MAE 563: Aircraft Propulsion

Professor Werner Dahm

Final Project Report

Chandler Hutchens

1216384494

Arizona State Unversity

Table of Contents

1. Introduction **3**
1.1 Project Description 3
1.2 Report Outline 3
2. Analysis Tool **4**
2.1 Description 4
2.2 Module 1: Free Stream (State 1) 4
2.3 Module 2: Inlet/Diffuser (State 2) 5
2.4 Module 3: Combuster (State 3) 5
2.5 Module 4: Converging Nozzle (State e) 7
2.6 Module 5: Flow Past Nozzle Exit (State 4) 8
2.7 Module 6: Various Performance Parameters 8
2.8 Module 7 and 8: Model Validation 10
3. Part A: T-s Diagrams **11**
4. Part B: Atmospheric Model Comparison **13**
5. Part C: Parameter Results With Varying M1 **15**
6. Part D: Parameter Results With Varying z **18**
7. Part E: Parameter Results With Varying z and M1 **21**
8. Part F: Parameter Results With Varying ηd **23**
9. Part G: Parameter Results With Varying ηn **26**
10. Part H: Parameter Results With Varying M2 **29**
11. Part I: Design Ram/Scram Jet **32**
12. Final Results/Discussion **35**
13. Appendix **36**
13.1 Module 1: Free Stream Matlab Function 36
13.2 Module 2: Inlet/Diffuser Matlab Function 38
13.3 Module 3: Combuster Matlab Function 40
13.4 Module 4: Converging Nozzle Matlab Function 42
13.5 Module 5: Flow Past The Nozzle Exit Matlab Function 44
13.6 Module 6: Various Parameters Matlab Function 46
13.7 Call Functions Matlab Code 48
13.8 Part A Matlab Code 52
13.9 Part B Matlab Code 54
13.10 Part C Matlab Code 56
13.11 Part D Matlab Code 58
13.12 Part E Matlab Code 60
13.13 Part F Matlab Code 62
13.14 Part G Matlab Code 64

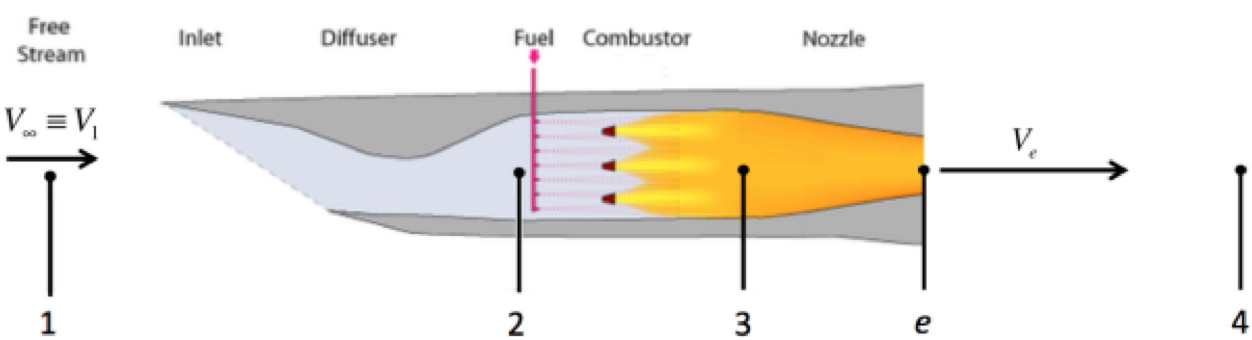
13.15 Part H Matlab Code 66
13.16 Part I Matlab Code 68

1. Introduction

1.1 Project Description

This project outlines the non-ideal flow path for a ram/scramjet engine system. This engine system is known as a ramjet, but when the flow speed becomes supersonic it is referred to as a scramjet engine. The engine consists of various sections shown in Figure 1-1. The ram/scramjet engine begins with the free stream (1) air entering the system. From here, the inlet/diffuser (2) takes in the incoming air and decelerates it to provide a uniform condition during its transfer to the combustor. Therefore, the next section is the combustor (3) where the air is combined with fuel and the resulting mixture is burned. The next section is the nozzle exit (e) which expells burned air/fuel mixture outwards from the engine to generate thrust. Finally, the last section is the resulting free stream behind the engine (4), which should be returned to the state before the engine after some given distance. A thermodynamic cycle (1-2-3-e-4) can be created and evaluated using these sections to represent the entire engine process. Therefore, solving various parameters in each section will give a total understanding of this thermodynamic cycle and overall engine performance.

Figure 1-1 Ram/Scramjet Propulsion System



1.2 Report Outline

This final project report will be organized starting with the analysis tool and then going into the given problem set (A-I), leading to a final discussion and conclusion to the overall project. Every calculation and plot was done and created using MATLAB, where the code for each section is separated below in the Appendix.

2. Analysis Tool

2.1 Description

The analysis tool is a function-based model developed in MATLAB, which performs a variety of calculations at the various sections of the ram/scramjet engine shown in Figure 1-1.

2.2 Module 1: Free Stream (State 1)

The first section of the thermodynamic cycle of the ram/scramjet is the free stream air before entering the engine. This section utilizes the isentropic model of the atmosphere that uses isentropic values of flight altitude (z^*), pressure (P_s), and temperature (T_s) (shown in Table 2-2-1) to calculate the temperature and pressure at this free stream position, shown in Equation 1 and 2.

Table 2-2-1 Isentropic Model of The Atmosphere Values

Altitude (z^*), m	Pressure (P_s), kPa	Temperature (T_s), K
8404	8404	288

$$\frac{T_1}{T_s} = \left[1 - \frac{\gamma-1}{\gamma}\left(\frac{z}{z^*}\right)\right] \tag{1}$$

$$\frac{P_1}{P_s} = \left[1 - \frac{\gamma-1}{\gamma}\left(\frac{z}{z^*}\right)\right]^{\frac{\gamma}{\gamma-1}} \tag{2}$$

However, this model is only taken to be true when the flight altitude (z) is less than 7958 m and therefore when the flight altitude is above 7958 the temperature (T_1) is taken to be constant at 210 K and the pressure is a function of z, as shown in Equation 3.

$$P_1 = 33.6e^{\left[\frac{z-7958}{6605}\right]} \tag{3}$$

Now that the values of T_1 and P_1 are calculated for various flight altitudes, the total-to-static equations can be used to calculate the total temperature and pressure for state 1, shown in Equations 4 and 5.

$$\frac{T_{t1}}{T_1} = \left[1 - \frac{\gamma-1}{2}M_1^2\right] \tag{4}$$

$$\frac{P_{t1}}{P_1} = \left[1 - \frac{\gamma-1}{2}M_1^2\right]^{\frac{\gamma}{\gamma-1}} \tag{5}$$

Continuing then, the final few calculations are for the specific heat capacity (C_p), speed of sound (a), and the velocity (v) at state 1, all shown in Equations 6, 7, and 8.

$$C_p = \frac{\gamma R}{\gamma - 1} \tag{6}$$

$$a = \sqrt{\gamma R T} \tag{7}$$

$$v = M \cdot a \tag{8}$$

2.3 Module 2: Inlet/Diffuser (State 2)

After the free stream, the air enters into the inlet/diffuser of the ram/scramjet engine. The air is subjected to ram compression and decelerated to a uniform state going into the next engine section. It begins by taking the total temperature at state 2 (T_{t2}) to be equal to the total temperature at state 1 (T_{t1}). From here, the total-to-static equation, shown in Equation 4, can be used to calculate T_2. Note, this is used inversely from section 1, where T_1 was known first and T_{t1} was calculated using Equation 4. Additionally, the equation will be using state 2's Mach number (M_2), which will be a design value for the inlet/diffuser.

Now that the total and static temperatures are known, the next step is to calculate the total pressure at state 2. With a given inlet/diffuser efficiency (η_d), the total pressure (P_{t2}) can be calculated using Equation 9.

$$\frac{P_{t2}}{P_1} = \left[1 - \eta_d \frac{\gamma - 1}{2} M_1^{\ 2}\right]^{\frac{\gamma}{\gamma - 1}} \tag{9}$$

Similar to the temperature above, using the total pressure (P_{t2}), the static pressure can be calculated using Equation 5. Again, this will be inversely used from state 1, as discussed for T_{t1} and T_1. Now the values of specific heat capacity (C_p), speed of sound (a), and the velocity (v) at state 2 can be calculated using Equations 6, 7, and 8.

Next, the entropy change (Δs) between states 2 and 1 can be calculated, as seen in Equation 10. This is due to the irreversibility of the inlet/diffuser, as seen with η_d.

$$\Delta s = C_p ln\left(\frac{T_{t2}}{T_{t1}}\right) - R ln\left(\frac{P_{t2}}{P_{t1}}\right) \tag{10}$$

2.4 Module 3: Combuster (State 3)

After the air has exited the inlet/diffuser uniformly, it arrives at the combuster chamber to be mixed with fuel and burned. It's at this point that the γ will be changed from the value that was used for just air in sections 1 and 2. This section begins by calculating the choked total temperature at state 3 ($(T_{t3,})_{choked}$), shown in Equation 12. This is calculated by setting M_3 equal to 1 in Equation 11.

$$\frac{T_{t3}}{T_{t2}} = \frac{M_3^{\ 2}}{M_2^{\ 2}} \frac{(1+\gamma M_2^{\ 2})^2}{(1+\gamma M_3^{\ 2})^2} \frac{(1+\frac{\gamma-1}{2}M_3^{\ 2})}{(1+\frac{\gamma-1}{2}M_2^{\ 2})} \quad (11)$$

$$(T_{t3})_{choked} = T_{t2}\left\{\frac{1}{2(\gamma+1)}\frac{1}{M_2^{\ 2}}(1 + \gamma M_2^{\ 2})^2(1 + \frac{\gamma-1}{2}M_2^{\ 2})^{-1}\right\} \quad (12)$$

Now, if the value of $(T_{t3,})_{choked}$ is less than the given $(T_{t3,})_{max}$, then the flow is considered choked. This results in the Mach number at state 3 (M_3) being equal to 1, and the total temperature (T_{t3}) being equal to $(T_{t3,})_{choked}$.

However, if the value of $(T_{t3,})_{choked}$ is greater than the given $(T_{t3,})_{max}$, the flow is not considered choked and the total temperature (T_{t3}) is equal to the $(T_{t3,})_{max}$. The Mach number (M_3) needs to be calculated using the quadratic equation, as shown in Equations 13 through 16.

$$C = \frac{T_{t3}}{T_{t2}} \frac{(1+\frac{\gamma-1}{2}M_2^{\ 2})}{(1+\gamma M_2^{\ 2})^2} M_2^{\ 2} \quad (13)$$

$$a = \left[C\gamma^2 - \frac{\gamma-1}{2}\right] \quad (14)$$

$$b = [2C\gamma - 1] \quad (15)$$

$$c = C \quad (16)$$

$$M_3^{\ 2} = \frac{-b \pm \sqrt{b^2 - 4ac}}{2a} \quad (16)$$

Using the resultant M_3, the static temperature (T_3) can be calculated using Equation 4. For the pressure, it can be seen that the static pressure at state 3 (P_3) is equal to the static pressure at state 2 (P_2), and therefore the total pressure (P_{t3}) can be calculated using Equation 5.

Continuing, the next value that can be calculated is the heat per unit mass (q_{23}) which is added to the combuster. This can be calculated using Equation 17, with the values of a and b being 986 and .179, respectively.

$$q_{23} = a\left[T_{t3} - T_{t2}\right] + \frac{1}{2}b\left[T_{t3}^{\ 2} - T_{t2}^{\ 2}\right] \quad (17)$$

Similar to state 2, the entropy change (Δs) can be calculated, this time from state 3 to state 2 and from state 3 to state 1, which can be calculated by using Equation 18.

$$\Delta s_{31} = \Delta s_{21} + \Delta s_{32} \tag{18}$$

Using the values of a and b above, a new equation can be used for the specific heat capacity (C_p), shown in Equation 19.

$$C_p = a + bT \tag{19}$$

Finally, the values for the speed of sound (a) and velocity (v) at state 3 can be calculated using Equations 6, 7, and 8.

2.5 MODULE 4: CONVERGING NOZZLE (STATE E)

Upon exiting the combustor, the resulting flow is then subjected to the converging nozzle of the ram/scramjet. This flow will continue with the γ found from the mixture in state 3. The first step is to calculate the prime Mach number (M'), as seen in Equation 20.

$$M' = \left\{ \frac{2}{\gamma-1} \frac{\eta_n \left[1-(\frac{P_1}{P_{t3}})^{\frac{\gamma-1}{\gamma}}\right]}{1-\eta_n \left[1-(\frac{P_1}{P_{t3}})^{\frac{\gamma-1}{\gamma}}\right]} \right\}^{\frac{1}{2}} \tag{20}$$

If the resulting M' is less than 1, than the nozzle is not choked and the Mach number at the exit (M_e) is equal to the prime Mach number (M'), and the pressure at the exit (P_e) is equal to the initial pressure (P_1).

However, if the resulting M' is greater than 1, then the nozzle is choked and the Mach number at the exit (M_e) will be equal to 1, and the pressure at the exit (P_e) will be equal to Equation 21.

$$\frac{P_e}{P_{t3}} = \left[1 - \frac{1}{\eta_n}(\frac{\gamma-1}{\gamma+1})\right]^{\frac{\gamma}{\gamma-1}} \tag{21}$$

Using the pressure at the exit (P_e), the total pressure at the exit (P_{te}) can now be calculated using Equation 5. For the temperature, it can be seen that the total temperature at the exit (T_{te}) is equal to the total temperature at state 3 (T_{t3}). Using this then, the static temperature at the exit can be calculated using Equation 4. Similarly, the values of specific heat capacity (C_p), speed of sound (a), and velocity (v) at the exit can be calculated using Equations 19, 7, and 8.

Continuing, exit mass flux ($\overline{m_e}$) can be calculated, as shown in Equation 22, where ρ_e is shown in Equation 23.

$$\overline{m_e} = \rho_e v_e A_e \tag{22}$$

$$\rho_e = \frac{P_e}{RT_e} \tag{23}$$

Finally, the entropy change (Δs) can be calculated from the exit to state 3 and from the exit to state 1, which can be calculated by using Equation 18, with the addition of Δs_{e3} at the end of the equation.

