Design of an Inlet for a Vertically-Cruising Ramjet

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by

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Abstract

A ramjet engine is a supersonic airbreathing engine that can fly at Mach numbers of 3 to 6 by reducing the flow using the inlet geometry. Additional design modifications produce a scramjet engine that can achieve hypersonic flight as well as supersonic. The ramjet engine inlet slows down the supersonic freestream flow to a subsonic range which then allows the combustion process to occur within the engine. The inlet is a crucial part of the design of a ramjet engine because it must produce subsonic speeds without a large total pressure loss which would drive down the efficiency of the engine and unstart the engine. In practice, ramjet engines are designed using constant dynamic pressure paths which result in horizontal flight throughout the operational range of the ramjet engine. In this project, a vertical flight trajectory is considered with a constant velocity for a short operational range that falls below the usual operational range of Mach 3. One of the more attractive and possibly farfetched applications of vertically flying ramjet engines could be as a booster stage for payloads to space. Without the need for oxidizer, the vehicle could reallocate the weight towards the fuel or the payload. For the purposes of this project, a pitot inlet and a two-dimensional oblique shock wave inlet were considered using an initial Mach number of 2.56 to produce a static geometry. As expected, the oblique shock wave inlet results in less total pressure losses for the given flight Mach number range.

Acknowledgements

I would like to thank Dr. Vergine for his guidance throughout this project and for inspiring me to pursue a MS in aerodynamics through the many undergraduate aerodynamic and propulsion courses I thought I would never pass much less ace. His patience and attention to detail made aerothermodynamics and propulsion seem very logical and so simple at times. I have always enjoyed learning and have found through the years that great teachers and professors are often hard to come by and Dr. Vergine is one of them. I would like to thank Edgar Aguilar and Michael Sandoval for convincing me to get a MS degree when I thought I was just about done with school. I could not have achieved a bachelor's degree without all the study sessions which in turn made the graduate courses seem much more bearable. To my parents, thank you for all the unwavering support throughout the years in which I was pursuing my higher education, you are the biggest reason I have been able to achieve this milestone in my life and I am eternally grateful for everything you have done for me. Last but certainly not least, I would like to thank Colin for all his support including helping me debug code and allowing me to talk through my thought process for problem solving even though he had no idea about what I was trying to solve. Thank you for sticking with me through all of the mental breakdowns and the stress of working a full-time job while pursuing my master's, now it's my turn to return the favor.

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β	Shock wave angle (º)
ρ	Density (kg/m^3)
γ	Specific heat ratio
θ	Turn angle (º)
a	Speed of sound (m/s)
$\dot{m}$	Mass flow rate (kg/s)
q	Dynamic pressure (Pa)
A	Area (cm^2)
M	Mach number
P	Pressure (atm)
R	Gas constant (J/kgK)
T	Temperature (K)
V	Velocity (m/s)

1 Introduction

1.1 Motivation

Turbojet engine-based aircrafts are strongly limited in the range of speeds that they can achieve due to the inevitable degradation of performance of this class of engines with increasing Mach numbers. Above Mach 2, in fact, the difference between the maximum combustion temperature (that is imposed by the presence of a turbine downstream) and that of the air downstream of the compressor may become so small that not enough heat could be provided to the flow before being accelerated through the nozzle with a consequent loss in thrust and efficiency.

An option to circumvent this issue are ramjets, which are airbreathing engines able to achieve higher supersonic speeds compared to turbojets as they do not rely on a compressor to compress the incoming air, rather on the shock waves naturally forming over the body of the aircraft and in the inlet. The absence of a turbine upstream of the nozzle not only permits the achievement of much higher temperatures in the combustor [1] – with consequently higher amounts of heat to be delivered to the flow – but also allows for the whole added energy to be employed for propulsive purposes (in fact, in a turbojet part of the added energy is extracted by the turbine to move the compressor). The main issues of this class of airbreathing engines are the necessity to either be "launched" from other vehicles or to be used in a combined cycle and the total pressure losses introduced by the inevitable presence of shock waves in the inlet. As the total pressure loss is directly related to a loss in thrust, the inlet of the ramjet is arguably the most important component as it effectively determines whether the engine will be able to function.

Combustion in ramjets occurs at subsonic speeds, thus limiting their operational range typically between Mach 2 and 6. The reason for this is twofold: above Mach 6 1) air enters the combustor already partially dissociated making the combustion ineffective and 2) the total pressure losses introduced by the normal shock in the inlet would become unacceptable.

As previously mentioned, one of the shortcomings of ramjet engines is the inability to function below supersonic speeds unless it is part of a turbine-based combine cycle, therefore many machines (i.e., aircraft or weapons) powered by this technology need to be launched from another aircraft and typically follow a constant dynamic pressure path. Ramjets are typically designed to fly at low angles of attack and constant dynamic pressure to set lift and drag at a specific level (since lift and drag coefficients only marginally depend on the freestream Reynolds number and are, therefore, nearly constant [2]). If the dynamic pressure is too large the vehicle will not be able to sustain the loads and if the dynamic pressure is too small the vehicle will need to be larger to compensate [3].

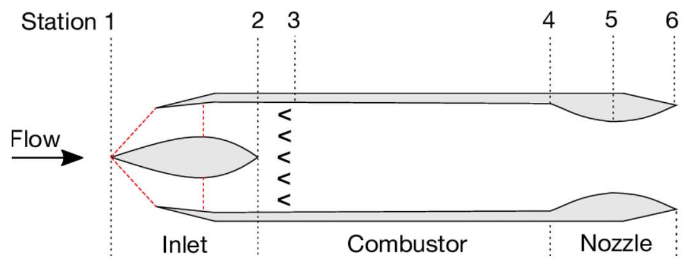

Figure 1.1: Notional schematic of a ramjet engine [4]

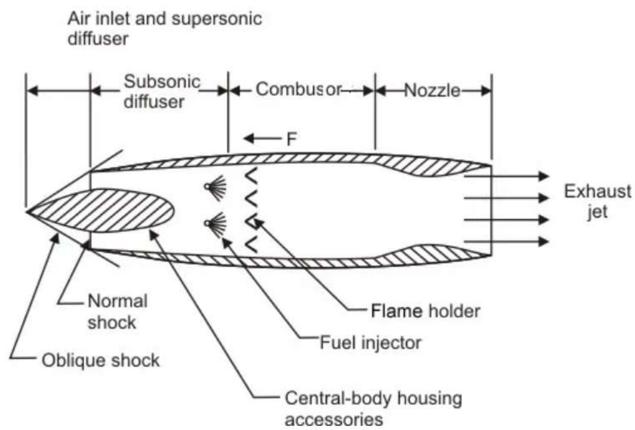

Figure 1.2: Components of a ramjet engine [5]

The focus of this project is to design a ramjet inlet that can match the combustion chamber requirements for a ranging altitude using a constant speed based on the vertical path. Airbreathing hypersonic engines, such as the ramjet, are highly desired vehicles because they do not need to carry oxidizer on board. The Saturn V rocket takeoff weight was 6 million lbf with 4 million lbf being the oxidizer weight and only 250,000 lbf being payload weight [3]. The weight of oxidizer can be reallocated for payload, fuel, and structure on airbreathing engines and would allow for a more thorough design at a smaller scale. The challenge in designing an airbreathing engine inlet for vertical flight lies in ensuring the engine has sufficient airflow going through during ascent considering the changes in air density as the altitude increases. There is a lot to gain from this research in terms of how vertical flight might be achieved for airbreathing hypersonic vehicles as well as any new findings of the challenges this entails.

1.2 Literature Review

Ramjets were first conceptualized in the early 1900s in Europe with the first designs being tested in the 1940s [6]. During World War II several countries, including the United States, Soviet Union and Germany, began to research and develop ramjets due to the lightweight and conceptually simple design which could be used for supersonic missiles. One of the first programs that was created to research and develop ramjet engines was the Bumblebee Program started by the United States Bureau of Ordnance [6]. The program developed and tested the Talos ramjet missile which was the second stage of a rocket. The first tests were conducted in 1945 using 6-in diameter ramjet vehicles to gather drag and aerodynamic performance telemetry. Eight months after the initial tests the demonstration of ramjet thrust higher than drag was conducted. The main roadblock of the first designs was the lack of stable combustion which resulted in the inability of the engines to operate at 60,000 feet at 2000 feet per second. To combat this problem the flameholder design was altered into a “can”-type flameholder which was based on turbojet technology. Following various tests on high and low altitude the design was implemented on the Talos combustion chambers successfully. The Talos engines were designed to operate in scenarios in which the shocks occur ahead of the cowl lip, subcritical, which is typical of engines that operate at rich fuel to air flow mixtures and results in maximum engine thrust [7].

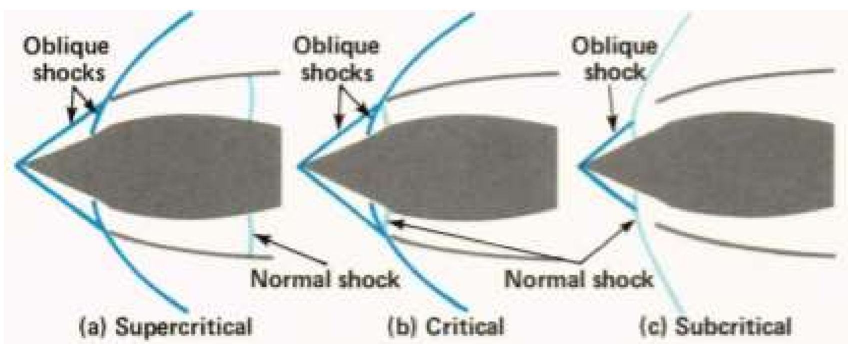

Figure 1.3: Demonstration of
a) Supercritical diffuser operation with normal shock downstream of the cowl lip
b) Critical diffuser operation with the normal shock near the cowl lip
c) Subcritical diffuser operation with the normal shock ahead of the cowl lip [7].

The tests on the inlet focused heavily on determining the limits of operation using wind tunnel tests and engine free jet tests. Of relevant importance was the redesign of the inlet cone from a single cone to a double cone to allow for maximum inlet capture area at high Mach numbers. The forward cone had a half angle of 25^o and a second half angle of 35^o. Overall, the Bumblebee program was able to successfully design and launch almost 1200 Talos ramjet engines with a 96% success rate and served as a basis for future designs [7].

Klaus Oswatitsch was a physicist that studied external compression intakes using multiple shocks. Oswatitsch found that there is a limit to internal compression above which the engine will not start, this is known as the Kantrowitz limit [8]. This phenomenon is explained in depth in [6] and summarized as follows, "shocks of a multi shock diffuser should have equal strength for maximum pressure. Due to the fact that entropy rise increases rapidly with the temperature ratio across a shock and if two shocks within a sequence were to have differing temperature ratios the change in entropy of the stronger shock would outweigh the change in the weaker shock when the flow is compressed to the same Mach number" [8]. To find the optimal cone angles Oswatitsch used upstream oblique shocks to find the normal component of the Mach number which is then used to find the freestream Mach that corresponds and therefore solve for the optimal pressure ratio across the shocks. At Mach numbers above three the flow is likely to separate the boundary layer therefore the second angle for the intake is steeper and leads to a steep cowl angle giving high drag [8]. Oswatitsch went on to define subcritical and supercritical operation as well as the noise from unstable subcritical operation, subsonic diffuser losses and the effect of boundary layer bleed and angle of attack which are all influenced by the inlet ramp angle. The flow before and after shocks can be analyzed using control volumes and initial conditions with the normal and oblique shock wave relations. This is of special importance in ramjet engines due to the requirement of subsonic flow with minimal aerodynamic losses in the combustion chamber which is best achieved isentropically. Taylor and Maccoll formulated the differential form of the equations describe and used by Adolf Busemann who specialized in supersonic flow and proved that internal compression inlets can use isentropic compression using one dimensional conical flow calculations with flow properties being only a function of the vertex angle [8]. The equations derived by Taylor and Maccoll are heavily used today to determine the flow past a cone to find the radial and angular velocity after a shock wave (Equation 1.1). Taylor and Maccoll defined a polar coordinate system at the point of the cone with the lines going through the point being in the radial direction and the angle theta being defined as the angle between the radial rays and the axis of the cone. The differential equation defined by Taylor and Maccoll was derived using the conservation of mass, momentum, and energy in terms of the velocity in the theta direction. To find the shock angle one must assume a value for it and solve the differential equation using numerical integration such as the Runge-Kutta scheme from the shock wave to the surface of the cone.

$$\frac{\gamma-1}{2}\left[1-V_r^2-\left(\frac{dV_r}{d\theta}\right)^2\right]\left[2V_r+\cot\theta\frac{dV_r}{d\theta}+\frac{d^2V_r}{d\theta^2}\right]-\frac{dV_r}{d\theta}\left[V_r\frac{dV_r}{d\theta}+\frac{dV_r}{d\theta}\frac{d^2V_r}{d\theta^2}\right] \tag{1.1}$$

where, $V_\theta=\frac{dV_r}{d\theta}$

The intake/inlet in hypersonic vehicles serve as the compression system for the engines by compressing the air flow to the pressure and temperature needed for combustion. Since the inlet is the first perturbance that the flow will encounter, it is important to determine how the flow will react within the inlet to ensure the downstream flow behaves as required. The inlet can determine whether the engine will start or unstart, resulting in subsonic flow before the combustion chamber which decreases the thrust and will require starting the engine. An engine unstart occurs when the mass flow rate through the inlet does not match up with the mass flow rate through the outlet which causes large differences in pressure and will push the shocks within the inlet out through the front causing the flow through the engine to be entirely subsonic. The unstarting of the engine can cause unstable flight conditions and compromise the maneuverability of the vehicle which can be detrimental. Arthur Kantrowitz studied the fluid dynamics of supersonic and hypersonic flow through nozzles and determined a theoretical concept for the contraction limit. The mass flow rate through a nozzle contracts as the nozzle converges which will cause the flow to slow down. When the flow reaches Mach 1 the flow chokes and will cause subsonic flow through the nozzle which is the unstarting condition for ramjet engines. Kantrowitz found that if the normal shock occurs after the throat of the inlet the flow is supersonic past the throat and will start the engine yet, when the normal shock moves forward ahead of the inlet the engine unstarts [9]. Through this finding, the contraction ratio was shown to be dependent on the operational Mach number. The Kantrowitz is defined in Equation 2 and will be used in this project to aid in determining the ratio of areas needed for the inlet based on the range of Mach numbers that can be achieved by the first stage [10].

