

WLT-1738

By: Team Locked In

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1.0 Abstract

Team Locked in was tasked to create a Ramjet that creates a positive net thrust. This jet's performance was assessed in the ranges between Mach 2.75 and 3.25. Assumptions to simplify and ease calculations were provided in the problem statement and made when necessary. There are 6 parts to the ramjet: an inlet, diffuser, flameholder, combustor, converging section, and a nozzle. The inlet was designed with the focus that all the oblique shocks converge at one point below the cowl. Having the height of the inlet allowed for the sizes of the other components to be calculated. The combustor and the nozzle areas are approximately the same height as the inlet to reduce the overall drag of the Ramjet. The Ramjet ended up being 3.5 meters in length, .45 meters in height, and 43 kN of net thrust.

2.0 Introduction

A Ramjet is a jet engine that creates thrust by efficiently combusting fuel in a stream of supersonic airflow. These engines do not require rotating machinery like compressors or turbines which are found in traditional jet engines. Ramjets are designed to operate at high speeds where they can take advantage of the compression of air entering the engine. They solely rely on the forward speed of the aircraft to compress the incoming air. The compressed air is then mixed with fuel and ignited in a combustor section. This produces a high temperature and pressure exhaust stream. The expansion of this exhaust stream propels this aircraft forward as it then goes through an exit nozzle. It is important to note Ramjets are only efficient at high speeds because the compression of the incoming air is directly related to the speed of the aircraft therefore, at higher speeds the airflow into the engine may not be enough to enable combustion to take place. The absence of moving parts like compressors or turbines make ramjets simpler than traditional jet engines. However, they require the aircraft to be traveling at supersonic speeds to function properly. Our project is to design a ramjet capable of generating net positive thrust.

3.0 Problem Statement

The goal of this final project is to design a Ramjet engine that can produce a positive net thrust. Its theoretical performance will be assessed in the range between Mach 2.75 and 3.25. General assumptions can be made that the vehicle is flying at 55,000 feet and is 2D. The Ramjet is composed of an inlet, diffuser, flameholder, combustor, converging, and nozzle section that need to be individually designed.

4.0 Assumptions Provided

There are general assumptions provided by the problem statement upon creation of the Ramjet. Our fuel is H₂ with a lower heating value (LHV) of 120 MJ/kg. The introduction of fuel only contributes heat to the bulk flow therefore, the additional mass and associated changes with it are negligible. The pressure gradient is constant and the specific heat ratio is 1.4 throughout the Ramjet. External flow is inviscid; however, pressure drag from entrance and exit areas is applied. The flameholder has an arbitrary pressure drop and the

combustor's burn time can be calculated with the equations $\frac{\Delta P}{P} = 0.81\gamma M^2$ and $\tau = 325P^{-1.6}e^{-\frac{0.81T_0}{1000}}$ respectively.

5.0 Design Approach

The design began with the creation of the inlet. The turning angle of the inlet and Mach number determines the ratio of length to height, as the geometry must be such that the oblique shocks intersect at the same point below the cowl. With the height of the inlet locked in, the sizes of the other portions of the vehicle were scaled accordingly. Both the combustor section area and the nozzle are prescribed to be approximately the same height as the inlet. This reduces the overall drag of the vehicle as the lower exterior section can have very shallow incident angles with respect to the flow. Figure 1 displays the entire finalized engine geometry.

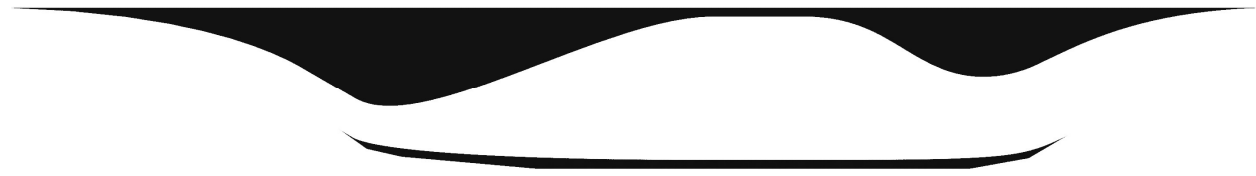


Figure 1: Engine Geometry

6.0 Section 1: Inlet Section Overview

The inlet of a ramjet serves to capture incoming air at supersonic speeds and direct it into the engine. Because of those high speeds, the air must be slowed down to subsonic speeds before entering the engine. The inlet does this by efficiently compressing incoming air at high speed without causing shockwaves that might disrupt the flow.

6.1 Inlet Section Assumptions

Assumptions were made for the inlet section to simplify and provide the team with a foundation when completing the problem. One assumption is that there was a terminating normal shock at the cowl which takes the flow from supersonic to subsonic at the last step of the inlet. This terminating shock also is assumed to generate the conditions of the flow going into the diffuser section.

The incoming flow properties were determined for flight at 55,000 feet above sea level (*U.S. Standard Atmosphere vs. Altitude*, 2003).

- $M_1 = [2.75, 3.25]$
- $P_1 = 9450 Pa$
- $T_1 = 216 K$
- $R_{Air} = 287 J/kg \cdot K$
- $\gamma = 1.4$

6.2 Inlet Design Approach

As mentioned previously, the inlet is designed such that all oblique shocks caused by inlet ramps intersect at a point ahead of the cowl. This is to avoid distorted flow from

entering the engine and invalidating the uniform flow assumption used in the diffuser. This design is completed at the maximum Mach in the range (3.25) so that when the flow is slower, the shocks extend further away from the cowl. The design is completed in a multi-stage process that determines the correct geometry to accomplish the desired effect. Initially, the total turning angle desired by the inlet is specified. This turning angle is divided into any number of turn increments which specifies the number of turn ramp elements in the inlet section. The properties of each oblique shock are calculated iteratively using the oblique shock relations, returning the beta angle of the shock with respect to the flow. With all the betas and thetas of the flow found, it is possible to construct the lengths of each inlet ramp such that all oblique shocks intersect at the same point. This is done using triangle geometry as detailed in Figure 2. Initially, the wavelength of the first oblique shock is prescribed to be 1 meter. After that, the side lengths of all triangles are found sequentially using the law of sines.

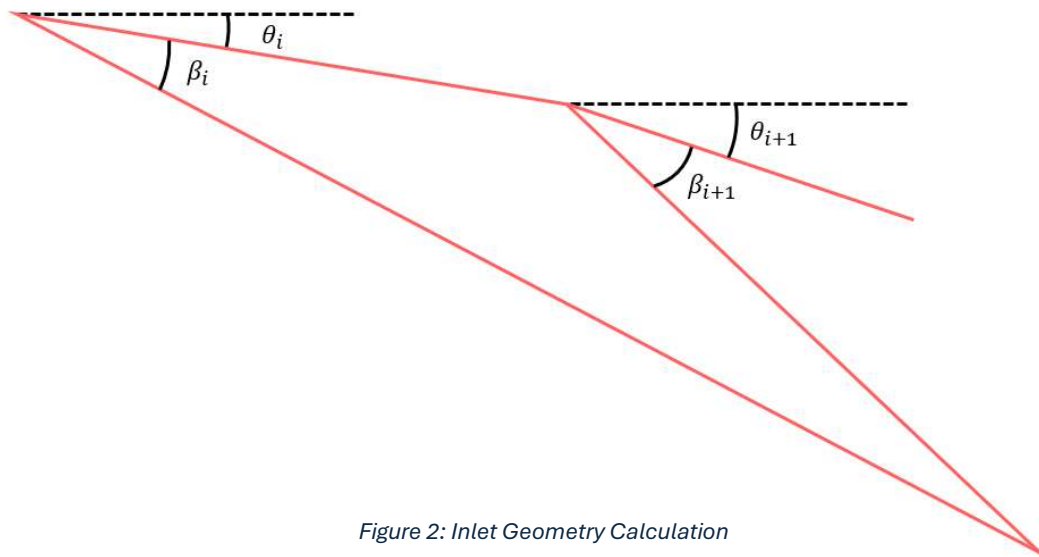


Figure 2: Inlet Geometry Calculation

6.3 Inlet Section Final Design

The inlet section goes through several iterations to find the total turning angle that would allow the design of the engine to create as much thrust as possible. The three angles tested were 20, 30, and 40 degrees. For each of them, the entire engine was redesigned to fit the height of the inlet and make the body of the engine as streamlined as possible. It was determined that a turning angle of 30 degrees resulted in the best performing engine out of the three. It can slow the flow to below Mach 2 before the cowl for the entire operating range of the vehicle, but at the same time does not create too much drag for the exterior of the vehicle. Figure 3 displays the final inlet geometry using a black curve. The oblique shocks are rendered in orange for the case of $M_1=3.25$. The cowl point is designed such that 95% of the flow after the final oblique shock enters the engine, while 5% escapes around the cowl. This allows for a safety margin in case of any disturbances at $M_1=3.25$ so

that distorted flow will not enter the engine. The normal shock at the cowl is indicated with a red line.

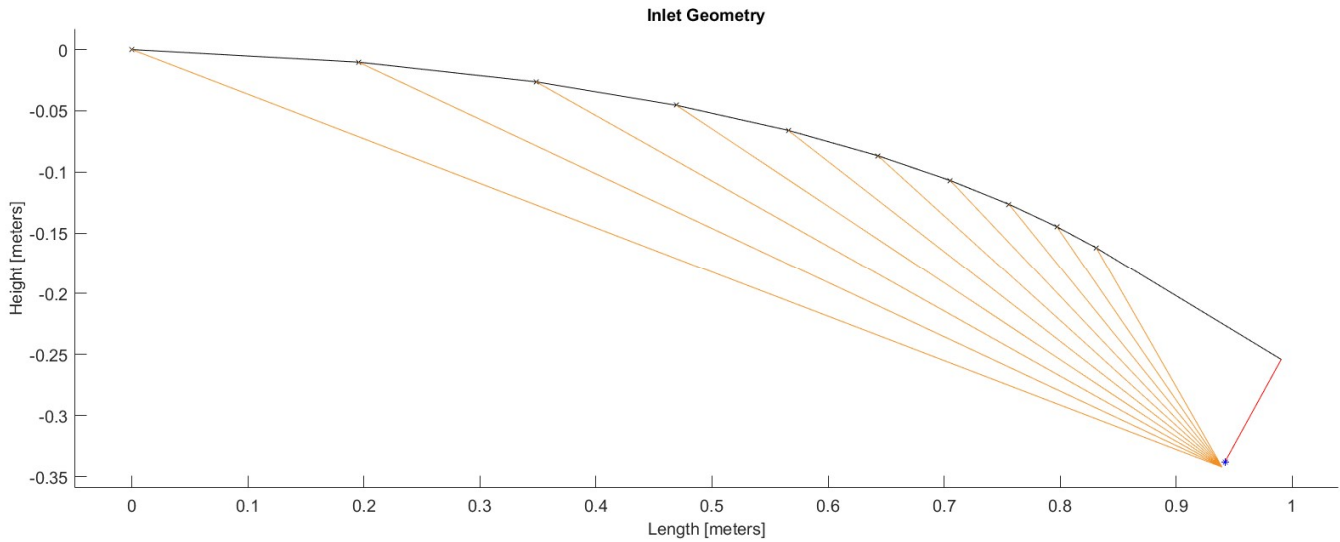


Figure 3: Final Inlet Geometry

7.0 Section 2: Diffuser Section Overview

The diffuser is designed to merge with the inlet and is used to decelerate the supersonic air to lower speeds therefore increasing its pressure before entering the combustion chamber. This is important because the combustion for ramjets requires high pressure and temperatures to mix with fuel to efficiently burn fuel.

7.1 Diffuser Section Assumptions

The diffuser section is to be treated as a quasi 1D isentropic flow.

7.2 Diffuser Design Approach

To create the diffuser section from 2 to 2p (before and after the normal shock), the cowl area, Mach, pressure, temperature, and mass flow rate of air are calculated. The mass flow rate of the air through the engine is calculated using this equation:

$$\dot{m}_{air} = P_2 * M_2 * A_{Cowl} * \sqrt{\frac{\gamma}{R_{Air} * T_2}} \dots (1)$$

To determine the state after the normal shock the following equations are used:

$$M_{2p} = \sqrt{\frac{1 + 5 * (\gamma - 1) * M_2^2}{(\gamma * M_2^2) - (5 * (\gamma - 1))}} \dots (2)$$

$$P_{2p} = P_2 * \left(1 + \frac{2 * \gamma * (M_2^2 - 1)}{\gamma + 1}\right) \dots (3)$$

$$T_{2p} = T_2 * \left(1 + \frac{2 * \gamma * (M_2^2 - 1)}{\gamma + 1}\right) * \left(\frac{2 + (\gamma - 1) * M_2^2}{(\gamma + 1) * M_2^2}\right) \dots (4)$$

To find the stagnation state after the normal shock, the following equations are used:

$$P_{02} = P_{2p} * (1 + \frac{1}{2} * (\gamma - 1) * M_{2p}^2)^{\frac{\gamma}{\gamma-1}} \dots (5)$$

$$T_{02} = T_{2p} * (1 + \frac{1}{2} * (\gamma - 1) * M_{2p}^2) \dots (6)$$

To create the diffuser section from section 2p to 3, the desired combustor area is specified. This is to ensure that the height of the vehicle remains equal to that of the inlet. The ratio of the actual area to the area at supersonic conditions $(\frac{A}{A^*})$ is calculated, which is used to numerically calculate the Mach number at each area section in the diffuser section. The stagnation temperature and pressure stay constant throughout the diffuser section which allows for easy determination of static temperature and pressure at each location.

7.3 Diffuser Section Final Design

The final design of the diffuser is based on the two area values previously defined. Since the height of the inlet is approximately 0.35 meters, the area of the combustor section is chosen to be 0.40 meters. The diffuser area next to the cowl is prescribed by the inlet geometry to be 0.097 meters. The flow at this area is turned down at an angle of 30 degrees and must be turned horizontal before reaching the combustor. This is accomplished by creating spline geometry in Fusion 360. A control point spline is used to generate a smooth expanding nozzle that would in theory not cause any flow separation even though we are operating under a Quasi 1D isentropic flow. At the start and end of the spline, a tangent condition is ensured so that the flow transition is smooth from the cowl to the combustor section. The expansion ends right where the flame holder and the combustor section begin making the diffuser approximately 1.058 meters in length.