2.6 MODULE 5: FLOW PAST NOZZLE EXIT (STATE 4)

The last part of the ram/scramjet engine cycle is the flow past the exit nozzle. It can be seen that the total temperature at state 4 (T_{t4}) is equal to the total temperature at the exit (T_{te}), and the static pressure at state 4 (P_4) is equal to the static pressure at the exit (P_e). Using the value of total temperature at the exit (T_{te}), the temperature at state 4 (T_4) can be calculated using Equation 24.

$$\frac{T_4}{T_{te}} = 1 - \eta_{n,\,ext}\left[1 - \left(\frac{P_1}{P_{te}}\right)^{\frac{\gamma-1}{\gamma}}\right] \tag{24}$$

Where $\eta_{n,\,ext}$ is equal to 1 when the prime Mach number (M') is less than 1, and equal to $M'^{(-.3)}$ when prime Mach number (M') is greater than or equal to 1.

Now, the Mach number at state 4 can be calculated using Equation 25.

$$M_4 = \left\{\frac{2}{\gamma-1}\left[\frac{T_{t4}}{T_4} - 1\right]\right\}^{\frac{1}{2}} \tag{25}$$

Then using the value of Mach number at state 4 (M_4), the total pressure at state 4 (P_{t4}) can be calculated using Equation 5. Now, the values of specific heat capacity (C_p), speed of sound (a), and velocity (v) at the exit can be calculated using Equations 19, 7, and 8.

Finally, the entropy change (Δs) can be calculated from state 4 to the exit and from state 4 to state 1, which can be calculated by using Equation 18, with the addition of Δs_{4e} at the end of the equation.

2.7 MODULE 6: VARIOUS PERFORMANCE PARAMETERS

Finally, various engine performance parameters can be calculated based on the values obtained from the previous sections. Starting with the inlet mass flux ($\overline{m}_i$), it can be calculated using Equation 26, where the $\overline{m}_e$ is a design value.

$$\overline{m}_i = \frac{\overline{m}_e}{1+\frac{q_{23}}{q_f}} \tag{26}$$

Using the given value of $\overline{m}_e$ and the calculated value of $\overline{m}_i$, the fuel mass flux ($\overline{m}_f$) can be calculated using Equation 27. Then the fuel-air mass ratio (f) can be calculated using Equation 28.

$$\overline{m}_f = \overline{m}_e - \overline{m}_i \tag{27}$$

$$f = \frac{\overline{m}_f}{\overline{m}_i} \tag{28}$$

From here, the thrust (T) can be calculated using Equation 29.

$$T = \overline{m}_i\left[(1 + f)v_e - v_1\right] + (P_e - P_1)A_e \tag{29}$$

Next, the thrust-specific fuel consumption (TSFC) can be calculated using Equation 30 and the specific impulse can be calculated using Equation 31. Where g is equal to 9.81 m/s^2

$$TSFC = \frac{\overline{m}_f}{T} \tag{30}$$

$$Isp = \frac{T}{\overline{m}_f g} \tag{31}$$

Now, the equivalent velocity (v_{eq}) by rearranging Equation 29, and can be calculated using Equation 32.

$$v_{eq} = v_e + (P_e - P_1)\frac{A_e}{\overline{m}_e} \tag{32}$$

The overall efficiency (η_o)of the engine can be calculated using Equation 33, where the propulsive efficiency (η_p) is shown in Equation 34, and the thermal efficiency (η_{th})is shown in Equation 35.

$$\eta_o = \eta_{th} \cdot \eta_p \tag{33}$$

$$\eta_p = \frac{2}{1+\frac{v_{eq}}{v_1}} \tag{34}$$

$$\eta_{th} = \frac{\left[\overline{m}_e \frac{1}{2} v_{eq}^{\ 2}\right] - \left[\overline{m}_i \frac{1}{2} v_1^{\ 2}\right]}{\overline{m}_i q_{23}} \tag{35}$$

Finally, the propulsive power (P) can be calculated using Equation 36.

$$P = T \cdot V_1 \tag{36}$$

2.8 Module 7 and 8: Model Validation

Using the sections above, the entire function-based model can be generated. This model was verified by performing two different test cases. The first is case #1, where the inputs can be seen in Table 2-8-1.

Table 2-8-1 Case #1 Inputs

Variable	Value
z	4300 m
M_1	2.4
η_d	.92
M_2	0.15
q_f	43.2 MJ/kg
$(T_{t3,})_{max}$	2400 K
η_n	.94
A_e	0.015 m^2
γ_{12}	1.4
γ_{34}	1.3
R	286.9 J/(kg-K)

Then case #2, where the Mach number at state 2 (M_2) is varied to run through a thermally choked scenario.

Table 2-8-2 Case #2 Inputs

Variable	Value
M_2	0.4
All Other Variables from Case #1	Same Value as Case #1

3. PART A: T-S DIAGRAMS

Based on the two cases seen above in section 2.8, a T-s diagram can be generated for the non-thermally choked and thermally choked cases. To generate a T-s diagram, the entire case must be run through the function to calculate the temperature (T) and the entropy (s) at each stage. In a T-s diagram, there are constant pressure curves that connect the entire cycle at two places. The first is a curve from state 2 to state 3, where it was seen that the pressure was constant. This curve can be calculated using Equation 37.

$$\left(\frac{dT}{ds}\right)_p = \frac{T}{C_p} \rightarrow dT = \frac{T}{a + bT} ds \quad (37)$$

Similarly, from state 4 to state 1 the pressure was taken to be constant and a curve can be generated using Equation 38. Where C_p is taken to 1004 J/(kg-K).

$$\left(\frac{dT}{ds}\right)_p = \frac{T}{C_p} \rightarrow dT = \frac{T}{C_p} ds \quad (38)$$

After generating the two curves and plotting the values of temperature and entropy the T-s diagrams can be seen in Figure 3-1-1 and 3-1-2.

Figure 3-1-1 T-s Diagram (Case #1: Non-Thermally Choked)

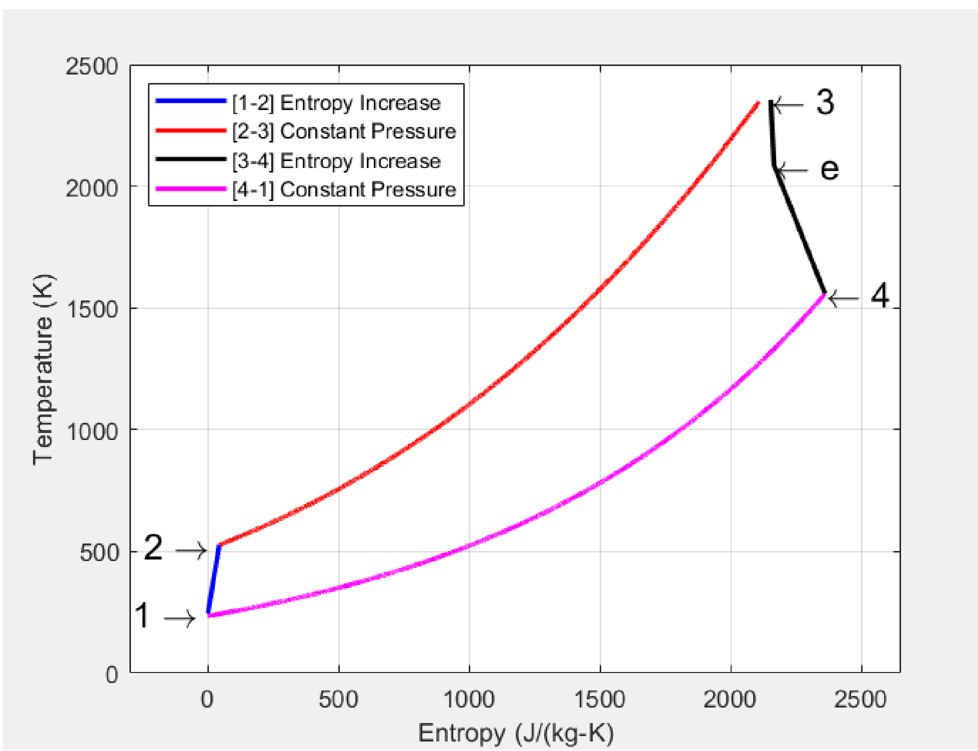
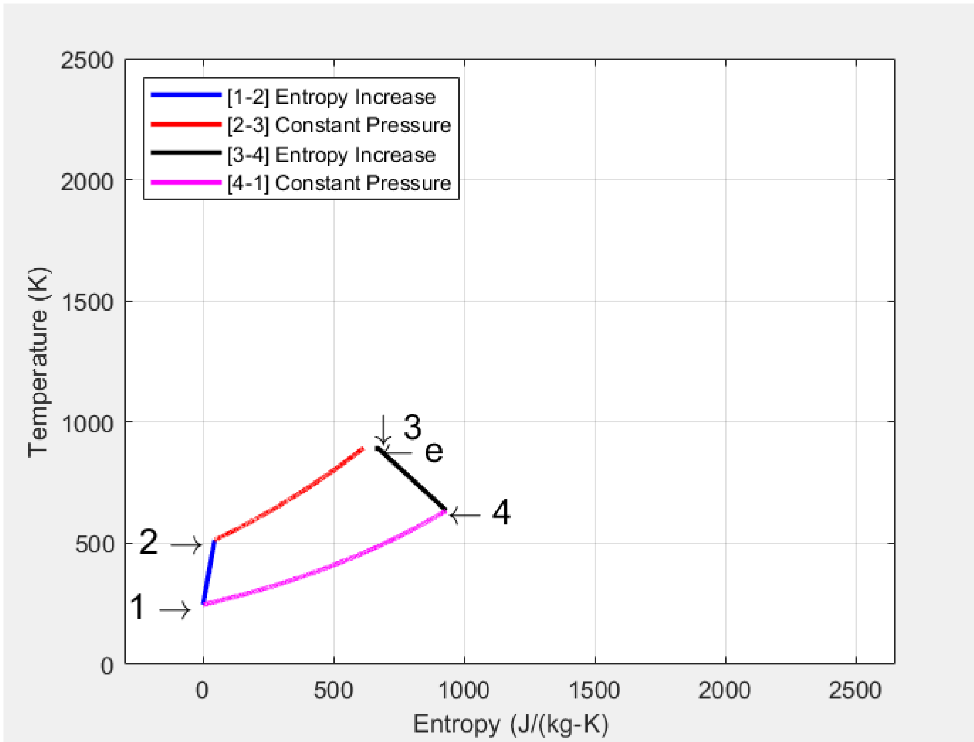


Figure 3-1-2 T-s Diagram (Case #2: Thermally Choked)



Based on the figures above, it can be seen that when the engine is non-thermally choked that the area enclosed is much larger than when the engine is thermally choked. The area enclosed on a T-s diagram is a visual representation of the net work (W_{net}), and therefore more work is generated when the engine is non-thermally choked. Theoretically, this makes sense because when the engine is thermally choked it limits the maximum temperature able to be obtained inside of the combuster.

4. Part B: Atmospheric Model Comparison

Now, it was assumed above in section 2.2 that the isentropic model of the atmosphere was an accurate representation of the temperature and pressure of the atmosphere but this should be compared against the data provided from the International Standard Atmosphere (ISA). As discussed in section 2.2 the isentropic model of the atmospheric uses Equations 1 and 2, while the ISA provides 42 data points from an altitude of 0 to 22000 m.

Figure 4-1-1 Altitude (z) vs Temperature Ratio (T/T_s) Comparison

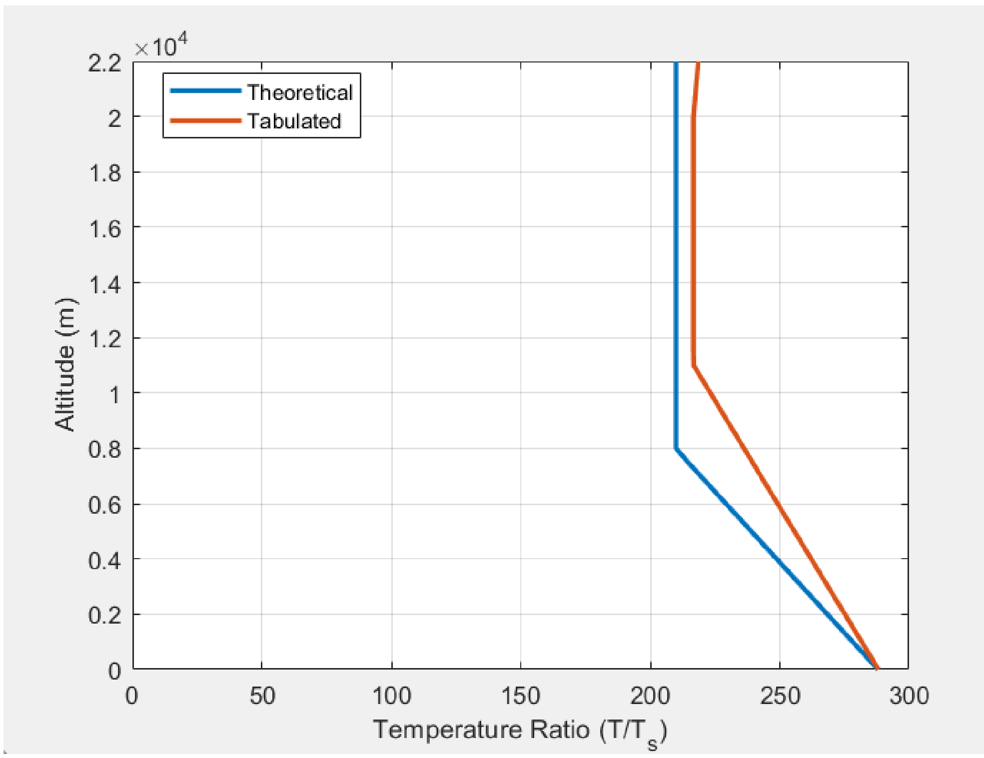
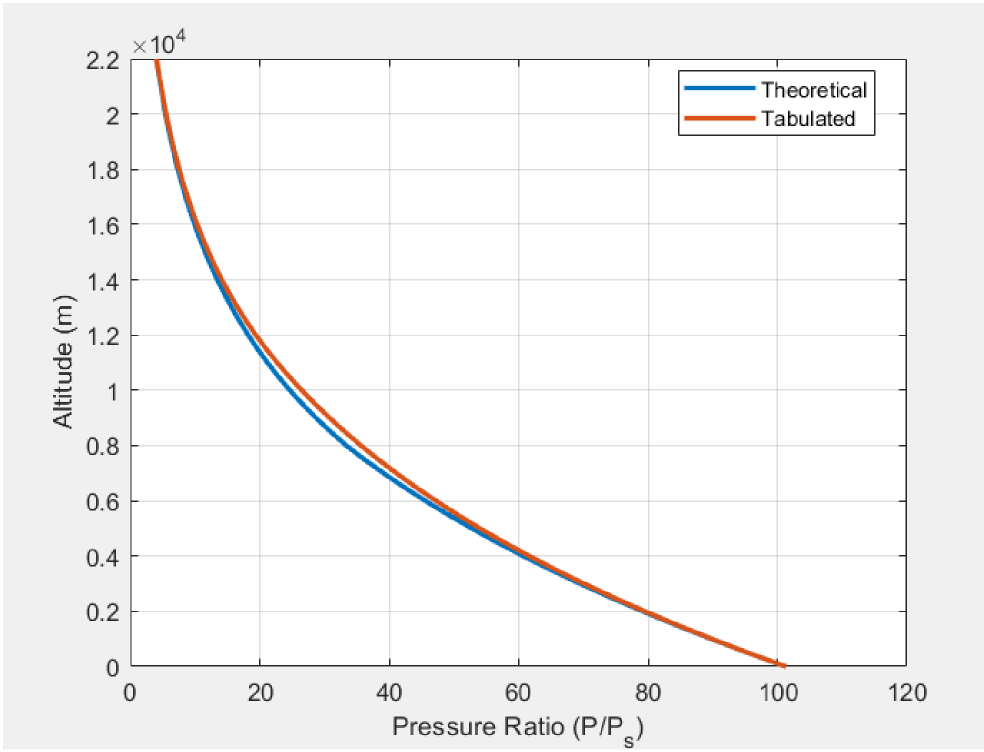


