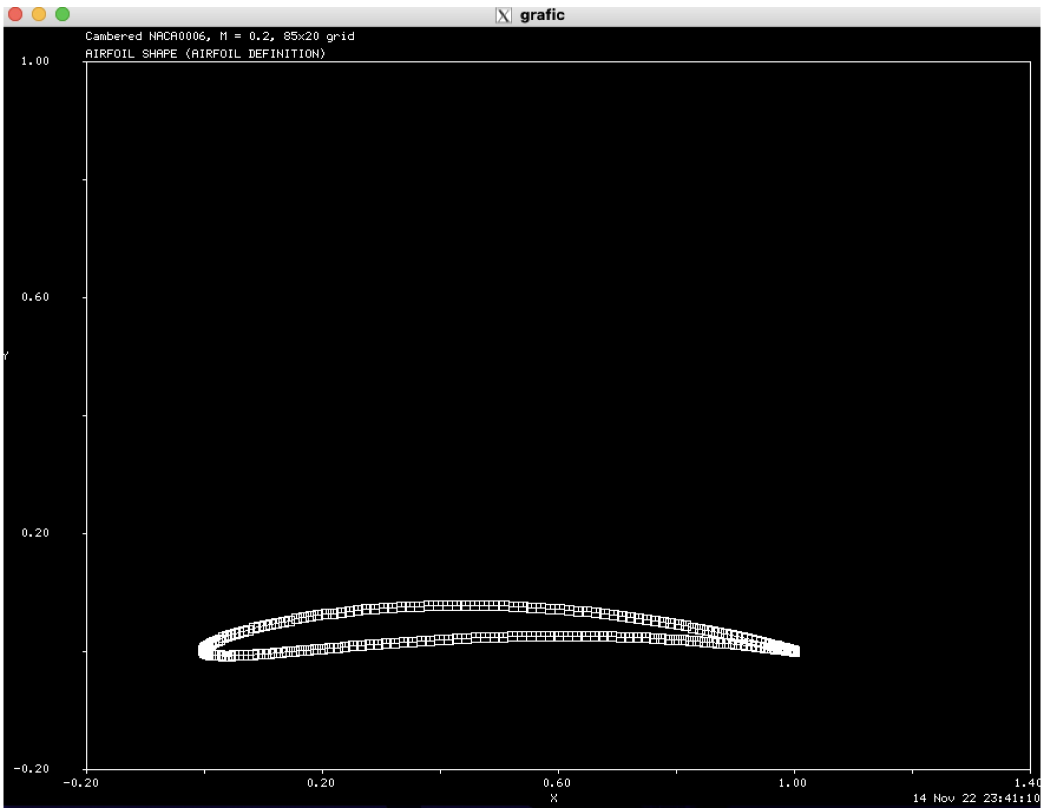

Figure 1: Plot of the airfoil used

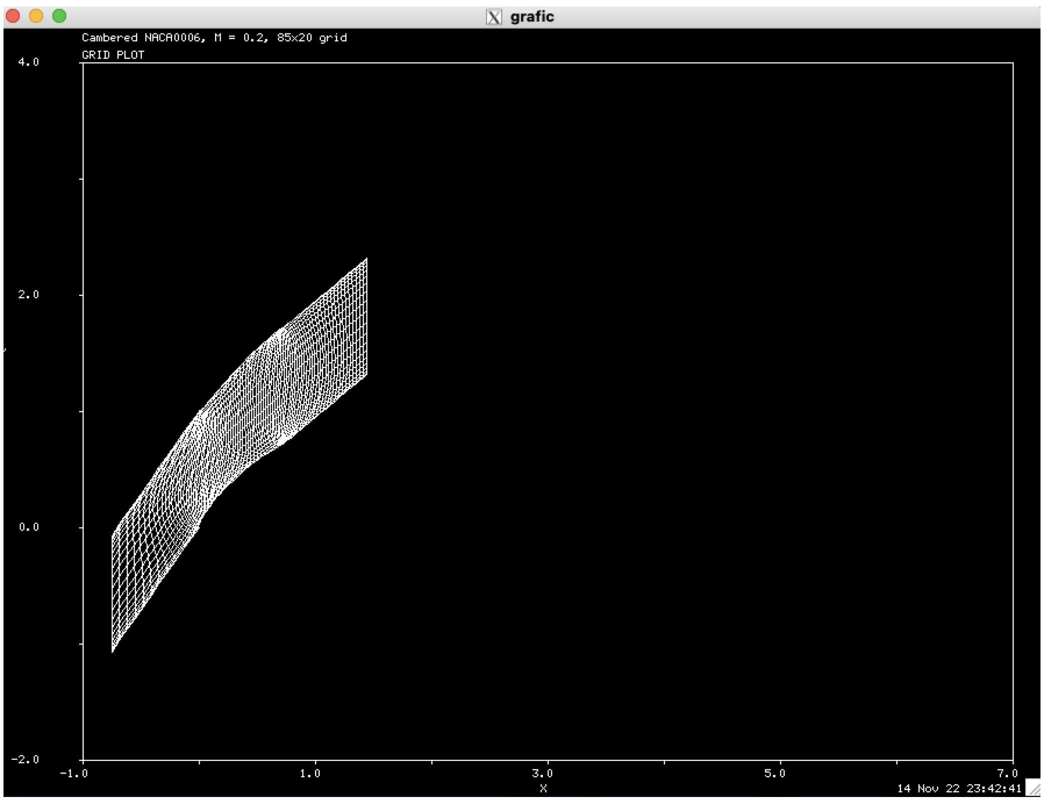

Figure 2: Plot of the