$$\frac{A}{A^*} = \left[\frac{(\gamma+1)M^2}{(\gamma-1)M^2+2}\right]^{0.5} \left[\frac{(\gamma+1)M^2}{2\gamma M^2-(\gamma-1)}\right]^{\frac{1}{\gamma-1}} \tag{1.2}$$

An important factor in determining how much thrust and how efficient the engine is will be the pressure and how well it is kept through the engine. As stated previously, the main goal of the inlet is to ensure the desired combustion pressure is reached without significant pressure losses. Through a shock wave the total pressure decreases and therefore the thrust decreases. Research done at the NASA Langley Mach 4 Blow Down Facility used a two-dimensional fixed geometry nozzle with initial ramp angles of 5°, 8°, and 12° and a moveable cowl that was adjustable to -8° into the inlet and 6° away from the inlet, all measured relative to the freestream (Figure 1.4) [11]. The tests involved using a throttling device to increase the back pressure of the inlet until the engine unstart occurred at several cowl angles as well as increasing the cowl angle during a test until unstart. An important note is that to start the engines the contraction ratio was decreased by reducing the cowl angle and later increased once started. This is how many ramjets are designed to increase the operational range. Emami et al. [12] conducted a test for a two-dimensional planar cowl length test using three different cowl lengths for the same varying cowl angles where it was found that all three inlet tests unstarted at roughly the same convergence cowl angle. This demonstrated the independence of the cowl length, mass flow capture and contraction ratio and indicated the shock boundary layer interactions that will cause the boundary layer to separate and push the shock out of the inlet. For this reason, Rodi et al. [11] tested cowl angles without varying the cowl length.

The mean and time accurate values for pressure were obtained and analyzed for the operations right before engine unstart and directly after. Figure 1.5 demonstrates the mean pressure readings along the length of the engine and shows that before the engine unstart (lines 1-4) the pressure begins to move upstream as the back pressure increases with the maximum back pressure being line 4. After the unstart the pressure increase is seen upstream of the cowl meaning the shock is detached and ahead of the cowl. It was found that the inlet contraction ratio had a major impact on the time-accurate and mean pressure measurements at maximum back pressure and the unstarted inlets redistributed the pressure field. The unstart of the inlet causes a high-pressure wave to move up through the inlet which can be detrimental when the vehicle is hypersonic and causes vibrations to occur through the engine [11].

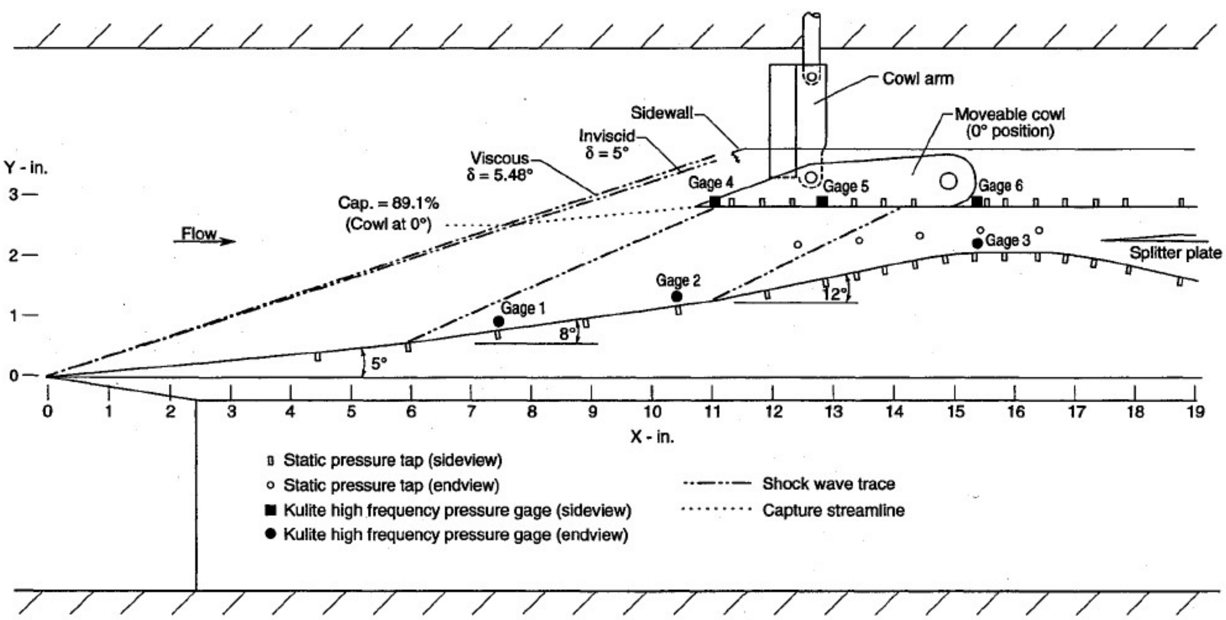

Figure 1.4: Ramjet/Scramjet inlet used in the pressure behavior experiments done by [11]

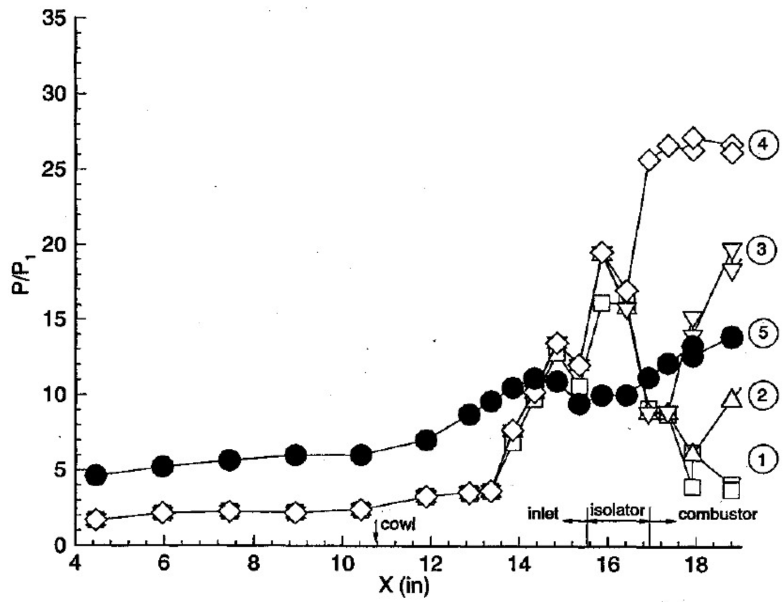

Figure 1.5: Mean pressure distribution along the length of the engine for 6° cowl angle before and after engine unstart [11]

The flow within a supersonic and hypersonic vehicle is often difficult to understand and calculate due to the many interactions between shocks, boundary layers, and expansion waves. It is crucial to understand the effects these interactions will have on the flow and how to design for them. Coratekin et al. [13] used turbulence models compared with experimental data to show whether two-dimensional inlet flows have turbulent effects as well as to investigate the asymmetric crossing shock interaction at Mach 3.95. Using a Mach 3 two ramp two-dimensional inlet, they found that two shocks occur after each corner and merge below the cowl lip Figure 1.6. Once inside the inlet the walls create shock wave and expansion wave patterns which is to be expected for supersonic flows. The flow is clearly turbulent due to the interaction of the boundary layer with the shock and expansion waves and can be seen in the pressure deviations along the walls of the engine Figure 1.7. The pressure increase is seen along the lip wall due to the double shocks ahead of the flow with less of the expansion wave interaction affect the region. The upper wall has a lower pressure distribution due to the shock boundary layer interaction that causes a small separation region. The three turbulence models shown are commonly used to study the aerodynamics associated with supersonic and hypersonic flow but are not entirely accurate for all interactions as can be seen in Figure 1.7. The study is clear example of the complexity within supersonic inlet flows and how quickly the flow becomes turbulent.

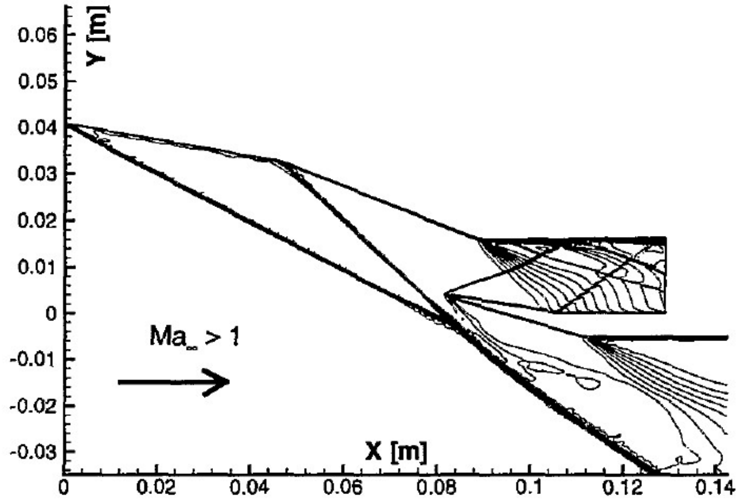

Figure 1.6: Mach isolines for double ramp two-dimensional inlet with freestream Mach 3 [13]

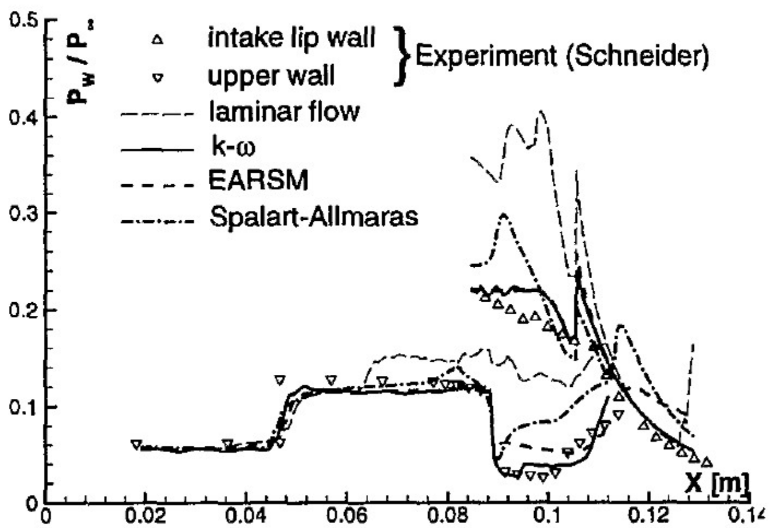

Figure 1.7: Surface pressure distribution along the double ramp inlet [13]

There are many ways to optimize ramjet inlets by making the geometry vary. As mentioned before, one way is to vary the cowl angle based on the freestream Mach number, another method of optimization which has proved to be promising is boundary-layer bleed. Boundary-layer bleed is achieved by creating openings along the walls of the inlet to allow the boundary layer to be pushed through a separate area while the flow proceeds through the inlet as designed. The benefit of a boundary-layer bleed system is an increased performance of the inlet and better stability of the pressure recovery which in turn makes the engine unstart occur at a higher operational range. Herrman et al. [14] found that by using a boundary-layer bleed system for a Mach 3 flow the pressure recovered 12.6% with the throttling stability point improving 15.9% and a mass flow decrease of 3.5%. It is important to note that despite these findings the flow internal to the inlet is very unsteady and the boundary-layer separation and shock wave interactions along with the back pressure can affect how efficient the boundary-layer bleed system is. The position chosen for the boundary-layer bleed entrance should be designed for the operational range due to the positioning of the shocks ahead of the entrance. By increasing the bleed entrance, the mass flow ratio can be varied and by increasing the bleed exit the pressure recovery increases and therefore improves the stability. Studies on boundary-layer bleed along with other optimization techniques for ramjet inlets will inevitably help prolong the operational range of ramjets yet the complexity of the turbulent flow is a big challenge to overcome.

1.3 Project Proposal

Hypersonic flight achieved with engines such as a ramjet engine is an efficient and cost-effective way of achieving rapid flight and because of this, research into how they can be adapted for vertical flight is attractive for space exploration. This project is carried out in collaboration with Engineering Space and will design a ramjet inlet for a vertical flight path using a constant speed as an initial condition and a set geometry. There are two primary forms of ramjet inlets, conical and square, the output of this project will compare the design of a conical inlet with a square inlet to determine which is best given the initial conditions.

1.4 Methodology

For this project, the goal is to design the inlet section for a ramjet that flies vertically at a constant speed. The first objective is to determine the interactions within the flow that will cause the engine to unstart such as having a normal shock before the throat and boundary layer interactions that will alter the intended geometry. To achieve this, Matlab will be used to find the flow parameters at various points within the inlet based on the design angles and shock wave equations. Using initial conditions, the code will output an inlet geometry based on whether the user wants an axis symmetric inlet or a 2D (square) inlet.