Figure 4-1-2 Altitude (z) vs Pressure Ratio (P/P_s) Comparison

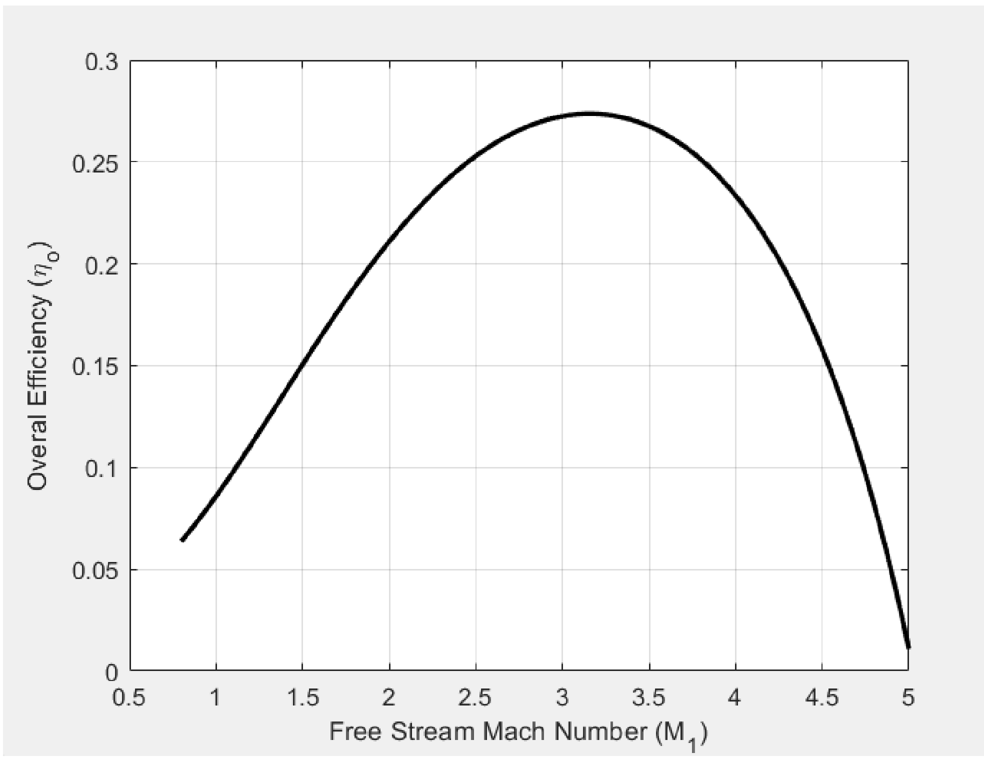


It can be seen from the figures above that the theoretical is fairly accurate when compared to the provided data from the ISA. Figure 4-1-1 shows more variation compared to Figure 4-1-2, which shows that the pressure is a little more accurate than the temperature. However, it should be noted that the data from the ISA's data can vary greatly based on the location, date, and time of day. Therefore, it can be assumed that the isentropic model of the atmosphere is an accurate representation of state 1.

5. Part C: Parameter Results With Varying M1

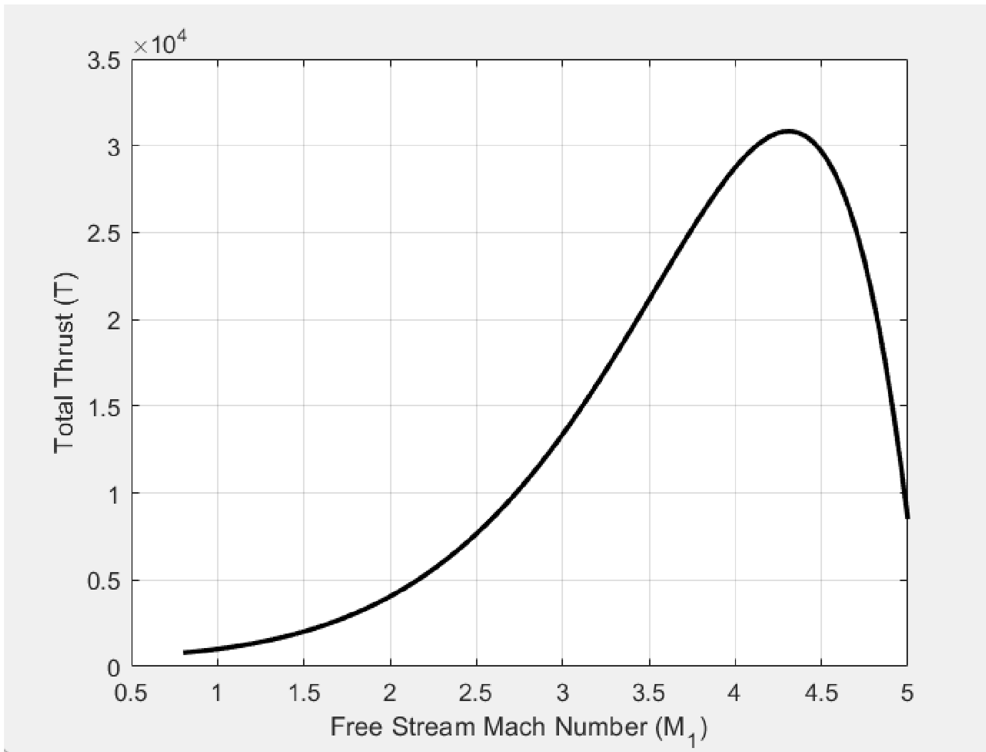
Using the non-thermally choked parameters (case #1), the function is run with varying M_1 from 0.8 to 5.0 with 100 total values. The requested output of the function was three graphs, Overall efficiency (η_o) vs Mach number at state 1 (M_1), Thrust (T) vs Mach number at state 1 (M_1), and Thrust-Specific Fuel Consumption (TSFC) vs Mach number at state 1 (M_1), as seen in Figures 5-1-1, 5-1-2, and 5-1-3.

Figure 5-1-1 Overall Efficiency (η_o) vs Free Stream Mach # (M_1)



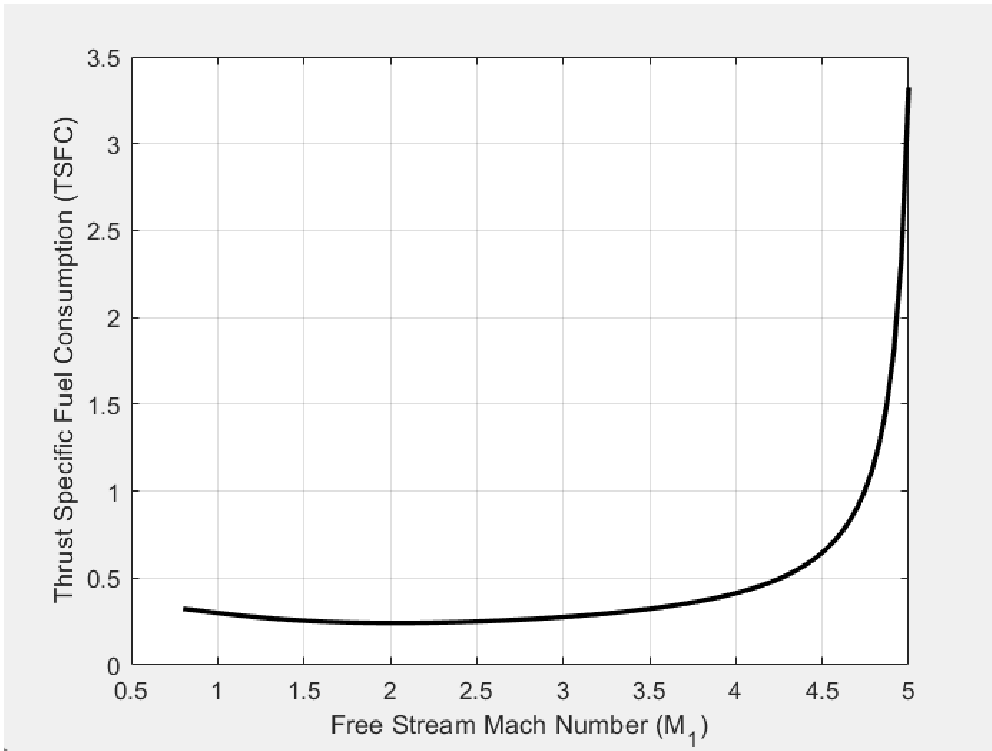
As seen in Figure 5-1-1, it can be seen that the maximum overall efficiency (η_o) is .2736 when the Mach number at state 1 (M_1) is at 3.175. However, this maximum doesn't last very long as either side of the curve descends quickly leading to a low overall efficiency (η_o) from Mach number at state 1 (M_1) .8 to 2.5 and 3.75 to 5. This leaves a small gap in the Mach number where the overall efficiency (η_o) is above .25. Theoretically, this makes sense because of the Mach number at state 2 (M_2) being held constant at .15, which would cause the air to slow down significantly and make the engine rather inefficient.

Figure 5-1-2 Total Thrust (T) vs Free Stream Mach # (M_1)



Looking at Figure 5-1-2, it can be seen that the Thrust has a lognormal curve which has a maximum Thurst of 30.83 kN at a Mach number at state 1 (M_1) of 4.321. The Thrust begins to climb slowly, then climbs rather quickly until reaching a maximum and having a steep decline. This decline potentially could be due to shocks in the oncoming air due to the supersonic speeds.

Figure 5-1-3 Thrust Specific Fuel Consumption (TSFC) vs Free Stream Mach # (M_1)

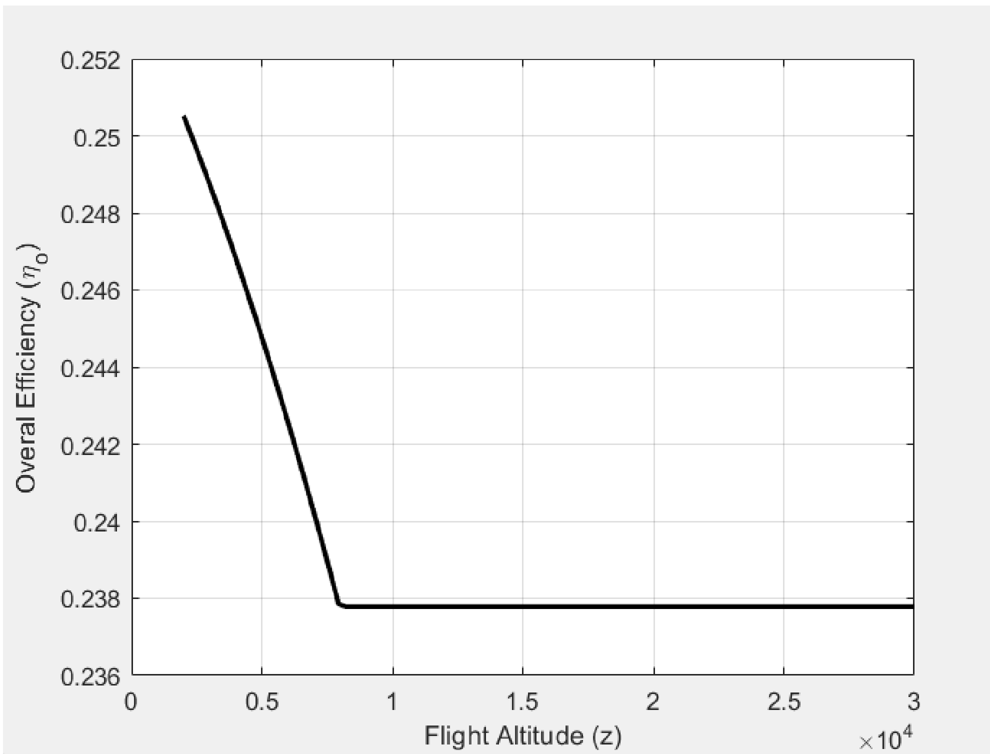


Finally, Figure 5-1-3 shows a relatively low TSFC until it begins to rapidly increase around Mach 4. The lowest value of TSFC appears to be at .241 where the Mach number at stage 1 (M_1) is 2.03. It is typical for the TSFC to be low when attempting to be fuel-efficient. This pattern is consistent with the graph above, where the fuel efficiency would go out the window when the engine's overall efficiency (η_o) is extremely low due to the high Mach number. Based on the lowest value of TSFC, it would be assumed that the most fuel-efficient flight Mach number at stage 1 (M_1) would be at or around 2.