Computational Grid used

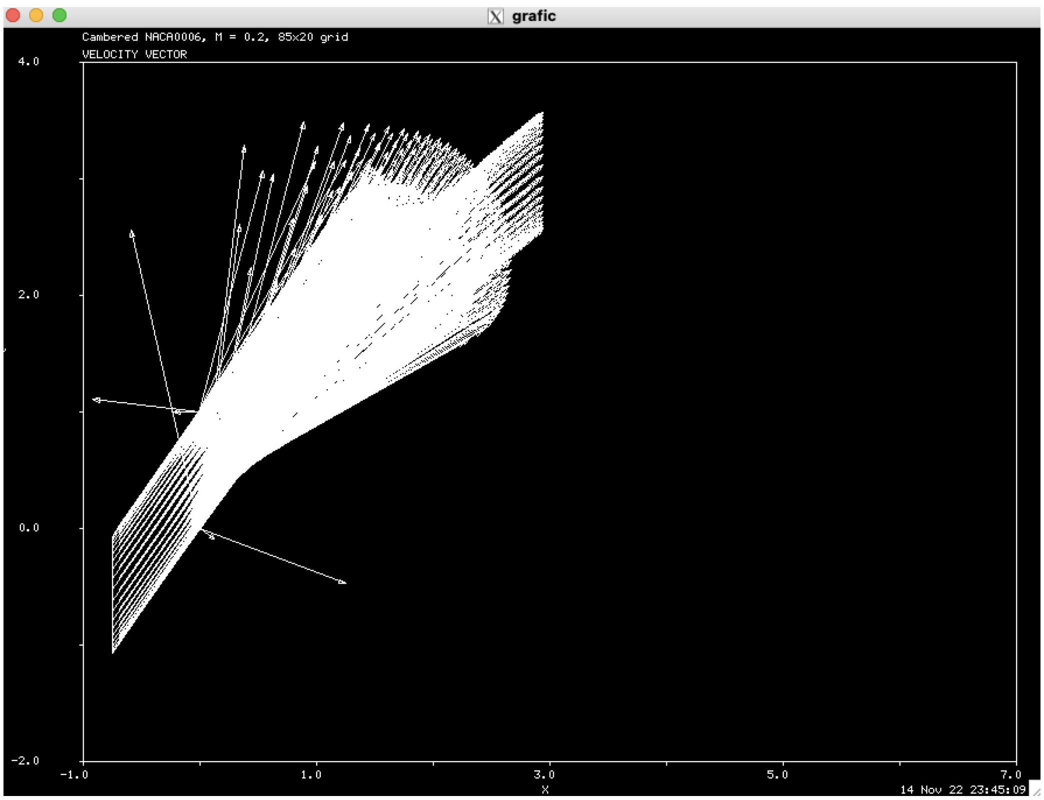

Figure 3: Plot the the Velocity Vectors

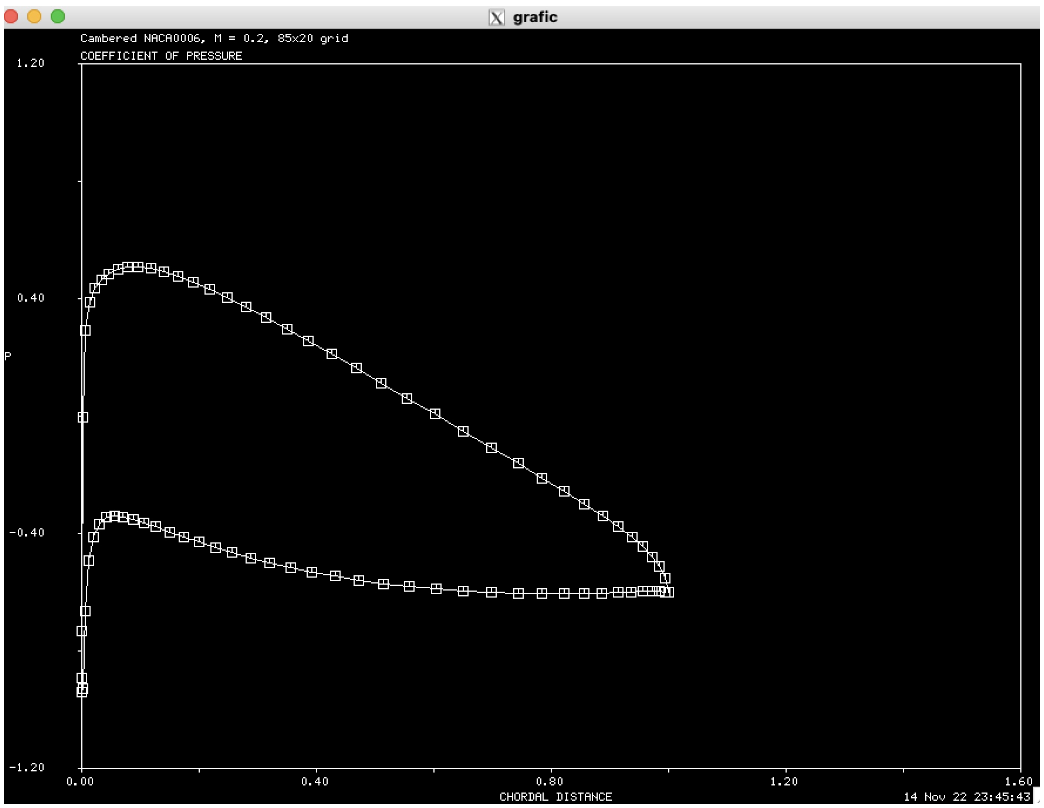