2 Preliminary Design Calculations

2.1 Pitot Inlet Design

A pitot intake is a circularized entrance in which air is designed to flow in and compress, in this case from supersonic to subsonic speeds. The pitot inlet generates a normal shock to decelerate the flow and to achieve the desired combustion chamber pressure while allowing for starting of the engine by ensuring that the shock occurs in the divergent section of the intake. Therefore, the area of the throat must be large enough to allow the supersonic flow to pass through, yet, small enough to ensure there is a shock in the divergent section. To calculate the throat area the Kantrowitz limit is used to determine the maximum area ratio, from the inlet to the throat, that a diffuser can have for a supersonic starting inlet.

The Kantrowitz limit is derived using a constant mass flow rate through the inlet and assuming a normal shock right after the inlet but before the throat, State 3 in Figure 2.1. The normal shock occurs at an area ahead of the throat such that the flow at the throat is sonic. Using the mass flow rate as constant through the inlet and the normal shock relations the equation for the area ratio becomes a function of the specific heat ratio and the freestream Mach number shown in Equation 6. For stable supersonic conditions, the normal shock must occur past the throat where the normal shock gives a minimum Mach number resulting in minimum pressure losses and resulting in higher engine efficiency, State 4 in Figure 2.1. The limit is a bounding case in which the engine will unstart with the normal shock ahead of the throat due to a large pressure loss from the freestream to the combustion chamber. The Kantrowitz limit is found and the area ratio is decreased by increasing the throat area to ensure the throat is large enough to allow for the supersonic flow to go through the throat and push the normal shock into the divergent section. In doing this, the flow becomes more stable and is less likely to unstart once in this condition. The calculations and findings are summarized below.

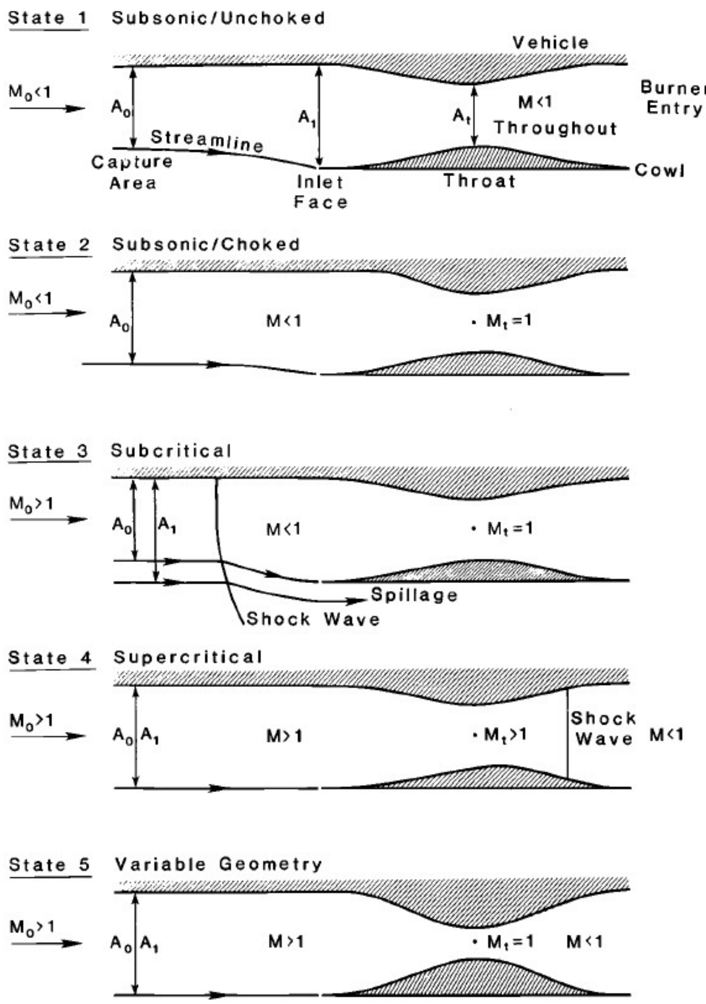

Figure 2.1: Pitot-Intake starting process

2.2 Preliminary Assumptions

The initial feasibility analysis for the geometry of the inlet involved several assumptions to calculate a simple fixed geometry that works for the Mach number range that is desired. For the freestream flow calculations, the atmospheric model based on the 1976 Standard Atmosphere was used. Using a given altitude range the static temperature, pressure, density, speed of sound, and viscosity are found in SI units. The incoming flow and the flow going through the inlet into the combustion chamber is calorically perfect and therefore the specific heat ratio is constant through the inlet. This is a reasonable assumption considering the temperature does not increase to the point where any chemical disassociation would take place that would cause the specific heat ratio to change. The altitude range that is used for this project is 5,500 meters – 20,000 meters based on preliminary calculations.

The main output needed for the design is the cross-sectional area of the inlet and the cross-sectional area of the throat based on given static and total pressures imposed by the flow in the combustor and considering the normal shock that needs to occur in the divergent section for the inlet to start. The first consideration when running a preliminary analysis is whether the ramjet should be designed for a constant dynamic pressure or a constant velocity flow path. Traditionally, ramjets are designed using constant dynamic pressure paths because these allow for aerodynamic forces to be nearly constant and therefore, predictable throughout the flight envelope. Unfortunately, to keep a constant dynamic pressure the flight Mach number must increase as the pressure decreases in the upper layers of the atmosphere. Within the engine, the pressure will then increase greatly to the point of reaching values that could greatly compromise the structural integrity of the aircraft. Considering the vertical flight path for this mission, the altitude range must be constrained using a constant velocity path as the values obtained for freestream Mach number and combustion chamber pressure would become too high reaching 800,000 Pascals (7.9 atm) at 10,000 kilometers of altitude using a constant dynamic pressure path.

The cross-sectional area was set based on the rocket booster which will likely have a six-inch diameter, thus the ramjet's size must be compatible with it. Preliminary calculations conducted by Dr. Vergine determine the flow parameters from the nozzle to the diffuser's exit and are bound by a desirable range of combustion pressure determined a compatible size of the inlet area. Using a desired exit Mach number of approximately 2 and a set nozzle area ratio, the flow parameters were calculated starting at the nozzle exit and going upstream into the combustor assuming a constant speed path. Using Rayleigh flow calculations in the combustor and setting the equivalence ratio for the fuel to air mixture to 0.6 and the combustor area to 12.5 by 12.5 cm^2, the pressure at the entrance of the combustor is found. This pressure would be considered the back pressure needed to determine where the normal shock will occur in the diffuser and was found to be within the range of 2 atm to 10 atm.

2.3 Freestream Conditions

The freestream pressure and temperature were found using the 1976 atmospheric model and are plotted below.

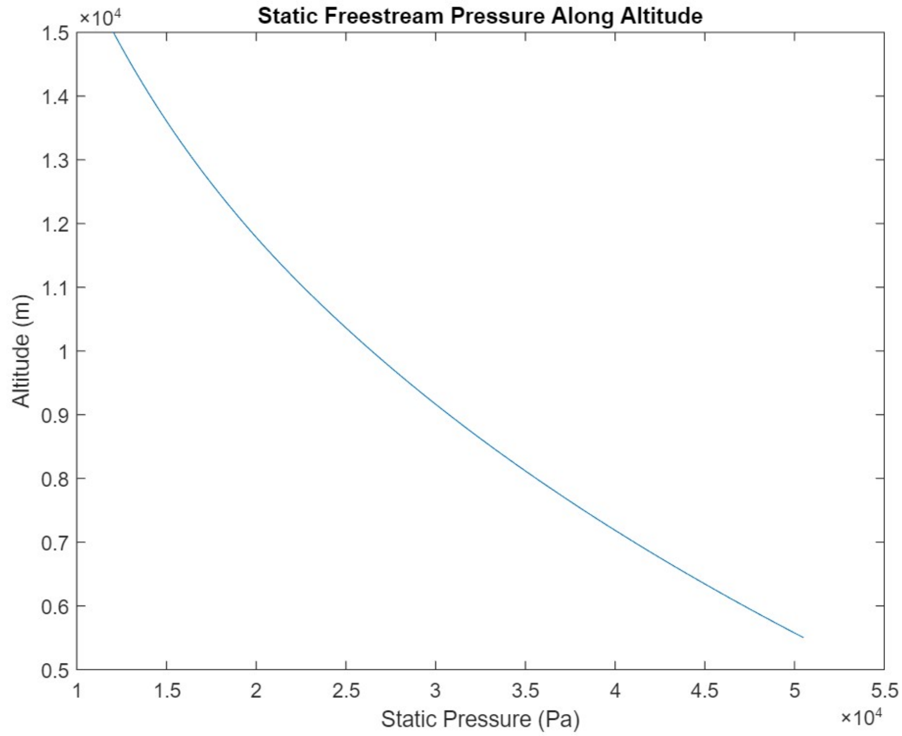

Figure 2.2: Static freestream pressure along altitude range

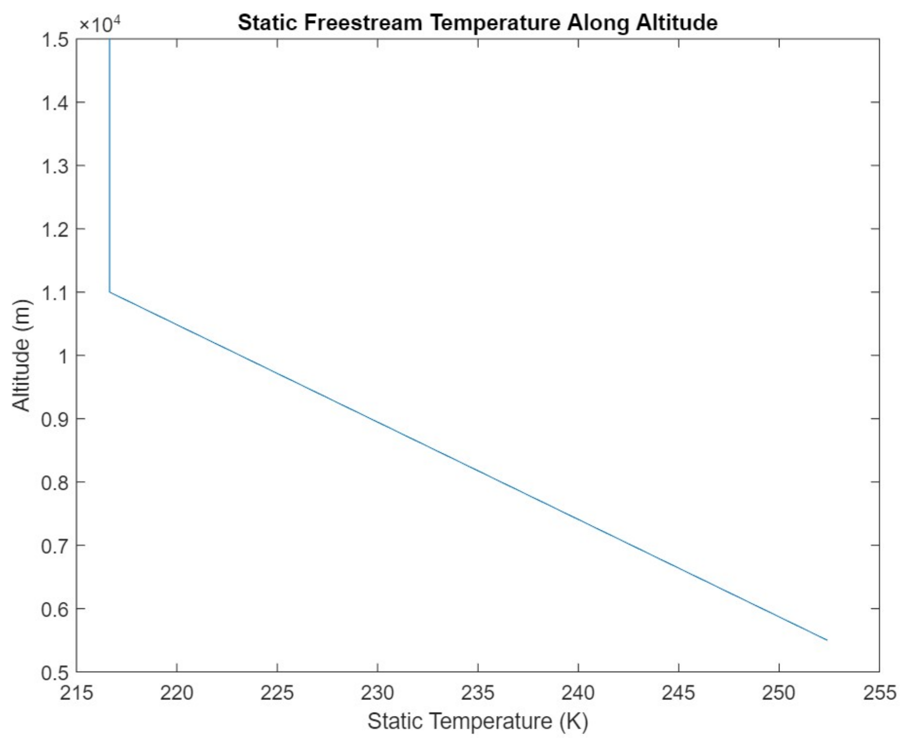

Figure 2.3: Static freestream temperature along altitude range

The freestream speed of sound a, mass flow rate, $\dot{m}$, and the dynamic pressure q (See Equation 2.1-2.3), were found using 1.4 as the specific heat ratio γ, 287 J/(kg·K) as the air gas constant R, and the freestream Mach number M, within a desired range given as an input.

$$\boldsymbol{a = \sqrt{(\gamma RT)}} \tag{2.1}$$

$$\boldsymbol{\dot{m} = \rho MA\sqrt{(\gamma RT)}} \tag{2.2}$$

$$\boldsymbol{q = \frac{1}{2}M^2P\gamma} \tag{2.3}$$

Starting from an initial flight condition at an altitude of 5500m and a Mach number of 2.56, the following plots show the freestream Mach number, mass flow rate and dynamic pressure as they vary with altitude with a constant velocity.

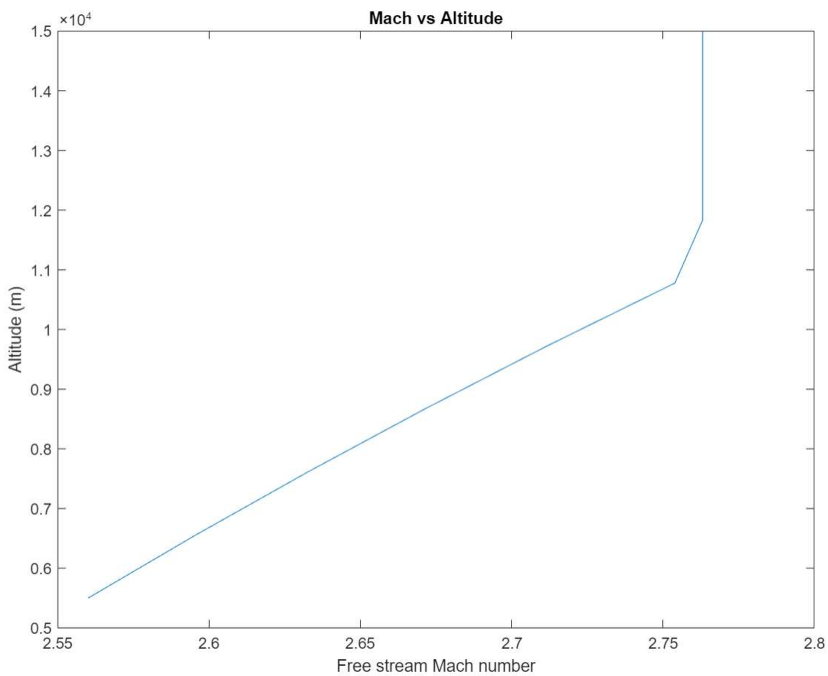

Figure 2.4: Freestream Mach number along altitude range

The Mach number is found by finding the velocity and altitude at which the rocket will reach maximum and calculating for the speed of sound along the altitude using the constant velocity. The constant Mach number after an altitude of around 11 km is a result of the constancy of the temperature in the tropopause.