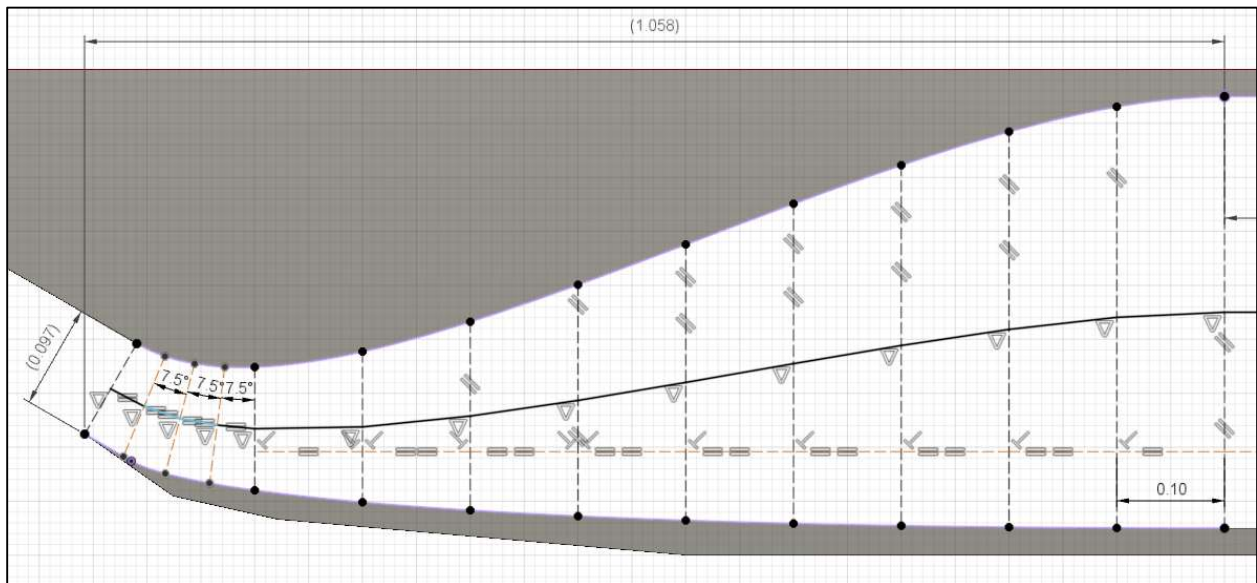


Figure 4: Diffuser Geometry

8.0 Section 3: Flameholder Section

Flameholders ensure the flame in ramjets are stable when enduring the subsonic airflow. This allows for an efficient and continuous combustion process. The flameholder pressure drop is given by the following equation. Since the process is adiabatic, the temperature stays constant across the flame holder.

$$P_{3p} = P_3 * (1 - 0.81 * \gamma * M_3^2) \dots (7)$$

$$T_{3p} = T_3 \dots (8)$$

9.0 Section 4 Combustor Section Overview

The combustor is designed to add heat flow by burning the fuel-air mixture. The heat addition is calculated to obtain the desired exit Mach number and ensures the nozzle can operate as per its range of operation. The combustor's length is sized based on average flow velocity and combustion time which ensures complete combustion of fuel before the gases exit into the nozzle.

The team defined that the equivalence ratio (φ) is then defined as $\varphi = 1$ at $M = 3.25$ when considering the flameholder. The stoichiometric fuel-to-air mass ratio is then calculated from here. This ratio represents the ideal balance of fuel and air for combustion within the combustion chamber. This is the ratio of the actual fuel-to-air ratio to the stoichiometric ratio and serves as a measure of how close the actual mixture is to stoichiometric conditions.

Regarding the equivalence ratio for lower Mach numbers, it reduces as the intake air speed reduces. Less fuel is added to achieve the optimal conditions through combustion to reach the correct conditions for the converging section to choke at its design point. Therefore, the slower the intake air Mach is, the less fuel added to combustion to reach the star state necessary for section 4 to 5.

Next the mass flow rate of the fuel is calculated for to determine the amount of fuel needed to achieve the desired equivalence ratio of 1 at mach 3.25. The mass flow rate of the fuel is calculated using this equation:

$$\dot{m}_{Fuel} = \dot{m}_{Air} * \varphi * \frac{M_{Air}}{M_{Fuel}} \dots (9)$$

$$\text{Where } \frac{M_{Air}}{M_{Fuel}} = \frac{1}{34.33}$$

Then the lower heating value of hydrogen is defined. This represents the amount of heat released per unit mass of hydrogen fuel during the combustion. With this value, the heat addition per unit mass of air is calculated using this equation:

$$q_{Addition} = LHV * \frac{\dot{m}_{Fuel}}{\dot{m}_{Air}} \dots (10)$$

9.1 Combustor Design Approach

The approach for the combustor is to keep the nozzle of the engine at the same performance over the operating range of the aircraft. Specifying the Mach number coming out of the combustor would achieve this. The combustor is specified with a phi of 1 for M1

3.25 so that at lower Mach numbers, less heat can be added, and the flow will choke at the same place. Designing the combustor starts off with a heat addition function that finds the Mach number, pressure, and temperature at the combustor. The average temperature and pressure between the third and fourth state are then calculated to determine the temperature and pressure gradients across the combustor. A combination of different combustion times is calculated based on the different points in the process to see how much they differ. Lastly, the required length for full combustion can be solved.

$$U = M * \sqrt{\gamma * R * T} \dots (11)$$

$$U_{Average} = \frac{U_1 + U_2}{2} \dots (12)$$

$$P_{Average} = \frac{P_1 + P_2}{2} \dots (13)$$

$$T_{Average} = \frac{T_1 + T_2}{2} \dots (14)$$

$$P_{04} = P_4 * (1 + \frac{1}{2} * (\gamma - 1) * M_4^2)^{\frac{\gamma}{\gamma - 1}} \dots (15)$$

$$T_{03} = T_3 * (1 + \frac{1}{2} * (\gamma - 1) * M_3^2) \dots (16)$$

$$\tau = 325 * (\frac{P_{State}}{101325})^{-1.6} * e^{\frac{-.8 * T_{State}}{1000}} * 10^{-6} \dots (17)$$

$$\tau_{Average} = 325 * (\frac{Pressure\ Gradient}{101325})^{-1.6} * e^{\frac{-.8 * Temperature\ Gradient}{1000}} * 10^{-6} \dots (18)$$

$$Required\ Length = U_{Average} * \tau_{Average} \dots (19)$$

9.2 Combustor Section Final Design

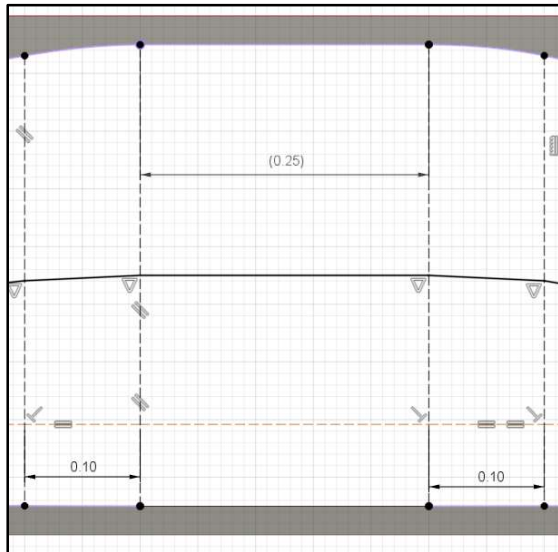


Figure 5: Combustor Section Geometry

The combustor is designed to enhance heat flow by burning the fuel-air mixture, achieving the desired exit Mach number for optimal nozzle operation. Its length is determined by the average flow velocity and the time required for complete combustion. Under the assumption of 1D isentropic flow and constant changes in velocity, pressure, and stagnation temperature, the combustor maintains nozzle performance across the aircraft's operating range. Originally, calculations suggested a minimal theoretical length for the combustor, but for practical implementation, it was set to 0.25 meters to ensure realism in the design. The required length calculated using the formula given seemed too small to be realistic.

The maximum length required was 0.006 meters from an average time constant of $2.89 * 10^{-5}$ seconds, and average velocity of 208.355 meters per second.

10.0 Section 5: Converging Section Overview

The main functions of the converging sections are to compress and accelerate the exhaust gasses or air flowing through it. This is because as the cross-sectional area of the nozzle decreases, the velocity of the fluid increases while its static pressure and temperature decreases.

10.1 Converging Section Assumptions

The nozzle section is treated as quasi 1D isentropic flow. The flow must reach Mach 1 at the throat.

10.2 Converging Section Design Approach

The approach when designing the converging section of the Ramjet is to first solve for the area ratios. Specifically, the ratio of the area after combustion to the area of the throat where the Mach number is 1. The ratio is calculated using the following equation:

$$A_4^* Ratio = \left(\frac{1}{M_4}\right) * \left(\frac{2+(\gamma-1)*M_4^2}{\gamma+1}\right)^{\frac{1}{2}*(\gamma+1)} \dots (20)$$

The area at the throat is then calculated using the ratio defined above and the area of state three. This helps determine the throat area where the flow becomes sonic after the combustion chamber. The temperature and pressures at the throat are then calculated using the following equations:

$$P_5 = \frac{P_{04}}{\left(1+\frac{1}{2}*(\gamma-1)*M_5^2\right)^{\frac{\gamma}{\gamma-1}}} \dots (21)$$

$$T_5 = \frac{T_{04}}{\left(1+\frac{1}{2}*(\gamma-1)*M_5^2\right)} \dots (22)$$

10.3 Converging Section Final Design

This converging section design is designed to meet the quasi 1D assumption. A control point spline is used again to connect the end of the combustor section with the throat. At the start and end of the spline, it is tangent to the flow at both sections. In between, a smooth curve of decreasing cross-sectional area transitions between the two points. In the diagram, points along the section are spaced at 0.1-meter intervals, and midpoints between these are connected to outline a general streamline. The lines are set to be tangent at both the beginning and the end of the section, ensuring that they are equally spaced angularly. These lines are used as points for the streamline analysis of property changes through the engine.

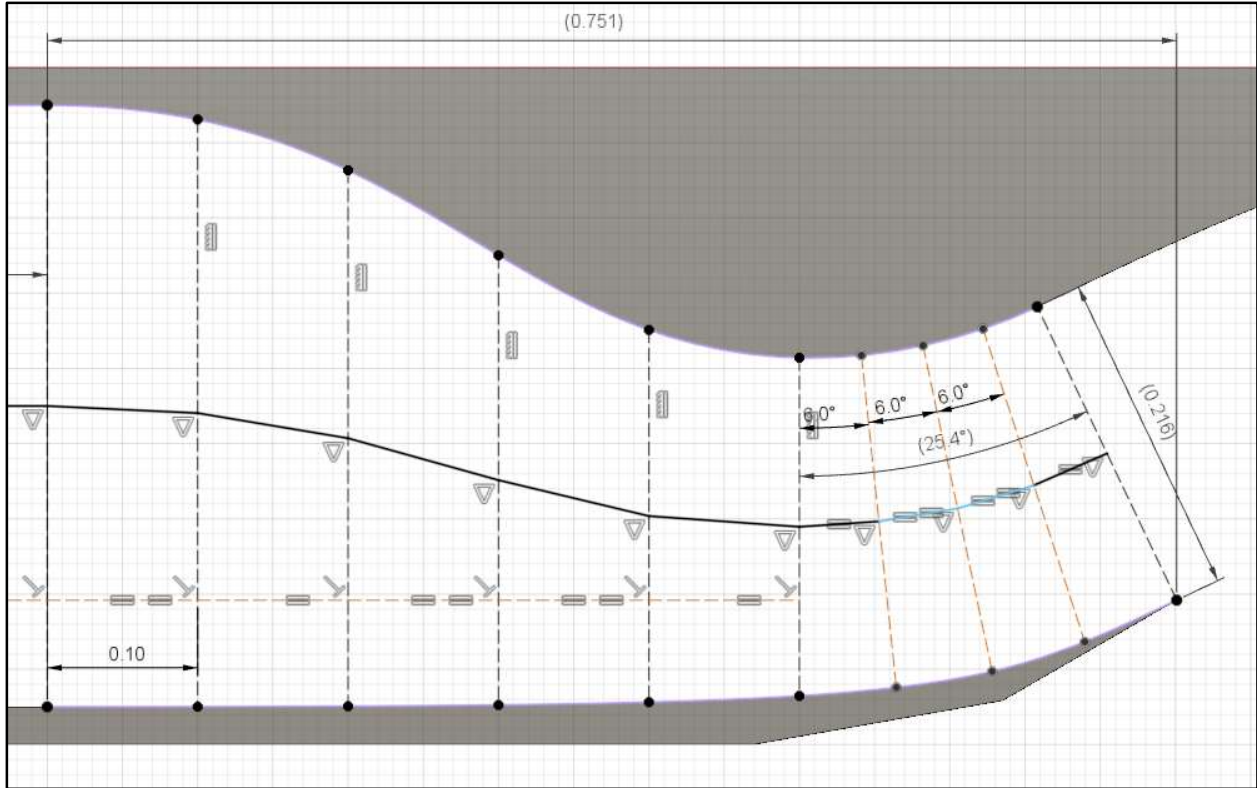


Figure 6: Converging Section Geometry

11.0 Section 6 Nozzle Section Overview

The nozzle is designed to generate thrust by converting high pressure and temperature gasses into kinetic energy. The nozzle uses expansion waves to expand and accelerate the flow to supersonic speeds. The final shape and size of the nozzle are created to ensure the exit Mach number matches atmospheric pressure to optimize thrust and minimize losses.

11.1 Nozzle Section Assumptions

The nozzle is assumed to be isentropic.

11.2 Nozzle Design Approach

As per the requirements of the final, the nozzle section is designed using the method of characteristics for steady 2D flow. The nozzle uses a single expansion wave to bring the flow parallel to the free stream flow, which allows it to only be made of a single curved wall. The exit pressure is specified such that the height of the nozzle matches approximately the height of the inlet. At $M_1=3.25$, this corresponds to a final pressure after expansion of 5 times the atmospheric pressure at this altitude. To calculate the curvature of the nozzle, the total turning angle is calculated from the Prandtl-Meyer equation with $M_5=1$ and M_6 specified by the equation below using $P_6 = 5 \cdot P_1$.

$$M_6 = \sqrt{\frac{2}{\gamma + 1} * \left(1 + \frac{1}{2} * (\gamma - 1) * M_5^2\right) * \frac{P_5^{\frac{\gamma-1}{\gamma}}}{P_6}}^{-1}$$

The total turning angle is divided into 5000 sections to allow for high accuracy in the calculation of the curvature. For each section, the Mach number and characteristic is calculated and used to find the points of the nozzle. Since the nozzle is isentropic, the temperature and velocity at each point can be found using the following equations:

$$T_6 = \frac{T_{04}}{\left(1 + \frac{1}{2} * (\gamma - 1) * M_6^2\right)}$$

$$U_6 = M_6 * \sqrt{\gamma * R * T_6}$$

11.3 Nozzle Final Design

The nozzle section of the ramjet engine is designed to accelerate high-temperature gases from the combustion chamber to supersonic speeds, assuming the exit pressure matches atmospheric conditions. The design process uses the method of characteristics for steady 2D flow, focusing on calculating the Mach number, temperature, and speed at state 6. This involves the equations to derive these values based on the conditions at the nozzle's throat (state 5) and adjusting for atmospheric exit pressure. The calculations ensure the nozzle effectively accelerates the flow to the desired Mach number while maintaining the necessary temperature and velocity profiles for optimal thrust. This approach helps to streamline the flow dynamics and maximizes the engine's thrust output under predefined assumptions.