6. Part D: Parameter Results With Varying z

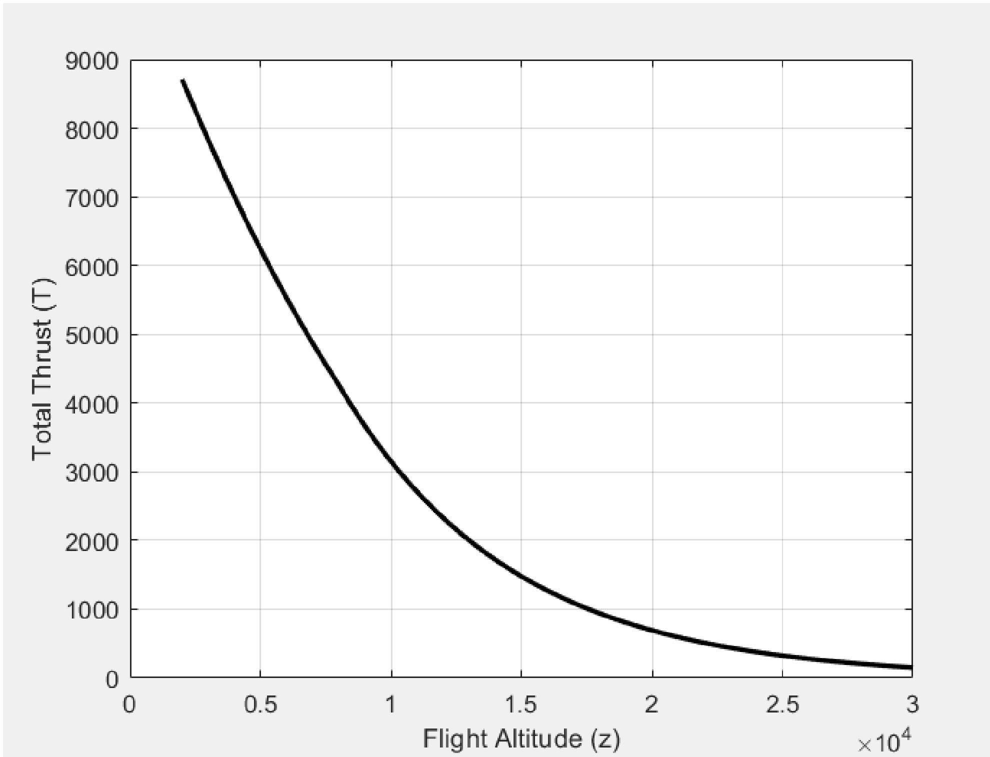
Using the non-thermally choked parameters (case #1), the function is run with varying z from 2000 m to 30000 m with 100 total values. The requested output of the function was three graphs, Overall efficiency (η_o) vs Flight Altitude (z), Thrust (T) vs Flight Altitude (z), and Thrust-Specific Fuel Consumption (TSFC) vs Flight Altitude (z), as seen in Figures 6-1-1, 6-1-2, and 6-1-3.

Figure 6-1-1 Overall Efficiency (η_o) vs Flight Altitude (z)



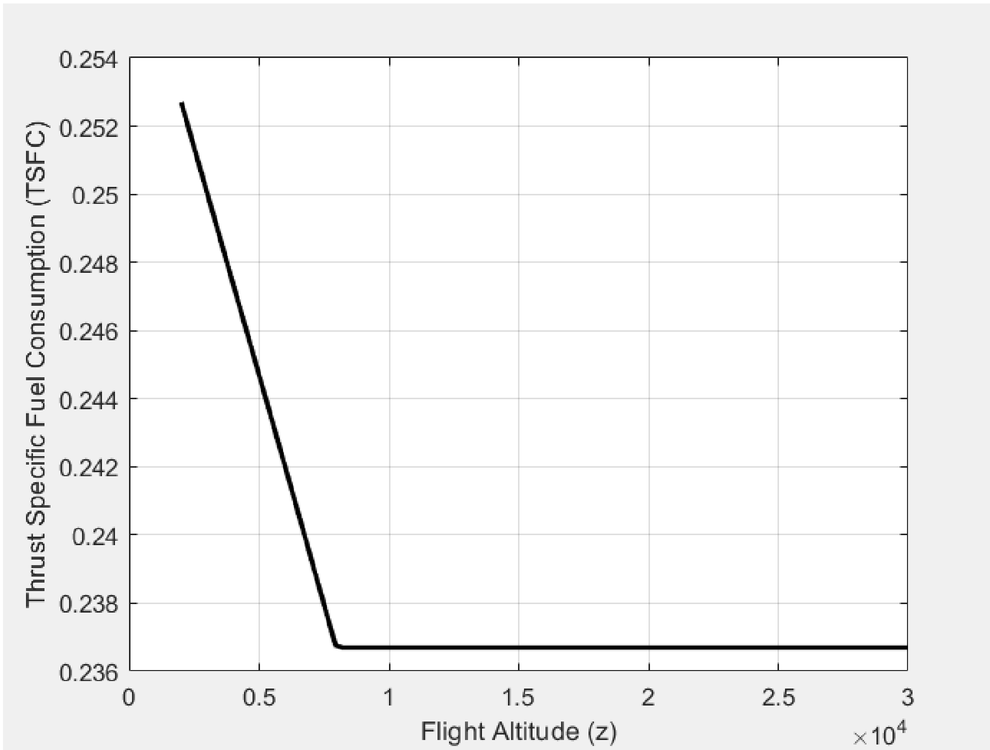
As seen in Figure 6-1-1, the overall efficiency (η_o) decreases in a linear line until reaching an efficiency of .237 and remains constant for the remainder of the flight altitude. This constant efficiency begins at around 8000 m and continues through 30000 m. This makes intuitive sense based on section 2.2 where the temperature becomes constant at about 8000 m. However, It is interesting that the overall efficiency (η_o) is decreasing the higher it is. This must be because there is less air in the atmosphere as the altitude rises, and thus the air entering the engine decreases causing the efficiency to decrease.

Figure 6-1-2 Total Thrust (T) vs Flight Altitude (z)



Looking at Figure 6-1-2, the Thurst decreases in an exponential line. Similar to the overall efficiency (η_o), it can be seen that because there is less air in the atmosphere as the altitude rises, and thus the air entering the engine decreases causing the thrust to decrease.

Figure 6-1-3 Thrust Specific Fuel Consumption (TSFC) vs Flight Altitude (z)

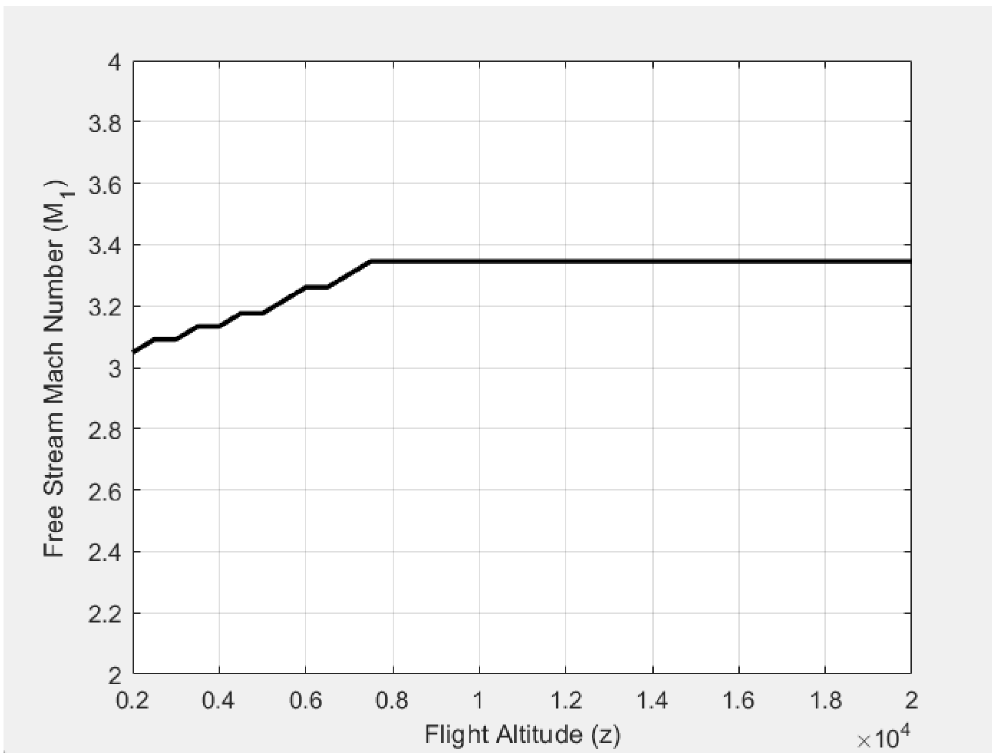


Finally, in Figure 6-1-3 a similar relationship for TSFC can be seen with overall efficiency (η_o). It decreases in a linear line until reaching a TSFC of .237 and remains constant for the remainder of the flight altitude. This constant TSFC begins at around 8000 m and continues through 30000 m.

7. Part E: Parameter Results With Varying z and M1

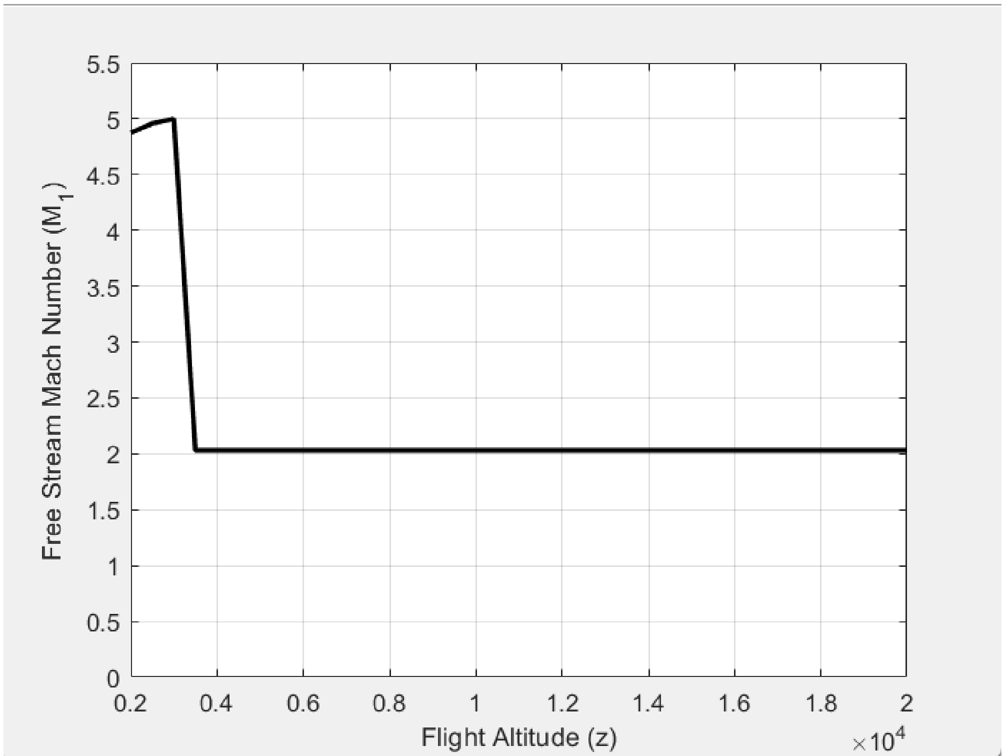
Using the non-thermally choked parameters (case #1), the function is run with varying z from 2000 m to 20000 m varying the steps by 500 m. For each of these values of z, vary M_1 from 0.8 to 5.0 with 100 total values. The requested output of the function was two graphs, Mach number at state 1 (M_1) that maximizes Overall Efficiency (η_o) vs Flight Altitude (z) and Mach number at state 1 (M_1) that minimizes Thrust-Specific Fuel Consumption (TSFC) vs Flight Altitude (z), as seen in Figures 7-1-1 and 7-1-2.

Figure 7-1-1 Free Stream Mach # (M_1) vs Flight Altitude (z) Optimized for Maximum η_o



Referencing Figure 7-1-1, it can be seen that as the optimized Mach number at state 1 (M_1) begins just over 3, and increases in a stepped manner until reaching a flight altitude of 7500 m, where the M_1 remains constant at 3.345 for the remaining altitudes. Similar to the rationale in section 6, this is about where the isentropic model of the atmosphere has temperature remaining constant.