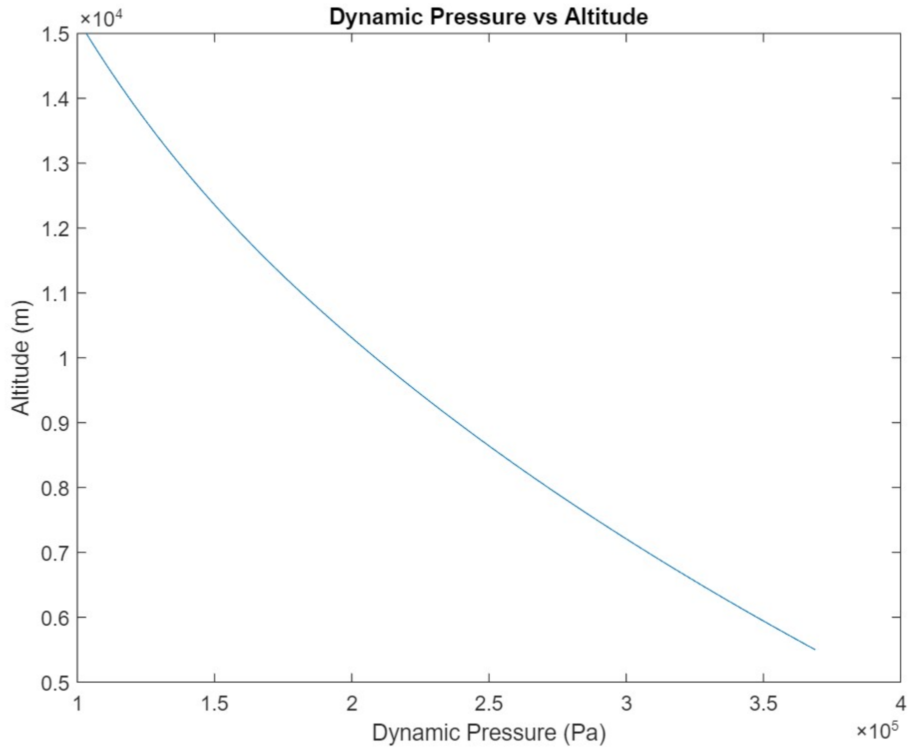

Figure 2.5: Dynamic pressure along altitude range

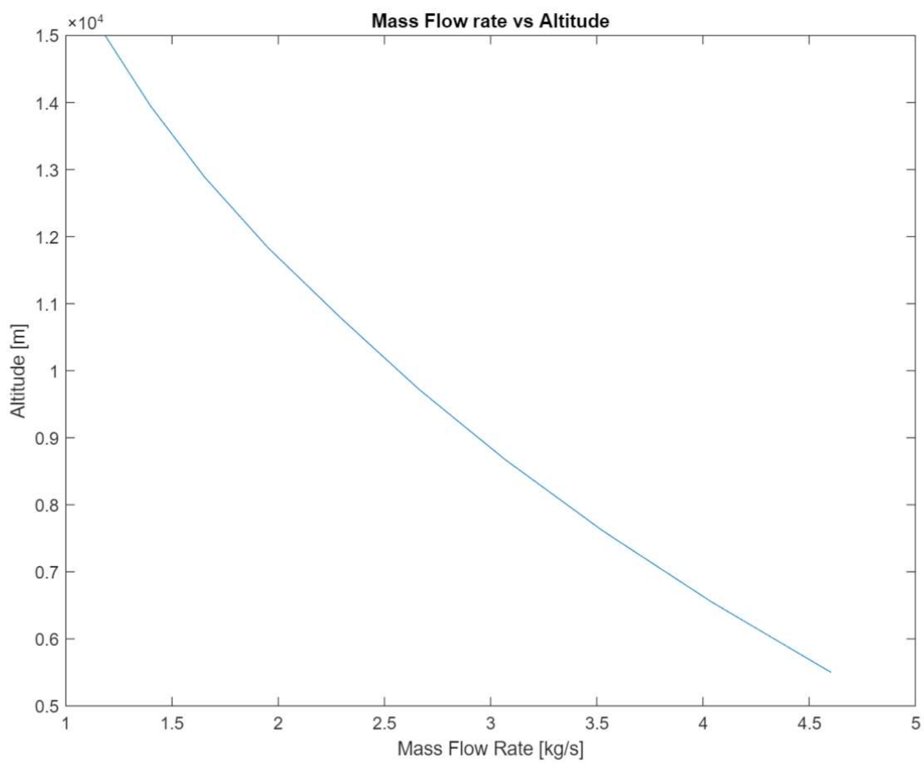

Figure 2.6: Mass flow rate along altitude range

When a constant dynamic pressure flight path is assumed the freestream Mach number and pressure range within the engine are higher as shown in Figure 2.7 and 2.8.

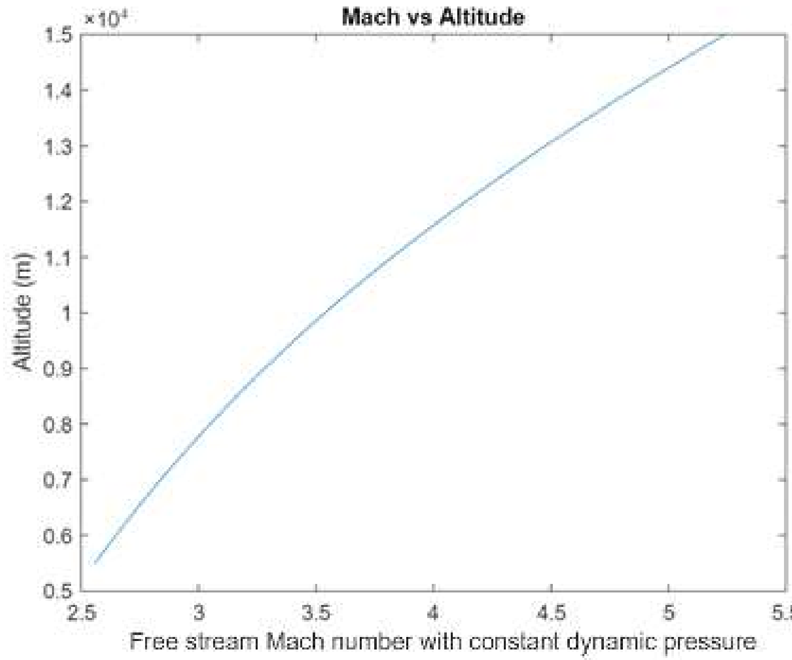

Figure 2.7: Mach number range when constant dynamic pressure is kept constant

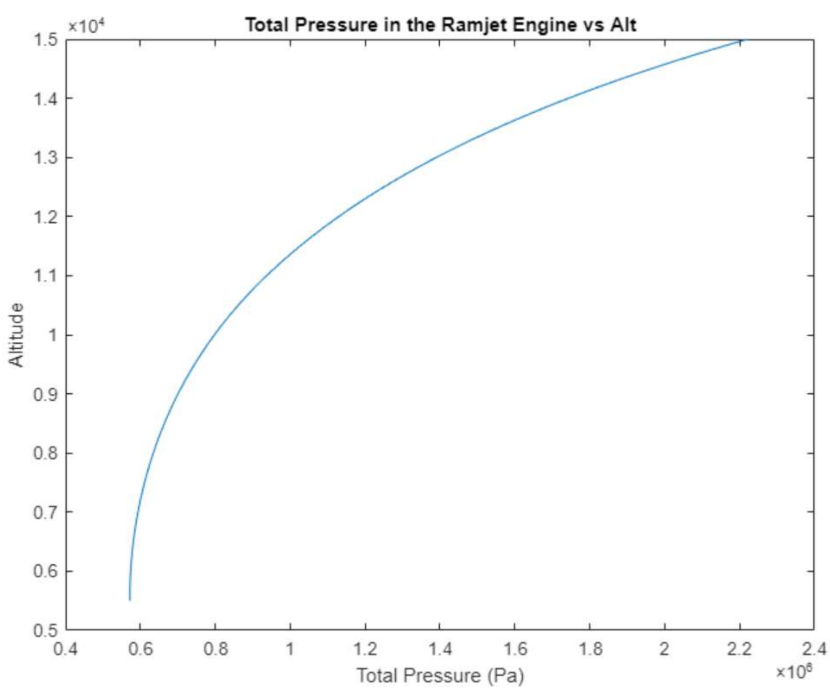

Figure 2.8: Total pressure through engine when dynamic pressure is kept constant

In Figure 2.8, the pressure within the engine increases as the vehicle ascends through the atmosphere which is undesirable for a static geometry design since the pressure in the combustion chamber will likely be too high for optimal combustion and will cause large pressure losses. Ramjet engines that use a constant dynamic pressure flight path fly along a horizontal trajectory to help keep the freestream values that vary with altitude relatively constant. The density, pressure and temperature vary with altitude therefore, keeping the dynamic pressure as a constant might also require a dynamic geometry that can change with altitude to increase the mass flow rate within the engine and therefore to better control the pressure and temperature within the engine. This is done by creating a moveable mechanism within the inlet that can increase the throat area as the Mach number increases in traditional ramjet engines. Since this project only considers a static geometry, this is not an option to reduce the effects of the constant dynamic pressure flight path meaning the range in which a vertically flying ramjet of this size would not operate for very long if at all.

2.4 Kantrowitz Limit Calculations

The Kantrowitz limit determines the maximum area ratio considering a configuration in which supersonic flow will go sonic at the throat given an initial Mach number. The limit is found using Equation 2.4 which is derived using constant mass flow rate through the inlet and assuming a normal shock right at the initial inlet area. The Kantrowitz limit acts as the minimum throat area that will create choked flow for a given freestream Mach number allowing for the normal shock to be swallowed in the divergent section. Therefore, to ensure starting of the inlet, the throat area must be increased to decrease the area ratio based on the operational Mach number range.

$$\frac{A_t}{A_0} = M_0\left(\frac{\gamma+1}{2+(\gamma-1){M_0}^2}\right)^{\frac{\gamma+1}{2(\gamma-1)}}\left[\frac{(\gamma+1){M_0}^2}{(\gamma-1){M_0}^2+2}\right]^{\frac{-\gamma}{\gamma-1}}\left[\frac{\gamma+1}{2\gamma {M_0}^2-(\gamma-1)}\right]^{\frac{-1}{\gamma-1}} \tag{2.4}$$

There is a simpler method of determining the minimum area at the throat for the diffuser in which the area ratio for the inlet Mach number is used with the total pressure ratio which is found by assuming a normal shock using the inlet Mach number shown in Equation 2.9 [2].

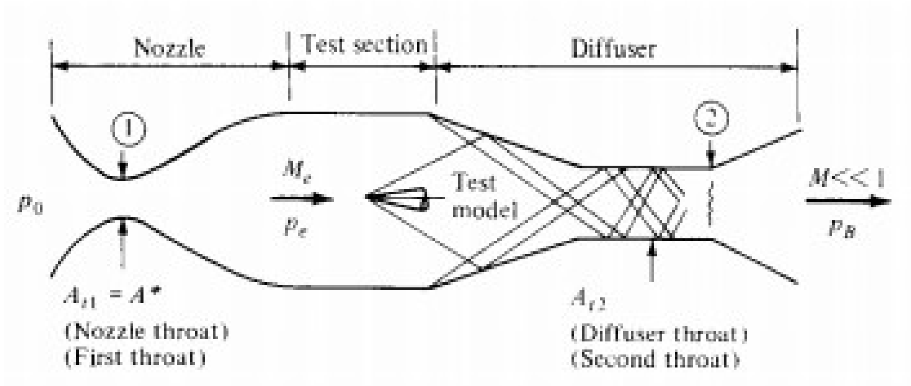

Figure 2.9: Supersonic wind tunnel

This relation was derived for supersonic wind tunnels assuming the critical area to be a throat area ahead of the test section, shown as first throat in Figure 2.9. The convergent-divergent nozzle generates the supersonic flow that feeds into the test section which holds the test model, in our case it would be the ramjet engine. The mass flow rate, $\dot{m}$, through the wind tunnel is constant for steady flow therefore, the critical mass flow rate in the first throat is equal to the mass flow rate through the second throat (Equation 2.5). Assuming choked flow at both throat areas gives the ratio shown in Equation 2.6, since the critical speed of sound, a^*, is constant for adiabatic flow. The critical temperature is also constant through adiabatic flow of a calorically perfect gas, giving the relation in Equation 2.7. Lastly, the critical pressures lead to a total pressure ratio through Equation 2.8 which leads to the simplified relation in Equation 2.9, where $A_{t,1}$ is the critical area of the flow, A^*.

$$\boldsymbol{\rho_1^* a_1^* A_{t,1} = \rho_2 u_2 A_{t,2}} \tag{2.5}$$

$$\frac{A_{t,2}}{A_{t,1}} = \frac{\rho_1^*}{\rho_2^*} \tag{2.6}$$

$$\frac{\rho_1^*}{\rho_2^*} = \frac{p_1^*/RT_1^*}{p_2^*/RT_2^*} = \frac{p_1^*}{p_2^*} \tag{2.7}$$

$$p_1^* = p_{0,1}\left(\frac{2}{\gamma+1}\right)^{\gamma/(\gamma-1)}$$
$$p_2^* = p_{0,2}\left(\frac{2}{\gamma+1}\right)^{\gamma/(\gamma-1)} \tag{2.8}$$

$$\frac{A_{t,2}}{A_{t,1}} = \frac{P_{0,1}}{P_{0,2}} \tag{2.9}$$

These two methods of determining the throat area have identical results as can be seen in Figure 2.10. The area that is found using the pressure ratio relation matches the Kantrowitz limit method exactly. The beginning of the trajectory is the most important point for a static design since the engine will need to start to remain in flight after this point. An option for ensuring an engine starts is doing an overspeed maneuver to push the normal shock that occurs in the inlet into the divergent section and then slightly decreasing the speed to the design range. This is not feasible for this project since the first stage rocket can only carry the engine up to the initial Mach number that the engine is designed for. Therefore, to account for this, the inlet and more importantly the throat area can be slightly undersized to create an overspeed maneuver by designing lower than the actual flight range.