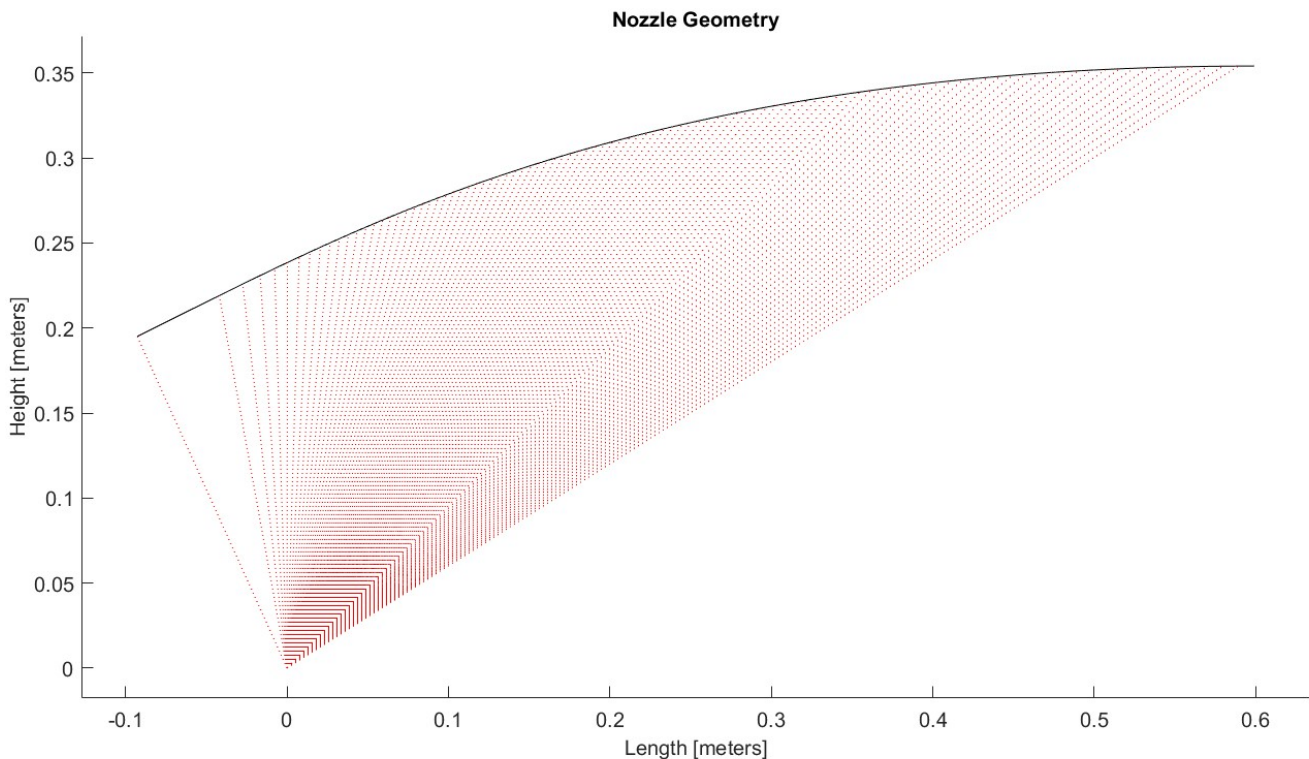


Figure 7: Nozzle Geometry

12.0 Overview of Final Design



Figure 8: Engine Geometry

Our final design ultimately was a blend of optimized and compromised components. The inlet was designed to sacrifice 5% of the potential mass flow rate it could theoretically achieve to have a safety margin between the oblique shock convergence point and the cowl edge. This margin only gets bigger as the speed drops below Mach 3.25. This does sacrifice more and more of the flow that is prepared by the shocks, but it is a necessary sacrifice.

The cowl is designed to have as low of a profile as possible without being overly complex to find the drag forces upon it. It suffices for the application and our design does produce excess thrust to overcome the drag force on the cowl.

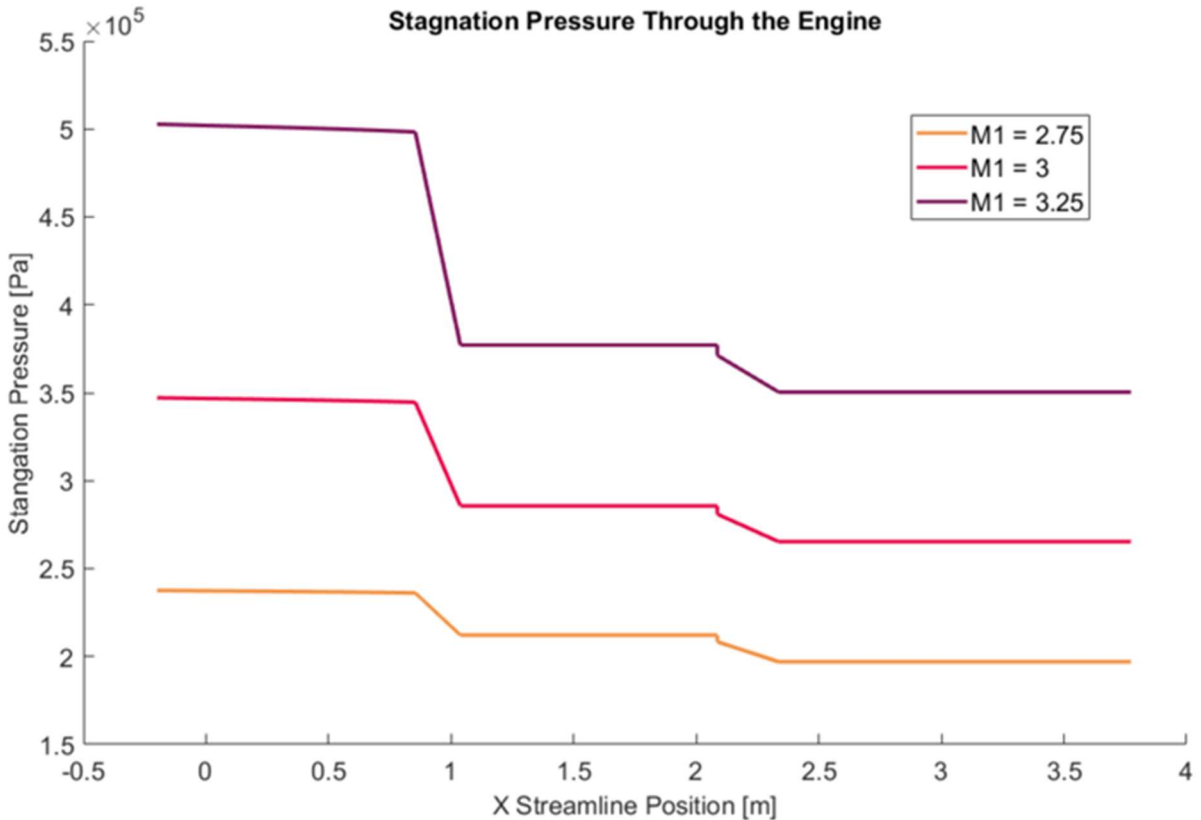
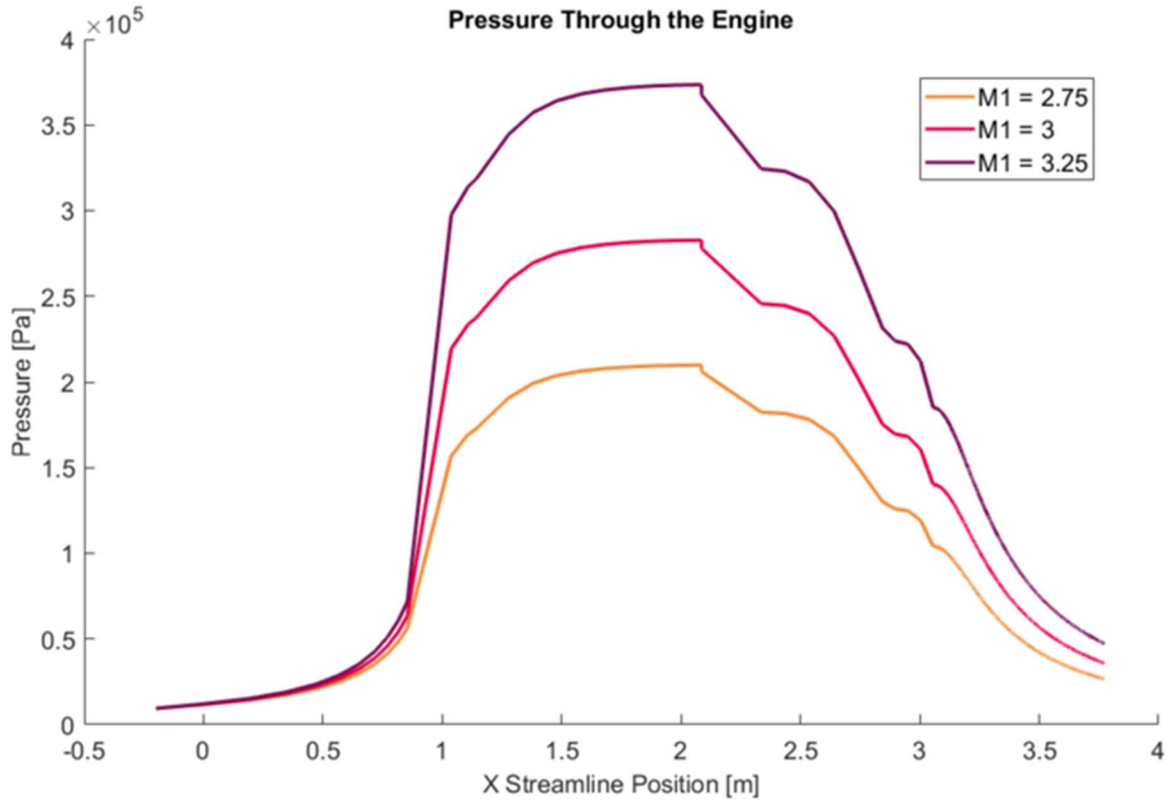
Our diffuser is an example of how we decided to balance the given assumptions with reality. In theory we could have made it an instant expansion out to 0.4 meters, and it would have fulfilled the quasi 1D isentropic assumption, but we felt the design should resemble reality. We gave it as smooth of a curve while also being considerate of the length.

The combustor is another example of balancing the assumptions and reality just like the diffuser section. While our calculations show that the combustor section could be incredibly short, borderline non-existent, this, we felt, was too much of a suspension of reality and decided to extend the combustor section out to 0.25 meters in length. We felt this length would be sufficient space for full combustion of the hydrogen fuel to occur. We do understand that this “feeling” for the distance is not based on any hard evidence.

The converging section is once again an example of the balance we had to strike with the previous two components however it had to be design in conjunction with the exit nozzle as the final angle of the throat would be depending on the angle at which the following nozzle would need to best match the ambient pressure at Mach 3. Our nozzle falls in the middle of the theoretical performance curve, allowing for effective use for speeds above and below Mach 3.

Finally, our nozzle did not follow the typical bell nozzle shape that is typically associated with a converging and diverging nozzle. It more closely resembles an aerospike engine relying on the local atmosphere to provide the necessary pressure to act as the artificial outer “bell” of the nozzle.

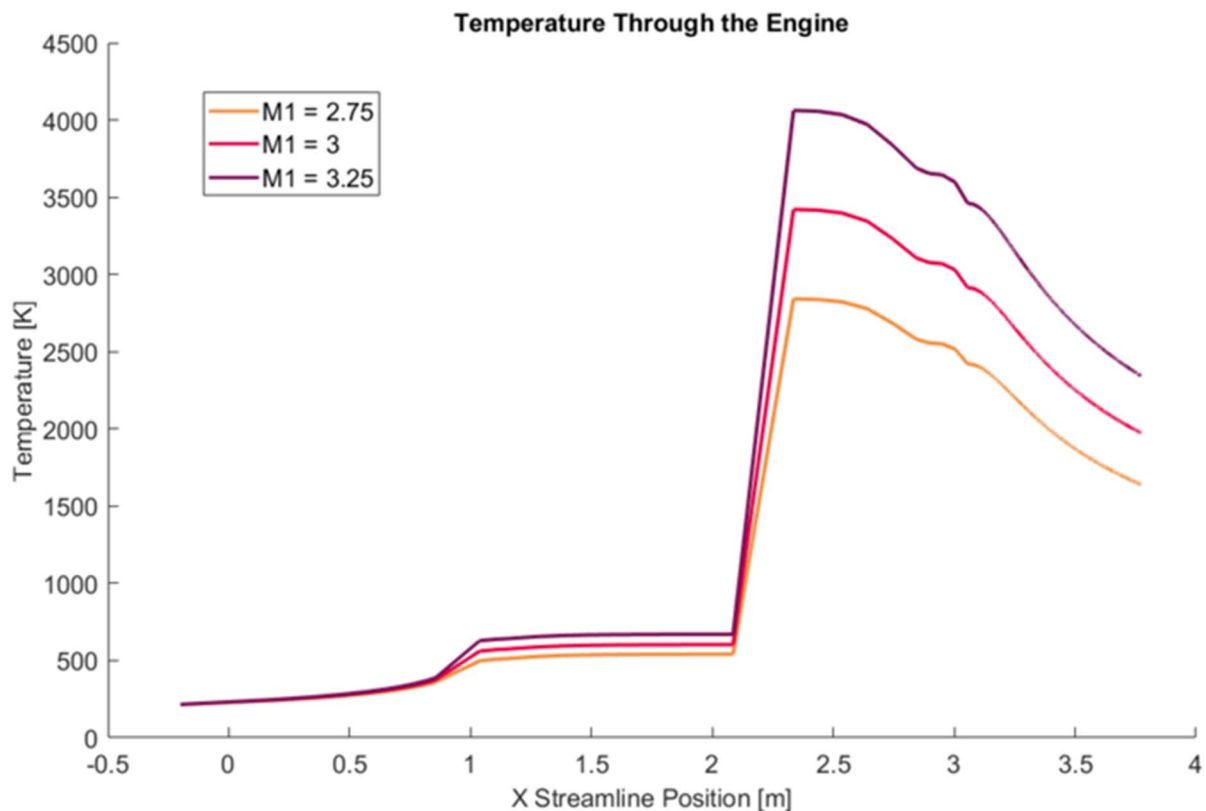
12.1 Final Design Outputs

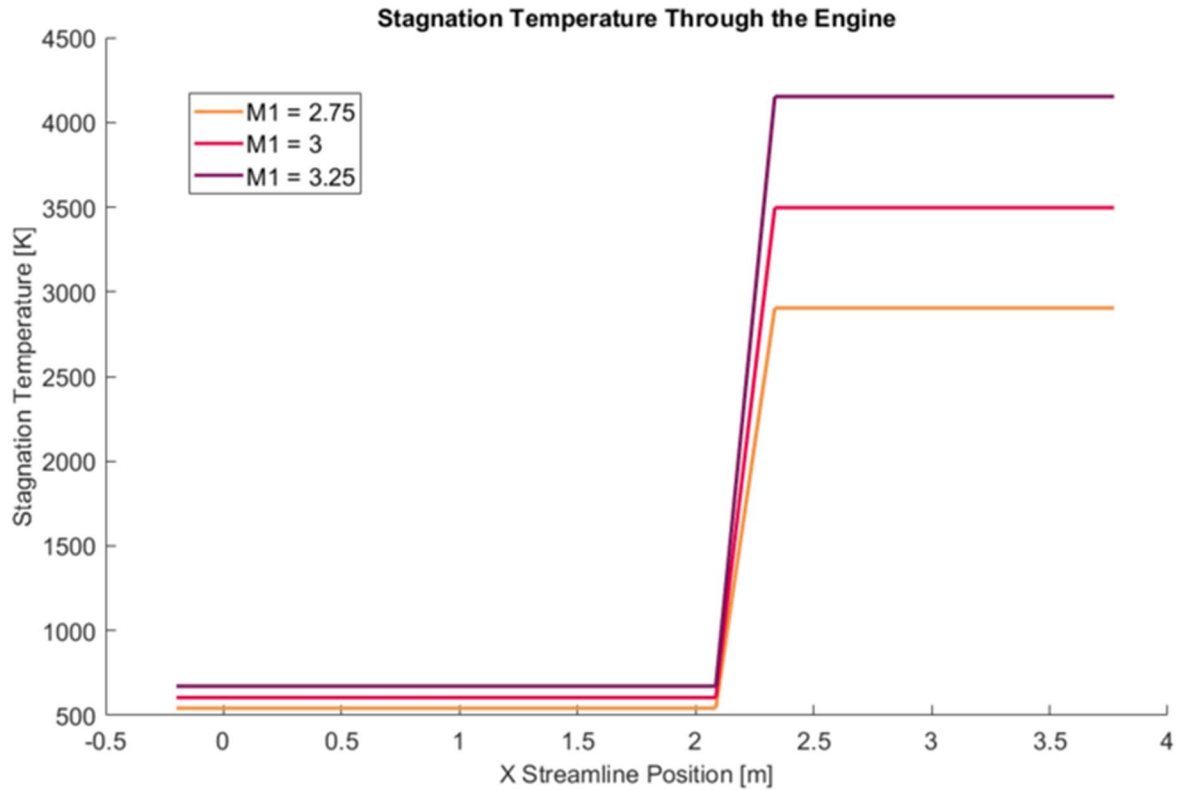


Pressure Along Streamline:

The stagnation pressure through the engine decreases as the air goes through the engine. The stagnation pressure stays relatively constant, drops, then stays constant again, drops, and finally is constant. The two drops occur at the inlet and at the converging section. Stagnation pressure drop at the inlet because there is an arbitrary pressure loss from a normal shock. It drops at the converging section because of the flow accelerating due to the decrease in area. When comparing the stagnation pressure through the engine for when M is 2.75, 3, and 3.25, they all display a relative pattern varying in magnitude from one another.