Figure 4: Plot of the Surface Pressure Coefficient

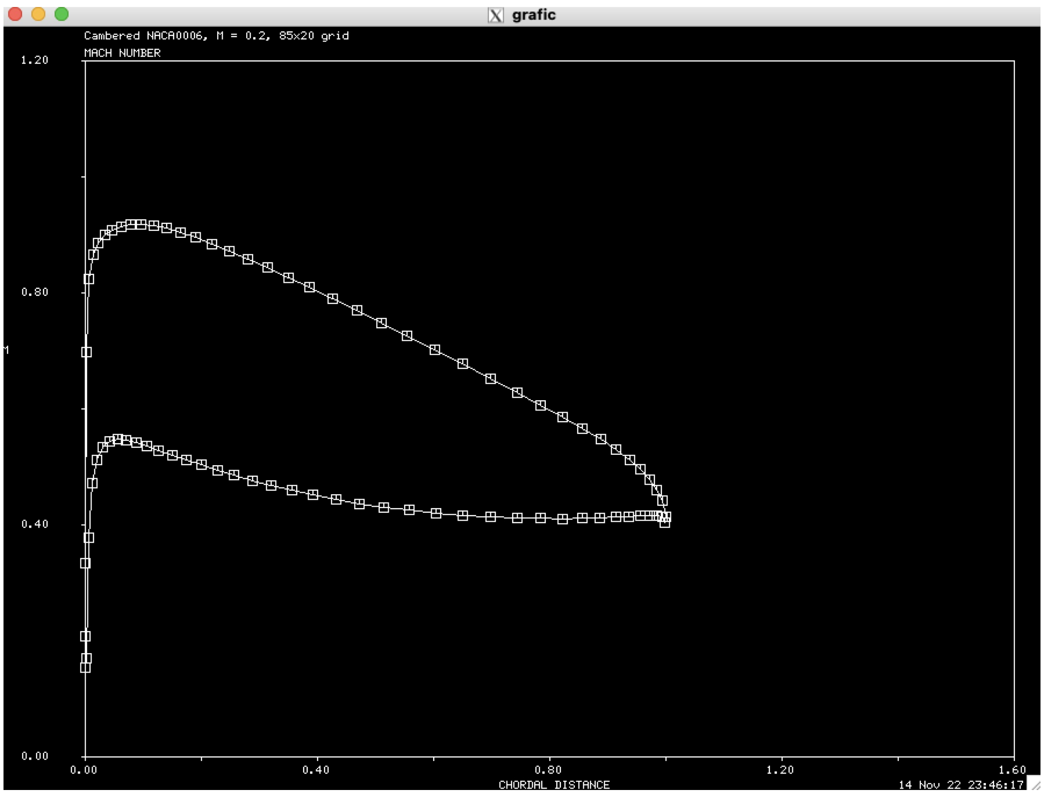

Figure 5: Plot of the Surface Mach Number