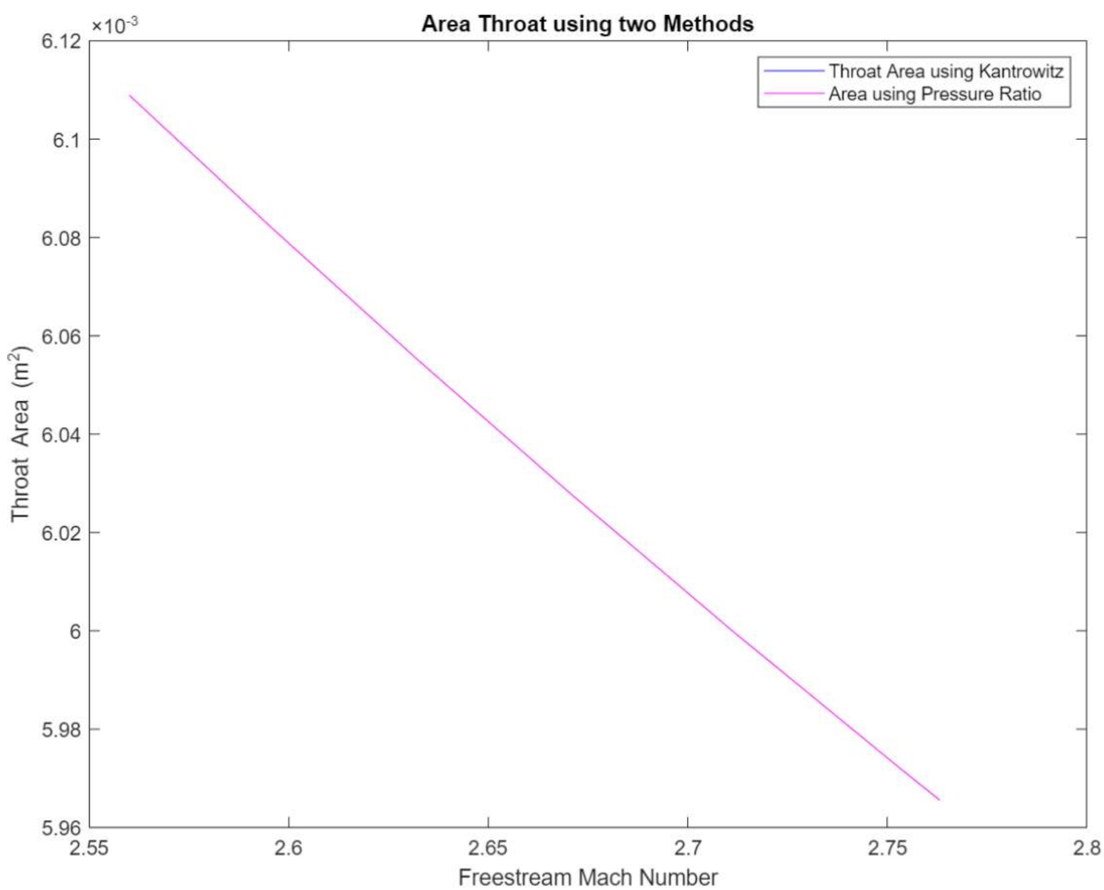

Figure 2.10: Minimum throat area using Kantrowitz limit and pressure area relations

The resulting throat area that is used through the rest of the calculations is a 7.9 cm by 7.9 cm cross section which is 62.41 cm^2, slightly larger than the first area for initial Mach number shown in the figure. The curve of the Kantrowitz limit is shown in Figure 2.11, where the y-axis is the maximum area (contraction) ratio based on the inlet Mach number on the x-axis. As can be seen at an entrance Mach number of 2.5 the maximum area ratio is around 1.35. For the area values chosen, area inlet of 9 x 9 cm^2 and area throat of 7.9 x 7.9 cm^2, the area ratio (area inlet/area throat) is 1.297 which less than the maximum contraction ratio.

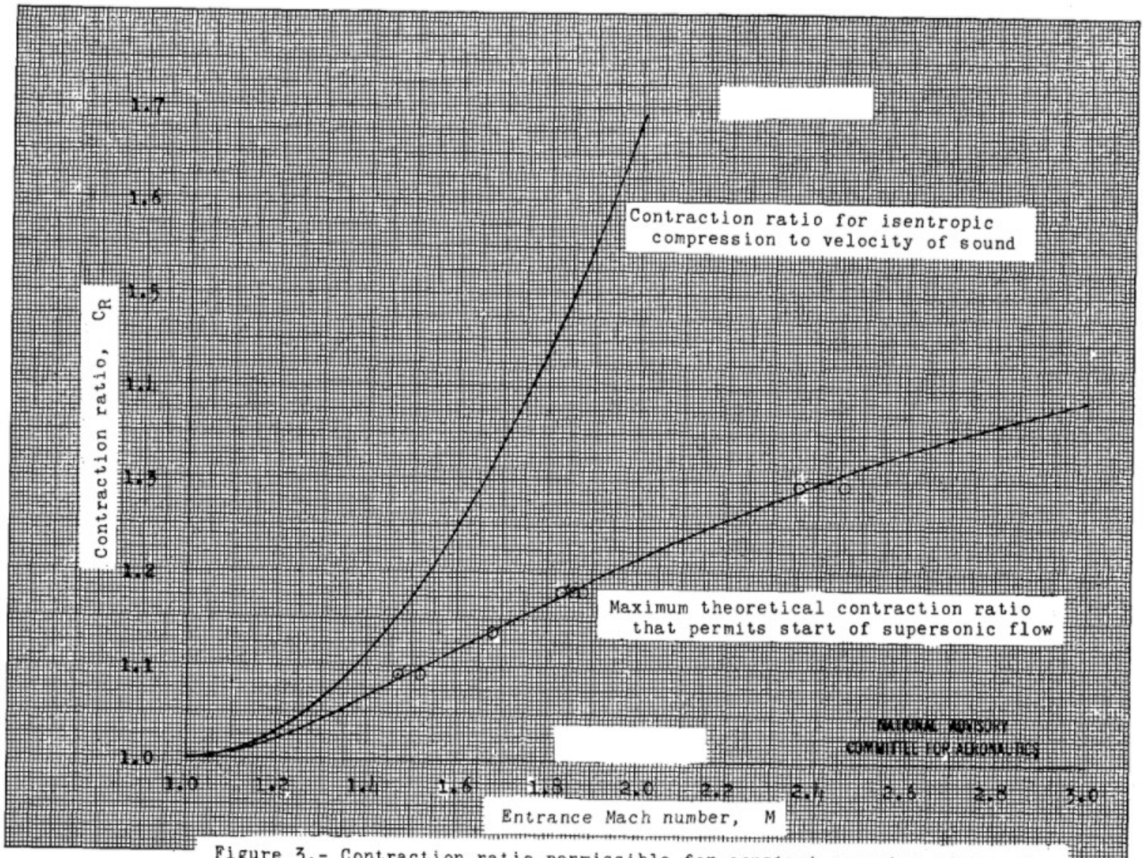

Figure 3.- Contraction ratio permissible for constant-geometry supersonic diffusers. Tailed symbols represent Mach numbers at which the diffusers failed to operate; plain symbols represent minimum Mach numbers at which diffusers operated.

Figure 2.11: Kantrowitz limit curve showing maximum contraction ratio (area ratio) for Mach number [9]

3 Normal Shock Calculations

3.1 Normal Shock Relations

A normal shock wave is a shock wave that is perpendicular to the flow that creates it and results in subsonic flow after the shock. In ramjet engines normal shock waves can allow the flow to go from supersonic to subsonic within the inlet to allow for subsonic flow in the combustion chamber. Normal shock waves are the strongest form of shock waves and are difficult to design for since a flow will likely go through an oblique shock wave or shock wave train to achieve the same result. Nevertheless, designing for a normal shock wave allows for simpler calculations of the desired flow parameters and the theory behind both types of waves is the same with oblique waves having the added angular component.

The following equations are used to determine the total and static pressure ratios, static temperature ratios and the Mach number after the normal shock.

$$\frac{P_2}{P_1} = \frac{2\gamma M^2 - (\gamma - 1)}{\gamma + 1} \tag{3.1}$$

$$\frac{P_{02}}{P_{01}} = \frac{(\gamma+1)M^2}{(\gamma-1)M^2+2}^{\frac{\gamma}{\gamma-1}} \frac{(\gamma+1)}{2\gamma M^2-(\gamma-1)}^{\frac{1}{\gamma-1}} \tag{3.2}$$

$$\frac{T_2}{T_1} = \frac{\left(2\gamma M^2-(\gamma-1)\right)(\gamma-1)M^2+2}{(\gamma+1)^2 M^2} \tag{3.3}$$

$$M_2 = \sqrt{\frac{(\gamma-1)M^2+2}{2\gamma M^2-(\gamma-1)}} \tag{3.4}$$

$$\frac{A}{A^*} = \frac{(\gamma+1)}{2}^{\frac{-(\gamma+1)}{2(\gamma-1)}} \frac{\left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{(\gamma+1)}{2(\gamma-1)}}}{M} \tag{3.5}$$

3.2 Mach Number at the Throat

The initial Mach number is known through the design research of the first stage rocket which reaches 2.56 at the separation altitude of 5.5 km. The area of the inlet and the area of the throat are set to 9 centimeters by 9 centimeters and 7.9 centimeters by 7.9 centimeters respectively. To find the Mach number at the throat of the inlet, the critical area (A*) was found using the isentropic relations, Equation 3.5, at the initial Mach number and using the critical area and solving for the Mach number using A* and the throat area. Figure 3.1 shows the Mach number at the throat of the diffuser which is slightly lower than the freestream Mach number since the flow is contracted.

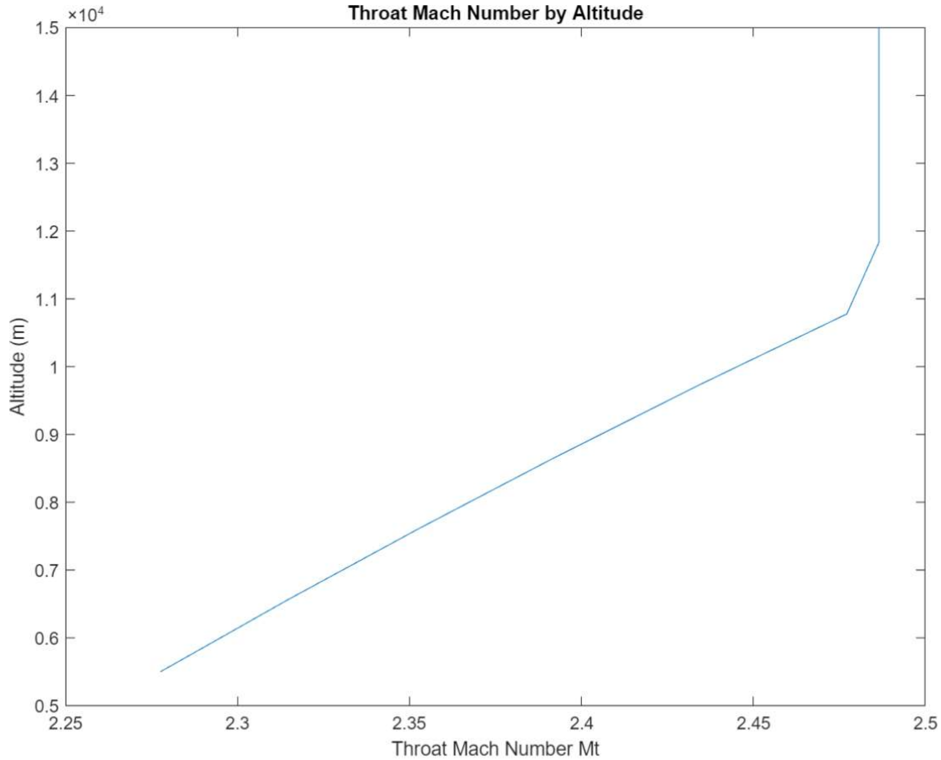

Figure 3.1: Mach number at the throat of the diffuser

3.3 Normal Shock Wave in Divergent Section

The calculations done for the normal shock wave use a Mach number directly upstream and directly downstream of the normal shock and the combustion pressure that was obtained by Dr. Vergine to locate the normal shock wave within the inlet. For the engine to function for the desired range of combustion pressures, the normal shock wave must occur in an area such that the pressure ratio (P3/P1) from the combustion chamber (P3) to the inlet (P1), which is known from preliminary analysis, is met. The normal shock wave in the divergent section of the diffuser must occur slightly after the throat yet not too far after the throat that the flow does not have enough time to expand to the desired combustion chamber pressure.