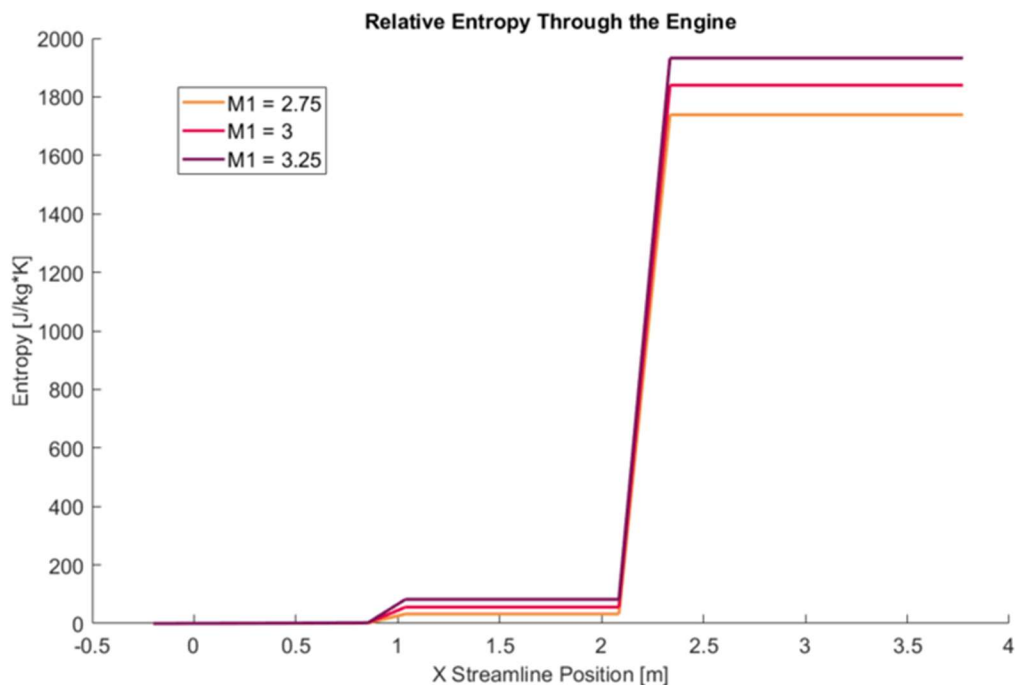
The pressure through the engine increases and decreases as the air goes through the engine. The pressure through the engine increases until it reaches the converging section. The pressure may seem “shaky” at that area due to the way it is discretized.





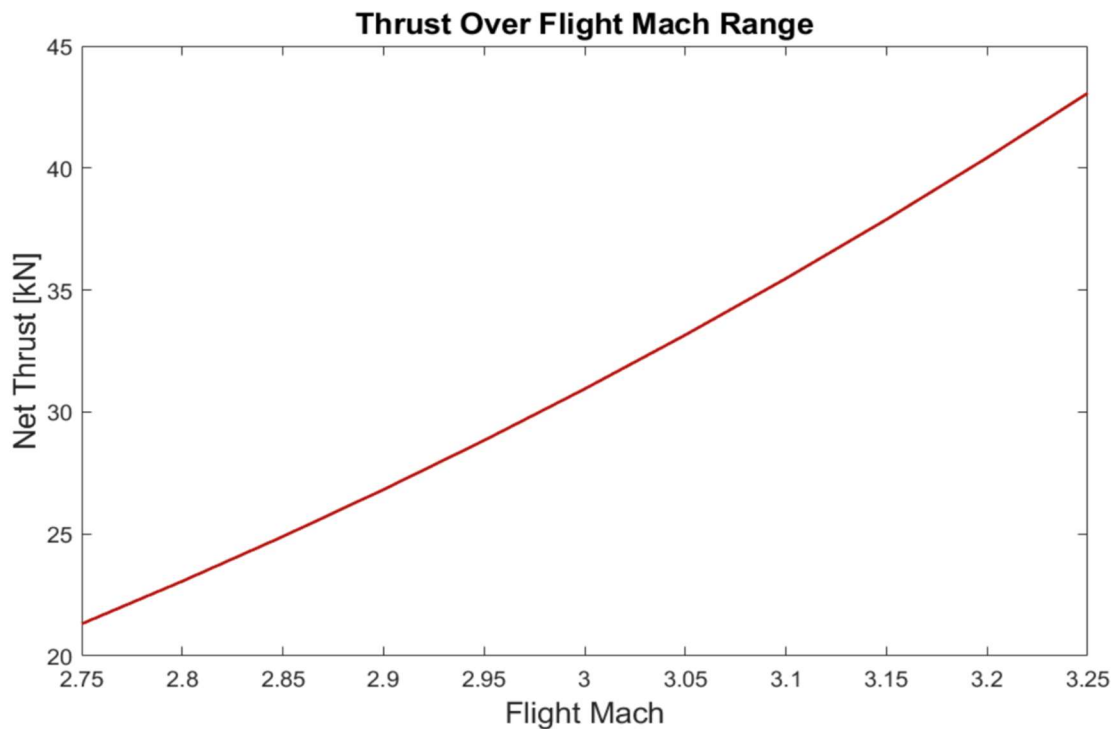
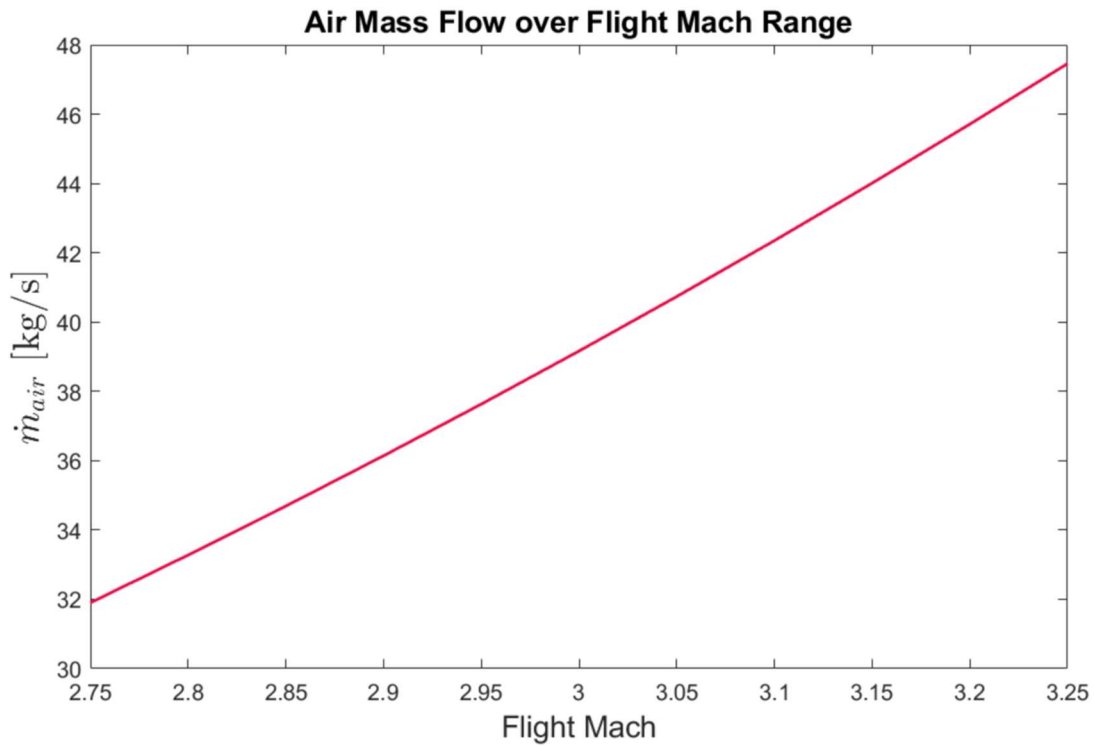
Temperature Along Streamline:

The temperature through the engine increases as the flow goes through the combustion chamber. The temperature increases as it goes through the engine initially because the air is moving into a tight space at the inlet which is then ignited and goes through combustion at the combustor.

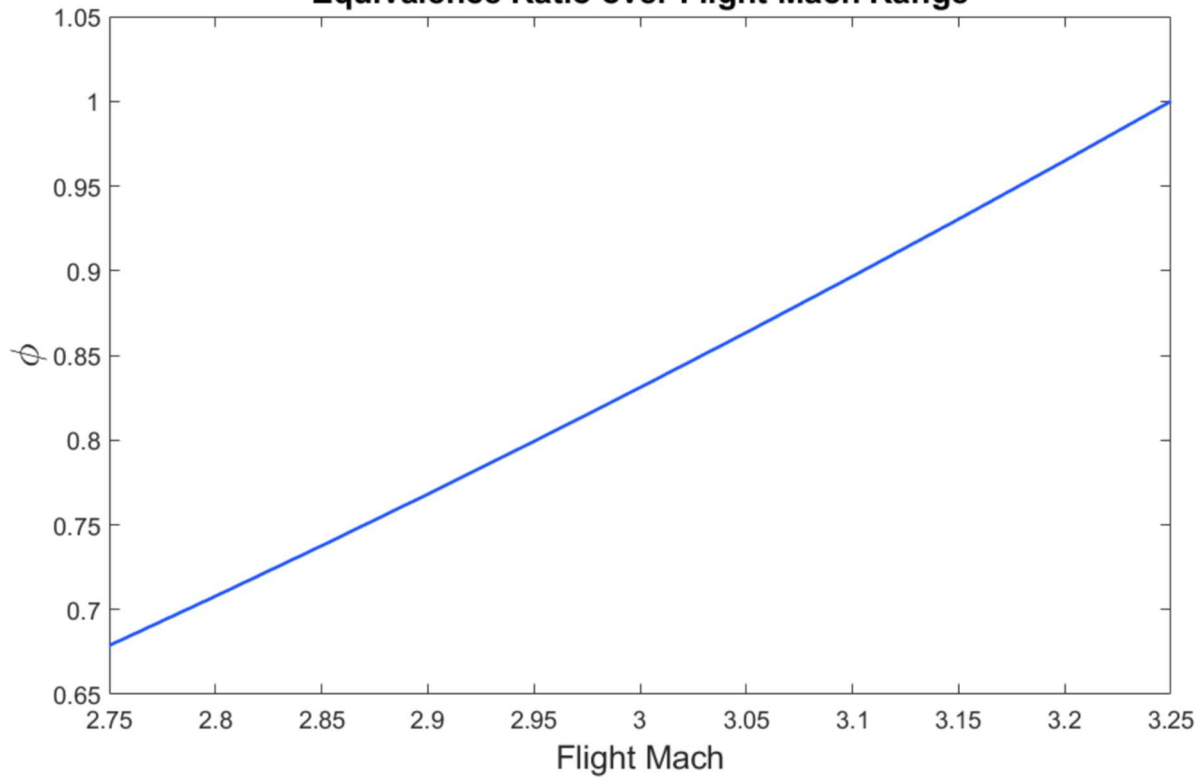


Relative Entropy Along Streamline:

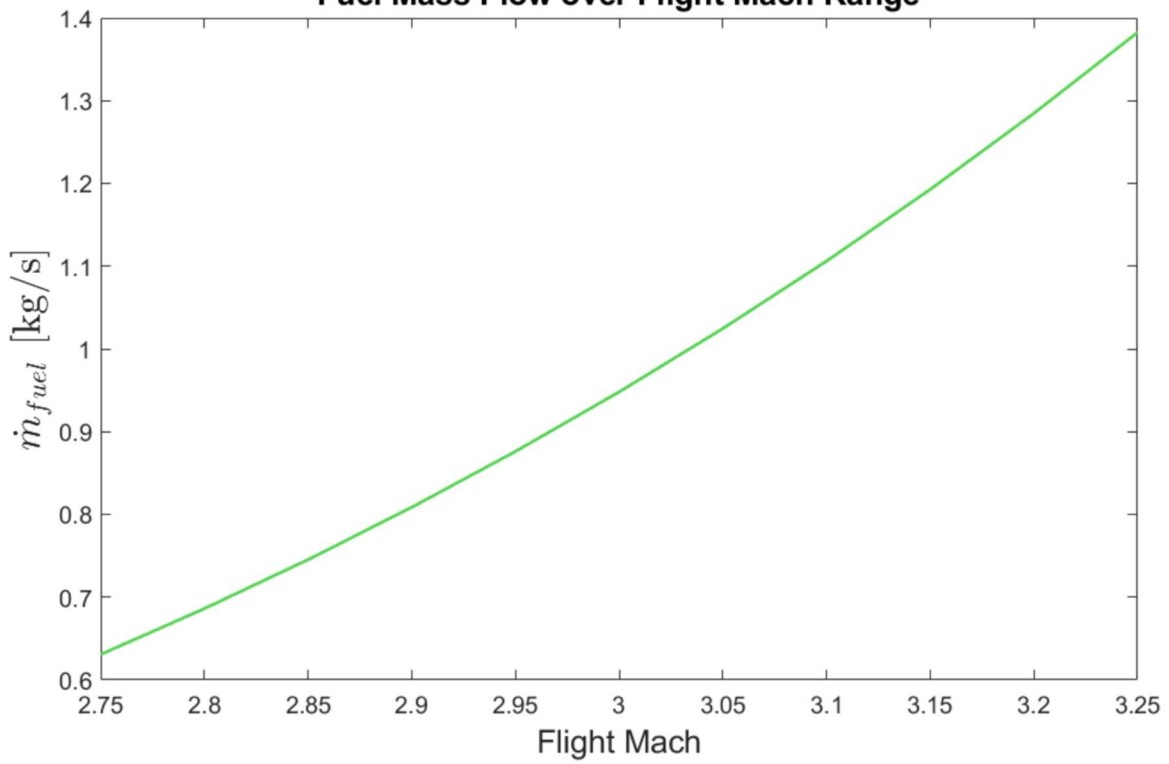
The relative entropy through the engine increases minimally as it went through the inlet section. Then it remains constant until the flow combusts at the combustor section and remains constant from there on out.



