Integrated Supersonic Inlet and Diffuser Design for a Mach 2.3 Fighter-Class Aircraft

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Project Objective:

The project focused on designing and validating a two-dimensional external-internal compression inlet for a fighter-class vehicle operating at Mach 2.3 and an altitude of 11 km. The inlet geometry was designed using classical oblique shock theory to achieve staged compression, followed by internal compression into a subsonic throat. Shock calculations were performed in MATLAB to determine ramp deflection angles and expected shock behavior. Finally, the geometry was modeled in Fusion 360 and analyzed using ANSYS Fluent to evaluate flow physics, shock structure, and inlet performance.

A density-based compressible solver was used to determine shock-dominated flow behavior. Turbulence was modeled using the SST k-ω model, and total pressure recovery was found. These results were then validated against the analytical oblique shock predictions. Inlet performance was quantified primarily through total pressure recovery at an internal engine-face cross-section.

Tools used:
- MATLAB (compressible flow and shock calculations)
- Fusion 360 (2D inlet geometry)
- ANSYS Fluent (CFD Validation)

Analytical Design and Theory:

Shock angle and post-shock properties were computed using oblique shock relations and solved numerically in MATLAB by utilizing root-finding methods. Shock calculations were performed to determine ramp deflection angles, oblique shock angles, and expected post-shock fluid tendencies.

```
%the downstream mach after ramp 2 becomes the mach speed at the terminal area
mach3 = downstream_flow_mach2;

% Downstream Mach after normal shock
downstream_mach_3to4 = sqrt((1+((gamma-1)/2)*mach3^2)/(gamma*mach3^2-((gamma-1)/2)));

% Static pressure ratio across normal shock
static_pressure_ratio_3to4 = 1 + (2*gamma/(gamma+1))*(mach3^2 - 1);

% Total pressure ratio across normal shock
total_pressure_ratio_3to4 = static_pressure_ratio_3to4* ...
((1 + ((gamma-1)/2)*downstream_mach_3to4^2)/(1 + ((gamma-1)/2)*mach3^2))^(gamma/(gamma-1));
```
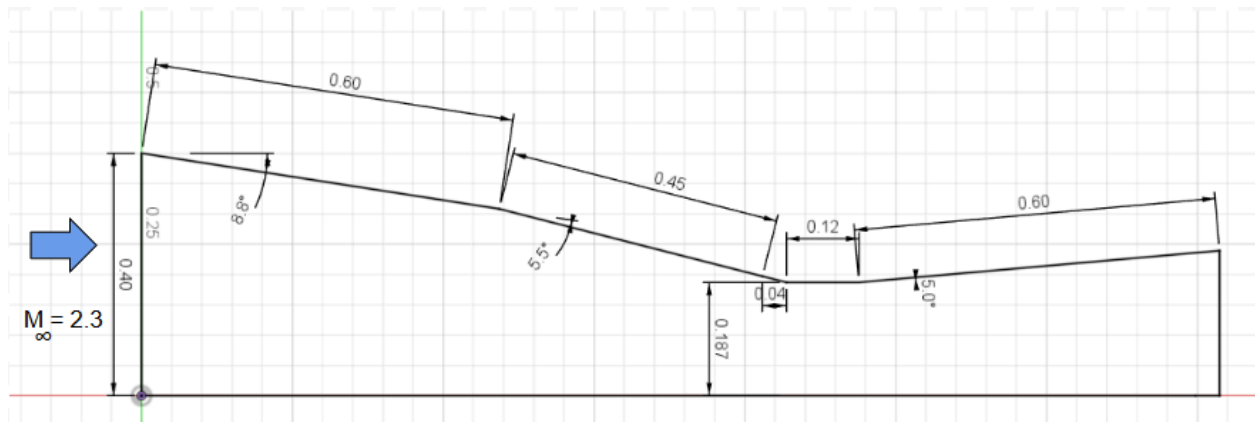
Table 1. Analytical Derived Inlet Performance Metrics at Mach 2.3

Quantity	Value
Freestream Mach	2.30
Ramp 1 Deflection Angle	8.8º
Ramp 1 Shock Angle	33.17º
Ramp 2 Deflection Angle	14.3º
Ramp 2 Shock Angle	35.54º
Total Pressure Recovery (Idealized)	0.8147

CAD - Inlet Geometry

2D inlet profile geometry was defined directly from oblique shock analysis. Ramp angles and throat location were selected to achieve a controlled compression prior to the terminal normal shock.