Figure 7-1-2 Free Stream Mach # (M_1) vs Flight Altitude (z) Optimized for Minimum TSFC

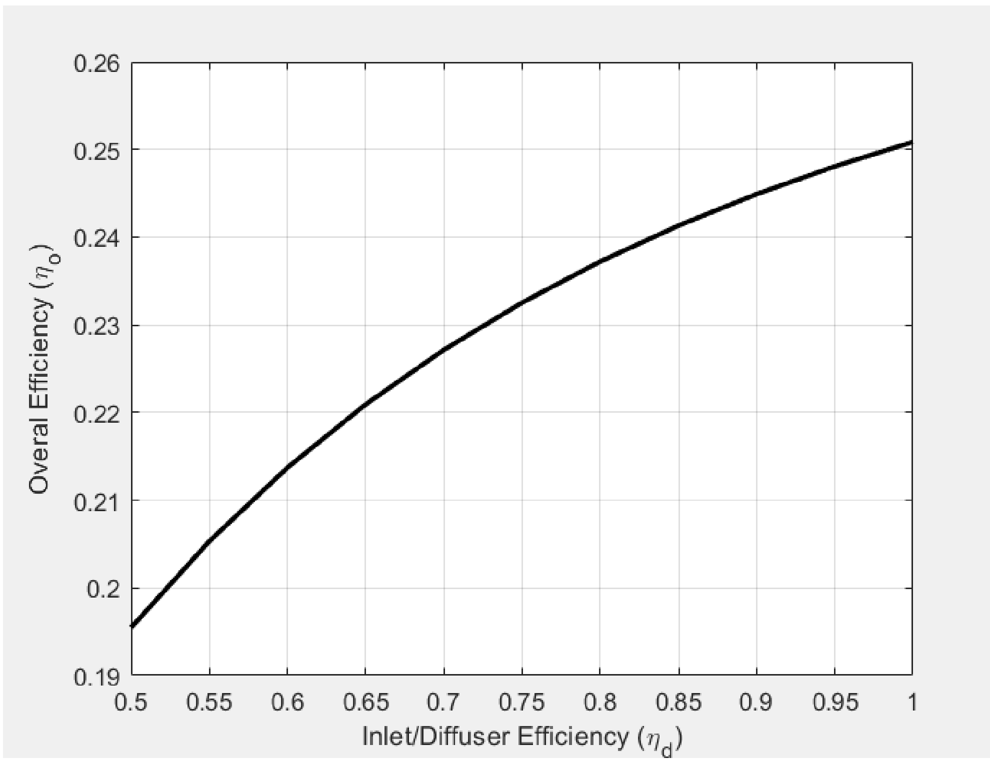


Looking at Figure 7-1-2, it is a much different graph than Figure 7-1-1. Beginning with an extremely high value of Mach number for the minimum TSFC, increasing a little, and then a steep decline until remaining constant at around 3500 m with a Mach number of about 2. Thinking about why the value of Mach is so large at the beginning, it can be remembered that air is in higher volumes at lower altitudes and the engine can produce more thrust without using as much fuel. The rapid decrease is surprising but when considering the decrease in air density and the cost to go faster, there appears to be this Mach number of 2 where the TSFC is minimized while still maintaining supersonic speeds.

8. Part F: Parameter Results With Varying η_D

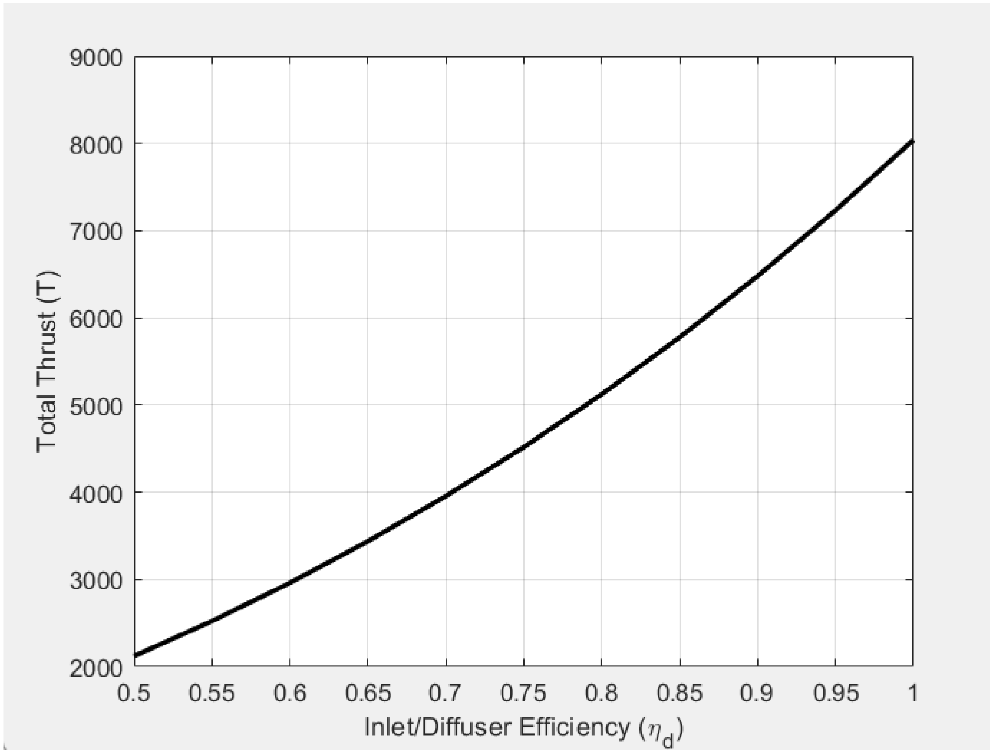
Using the non-thermally choked parameters (case #1), the function is run with varying inlet/diffuser efficiency (η_d) from .5 to 1 with steps of .05. The requested output of the function was three graphs, Overall efficiency (η_o) vs inlet/diffuser efficiency (η_d), Thrust (T) vs inlet/diffuser efficiency (η_d), and Thrust-Specific Fuel Consumption (TSFC) vs inlet/diffuser efficiency (η_d), as seen in Figures 8-1-1, 8-1-2, and 8-1-3.

Figure 8–1-1 Overall Efficiency (η_o) vs Inlet/Diffuser Efficiency (η_d)



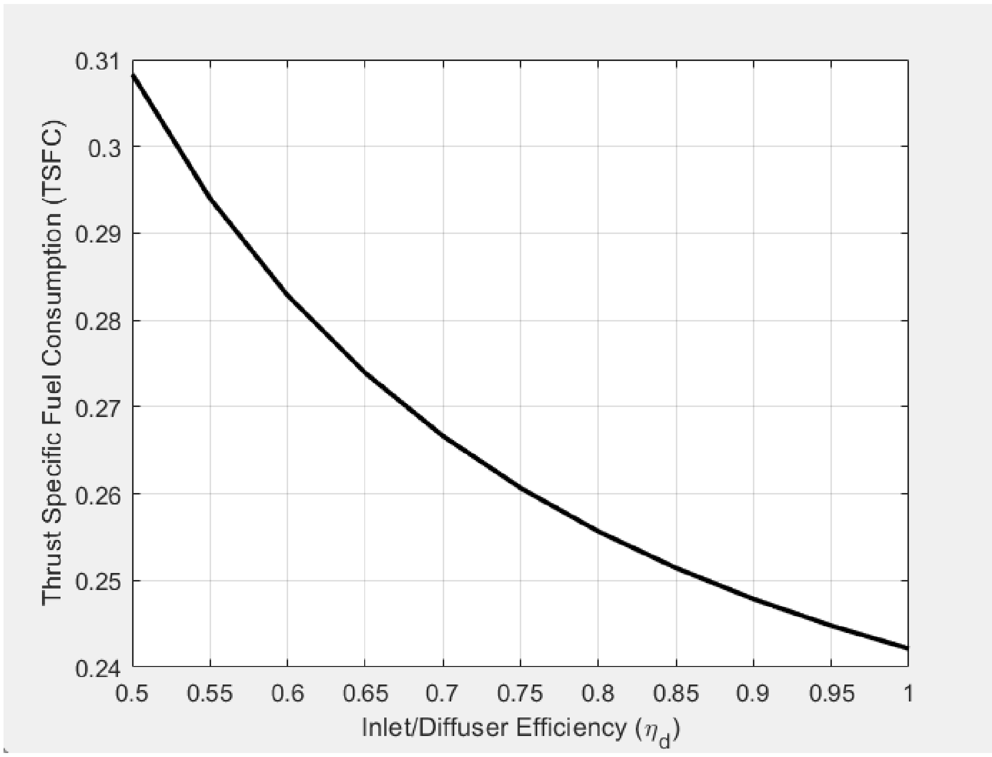
As seen in Figure 8-1-1, an expected curve is seen with increasing inlet/diffuser efficiency (η_d) the overall efficiency (η_o) also increases. As can be expected, the highest overall efficiency is seen when the inlet/diffuser efficiency is 1, which would make the system reversible.

Figure 8-1-2 Total Thrust (T) vs Inlet/Diffuser Efficiency (η_d)



In Figure 8-1-2, a positive increase in thrust can be seen as the inlet/diffuser efficiency (η_d) increases. Again, this is expected because as the inlet/diffuser becomes closer to reversible, the ram compression can retain higher pressure.

Figure 8-1-3 Thrust Specific Fuel Consumption (TSFC) vs Inlet/Diffuser Efficiency (η_d)

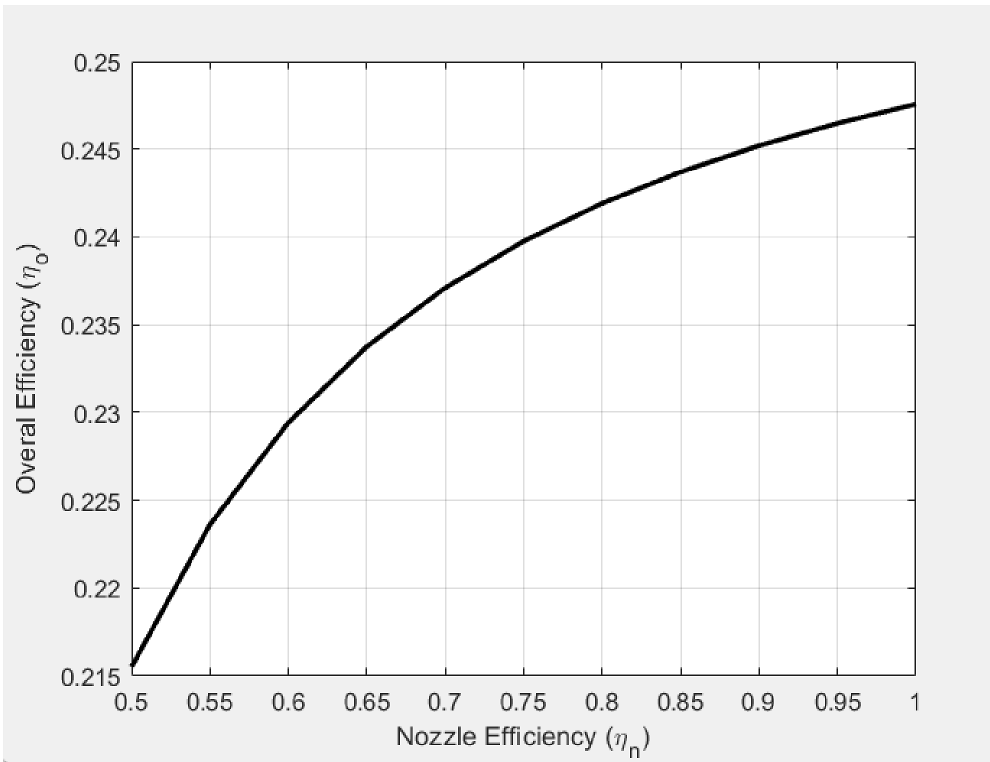


Looking at Figure 8-1-3, a negative exponential curve is seen showing that as the inlet/diffuser efficiency increases, the TSFC decreases. As above, this makes intuitive sense because as the inlet/diffuser becomes more efficient, so does the ram/scramjet fuel efficiency.

9. Part G: Parameter Results With Varying η_N

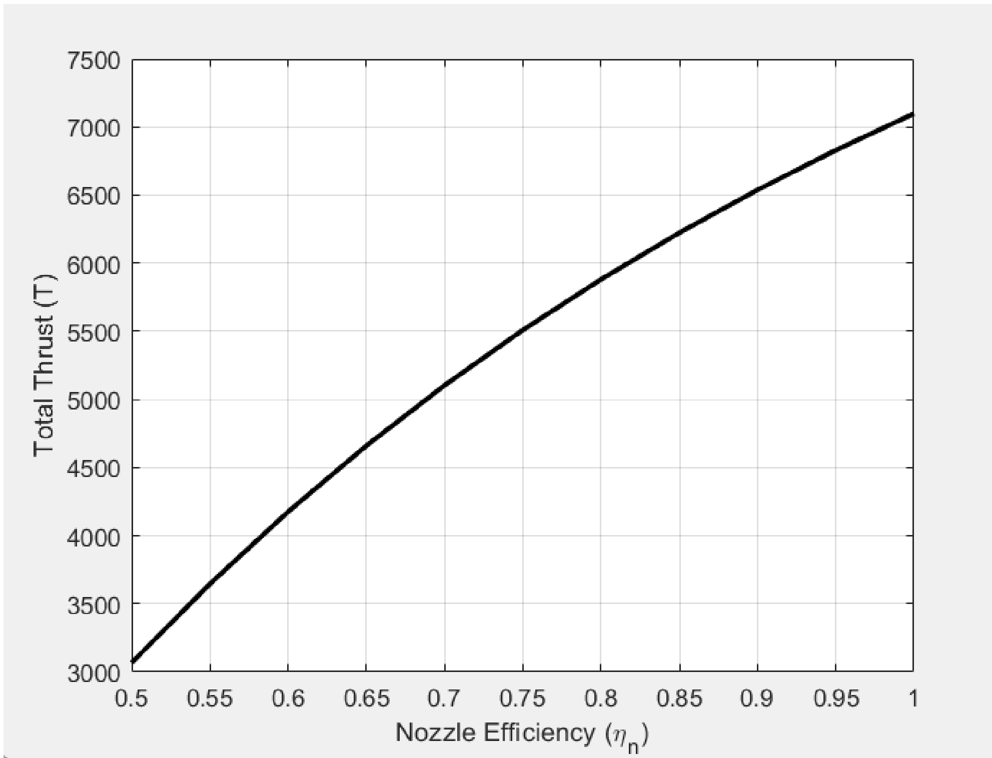
Using the non-thermally choked parameters (case #1), the function is run with varying nozzle efficiency (η_n) from .5 to 1 with steps of .05. The requested output of the function was three graphs, Overall efficiency (η_o) vs nozzle efficiency (η_n), Thrust (T) vs nozzle efficiency (η_n), and Thrust-Specific Fuel Consumption (TSFC) vs nozzle efficiency (η_n), as seen in Figures 9-1-1, 9-1-2, and 9-1-3.