Equivalence Ratio over Flight Mach Range



Fuel Mass Flow over Flight Mach Range



Mass Flow and Thrust:

The net thrust and mass flow rate of the air increase as the flight Mach increases.

Full Schedule Over Range:

The equivalence ratio and the mass flow rate of the fuel increases as the flight Mach increase.

13.0 Concluding Thoughts

In conclusion, our ramjet design meets the design requirements that were laid out for the team. The design generates net positive thrust throughout the Mach range at 55,000 feet. It also successfully achieves our design goal of having the lowest cross-sectional area to reduce overall drag on the vehicle. Each component of the design fits within and does not violate any of the given constraints or assumptions.

References

Jr, A. J. D. (2021). *Modern Compressible Flow: With Historical Perspective*. McGraw-Hill Higher Education. (Original work published 2002)

U.S. Standard Atmosphere vs. Altitude. (2003, January 26). The Engineering ToolBox. https://www.engineeringtoolbox.com/standard-atmosphere-d_604.html

Appendix: MATLAB Scripts Used

The MATLAB scripts used in the completion of this project are listed below and attached to the assignment as a zip file.

[EngineDesign.m](#)

[EnginePerformance.m](#)

[ObliqueShock.m](#)

[ExpansionWave.m](#)

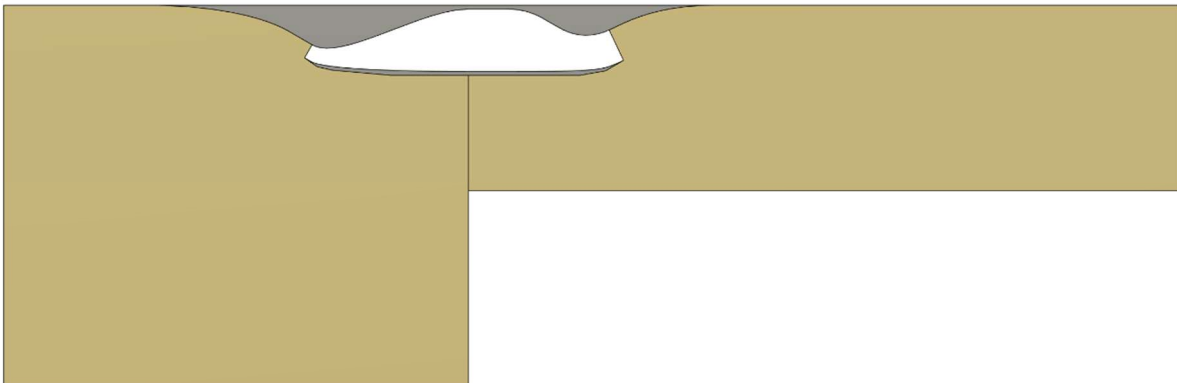
[HeatAddition_1D.m](#)

[PrandtlMeyerD.m](#)

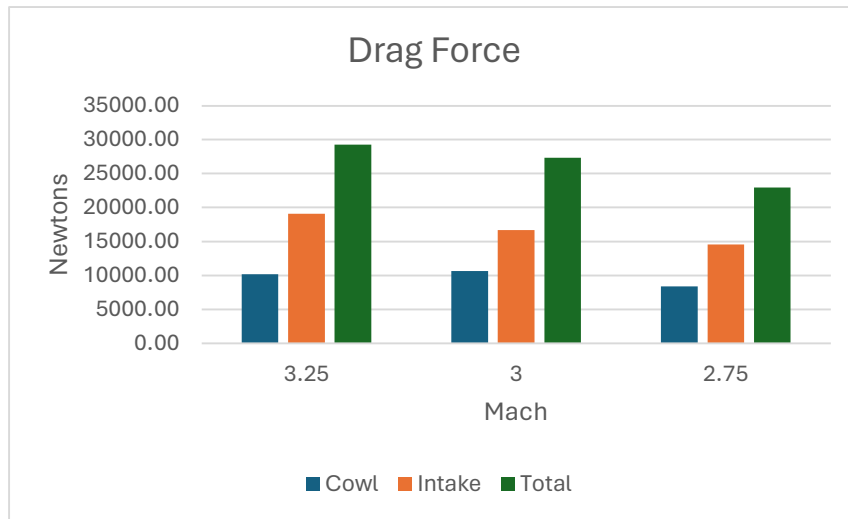
[PrandtlMeyerInverseD.m](#)

Appendix: Simulation Data Archive

Fluid Volume

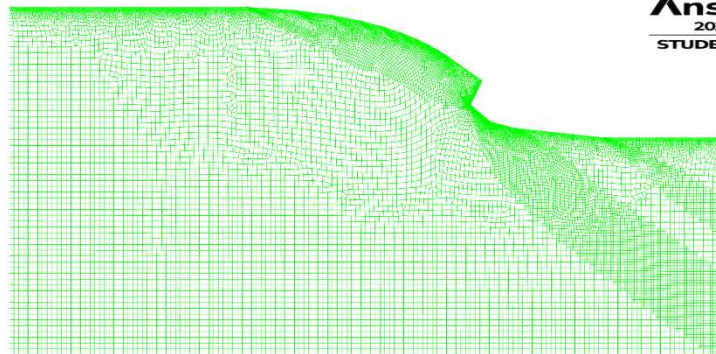


Drag Values from Fluent

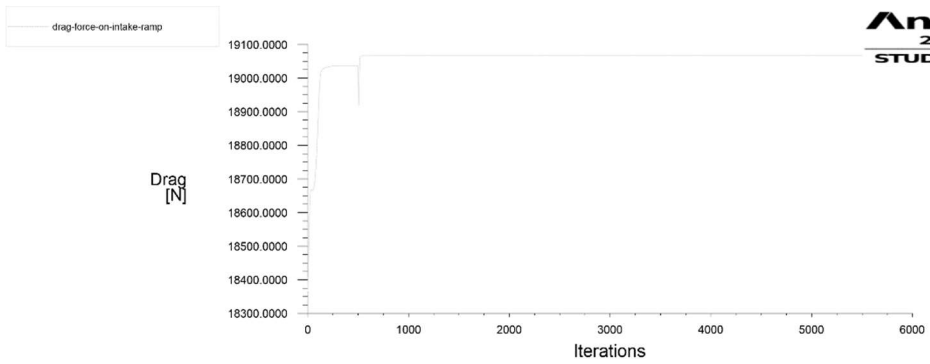
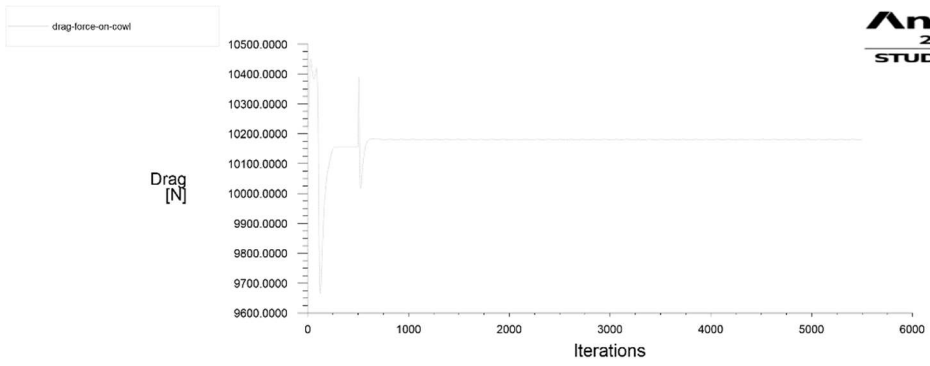
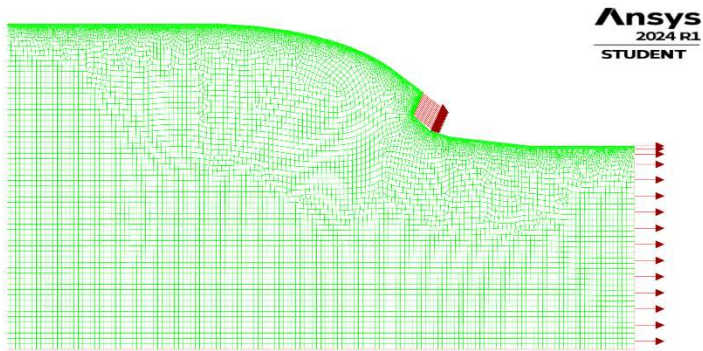
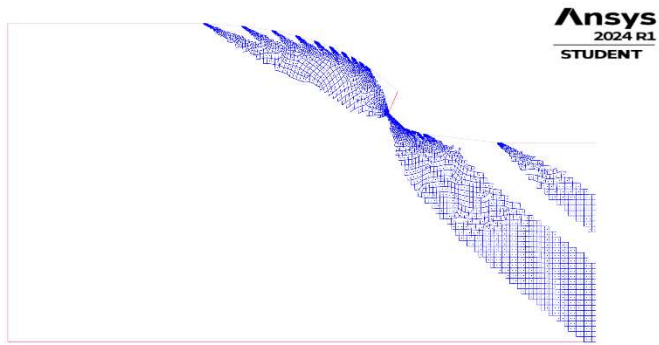


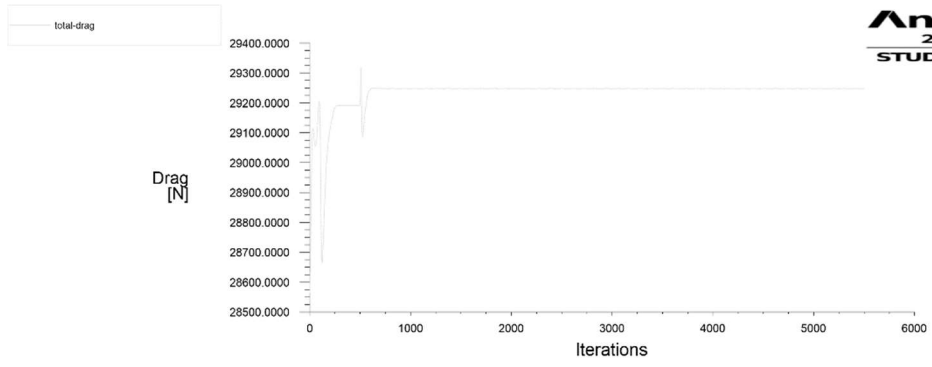
Inlet Mach 3.25 Simulation Data

Forces						
Zone	Forces [N]		Viscous	Total	Coefficients	
	Pressure	Total			Pressure	
Viscous						
wall_cowl	(10180.998 23872.849 0)	(0.58325856 1.3676502 0)	(0 0 0)	(10180.998 23872.849 0)	(0.58325856 1.3676502 0)	
wall_intakeramp	(19066.53 44124.771 0)	(1.0923013 2.5278614 0)	(0 0 0)	(19066.53 44124.771 0)	(1.0923013 2.5278614 0)	
Net	(29247.528 67997.621 0)	(1.6755598 3.8955116 0)	(0 0 0)	(29247.528 67997.621 0)	(1.6755598 3.8955116 0)	
Forces - Direction Vector (1 0 0)						
Zone	Pressure	Viscous	Total	Pressure	Viscous	Total
wall_cowl	10180.998	0	10180.998	0.58325856	0	0.58325856
wall_intakeramp	19066.53	0	19066.53	1.0923013	0	1.0923013
Net	29247.528	0	29247.528	1.6755598	0	1.6755598

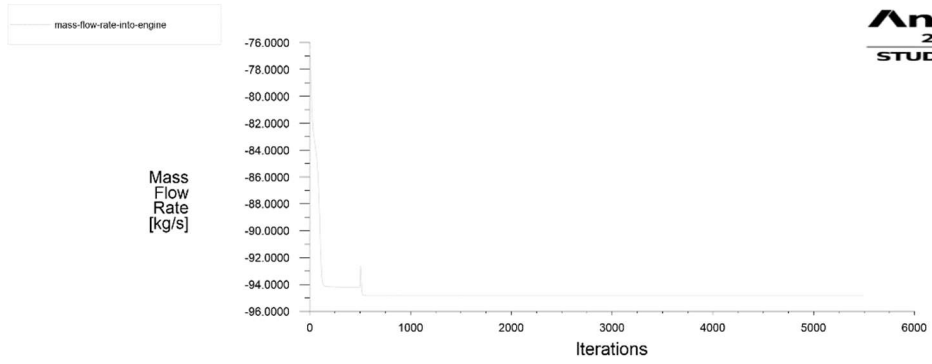


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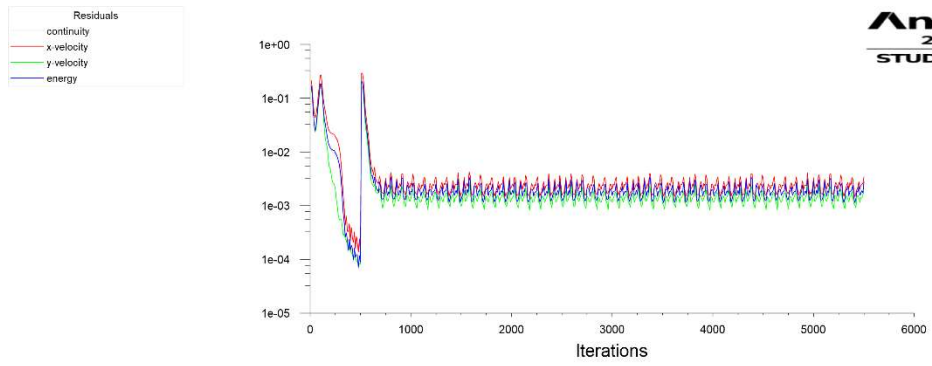