CFD - Flow Validation and Visualization

- Density-based solution, compressible flow
- Boundary conditions
  - Inlet: Pressure Far-field
  - Outlet
- Total pressure recovery was evaluated using a mass-weighted average on an internal engine-face plane, located downstream of the internal shock system to avoid outlet boundary contamination
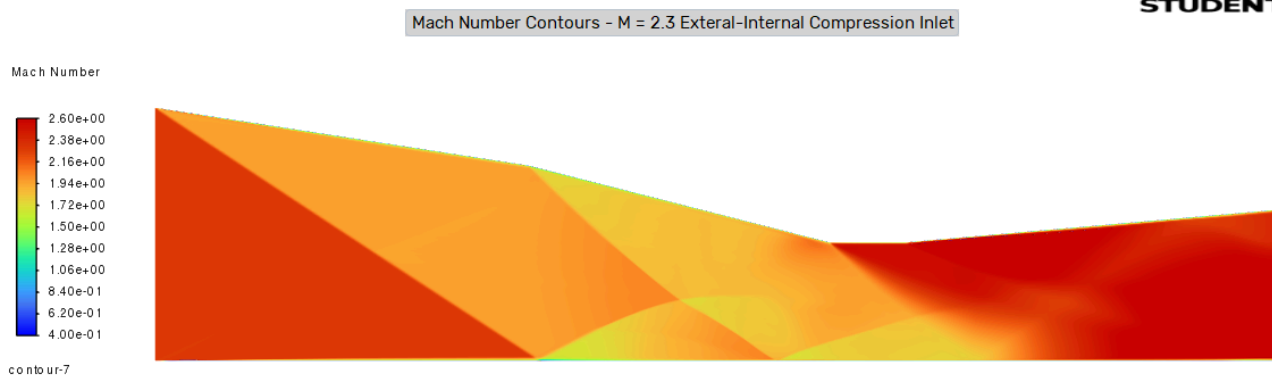
Figure 1: Mach Number Contour at Mach 2.3, showing staged external compression via oblique shocks followed by internal compression into a subsonic throat
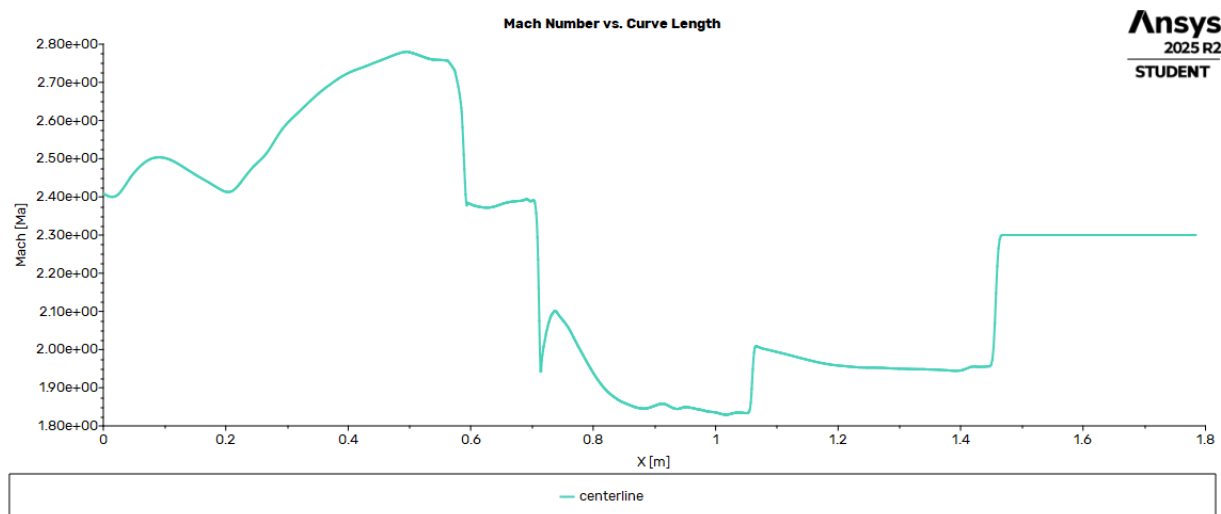