Figure 9-1-1 Overall Efficiency (η_o) vs Nozzle Efficiency (η_n)



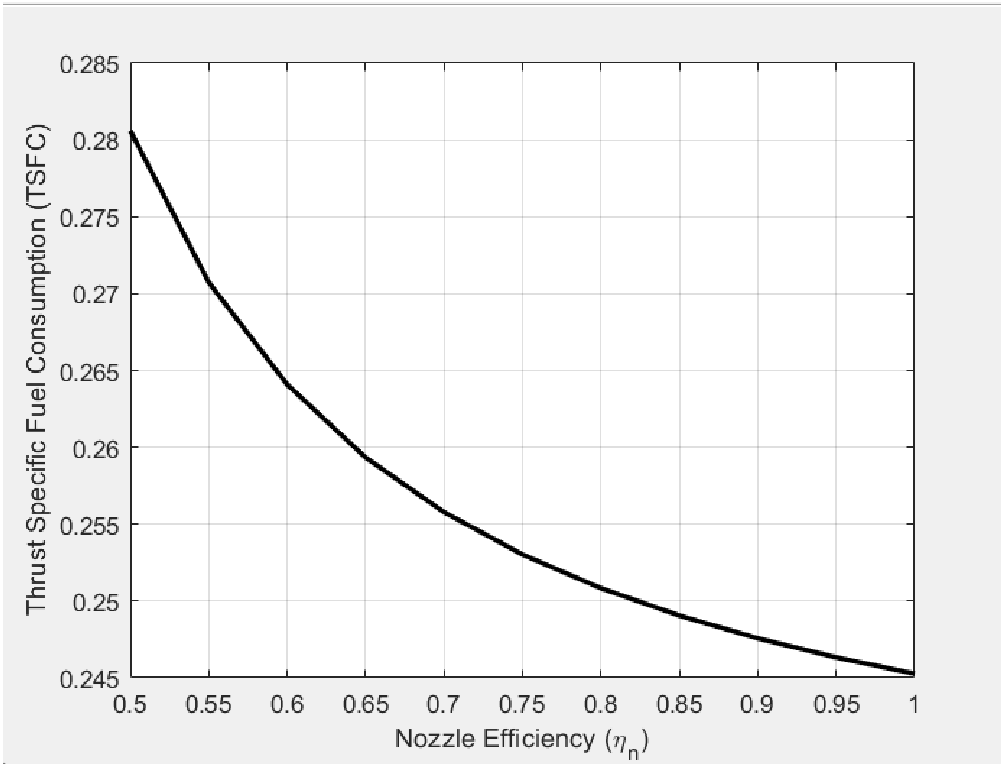
Looking at Figure 9-1-1, it can be seen that this follows the same trend above as seen in Figure 8-1-1. Therefore, as the nozzle efficiency (η_n) increases towards reversible, the Overall efficiency (η_o) increases.

Figure 9–1-2 Total Thrust (T) vs Nozzle Efficiency (η_n)



Again, looking at Figure 9-1-2 it can be seen that this follows the same trend above as seen in Figure 8-1-2. Therefore, as the nozzle efficiency (η_n) increases towards reversible, the thrust increases even more positively than in Figure 9-1-1. It should be noted that in comparison to the variation of inlet/diffuser efficiency (η_d), the nozzle efficiency (η_n) has slightly less impact on the thrust with the max thrust being about 800 N less.

Figure 9-1-3 Thrust Specific Fuel Consumption (TSFC) vs Nozzle Efficiency (η_n)

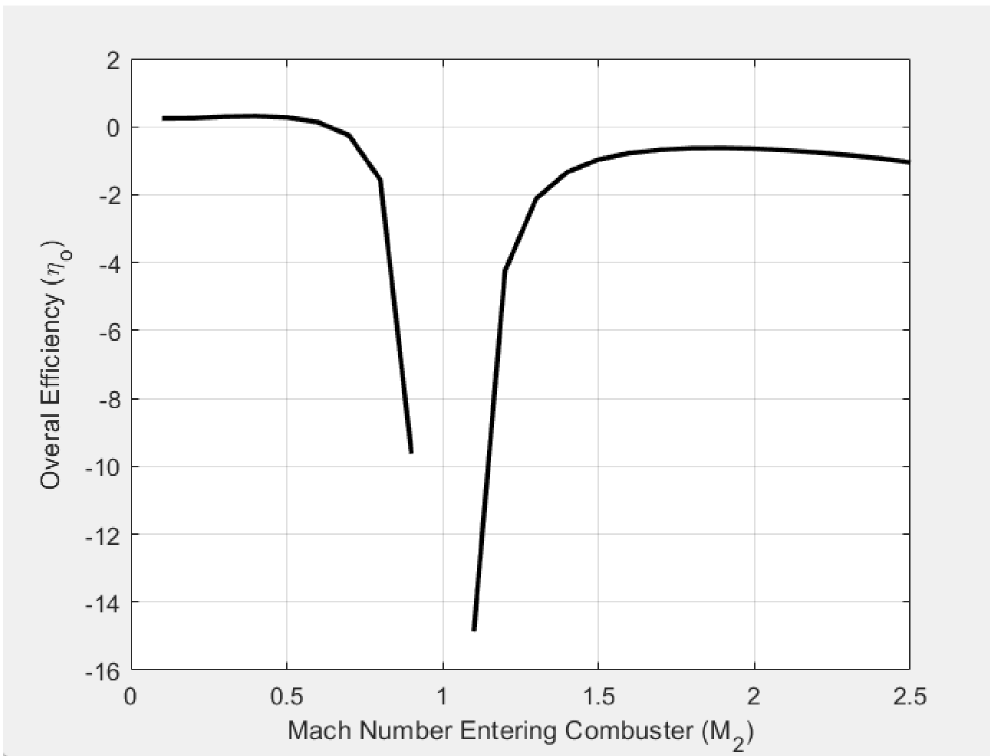


Finally, Figure 9-1-3 can be seen that this follows the same trend above as seen in Figure 8-1-3. As the nozzle efficiency (η_n) increases towards reversible, the TSFC decreases leading to a more fuel-efficient engine. It should be noted that in comparison to the variation of inlet/diffuser efficiency (η_d), the nozzle efficiency (η_n) has slightly less impact on the TSFC with a change of about .035 compared to about .6.

10. Part H: Parameter Results With Varying M2

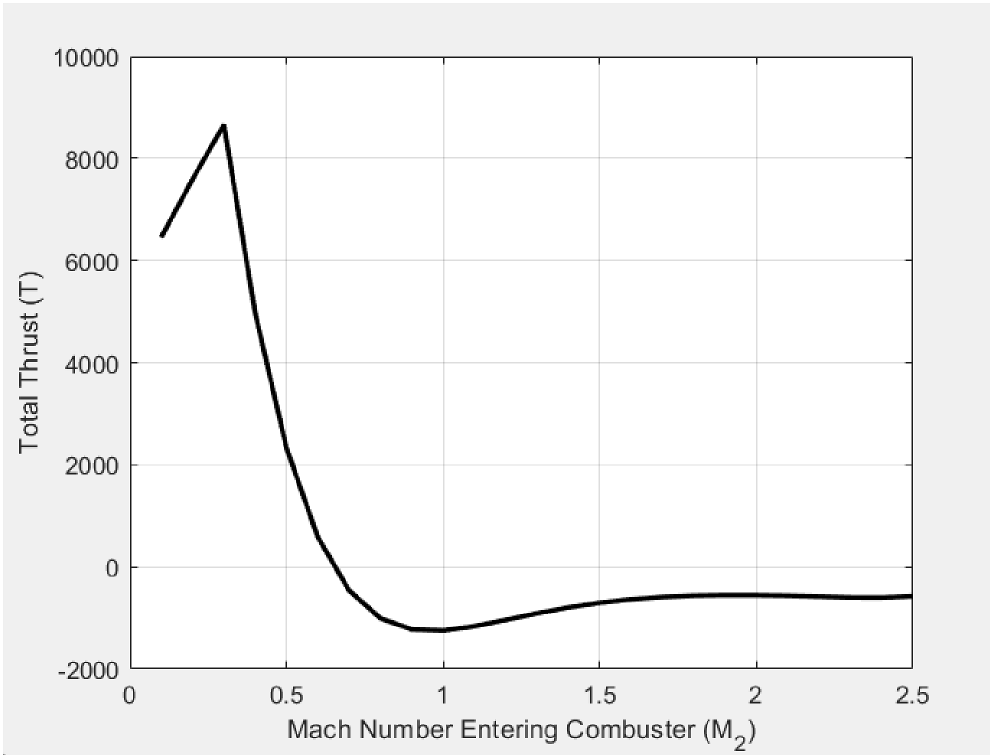
Using the non-thermally choked parameters (case #1), the function is run with varying Mach number at state 2 (M_2) from .1 to 2.5 with steps of .1. The requested output of the function was three graphs, Overall efficiency (η_o) vs Mach number at state 2 (M_2), Thrust (T) vs Mach number at state 2 (M_2), and Thrust-Specific Fuel Consumption (TSFC) vs Mach number at state 2 (M_2), as seen in Figures 10-1-1, 10-1-2, and 10-1-3.

Figure 10-1-1 Overall Efficiency (η_o) vs Mach # Entering Combuster (M_2)



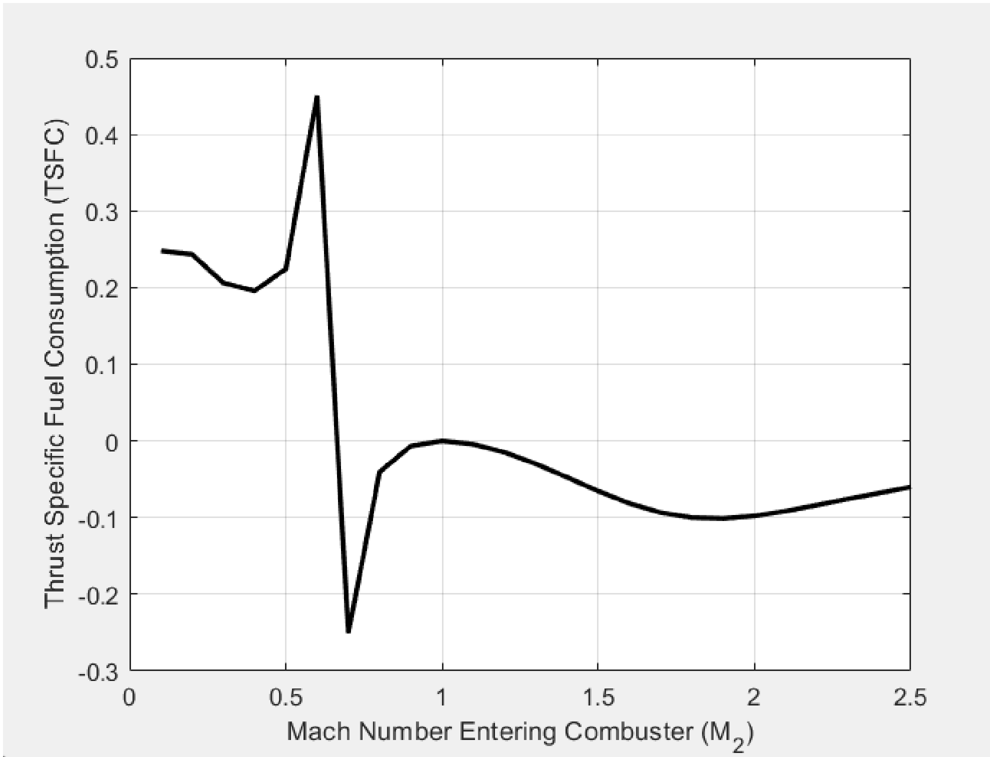
Looking at Figure 10-1-1, we see that the Overall efficiency (η_o) is mostly negative for every value of Mach number at state 2 (M_2), except for the range of .1 to .6. When M_2 approaches 1, it appears to be an asymptote making the Overall efficiency (η_o) go to negative infinity. Even after M_2 goes beyond 1, it remains with a negative value for Overall efficiency (η_o). The reasoning for this is most likely due to the thermal choking which results when M_2 becomes too large forcing M_3 to be equal to 1.

Figure 10-1-2 Total Thrust (T) vs Mach # Entering Combuster (M_2)



Looking at Figure 10-1-2, we see that the thrust initially increases until reaching a value of M_2 of .3, from here the thrust sharply decreases until reaching a minimum at 1. From here, the thrust rises slightly becoming fairly constant with a value of around -550 N. Again, this is most likely due to the combuster being thermally choked and resulting in very low thrust performance. Going back to Figure 3-1-2, it can be seen that the net work is minimal for the thermally choked case and now comparing the thermally choked scenarios thrust performance to non-thermally choked it is easy to see the drastic performance downgrade.

Figure 10-1-3 Thrust Specific Fuel Consumption (TSFC) vs Mach # Entering Combuster (M_2)



Now, Figure 10-1-3 shows an interesting graph with a lot of changes in TSFC based on M_2. It initially starts high at .25 and then climbs higher to .45, where the M_2 is .6. At this time, the TSFC rapidly decreases to a negative value and remains negative for the remainder of the M_2 values. These don't practically make much sense, but it is most likely caused by the thermally choked combuster.

11. Part I: Design Ram/Scram Jet

Finally, the last section uses the function generated above to design a ram/scramjet engine system that has an optimal design of maximum positive thrust and maximum overall efficiency. Using the given input parameters in Table 11-1-1, vary the Mach number at state 2 (M_2) and the combuster exit total temperature (T_{t3}) to find the optimal design.

Table 11-1-1 Design Inputs

Variable	Value
M_1	5
z	27400 m
η_d	.92
q_f	43.2 MJ/kg
$(T_{t3,})_{max}$	2400 K
η_n	.94
A_e	0.015 m^2
γ_{12}	1.4
γ_{34}	1.3
R	286.9 J/(kg-K)

Now, to determine the optimal design for maximum positive thrust and maximum overall efficiency, it was requested to vary M_2 and T_{t3}. To complete this M_2 was varied from .1 to 2.5, as was previously done above in section 10. Varying M_2, this will directly affect T_{t3} based on section 2.4 where the combuster $(T_{t3})_{choked}$ is calculated. After varying M_2 and storing the resulting T_{t3}, Overall efficiency (η_o), and Thrust (T), a plot showing the 3-axis can be generated for both Overall efficiency (η_o) vs combuster exit total temperature (T_{t3}) vs Mach number at state 2 (M_2), Thrust (T) vs combuster exit total temperature (T_{t3}) vs Mach number at state 2 (M_2), as shown in Figure 11-1-1 and 11-1-2.