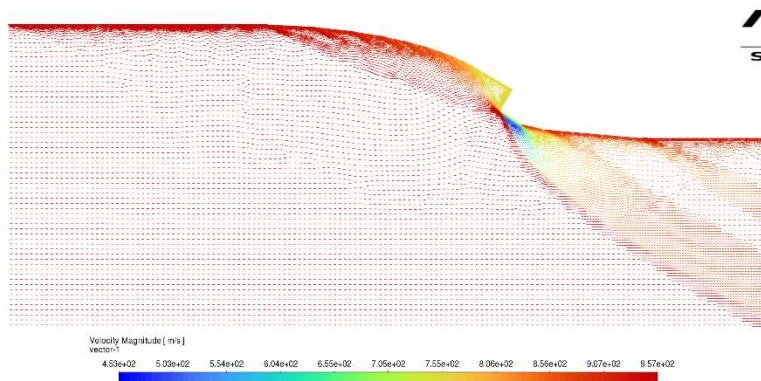
Ansys
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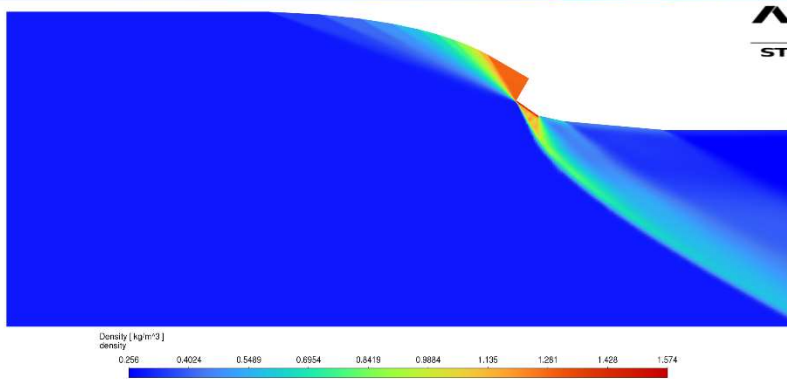
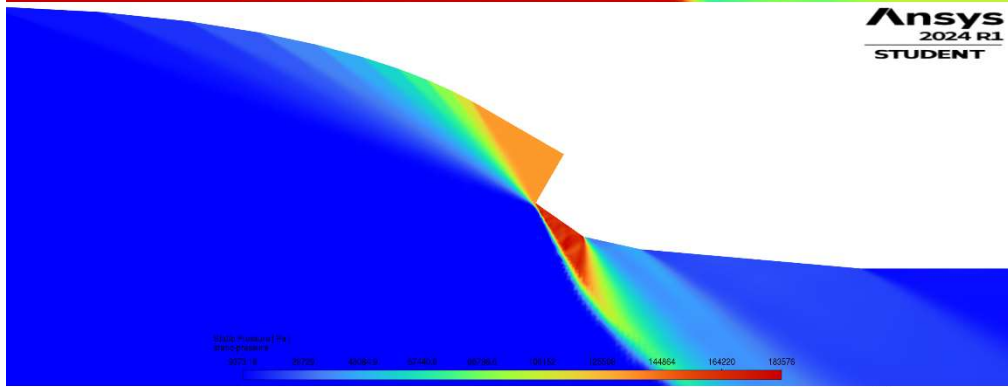
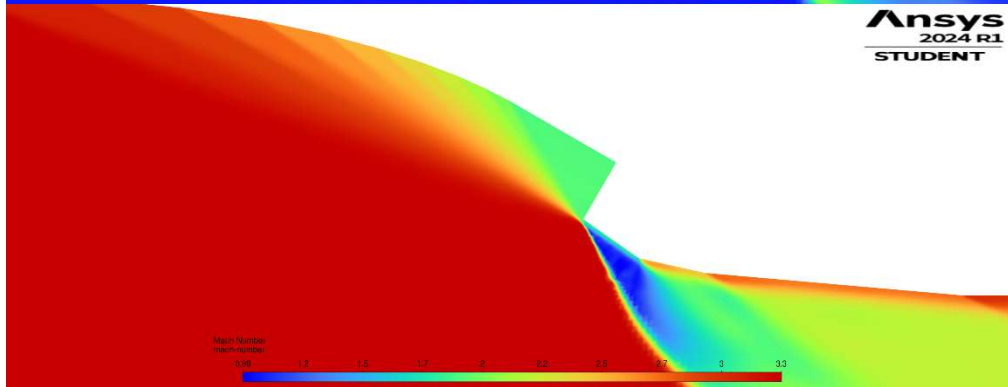
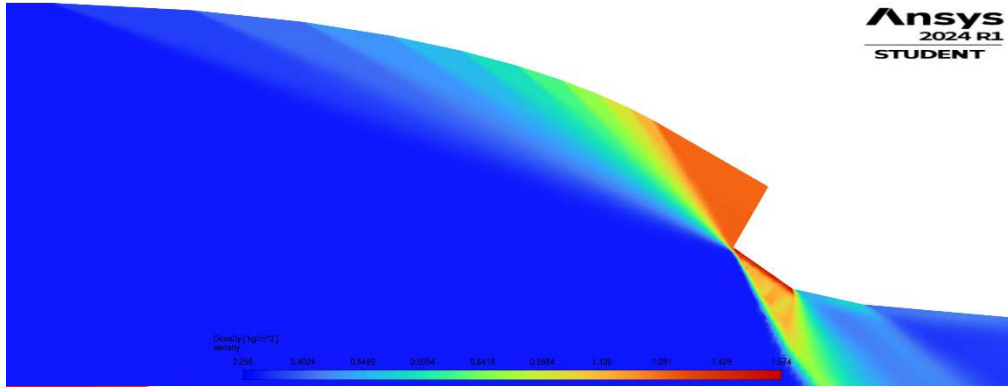
Ansys
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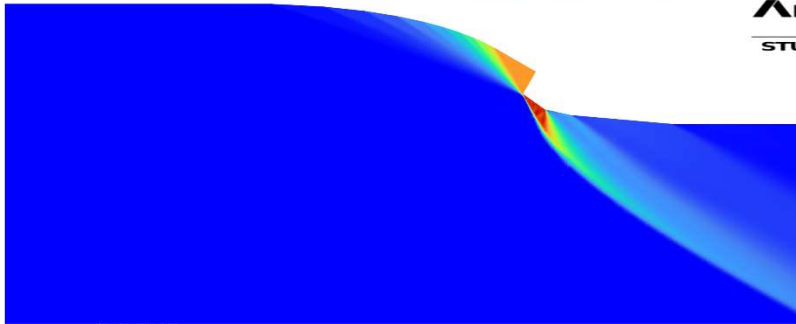
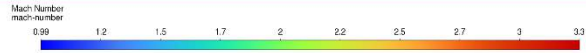
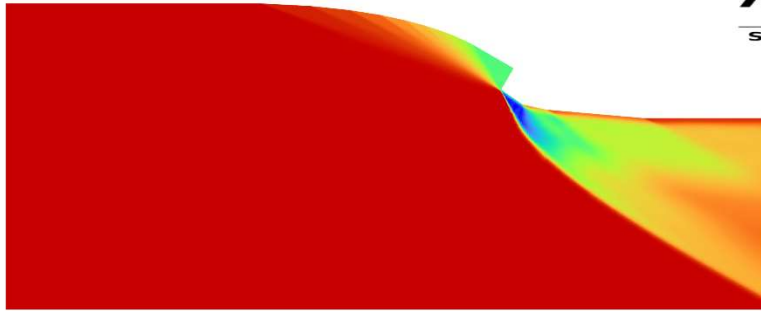


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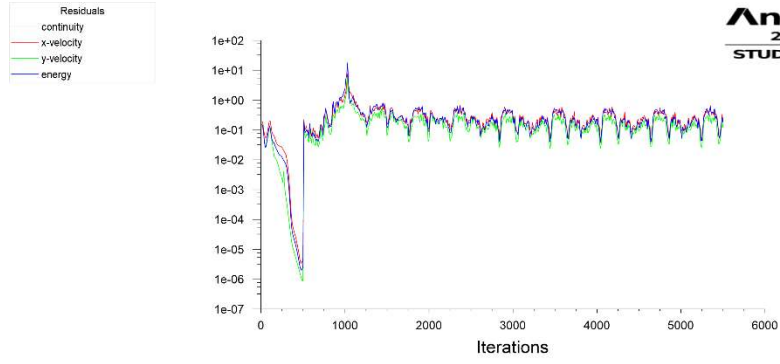


```
Area-Weighted Average
Static Pressure [Pa]
-----
pressure_outlet_1_engine 134045.96
pressure_outlet_2_exit 22009.997
wall_cowl 34389.098
wall_intakeramp 56118.366
-----
Net 42116.732

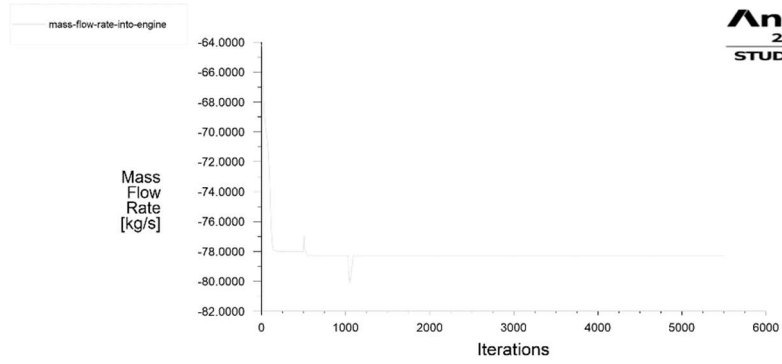
Area-Weighted Average
Mach Number
-----
pressure_outlet_1_engine 1.9224987
pressure_outlet_2_exit 2.6269396
wall_cowl 2.7326871
wall_intakeramp 2.5590899
-----
Net 2.6187115

Area-Weighted Average
Static Temperature [K]
-----
pressure_outlet_1_engine 386.31421
pressure_outlet_2_exit 283.02309
wall_cowl 273.23167
wall_intakeramp 296.62913
-----
Net 287.59135
```

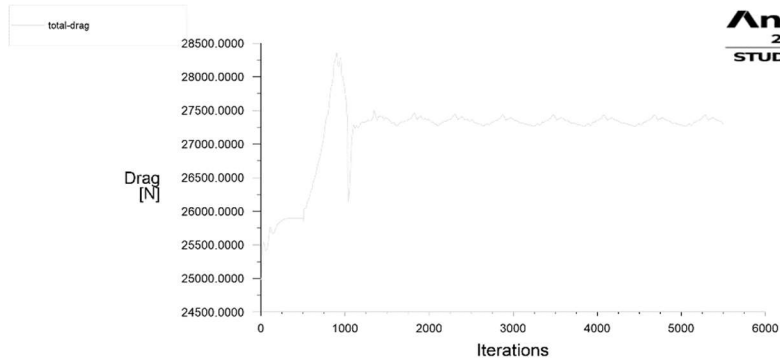
Inlet Mach 3 Simulation Data



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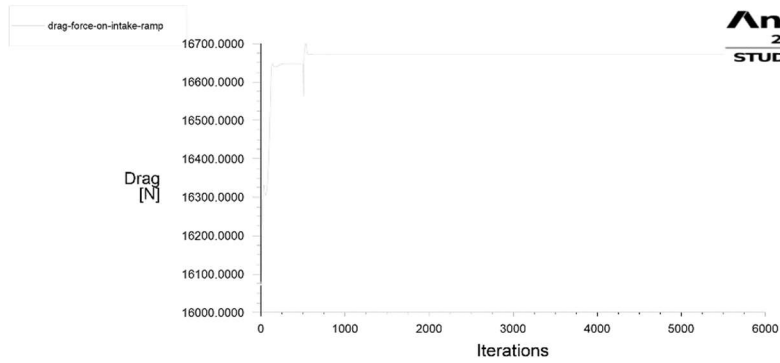


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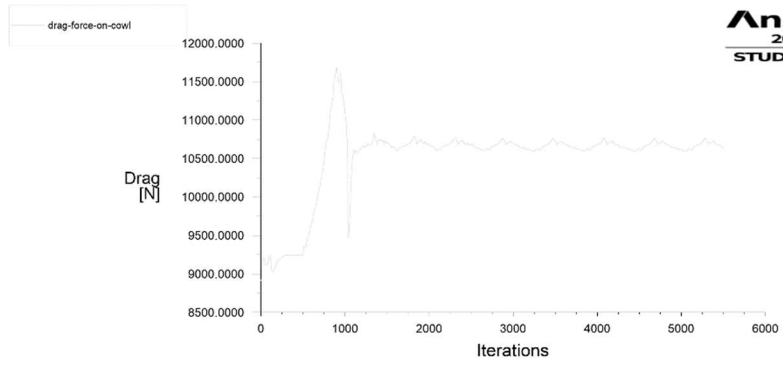


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2024 R1
STUDENT

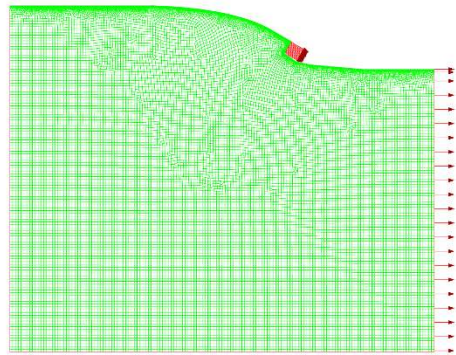
2



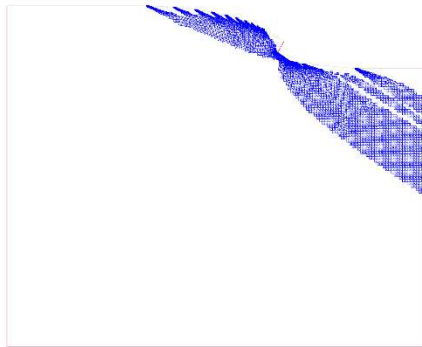
Ansys
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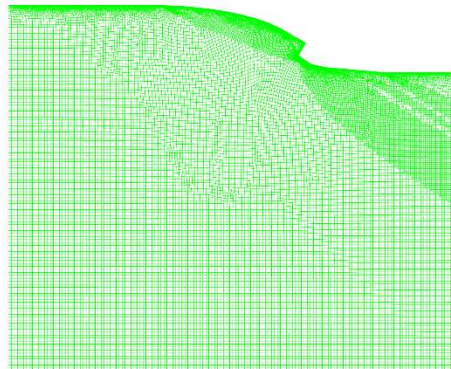
Ansys
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STUDENT

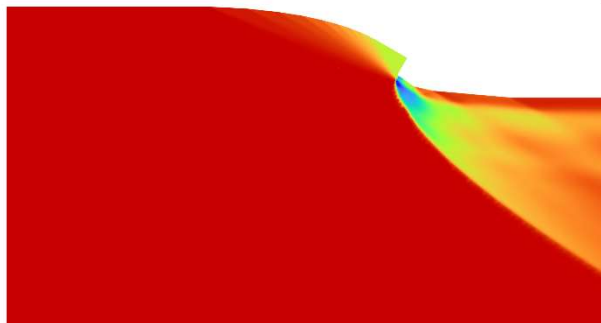
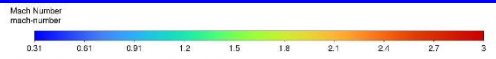
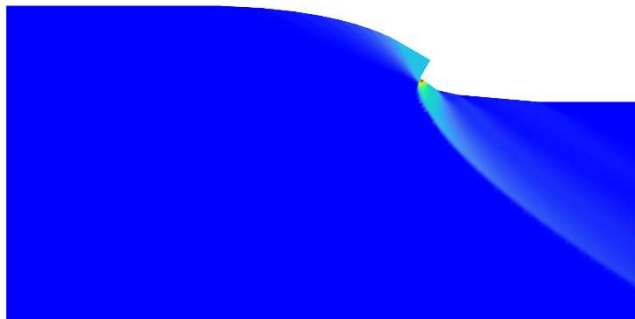
Force		Force [N]		Viscous		Total		Coefficients	
zone	Pressure	Viscous	Total	Pressure	Viscous	Pressure	Viscous	Pressure	Viscous
Viscous	(10634.529 22610.735 0)	(0 0 0)	(10634.529 22610.735 0)	(0.60924086 1.209345 0)	(0.95504938 2.2269086 0)	(10634.529 22610.735 0)	(0.60924086 1.209345 0)	(0.60924086 1.209345 0)	(0.95504938 2.2269086 0)
wall_cowl	(10634.529 22610.735 0)	(0 0 0)	(10634.529 22610.735 0)	(0.60924086 1.209345 0)	(0.95504938 2.2269086 0)	(10634.529 22610.735 0)	(0.60924086 1.209345 0)	(0.60924086 1.209345 0)	(0.95504938 2.2269086 0)
wall_intakeramp	(16670.746 38972.924 0)	(0 0 0)	(16670.746 38972.924 0)	(0.65504938 1.4642902 0)	(0.95504938 2.2269086 0)	(16670.746 38972.924 0)	(0.65504938 1.4642902 0)	(0.65504938 1.4642902 0)	(0.95504938 2.2269086 0)
Net	(27305.275 61483.659 0)	(0 0 0)	(27305.275 61483.659 0)	(1.5642902 3.5223336 0)	(1.5642902 3.5223336 0)	(27305.275 61483.659 0)	(1.5642902 3.5223336 0)	(1.5642902 3.5223336 0)	(1.5642902 3.5223336 0)