Figure 2: Centerline Mach number distribution along the inlet length, showing staged deceleration from freestream Mach number through external and internal compression
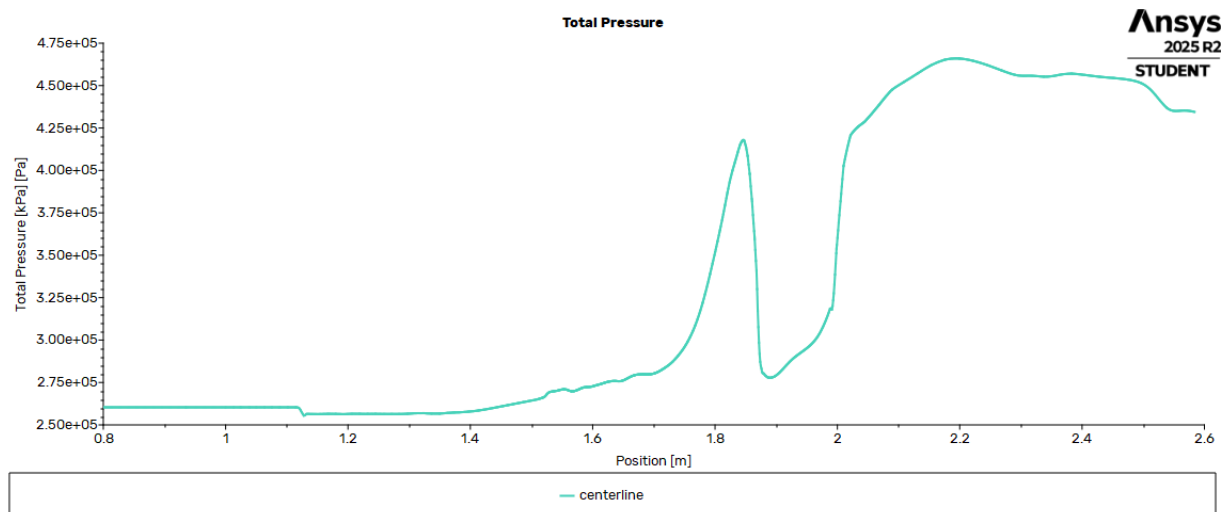

Figure 3: Mass-weighted total pressure evaluated at an internal engine face cross-section downstream of the shock system

Table 2. Comparison of Analytical and CFD-Predicted Performance Metrics at Mach 2.3

Quantity	Calculated Value	Simulated Value
Freestream Total Pressure	-	283 kPa
Engine-Face Total Pressure	-	256 kPa
Total Pressure Recovery	0.8147	0.90
Ramp 1 Shock Angle	33.17º	33.7º

Results and Validation

The inlet performance simulated by compressible CFD showed a strong agreement with the analytical fluid dynamics calculations. The simulated Total Pressure Recovery was found to be **0.90**, compared to the theoretical value of **0.8147**. The higher recovery rate observed in the CFD simulated value can be attributed to the simulation being run in perfectly ideal conditions, as well as the absence of real-world conditions like surface roughness.

The measured shock angle of the first compression ramp from the CFD Mach Contour was approximately **33.7º**, which almost exactly matches the theoretical value of **33.17º**. These results have less than a 1% error, and this close agreement validates the inlet geometry and confirms that the dominant oblique shock is correctly modeled by the simulation. Overall, the results demonstrate the effective staged compression is reached by the inlet, and has an acceptable total pressure recovery when operating at Mach 2.3 Conditions.

Design Takeaway

This project highlighted the importance of finding performance metrics at physically meaningful locations. Early attempts to quantify data at incorrect locations led to misleading results, which eventually led to the need of an engine-face based measurement to ensure accurate measurements in supersonic inlet analysis.