The combustion chamber area was set to 12.5 centimeters by 12.5 centimeters making the back end of the inlet larger than the front end of the inlet. The normal shock location is not known initially yet it is clear, through the supersonic Mach number at the throat, that it should occur in the diverging section of the diffuser. The diverging section of the diffuser was broken up into sections from the throat to the combustor. At each section, a normal shock calculation was done by using the critical area to find the Mach number and using Equation 15 to solve for the Mach number after the normal shock. The critical area is found for the flow downstream of the normal shock and is used with the combustion chamber area to find the static to total pressure ratio (P3/P1) at the end of the diffuser. P3/P1 is known from the preliminary calculations and Figure 3.2 shows the static pressure ratio (P3/P1) based on the local area ratio (A/A*) that can occur with a normal shock at each location. Matching the values of P3/P1 from Figure 3.2 to the P3/P1 found by Dr. Vergine gives the location of the normal shock.

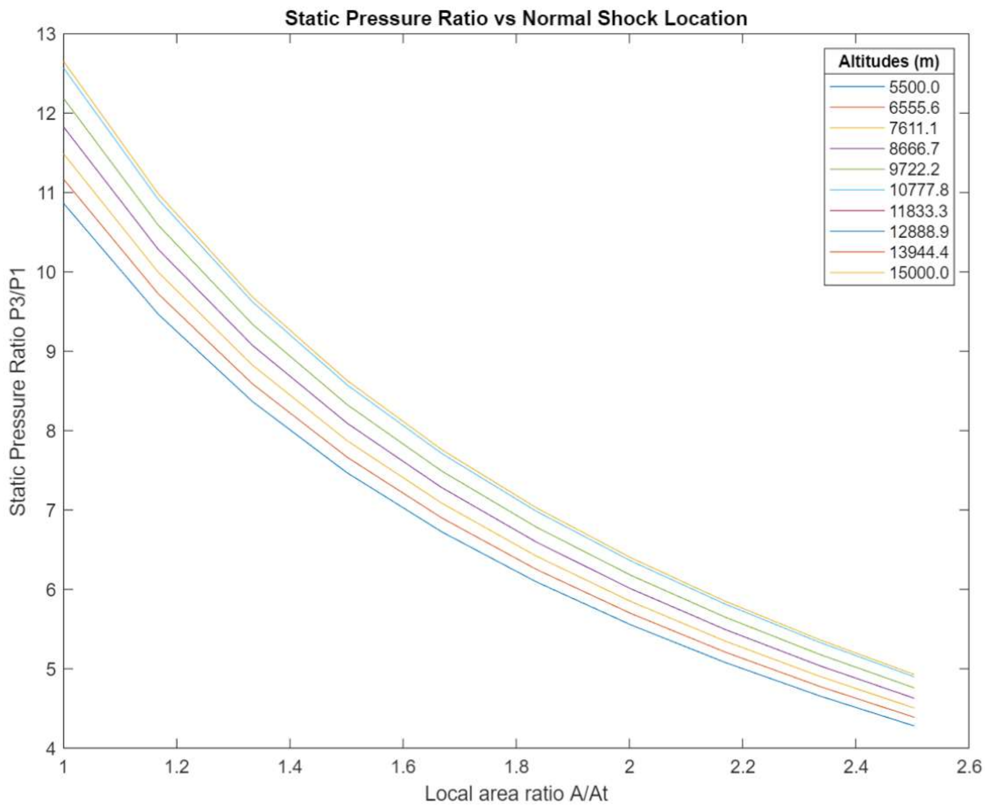

Figure 3.2: Static pressure ratio (Pc/Pi) variation along the diverging section of the inlet

The pressure in the combustion chamber that is desired is in the range of 2-5 atm depending on the altitude, based on the preliminary analysis. The calculations were run for the altitude range of 5.5 km to 15 km with the normal shock that gives the desired combustion chamber pressure occurring around an area ratio (A/At) of 1.2 as can be seen in Figure 3.3. The shock occurs shortly after the throat which is desired for stable flow and gives enough space for the flow to expand into the desired pressure range for the combustion chamber.

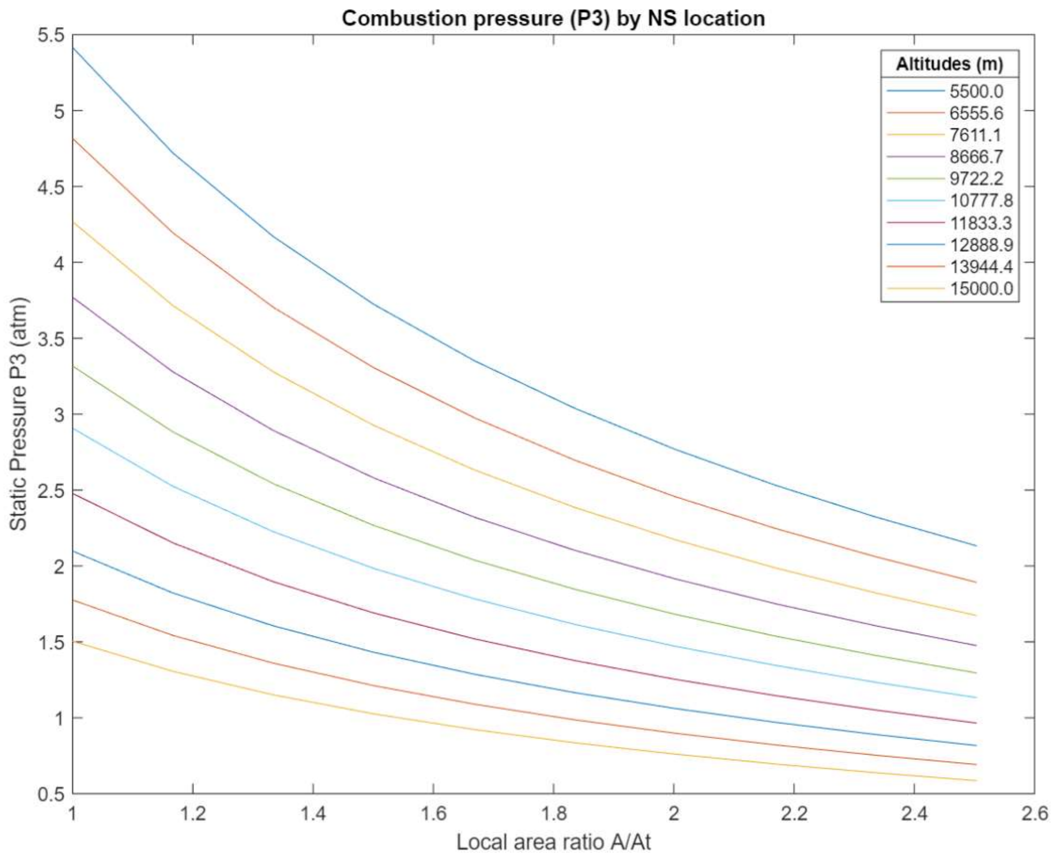

Figure 3.3: Static combustion pressure as the normal shock varies through the inlet

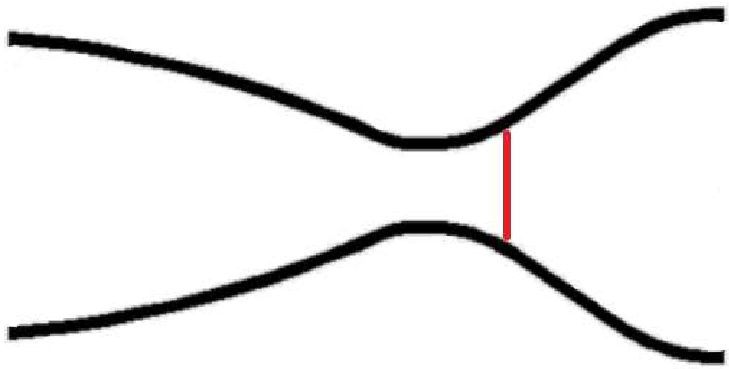

Figure 3.4: Preliminary schematic of normal shock location

Figure 3.4 shows a preliminary schematic with a location of the normal shock wave at a shock to throat area ratio of 1.2. The location of the shock will change with altitude as the freestream pressure decreases. Once the engine starts, by having the normal shock in the divergent section as shown, it is stable and should remain started even though the location of the normal shock will shift with altitude.

4 Oblique Shock Waves

4.1 Oblique Shock Waves

As stated in the previous chapter, normal shock wave diffusers require a shock in the divergent section of the diffuser for stable flow. Yet, to achieve the best total pressure recovery for a normal shock diffuser, the shock would need to occur at the throat which would be undesirable due to the instability of the flow when the shock occurs upstream or at the throat [15]. Designs using oblique shock waves or conical shocks ahead of the inlet are more common practice when designing ramjet engines rather than pitot intakes due to the reduction in pressure losses which are crucial when flying in the supersonic range. The design forces a shock or shocks to occur ahead of the inlet which allows for more controlled manipulation of the flow prior to entering the engine. The difficulty in designing a pitot intake is that the flow will naturally start to form an oblique shock wave before the throat of the inlet if the flow is not strong enough to push past the throat and form a normal shock in the divergent section. In practice it is much easier to design for the more likely case which would be an oblique shock wave which will be discussed in this chapter.

4.2 Equations

Oblique shock waves occur when a two-dimensional supersonic flow encounters an obstruction such as a ramp or, in the case of a ramjet, an inlet cone. Oblique shock waves are more likely to occur in the inlet being studied due to the flow encountering the surface of the inlet which is converging and diverging. Normal shock waves are the strongest form of shock waves which will slow down supersonic flow to subsonic flow producing large pressure losses therefore, oblique shock waves are more favorable. Unlike the flow following a normal shock wave, an oblique shock wave can produce supersonic flow or subsonic flow depending on the incoming Mach number and whether the shock wave is strong or weak, although generally a weak shock will occur. To produce subsonic flow using oblique shock waves, several shock waves could be used, or a combination of oblique and normal shocks can be used. Oblique shock waves will reflect or bounce off surfaces including other shock waves to create a shock train that will usually occur in higher supersonic flows or hypersonic flows. For the incoming Mach number of 2.56 being considered for this project, the flow will likely produce an oblique shock and a normal shock as can be seen in Figure 4.1.

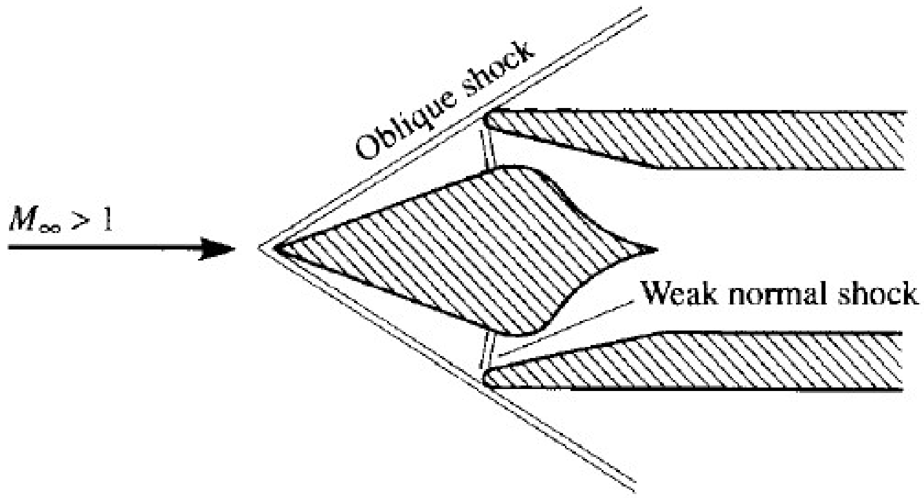

Figure 4.1: Supersonic conical inlet with resulting oblique shock wave and normal shock wave [2]

Oblique shock wave analysis is like normal shock wave analysis with the added component of the turn angle and wave angle. In essence, the flow will encounter the corner or ramp which will produce an oblique shock wave that turns the flow so that it runs parallel to the angle it encountered as can be seen in Figure 4.2: Oblique shock wave geometryFigure 4.2. The flow across the shock can be broken into the components that are normal to the shock and parallel to the flow as shown in Figure 4.3.