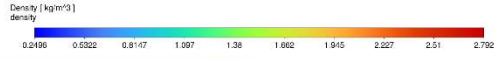
Force - Direction Vector (1 0 0)		Force [N]		Coefficients		
zone	Pressure	Viscous	Total	Pressure	Viscous	Total
wall_cowl	10634.529	0	10634.529	0.60924086	0	0.60924086
wall_intakeramp	16670.746	0	16670.746	0.65504938	0	0.65504938
Net	27305.275	0	27305.275	1.5642902	0	1.5642902

Area-Weighted Average Static Pressure [Pa]	
pressure_outlet_1_engine	117487.41
pressure_outlet_2_exit	15637.249
wall_cowl	33406.928
wall_intakeramp	50561.469
Net	31128.018

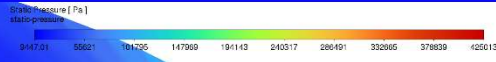
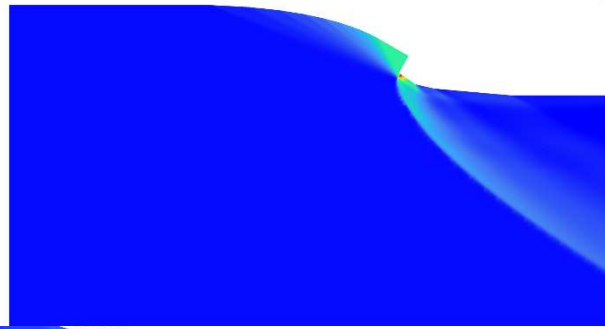
Area-Weighted Average Mach Number	
pressure_outlet_1_engine	1.7621088
pressure_outlet_2_exit	2.719706
wall_cowl	2.4775435
wall_intakeramp	2.3627205
Net	2.5477727

Area-Weighted Average Static Temperature [K]	
pressure_outlet_1_engine	372.86457
pressure_outlet_2_exit	246.68257
wall_cowl	275.5434
wall_intakeramp	290.19792
Net	267.6892

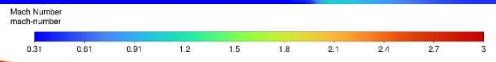
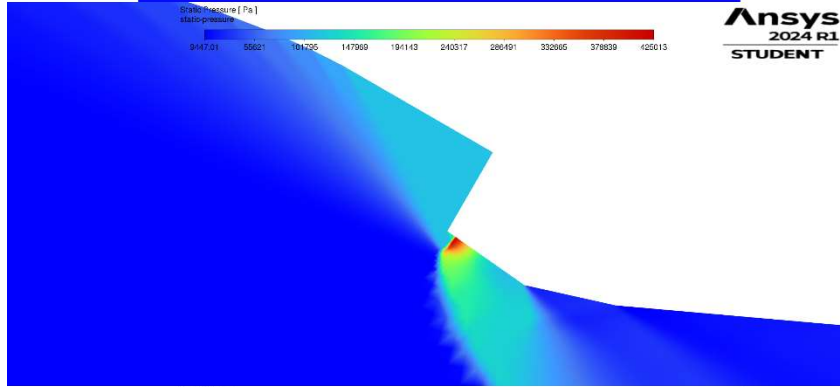




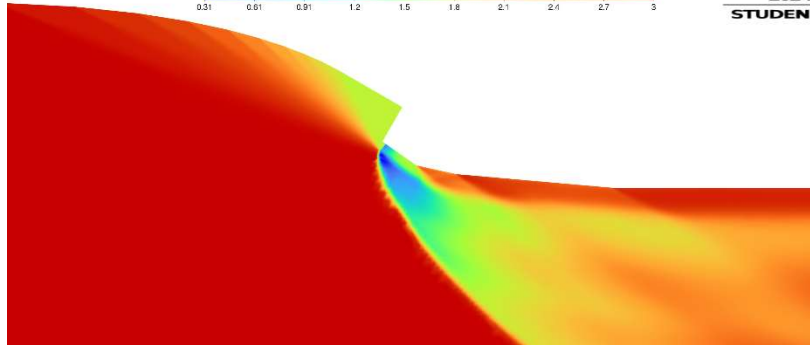
Ansys
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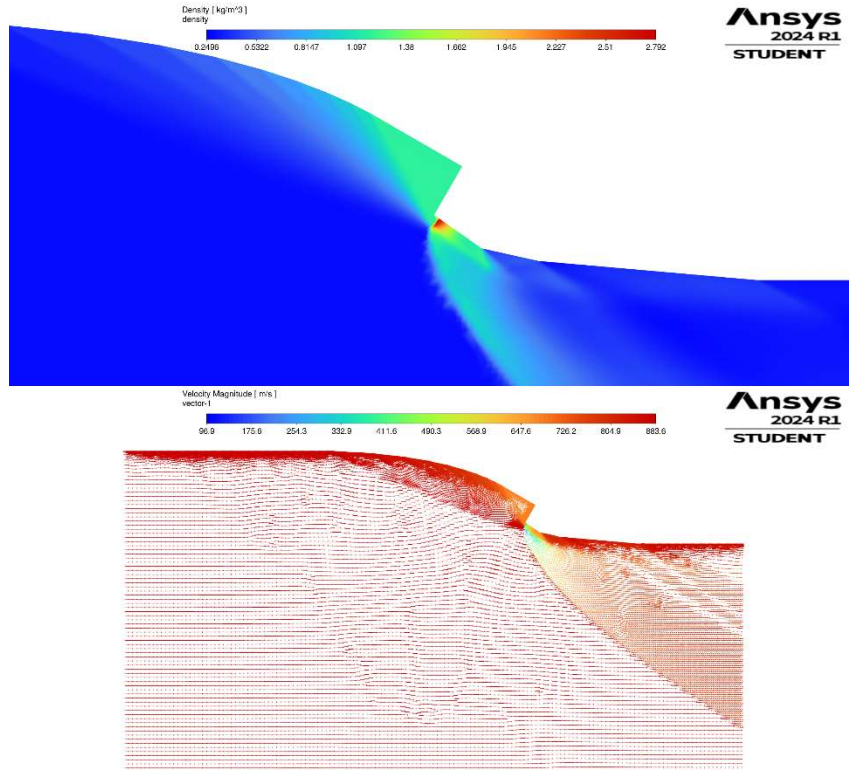


Ansys
2024 R1
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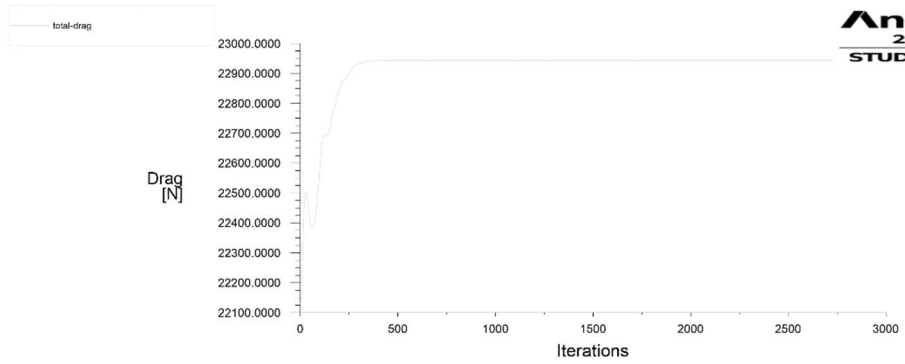
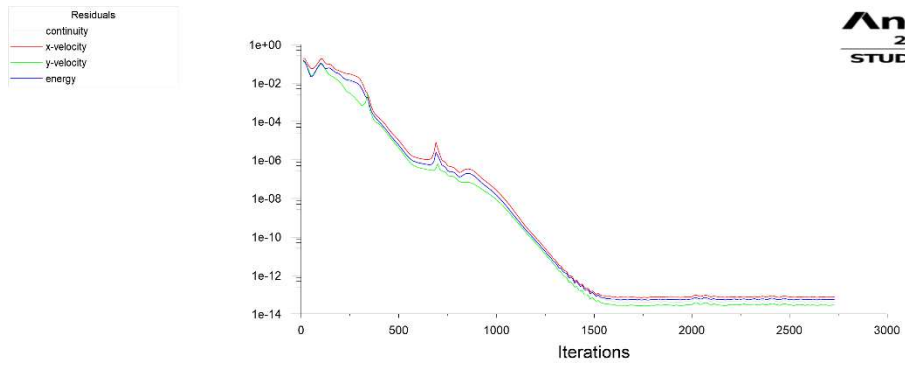


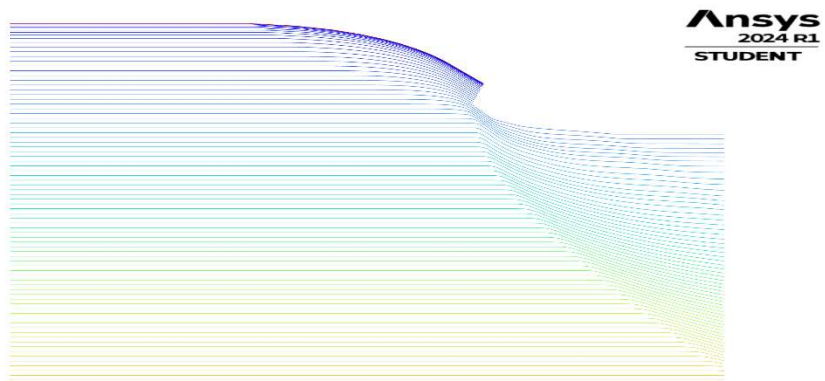
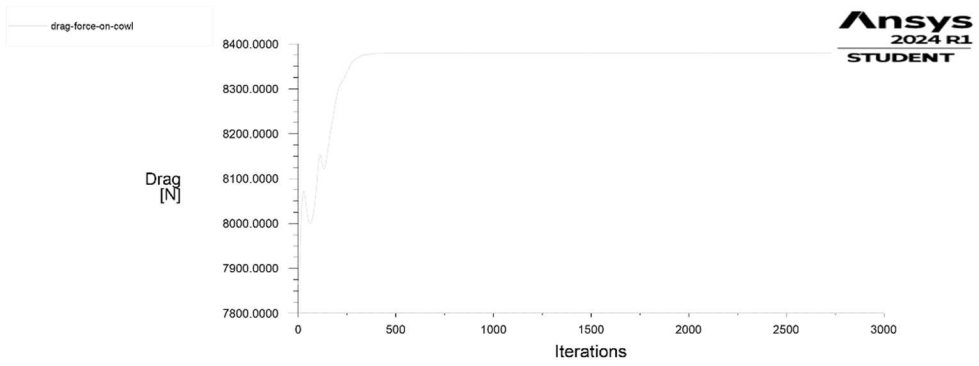
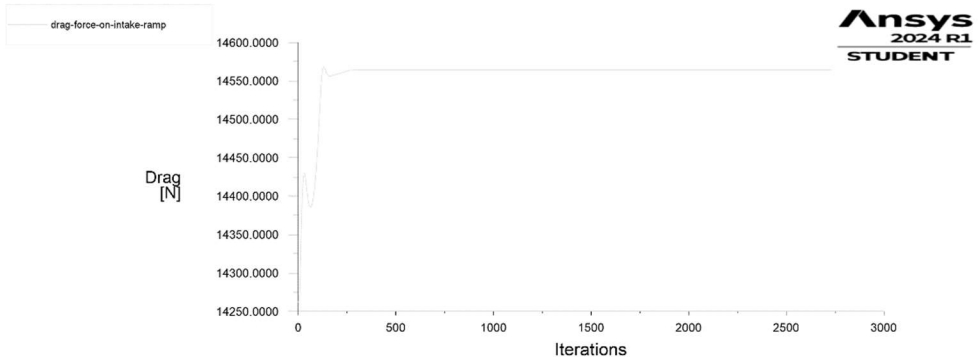
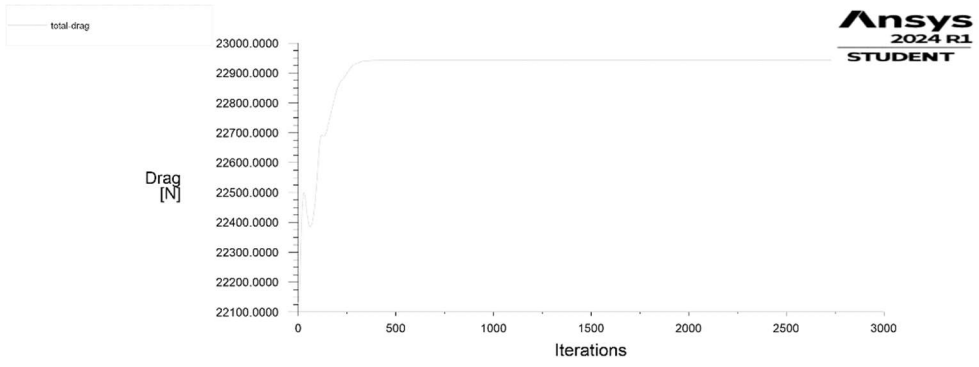
Ansys
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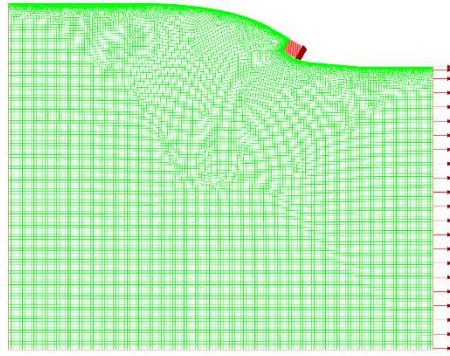




Inlet Mach 2.75 Simulation Data







```

Force
-----
Coefficient
Zone          Force [N]
-----
Pressure
Zone          Pressure          Viscous          Viscous          Total          Total
-----
Pressure
wall_cowl     (8379.1784 18479.814 0)    (0 0 0)          (0 0 0)          (0.48003424 1.0500005 0)    (8379.1784 18479.814 0)
(0.48003424 1.0500005 0)
wall_intakeramp (14564.105 34201.956 0)    (0 0 0)          (0 0 0)          (0.83436212 1.9093961 0)    (14564.105 34201.956 0)
(0.83436212 1.9093961 0)
-----
Net
(22943.283 52681.809 0)    (0 0 0)          (0 0 0)          (1.3143964 3.018085 0)    (22943.283 52681.809 0)
(1.3143964 3.018085 0)

Force - Directional Values (1 0 0)
-----
Zone          Force [N]          Coefficients
-----
Zone          Pressure          Viscous          Total          Pressure          Viscous          Total
-----
wall_cowl     8379.1784          0          8379.1784          0.48003424          0          0.48003424
wall_intakeramp 14564.105          0          14564.105          0.83436212          0          0.83436212
-----
Net          22943.283          0          22943.283          1.3143964          0          1.3143964