Figure 11-1-1 Overall Efficiency (η_o) vs Combuster Exit Total Temperature (T_{t3}) vs Mach # Entering Combuster (M_2)

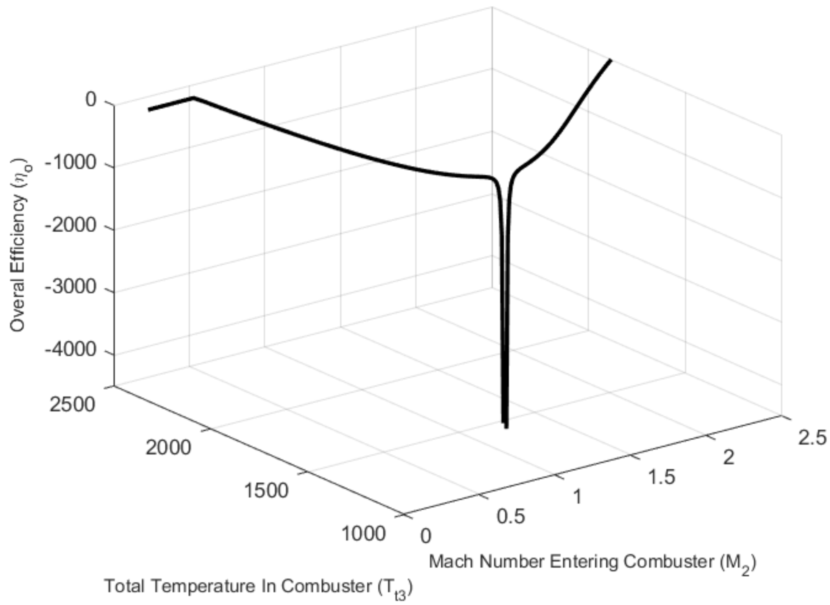
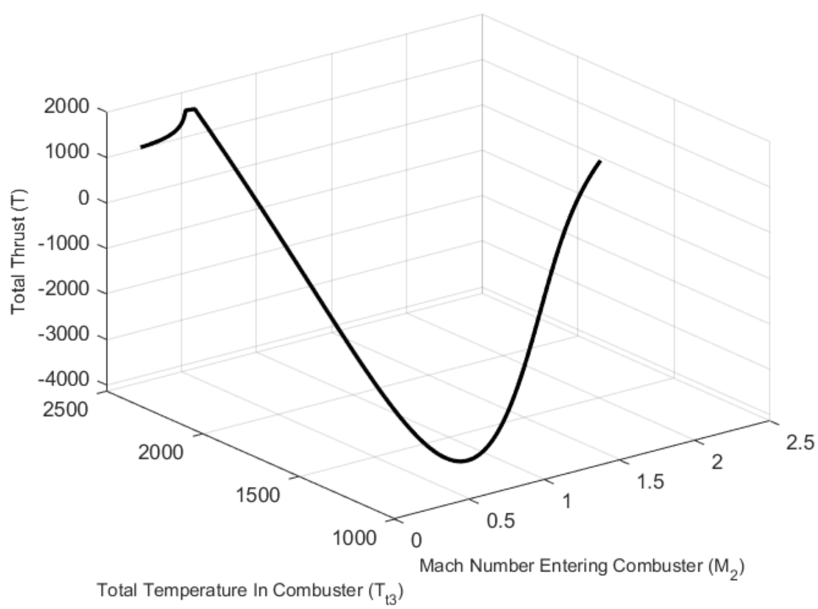


Figure 11-1-2 Total Thrust (T) vs Combuster Exit Total Temperature (T_{t3}) vs Mach # Entering Combuster (M_2)



The results generated from the graphs above found an optimal design for maximum overall efficiency (η_o) and thrust based on Mach number at state 2 (M_2) and combuster exit total temperature (T_{t3}), as seen in Tables 11-1-2 and 11-1-3.

Table 11-1-2 Optimal Design #1: Overall Efficiency

Variable	Value
M_2	0.4
T_{t3}	2400 K
η_o	.1304

Table 11-1-3 Optimal Design #2: Thrust

Variable	Value
M_2	.41
T_{t3}	2359.94 K
T	1970.41 N

When referencing the resulting figures above, it can be seen that when rotating to be in the x-z reference frame that the results are exact of Figures 10-1-1 and 10-1-2. The difference comes with the corresponding y-axis of the resultant T_{t3}, where we get the optimal results in Table 11-1-2 and 11-1-3.

Additional methods were attempted, the most notable was varying M_2 and the fuel-air ratio (f) into the combuster. This was a rather practical method, which resulted in the same results shown in Tables 11-1-2 and 11-1-3. By completing this other method, it was confirmed that the optimal design values above were validated.

12. Final Results/Discussion

Upon completing the function created in sections 2.2 to 2.8, various inputs could be tested to obtain results that model the ram/scramjet. After performing a variety of these inputs in sections 3 to 10, it can be concluded that changing a single parameter can have a significant impact on the engine performance. Depending on the scope of the aircraft or mission that is being performed, certain design or achieved parameters should be taken into consideration, such as flight Mach number (M_1), Mach number at state 2 (M_2), inlet/diffuser efficiency (η_d), nozzle efficiency (η_n), flight altitude (z), and others. Throughout this report, a variety of graphs have shed light on the complexity of the ram/scramjet design and how it performs for those various parameters.

After reviewing all the results above, it was particularly interesting to the see effect of thermal choking, for overall efficiency and total thrust in particular. When performing sections 10 and 11, it was strange to see a majority of values result negatively for overall efficiency, thrust, and TSFC. Continuing after section 10 and going into other input parameters, or furthering the limits for the parameters used in sections 5 to 10 would be interesting to see if the results continue as expected or alter after extreme circumstances.

In summary, this report details the equations necessary to generate a function for the ram/scramjet engine cycle, along with a variety of calculations using the function to generate output performance parameters and compare the results from a variety of input parameters. An optimal design was calculated based on a few specific input parameters. The calculations and graph results were all completed using a function in MATLAB, which is detailed in the Appendix below.

13. Appendix

13.1 Module 1: Free Stream Matlab Function

```
function [T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z,M1,g12,R,zs,Ps,Ts)
%
% Module_1 solves for the conditions at state 1 of the ram jet engine.
% State 1 represents the free stream.
%
% Input:
%   z: Flight Altitude
%   M1: Mach Number at State 1
%   g12: Gamma before the combuster
%   R: Gas Constant
%   zs: Isentropic Flight Altitude
%   Ps: Isentropic Pressure
%   Ts: Isentropic Temperature

% Ouput:
%   T1: Temperature at State 1
%   P1: Pressure at State 1
%   Tt1: Total Temperature at State 1
%   Pt1: Total Pressure at State 1
%   Cp1: Specific Heat at State 1
%   a1: Speed of Sound at State 1
%   V1: Velocity at State 1

  if z <= 7958

    T1 = Ts*(1-((g12-1)/g12)*(z/zs)) ; % K
    P1 = Ps*(1-((g12-1)/g12)*(z/zs))^(g12/(g12-1)) ; % kpa

  else

    T1 = 210 ; % K
    P1 = 33.6*exp((z-7958)/6605) ; % kpa

  end

  Tt1 = T1*(1+((g12-1)/2)*M1^2) ; % K

  Pt1 = P1*(1+((g12-1)/2)*M1^2)^(g12/(g12-1)) ;
```

```
    Cp1 = (g12*R)/(g12-1) ; % J/(kg-K)

    a1 = sqrt(g12*R*T1) ; % m/s

    V1 = M1*a1 ; % m/s

end
```

13.2 Module 2: Inlet/Diffuser Matlab Function

```
function [Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1,M2,Tt1,P1,Pt1,Cp1,eta_d)
%
% Module_2 solves for the conditions at state 2 of the ram jet engine.
% State 2 represents the Inlet/Diffuser.
%
% Input:
%   g12: Gamma before the combuster
%   R: Gas Constant
%   M1: Mach at State 1
%   M2: Mach at State 2
%   Tt1: Total Temperature at State 1
%   P1: Pressure at State 1
%   Pt1: Total Pressure at State 1
%   eta_d: Diffuser efficiency
%   Cp1: Specific Heat at State 1

% Ouput:
%   Tt2: Total Temperature at State 2
%   T2: Temperature at State 2
%   Pt2: Total Pressure at State 2
%   P2: Pressure at State 2
%   Cp2: Specific Heat at State 2
%   a2: Speed of Sound at State 2
%   V2: Velocity at State 2
%   ds12: Entropy Change From State 1 to State 2

  Tt2 = Tt1 ; % K

  T2 = Tt2/(1+((g12-1)/2)*M2^2) ; % K

  Pt2 = P1*(1+eta_d*((g12-1)/2)*M1^2)^(g12/(g12-1)) ; % kpa

  P2 = Pt2/((1+((g12-1)/2)*M2^2)^(g12/(g12-1))) ; % kpa

  Cp2 = (g12*R)/(g12-1) ; % J/(kg-K)

  a2 = sqrt(R*g12*T2) ; % m/a

  V2 = M2*a2 ; % m/a
```

```
    ds12 = Cp2*log(Tt2/Tt1) - R*log(Pt2/Pt1) ;
end
```

13.3 Module 3: Combuster Matlab Function

```
function [Tt3, M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2,Tt2,P2,Pt2,a,b,ds12)
%
% Module_3 solves for the conditions at state 3 of the ram jet engine.
% State 3 represents the Combuster.
%
% Input:
%   g34: Gamma After the combuster
%   R: Gas Constant
%   Tt3_max: Max Temperature in the combuster
%   M2: Mach at State 2
%   T2: Temperature at State 2
%   Tt2: Total Temperature at State 2
%   P2: Pressure at State 2
%   Pt2: Total Pressure at State 2
%   a: Value for Cp expression
%   b: Value for Cp expression
%   ds12: Entropy Change From State 1 to State 2

% Ouput:
%   Tt3: Total Temperature at State 3
%   M3: Mach # at State 3
%   q23: Heat input from State 2 to State 3
%   T3: Temperature at State 3
%   P3: Pressure at State 3
%   Pt3: Total Pressure at State 3
%   Cp3: Specific Heat at State 3
%   a3: Speed of Sound at State 3
%   V3: Velocity at State 3
%   ds23: Entropy Change From State 2 to State 3
%   ds13: Entropy Change From State 1 to State 3

  Tt3_choked = Tt2*((1/(2*(g34+1)))*(1/M2^2)*((1+g34*M2^2)^2)*((1+((g34-1)/2)*M2^2)^-1)) ;

  if Tt3_choked < Tt3_max % Choked Flow

     M3 = 1 ;
     Tt3 = Tt3_choked ; % K

  else % Not Choked
```

```
        Tt3 = Tt3_max ; % K

        C = (Tt3/Tt2)*((1+((g34-1)/2)*M2^2)/((1+g34*M2^2)^2))*M2^2 ;
        X = C*g34^2-((g34-1)/2) ;
        Y = 2*C*g34-1 ;
        % Z1 = (-Y+sqrt(Y^2-4*X*C))/(2*X) ;
        Z2 = (-Y-sqrt(Y^2-4*X*C))/(2*X) ;

        M3 = sqrt(Z2) ;

    end

    q23 = (a*(Tt3-Tt2)+.5*b*(Tt3^2-Tt2^2))/1e6 ; % MJ/kg

    T3 = Tt3/(1+((g34-1)/2)*M3^2) ; % K

    P3 = P2 ; % kpa

    Pt3 = P3*((1+((g34-1)/2)*M3^2)^(g34/(g34-1))) ; % kpa

    Cp3 = a+b*T3 ; % J/(kg-K)

    a3 = sqrt(R*g34*T3) ; % m/a

    V3 = M3*a3 ; % m/a

    ds23 = Cp3*log(Tt3/Tt2) - R*log(Pt3/Pt2) ;

    ds13 = ds12 + ds23 ;

end
```

13.4 Module 4: Converging Nozzle Matlab Function

```
function [Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n,Tt3,P1,Pt3,a,b,ds12,ds23)
%
% Module_4 solves for the conditions at state e of the ram jet engine.
% State e represents the Converging Nozzle.
%
% Input:
%   g34: Gamma After the combuster
%   R: Gas Constant
%   Ae: Area of the nozzle exit
%   eta_n: Nozzle Efficiency
%   Tt3: Total Temperature at State 3
%   P1: Pressure at State 1
%   Pt3: Total Pressure at State 3
%   a: Value for Cp expression
%   b: Value for Cp expression
%   ds12: Entropy Change From State 1 to State 2
%   ds23: Entropy Change From State 2 to State 3

% Ouput:
%   Me: Mach # at State e
%   Pte: Total Pressure at State e
%   Pe: Pressure at State e
%   Tte: Total Temperature at State e
%   Te: Temperature at State e
%   Cpe: Specific Heat at State e
%   ae: Speed of Sound at State e
%   Ve: Velocity at State e
%   m_dote: Mass Flow Rate at State e
%   ds3e: Entropy Change From State 3 to State e
%   ds1e: Entropy Change From State 1 to State e

   M_prime =
((2/(g34-1))*((eta_n*(1-(P1/Pt3)^((g34-1)/g34))/(1-eta_n*(1-(P1/Pt3)^((g34-1)/g34))))))^.5 ;

   if M_prime < 1 % Not Choked

      Me = M_prime ;

      Pe = P1 ; % kpa
```

```
    else % Choked
        Me = 1 ;
        Pe = Pt3*(1-(1/eta_n)*((g34-1)/(g34+1)))^(g34/(g34-1)) ; % kpa
    end
    Tte = Tt3 ; % K
    Te = Tte/(1+((g34-1)/2)*Me^2) ; % K
    Pte = Pe*((1+((g34-1)/2)*Me^2)^(g34/(g34-1))) ; % kpa
    Cpe = a+b*Te ; % J/(kg-K)
    ae = sqrt(R*g34*Te) ; % m/a
    Ve = Me*ae ; % m/a
    m_dote = ((Pe/(R*Te))*Ve*Ae)*1000 ; % kg/s
    ds3e = Cpe*log(Tte/Tt3) - R*log(Pte/Pt3) ;
    ds1e = ds12 + ds23 + ds3e ;
end
```