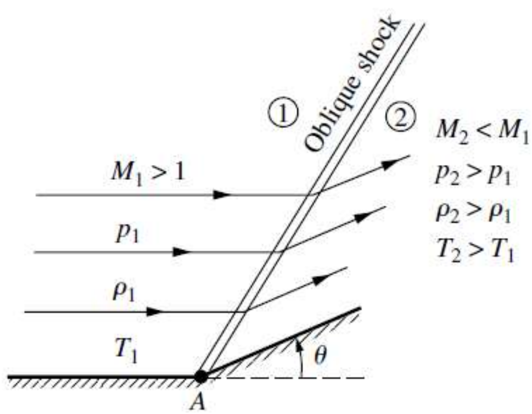

Figure 4.2: Oblique shock wave geometry

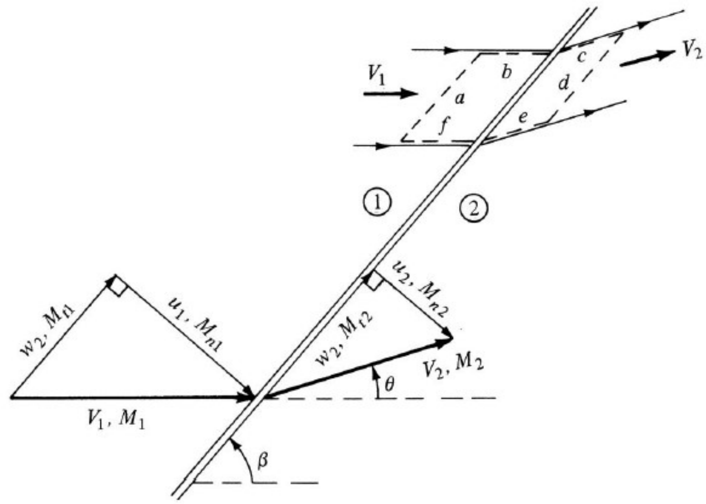

Figure 4.3: Oblique shock wave velocity and Mach number components

The calculation for the components normal to the shock are the same as normal shock wave analysis yet the component that is parallel to the flow has the added turn angle, theta, and wave angle, beta. To solve for the Mach number, static pressure ratio and temperature ratio across the oblique shock wave the following equations are used:

$$M_{n,1} = M_1 sin\beta \tag{4.1}$$

$$M_{n,2}^2 = \frac{1+[(\gamma-1)/2]M_{n,1}^2}{\gamma M_{n,1}^2-(\gamma-1)/2} \tag{4.2}$$

$$M_2 = \frac{M_{n,2}}{\sin(\beta-\theta)} \tag{4.3}$$

$$\frac{p_2}{p_1} = 1 + \frac{2\gamma}{\gamma+1}(M_{n,1}^2 - 1) \tag{4.4}$$

$$\frac{T_2}{T_1} = \frac{p_2}{p_1}\frac{\rho_1}{\rho_2} \tag{4.5}$$

Oblique shock waves have a maximum deflection angle that is dependent on the freestream Mach number. If the turn angle, θ, is greater than the maximum deflection angle the shock wave will curve and detach from the body the flow is encountering. If the turn angle is less than the maximum deflection, the flow will result in either a strong or weak shock solution. As stated previously, the resulting shock for the cases being considered in this project will be weak due to the requirement of a high pressure ratio for a strong shock wave which is less likely to occur in nature. For the purposes of this project, only two-dimensional calculations were used to simplify the calculations.

4.3 Benefits

The main concern with designing a ramjet engine inlet is ensuring the pressure losses are not so great that the engine does not start or starts and slows down to the point it unstarts. Using oblique shock waves ahead of the inlet slows down the flow before entering the diffuser using external compression, therefore reducing the need for internal compression, and allowing for a more efficient process to achieve the same result. A pitot intake looks much simpler in design but, in this case, did not show the viscous forces, boundary layer thickness and pressure losses that can complicate and prevent the flow from achieving combustion efficiently.

An example of the use of conical inlets in a supersonic engine is the SR-71 designed by Lockheed Corporation in the 1960s which reached flight at up to Mach number 3.2 [16]. The SR-71 used a moveable conical inlet to maintain the shock waves within the desired region by moving up to 0.66 meters in the axial direction during flight. The plane could reach up to 83,000 feet at Mach number 3.2 while being able to refuel during flight to account for the large amount of fuel needed to operate. The conical inlet and the engine allowed for airflow to go through and be used to cool the engine, reduce drag, and increase air flow.

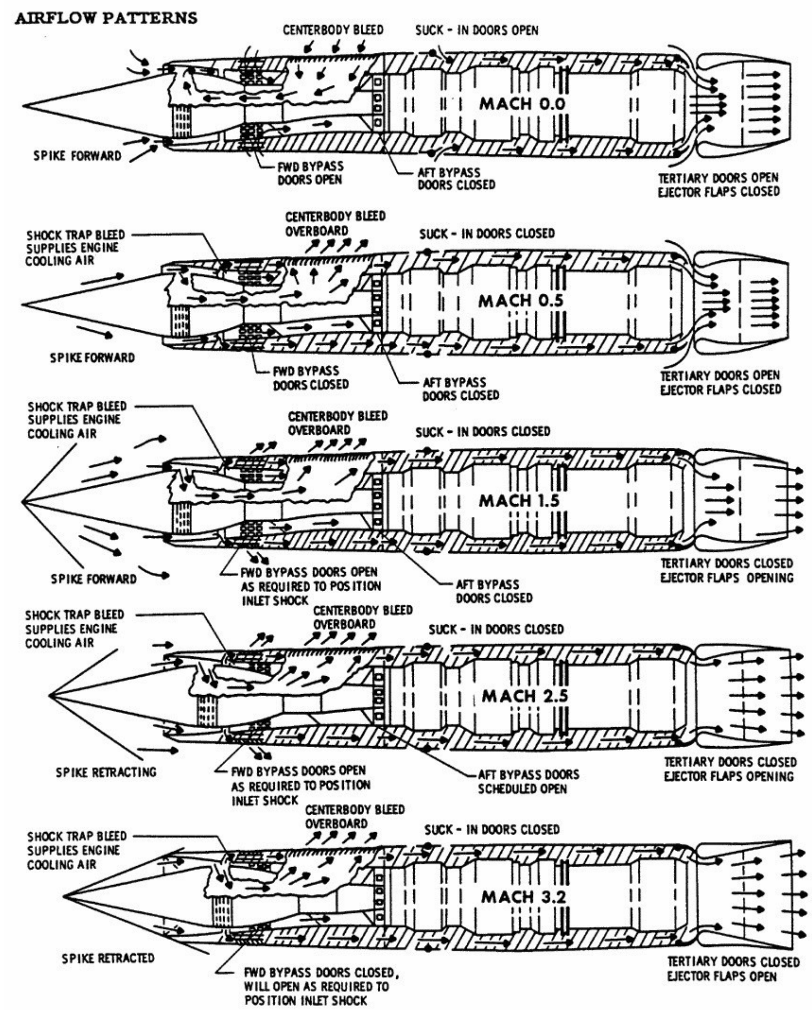

Figure 4.4: SR-71 engine diagram describing flow pattern at different velocities [17]

Figure 4.4 shows the complex internal and external design considerations that allow the engine to function throughout the entire range of velocity. Focusing on the conical inlet at Mach numbers of 1.5 and above, the airflow starts producing a relatively high shock wave angle creating a detached shock between the inlet and the freestream which adds drag and prevents the engine from starting due to the mass flow that does not enter the engine. As the flow increases the engine will begin to gather more air flow through the inlet and the retraction of the cone allows the inlet area to increase to an optimal area in which the edge of the shock wave moves towards the edge of the inlet. In the final stage at Mach number 3.2 the shock wave meets the edge of the inlet, allowing a maximum amount of air flow into the engine. An important observation is the use of ducts in the SR-71 to bypass the air flow at the different stages of flight. For long range supersonic flight this is a useful method used to manipulate and reduce the boundary layer while also allowing the air to cool the engine. The ducts and the use of bypass doors also aid in manipulating the position of the shocks waves during flight which can allow the engine to remain started for a wider range of altitudes and velocities. In the case of this project, the geometry is kept constant and therefore the range of altitude and Mach numbers that can be achieved will be minimal. By using a pointed two-dimensional geometry for the inlet, the incoming air flow can be more easily manipulated ahead of the inlet to achieve a greater result than a pitot inlet.

4.4 Assumptions

To simplify the design process needed for a cone shaped inlet a few assumptions were made to produce the geometry. The design that will be discussed is an example of a two-dimensional pointed inlet that can achieve a more optimal combustion chamber pressure as the pitot inlet discussed in chapter three. To reiterate, although the design appears to be a conical inlet based on Figure 4.5, only the cross section of the cone was analyzed using oblique shock wave analysis rather than the conical shock calculations. Conical shock waves, although similar in cross sectional design are more complex to analyze and produce quite different shock waves. In practice, conical shocks are more desirable but for the purpose and simplification of this project only the cross section of a cone was analyzed.

As previously stated, a static geometry is being considered for this project to simplify the design process therefore, one half angle must be chosen to do the oblique shock wave analysis which will feed into the engine. To achieve the best result, a normal shock must occur at the edge of the inlet, as shown in Figure 4.5, to produce combustion pressure that is within the operational range. Since a constant speed vertical trajectory is being considered, the operational Mach number range is small enough to assume the shock wave angle will shift minimally during flight. A constant area is used in the diffuser section due to the optimization of the shock waves such that the flow entering the diffuser is at the desired combustion chamber pressure range and therefore will not need to be further expanded.

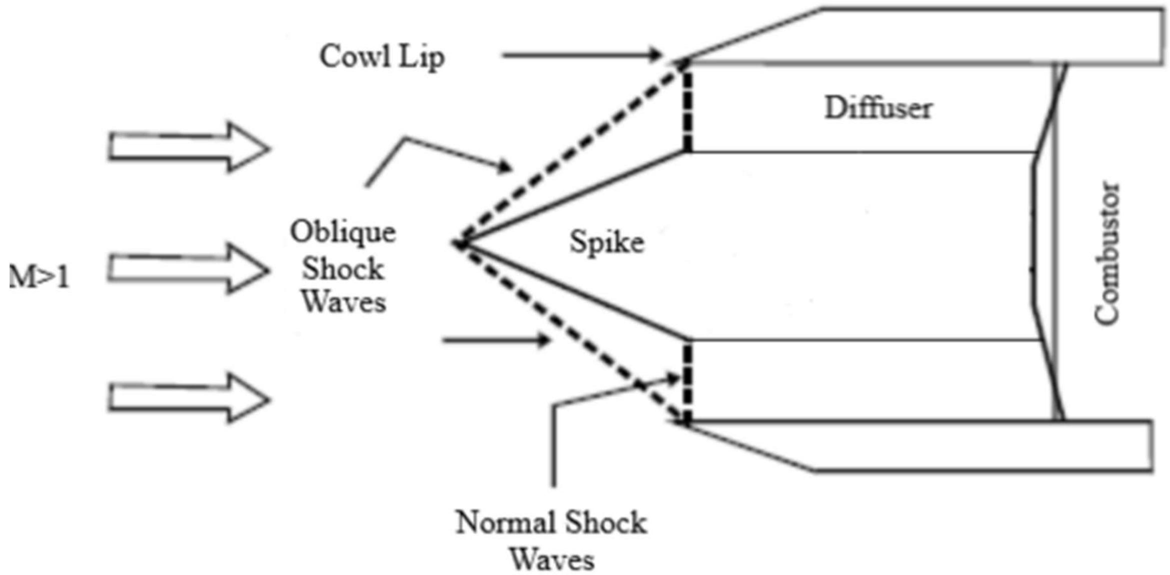

Figure 4.5: Two-dimensional pointed inlet diagram with oblique and normal shock waves

4.5 Two-dimensional Pointed Inlet Results

The following results were produced using a turn angle range of 0 to 30 degrees for the range of Mach numbers that will be seen in flight. In the following figures, each line represents a calculation using the same Mach number. The Mach number values were obtained in Chapter 2 shown in Figure 2.4.

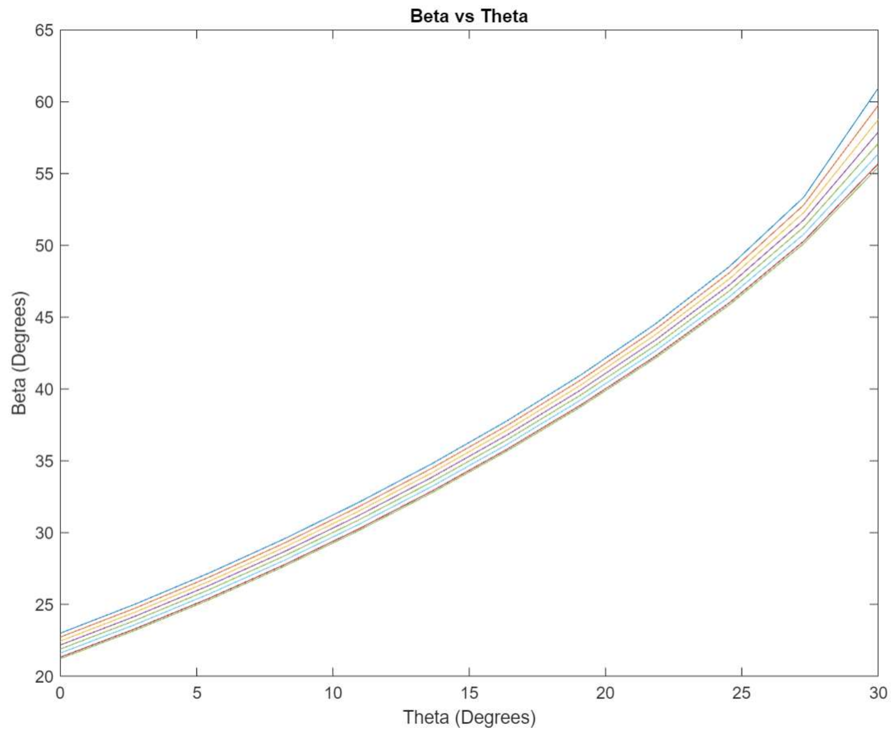

Figure 4.6: Shock wave angle as a result of operational Mach number and turn angle

Figure 4.6 shows the resulting shock wave angle based on the given turn angle and Mach number. As stated before, the narrow Mach number range shows the slight variation that will be seen during flight therefore, the median value of Mach number can be used to determine the geometry. This is a verification that the position of the shock waves will not shift greatly compared to the pitot inlet.