```

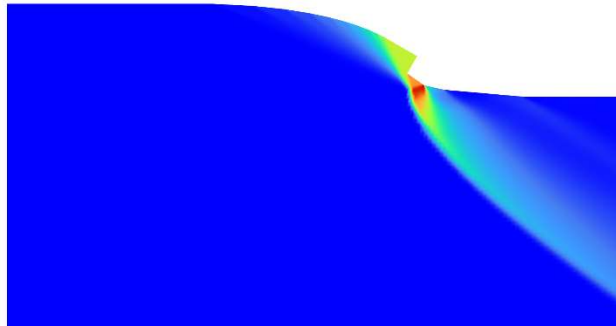
```

Area-Weighted Average
Static Temperature [K]
-----
pressure_outlet_1_engine 360.51204
pressure_outlet_2_exit   245.63525
wall_cowl                 262.90279
wall_intakeramp          284.379
-----
Net                       262.23692

Area-Weighted Average
Mach Number
-----
pressure_outlet_1_engine 1.5880436
pressure_outlet_2_exit   2.4787656
wall_cowl                 2.3330716
wall_intakeramp          2.15867
-----
Net                       2.3421685

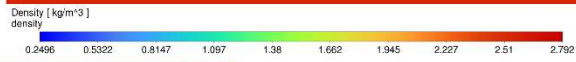
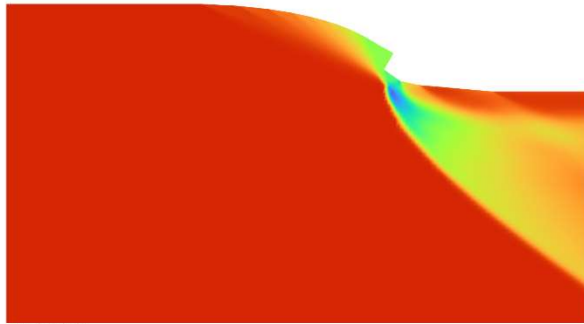
Area-Weighted Average
Static Pressure [Pa]
-----
pressure_outlet_1_engine 103108.7
pressure_outlet_2_exit   16272.158
wall_cowl                 28952.118
wall_intakeramp          45553.327
-----
Net                       28724.638

```

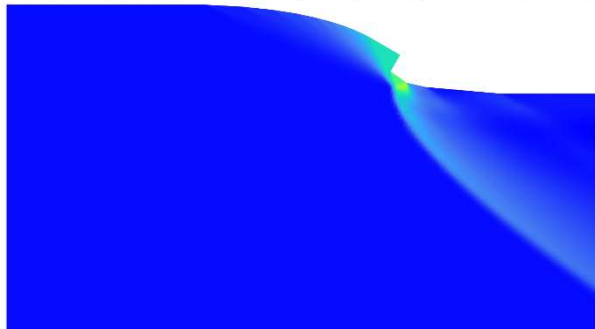




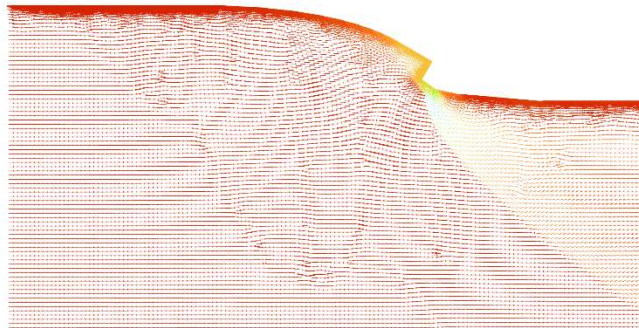
Ansys
2024 R1
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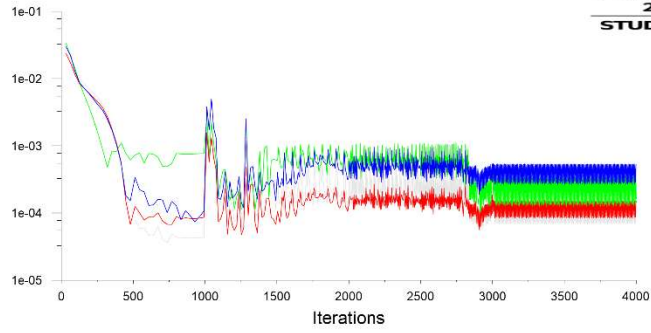
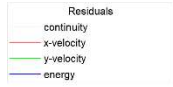
Ansys
2024 R1
STUDENT



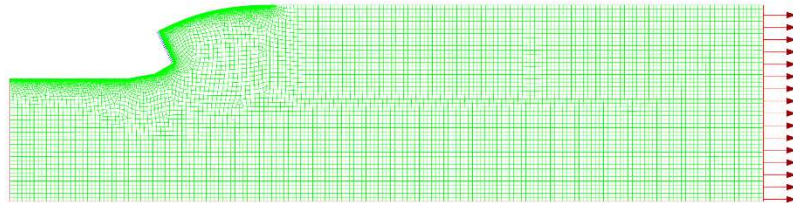
Ansys
2024 R1
STUDENT



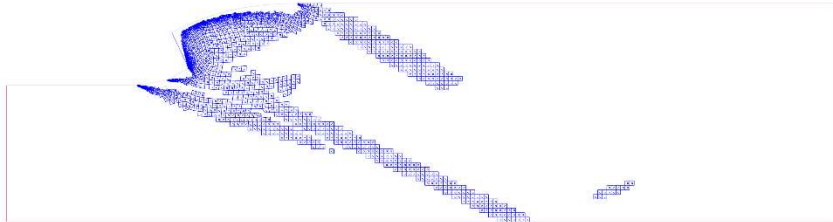
Outlet Mach 3.25 Simulation Data



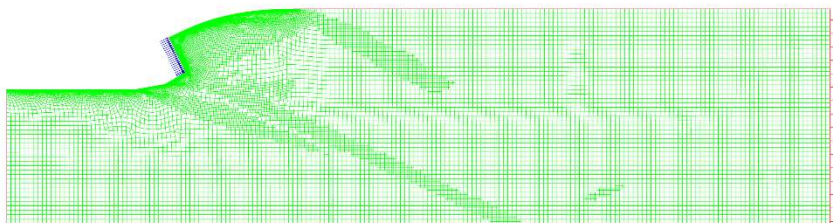
Ansys
2024 R1
STUDENT



Ansys
2024 R1
STUDENT



Ansys
2024 R1
STUDENT



Ansys
2024 R1
STUDENT

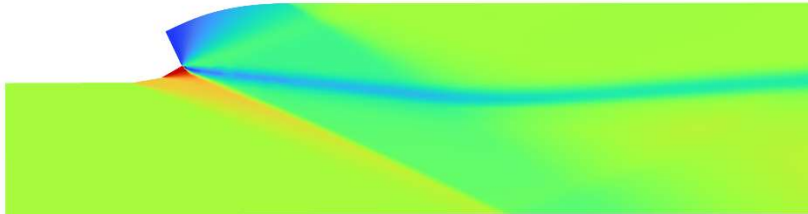
Static Pressure [Pa]
static-pressure
1539.9 19918.8 38297.7 56676.5 75055.4 93434.3 111813.2 130192.1 148570.9 166949.8 185328.7

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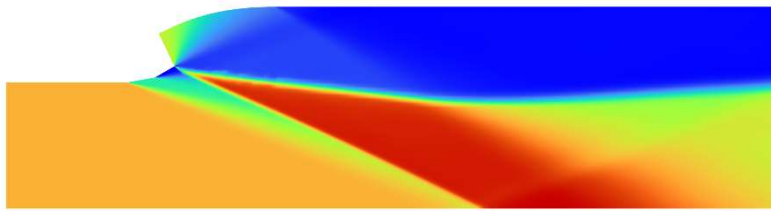
Mach Number
mach-number
0.7 1.1 1.5 1.9 2.2 2.6 3.0 3.4 3.7 4.1 4.5

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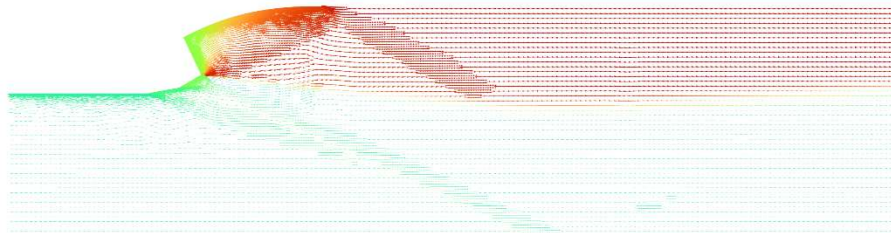
Density [kg/m^3]
density
0.0 0.1 0.1 0.1 0.2 0.2 0.2 0.3 0.3 0.3 0.4

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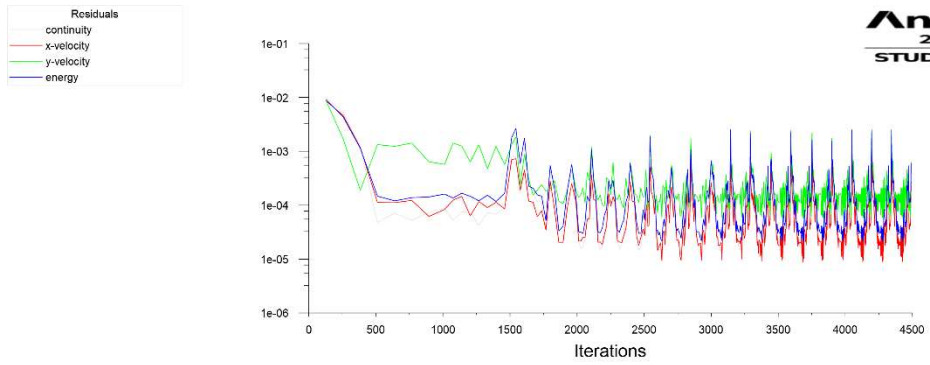


Velocity Magnitude [m/s]
velocity
232 410 588 767 945 1124 1302 1481 1659 1838 2016

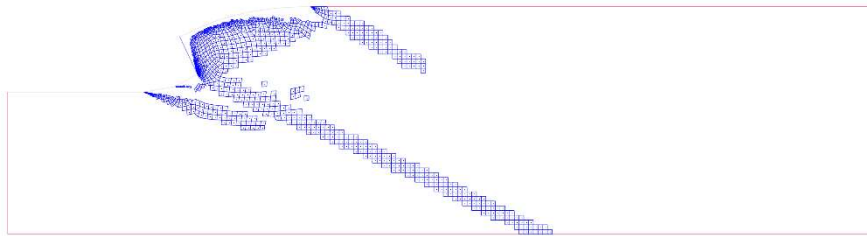
Ansys
2024 R1
STUDENT



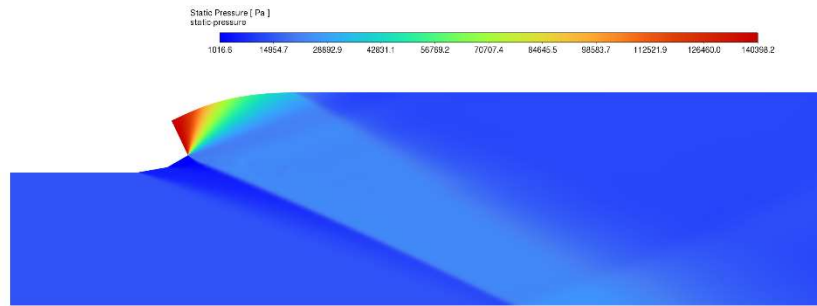
Outlet Mach 3 Simulation Data



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2024 R1
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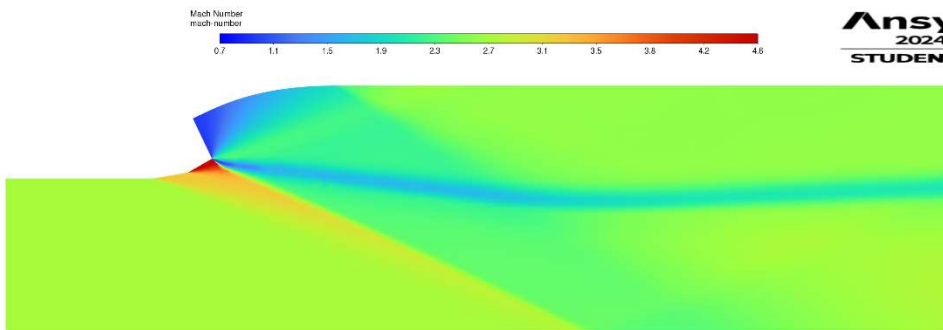


Ansys
2024 R1
STUDENT

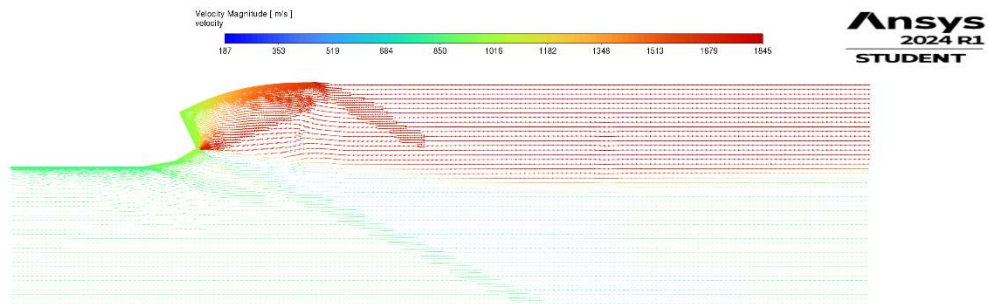
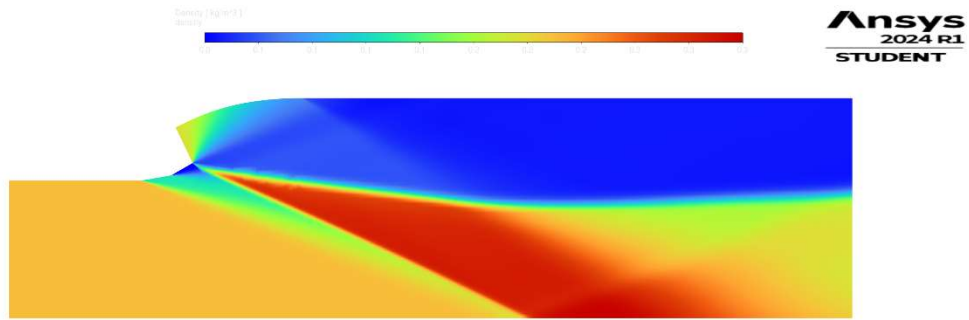


2

Ansys
2024 R1
STUDENT



Ansys
2024 R1
STUDENT



Outlet Mach 2.75 Simulation Data