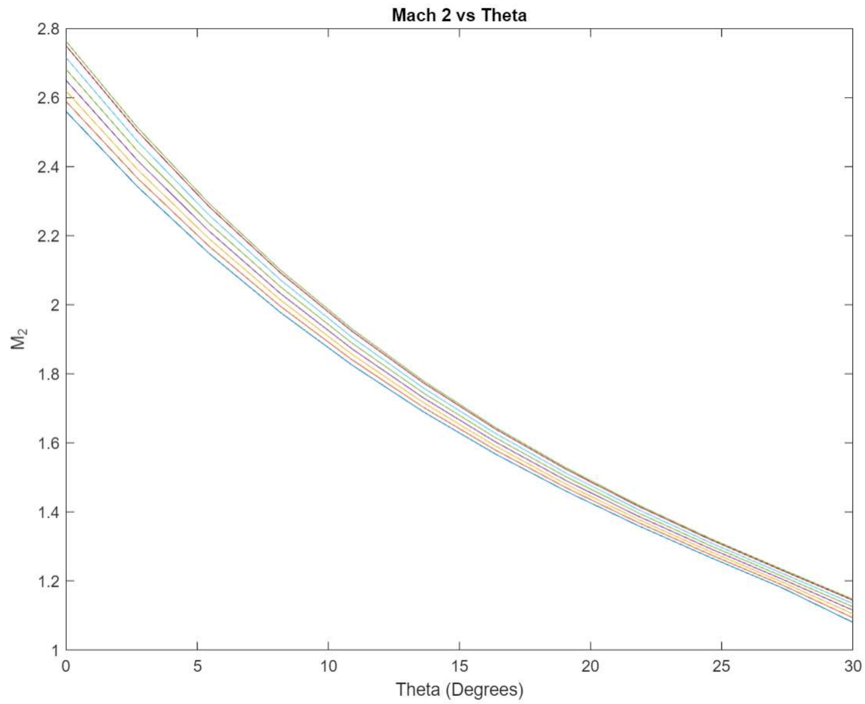

Figure 4.7: Mach number after oblique shock wave

Figure 4.7 shows the resulting Mach number after the oblique shock wave that results from the turn angle. As expected, the smaller turn angles result in higher Mach numbers due to the flow slowing down only slightly which then results in a stronger normal shock at the inlet entrance as shown in Figure 4.8. Based on the Mach number range achieved after the oblique shock wave, using smaller turn angles is likely not ideal because the resulting normal shock wave downstream would need to be stronger.

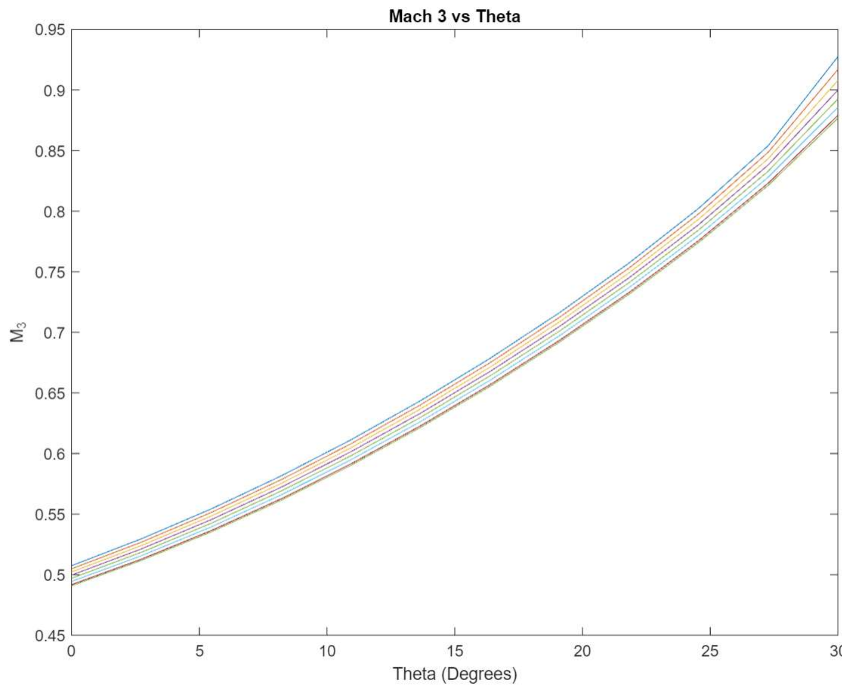

Figure 4.8: Mach number at the inlet after the normal shock wave

In Figure 4.8 we see that indeed the smaller turn angles produced a stronger shock while the larger turn angles only slightly decreased in Mach number. The inlet must reduce the flow to subsonic as efficiently as possible to reduce the total pressure losses. To quantify this, the total pressure ratio from the inlet to the freestream is plotted for each turn angle to determine the best combination of oblique shock wave to normal shock wave in Figure 4.9. In an ideal scenario, where total pressure is completely conserved, the total pressure at the inlet and at the freestream would be equal resulting in a ratio value of 1. As shown above, the highest ratio of roughly 0.8 is obtained around a turn angle of 17 degrees. The values at a turn angle of 0 degrees in Figure 4.9 are the total pressure ratio obtained for a pitot intake since the flow enters the inlet without being deflected and is reduced with the normal shock wave. This produces the lowest total pressure ratio and consequently higher total pressure losses. This direct comparison shows the benefit of a conical inlet by reducing the pressure losses which would lead to a more efficient thrust profile.

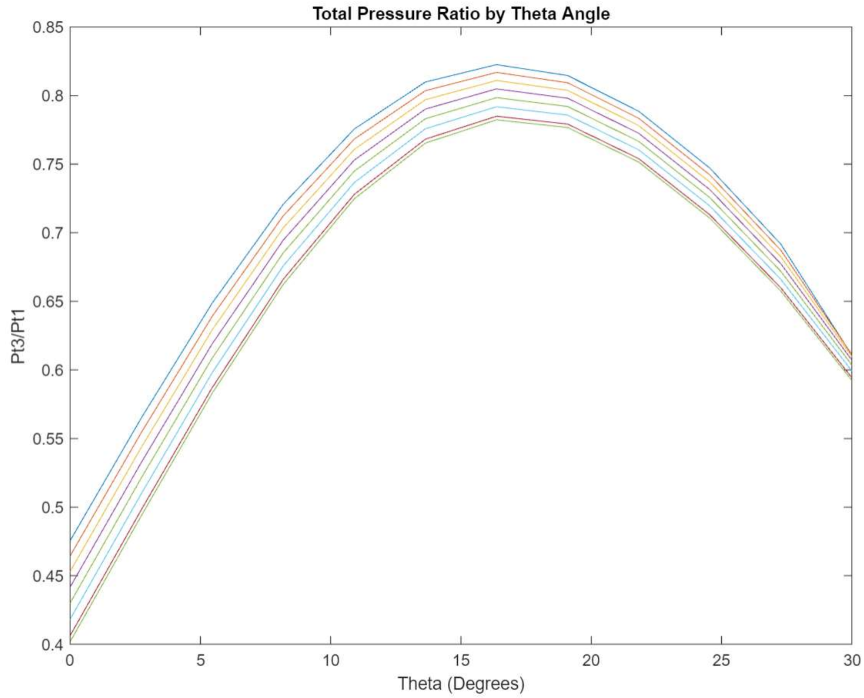

Figure 4.9: Total pressure ratio from the inlet to the freestream

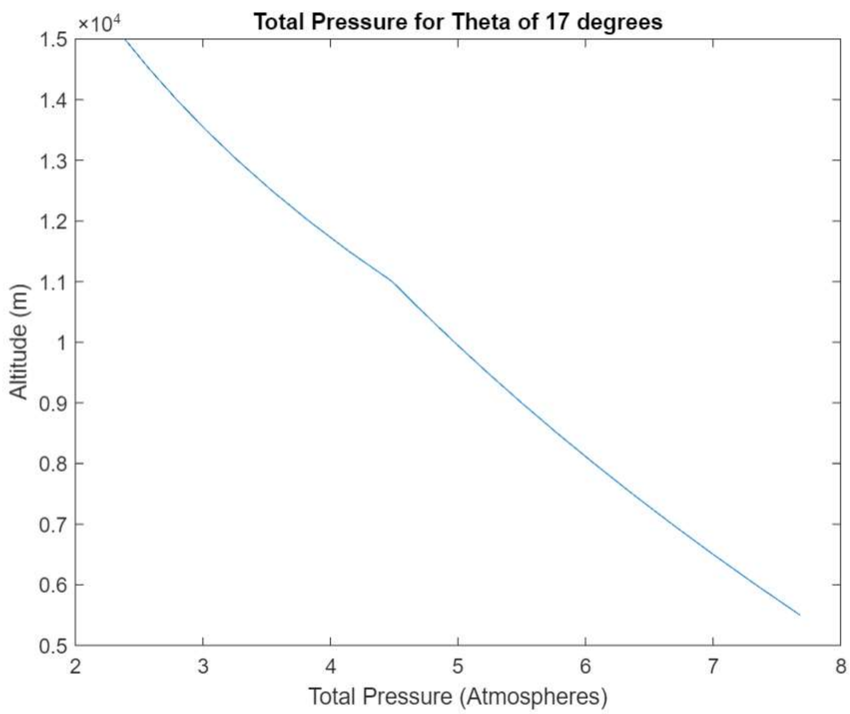

Figure 4.10: Total pressure (atm) at the inlet after the normal shock wave

The total pressure values at the entrance of the engine throughout the operational range are shown in Figure 4.10. These values are comparable to the range of values that Dr. Vergine obtained for combustion chamber pressure although slightly higher and more favorable. The geometry of the conical inlet can be gathered using the sine and cosine of the turn angle and shock wave angle using the values obtained at 17 degrees. The rocket geometry that is being considered will have a diameter of roughly 6 inches in which the ramjet will need to interface. The inlet can be capped at a 6-inch diameter and the remaining geometry is designed around the axis of the inlet. The Beta angle that is obtained at a turn angle of 17 degrees is around 35 degrees which would give a 2-dimensional cone base of roughly 2.6 inches with a cone height of 4.3 inches and the half angle being 17 degrees at the tip of the cone. This is a rough estimate of the geometry that can be obtained for a conical inlet that would improve the performance of the engine by reducing the pressure losses, which does not account for any boundary layer interactions that might add to the angles that were obtained.

5 Conclusion

5.1 Conical vs. Pitot Inlet

Supersonic and hypersonic engines are easy to understand in theory but difficult to design in practice. Although this project might seem outlandish, the overall outcome through the research and analysis done is that a vertically flying supersonic airbreathing engine could be achieved. With the aerospace industry expanding and developing new state of the art technology for larger missions it is possible that a vertically flying airbreathing engine might aid in achieving space flight through a more efficient manner than traditional methods. There is still a lot of research and testing to be done on supersonic and hypersonic flight and airbreathing engine design but this small project highlights the plausibility and flexibility of these designs. Under a more dedicated and rigorous approach, this project could achieve more promising results with several different applications to consider.

There are several areas of constraint which are to be expected for a university project with the main one being a small budget in which to make designing and testing components possible. The team working on the first stage booster rocket found it difficult to achieve anything higher than Mach 2.7 using off the shelf rocket engines which the team is limited to due to safety concerns from the university. Through the design process of a large project like this, several factors come in to play at different times which is expected in any systems engineering type of analysis. Firstly, the ramjet engine was constrained to the diameter of the rocket engines which are set by the manufacturer, there is no getting around this. Therefore, the ramjet engine was always going to be small which might appear to be an easier approach but is more susceptible to boundary layer interactions and constrains the design to a static geometry. Secondly, the rocket engines can only produce a specific amount of thrust which became a problem when considering the weight of the ramjet engine along with the first stage components. Lastly, the static geometry is not an ideal design choice due to the reduced operational range and the higher possibility of an engine unstart without the help of a shifting geometry to accommodate the positioning of the shock waves needed.

A dynamic inlet design allows for a larger flight range and can aid in the reduction of boundary layer interactions but requires larger geometries to be able to place all the moving parts. The two-dimensional pointed inlet design that was found in chapter 4 is a theory-based example of why pitot inlets are less favored in practice. The total pressure losses in a pitot inlet create a less efficient engine and the design is susceptible to having the normal shock wave occur ahead of the throat rather than downstream of it where it is desired. Further improvements can be implemented to improve the performance of both inlet designs with one of the more common ones being boundary layer bleed ducts that allow the flow to perform closer to theoretical cases. The other consideration would be a dynamic geometry that would shift in flight to better accommodate the changing freestream flow parameters. This could be done using a varying throat area for the pitot inlet and using a shifting cone length for the conical inlet, both of which are used and needed in practice. Considering the limited flexibility within this project, the results were still promising for a low supersonic ramjet engine and addressing the constraints in future work would produce even better results.

The simplifications explained within this report could cause significant differences between what is expected and what can be achieved as was studied during the first supersonic flights. Over time, the aerospace community evolved and adapted through testing of supersonic flow. This project pushes the boundaries of what has been done and explores the limits of ramjet engine design. Although many assumptions and simplifications were made in this design process, the takeaway if that vertical flight is plausible for ramjet engines. Further iterations of this design would need to go into the finer details of optimization techniques that would be necessary to extend the operational range as well as to outline any aspects of supersonic flow that were not considered and would have a considerable effect on the performance of the engine. There is much work to be done to achieve a fully functional vertically flying ramjet engine inlet design but this is a baseline effort on how that can be achieved.

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