13.5 Module 5: Flow Past The Nozzle Exit Matlab Function

```
function [M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e)
%
% Module_5 solves for the conditions at state 4 of the ram jet engine.
% State 4 represents the flow past the nozzle exit.
%
% Input:
%   g34: Gamma After the combuster
%   R: Gas Constant
%   Ae: Area of the nozzle exit
%   eta_n: Nozzle Efficiency
%   Tt3: Total Temperature at State 3
%   P1: Pressure at State 1
%   Pt3: Total Pressure at State 3
%   a: Value for Cp expression
%   b: Value for Cp expression
%   ds12: Entropy Change From State 1 to State 2
%   ds23: Entropy Change From State 2 to State 3

% Output:
%   M4: Mach # at State 4
%   Pt4: Total Pressure at State 4
%   P4: Pressure at State 4
%   Tt4: Total Temperature at State 4
%   T4: Temperature at State 4
%   Cp4: Specific Heat at State 4
%   a4: Speed of Sound at State 4
%   V4: Velocity at State 4
%   dse4: Entropy Change From State e to State 4
%   ds14: Entropy Change From State 1 to State 4

  if M_prime < 1

    eta_n_ext = 1 ;

  else

    eta_n_ext = M_prime^(-.3) ;

  end
```

```
    Tt4 = Tte ; % K

    T4 = Tte*(1-eta_n_ext*(1-(P1/Pte)^((g34-1)/g34))) ; % K

    M4 = ((2/(g34-1)*((Tt4/T4)-1)))^.5 ;

    P4 = P1 ; % kpa

    Pt4 = P4*((1+((g34-1)/2)*M4^2)^(g34/(g34-1))) ; % kpa

    Cp4 = a+b*T4 ; % J/(kg-K)

    a4 = sqrt(R*g34*T4) ; % m/a

    V4 = M4*a4 ; % m/a

    dse4 = Cp4*log(Tt4/Tte) - R*log(Pt4/Pte) ;

    ds14 = ds12 + ds23 + ds3e + dse4 ;

end
```

13.6 Module 6: Various Parameters Matlab Function

```
function [m_dotf, m_doti, f, J_T, P_T, T_T, Veq, TSFC, Isp, eta_th, eta_p, eta_o, P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1)
%
% Module_6 solves for various parameters of the ram jet engine.
%
% Input:
%   qf: Heating Value
%   q23: Incoming Heat at State 2 to State 3
%   m_dote: Mass Flow Rate at State e
%   Pe: Total Pressure at State e
%   P1: Pressure at State 1
%   Ae: Area of the nozzle exit
%   Ve: Velocity at State e
%   V1: Velocity at State 1

% Ouput:
%   m_dotf: Fuel Mass Flow Rate
%   m_doti: Inital Mass Flow Rate
%   f: Fuel-Air Mass Ratio
%   J_T: Jet Thrust
%   P_T: Pressure Thrust
%   T_T: Total Thrust
%   Veq: Equivalent Velocity
%   TSFC: Thrust Specific Fuel Consumption
%   Isp: Specific Impulse
%   eta_th: Thermal Efficiency
%   eta_p: Propulsive Efficiency
%   eta_o: Overall Efficiency
%   eta_o: Propulsive Power

  m_doti = m_dote/(1+(q23/qf)) ; % kg/s

  m_dotf = m_dote - m_doti ; % kg/s

  f = m_dotf/m_doti ;

  J_T = m_doti*((1+f)*Ve-V1) ; % N

  P_T = (Pe-P1)*Ae*1000 ; % N
```

```
    T_T = J_T + P_T ; % N

    TSFC = (m_dotf/T_T)*60*60 ; % (kg/hr)/N

    Isp = T_T/(m_dotf*9.81) ; % s

    Veq = Ve + (Pe-P1)*1000*(Ae/m_dote) ; % m/s

    eta_th = ((m_dote*.5*Veq^2)-(m_doti*.5*V1^2))/(m_doti*(q23*1000000)) ;

    eta_p = 2/(1+(Veq/V1)) ;

    eta_o = eta_th*eta_p ;

    P = (T_T*V1)/1000000 ; % MW

end
```

13.7 Call Functions Matlab Code

```
%% MAE 563 ; Chandler Hutchens

% Final Project

format compact ;
format long ;
close all ;
clear all ;
clc ;
dbstop if error ;

%% Test Case (Section 7/8) Variables

z = 4300 ; % m
M1 = 2.4 ;
eta_d = .92 ;
% M2 = .15 ; % Section 7
M2 = .4 ; % Section 8 -> Thermally Choked
Tt3_max = 2400 ; % K
qf = 43.2 ; % MJ/kg
eta_n = .94 ;
Ae = .015 ; % m^2
g12 = 1.4 ;
g34 = 1.3 ;
R = 286.9 ; % J/(kg-K)
a = 986 ;
b = .179 ;

%% Atmospheric Variables

zs = 8404 ; % m
Ps = 101.3 ; % kpa
Ts = 288 ; % K

%% Module 1: Solution at State 1 (Free Stream)

[T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z,M1,g12,R,zs,Ps,Ts) ;

% fprintf('Module 1: Solution at State 1 (Free Stream) \n')
% fprintf('Temperature at State 1 (T1) = %.3f K \n',T1)
```

```
% fprintf('Pressure at State 1 (P1) = %.3f kpa \n',P1)
% fprintf('Total Temperature at State 1 (Tt1) = %.3f K \n',Tt1)
% fprintf('Total Pressure at State 1 (Pt1) = %.3f kpa \n',Pt1)
% fprintf('Specific Heat at State 1 (Cp1) = %.3f J/(kg-K) \n',Cp1)
% fprintf('Speed of Sound at State 1 (a1) = %.3f m/s \n',a1)
% fprintf('Velocity at State 1 (V1) = %.3f m/s \n',V1)

%% Module 2: Solution at State 2 (Inlet/Diffuser)

[Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1,M2,Tt1,P1,Pt1,Cp1,eta_d) ;

% fprintf('Module 2: Solution at State 2 (Inlet/Diffuser) \n')
% fprintf('Total Temperature at State 2 (Tt2) = %.3f K \n',Tt2)
% fprintf('Temperature at State 2 (T2) = %.3f K \n',T2)
% fprintf('Total Pressure at State 2 (Pt2) = %.3f kpa \n',Pt2)
% fprintf('Pressure at State 2 (P2) = %.3f kpa \n',P2)
% fprintf('Specific Heat at State 2 (Cp2) = %.3f J/(kg-K) \n',Cp2)
% fprintf('Speed of Sound at State 2 (a2) = %.3f m/s \n',a2)
% fprintf('Velocity at State 2 (V2) = %.3f m/s \n',V2)
% fprintf('Entropy Change From State 1 to State 2 (s2-s1) = %.3f J/(kg-K) \n',ds12)

%% Module 3: Solution at State 3 (Combuster)

[Tt3, M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2,Tt2,P2,Pt2,a,b,ds12) ;

% fprintf('Module 3: Solution at State 3 (Combuster) \n')
% fprintf('Total Temperature at State 3 (Tt3) = %.3f K \n',Tt3)
% fprintf('Mach # at State 3 (M3) = %.4f \n',M3)
% fprintf('Heat Input From State 2 to State 3 (q23) = %.3f MJ/kg \n',q23)
% fprintf('Temperature at State 3 (T3) = %.3f K \n',T3)
% fprintf('Pressure at State 3 (P3) = %.3f kpa \n',P3)
% fprintf('Total Pressure at State 3 (Pt3) = %.3f kpa \n',Pt3)
% fprintf('Specific Heat at State 3 (Cp3) = %.3f J/(kg-K) \n',Cp3)
% fprintf('Entropy Change From State 2 to State 3 (s3-s2) = %.3f J/(kg-K) \n',ds23)
% fprintf('Entropy Change From State 1 to State 3 (s3-s1) = %.3f J/(kg-K) \n',ds13)

%% Module 4: Solution at State e (Converging Nozzle)

[Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n,Tt3,P1,Pt3,a,b,ds12,ds23) ;
```

```
% fprintf('Module 4: Solution at State e (Coverging Nozzle) \n')
% fprintf('Mach # at State e (Me) = %.4f \n',Me)
% fprintf('Total Pressure at State e (Pte) = %.3f kpa \n',Pte)
% fprintf('Pressure at State e (Pe) = %.3f kpa \n',Pe)
% fprintf('Total Temperature at State e (Tte) = %.3f K \n',Tte)
% fprintf('Temperature at State e (Te) = %.3f K \n',Te)
% fprintf('Specific Heat at State e (Cpe) = %.3f J/(kg-K) \n',Cpe)
% fprintf('Speed of Sound at State e (ae) = %.3f m/s \n',ae)
% fprintf('Velocity at State e (Ve) = %.3f m/s \n',Ve)
% fprintf('Mass Flow Rate at State e (m_dote) = %.3f kg/s \n',m_dote)
% fprintf('Entropy Change From State 3 to State e (se-s3) = %.3f J/(kg-K) \n',ds3e)
% fprintf('Entropy Change From State 1 to State e (se-s1) = %.3f J/(kg-K) \n',ds1e)

%% Module 5: Solution at State 4 (Past Nozzle Exit)

[M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e) ;

% fprintf('Module 5: Solution at State 4 (Past Nozzle Exit) \n')
% fprintf('Mach # at State 4 (M4) = %.4f \n',M4)
% fprintf('Total Pressure at State 4 (Pt4) = %.3f kpa \n',Pt4)
% fprintf('Pressure at State 4 (P4) = %.3f kpa \n',P4)
% fprintf('Total Temperature at State 4 (Tt4) = %.3f K \n',Tt4)
% fprintf('Temperature at State 4 (T4) = %.3f K \n',T4)
% fprintf('Specific Heat at State 4 (Cp4) = %.3f J/(kg-K) \n',Cp4)
% fprintf('Speed of Sound at State 4 (a4) = %.3f m/s \n',a4)
% fprintf('Velocity at State 4 (V4) = %.3f m/s \n',V4)
% fprintf('Entropy Change From State e to State 4 (s4-se) = %.3f J/(kg-K) \n',dse4)
% fprintf('Entropy Change From State 1 to State 4 (s4-s1) = %.3f J/(kg-K) \n',ds14)

%% Module 6: Solution of Thrust And Other Parameters

[m_dotf, m_doti, f, J_T, P_T, T_T, Veq, TSFC, Isp, eta_th, eta_p, eta_o, P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1) ;

fprintf('Module 6: Solution of Thrust And Other Parameters \n')
fprintf('Fuel Mass Flow Rate (m_dotf) = %.3f kg/s \n',m_dotf)
fprintf('Inital Mass Flow Rate (m_doti) = %.3f kg/s \n',m_doti)
fprintf('Fuel-Air Mass Ratio (f) = %.3f \n',f)
fprintf('Jet Thrust (J_T) = %.3f N \n',J_T)
fprintf('Pressure Thrust (P_T) = %.3f N \n',P_T)
fprintf('Total Thrust (T_T) = %.3f N \n',T_T)
```

```
fprintf('Equivalent Velocity (Veq) = %.3f m/s \n',Veq)
fprintf('Thrust Specific Fuel Consumption (TSFC) = %.3f (kg/hr)/N \n',TSFC)
fprintf('Specific Impulse (Isp) = %.3f s \n',Isp)
fprintf('Thermal Efficiency (eta_th) = %.3f \n',eta_th)
fprintf('Propulsive Efficiency (eta_p) = %.3f \n',eta_p)
fprintf('Overall Efficiency (eta_o) = %.3f \n',eta_o)
fprintf('Propulsive Power (P) = %.3f MW \n',P)
```

13.8 Part A Matlab Code

```
%% Part A

% State 1 to State 2
T12 = [T1,T2] ;
s12 = [0,ds12] ;

% State 2 to State 3
s23 = linspace(ds12,ds23,10000) ;
T23(1) = T2 ;
ds = .18 ; % s23(101)-s23(100) -> Section 7
% ds = .0615 ; % s23(101)-s23(100) -> Section 8

for i = 1:(length(s23)-1)

   T23(i+1) = T23(i) + ((T23(i))/(a+b*T23(i)))*ds ;

end

% State 3 to State 4
T34 = [T3,Te,T4] ;
s34 = [ds13,ds1e,ds14] ;

% State 4 to State 1
s41 = linspace(ds14,0,10000) ;
T41(1) = T4 ;
ds = -.19 ; % s41(101)-s41(100) -> Section 7
% ds = -.095 ; % s41(101)-s41(100) -> Section 8

for i = 1:(length(s41)-1)

   T41(i+1) = T41(i) + ((T41(i))/Cp1)*ds ;

end

figure ;
plot(s12,T12,'b-','lineWidth',2)
hold on
plot(s23,T23,'r','lineWidth',2)
text(ds12,T2,'2 \rightarrow ','FontSize',15,'HorizontalAlignment','right')
text(ds13,T3,'\leftarrow 3','FontSize',15,'HorizontalAlignment','left')
```

```
plot(s34,T34,'k-','lineWidth',2)
text(ds1e,Te,'\leftarrow e','FontSize',15,'HorizontalAlignment','left')
text(ds14,T4,'\leftarrow 4','FontSize',15)
plot(s41,T41,'m','lineWidth',2)
text(0,T1,'1 \rightarrow ','FontSize',15,'HorizontalAlignment','right')
% title('T-s Diagram (Case #1)')
xlabel('Entropy (J/(kg-K)')
ylabel('Temperature (K)')
legend('[1-2] Entropy Increase','[2-3] Constant Pressure','[3-4] Entropy Increase','[4-1] Constant
Pressure','location','northwest')
xlim([-300,2650])
ylim([0,2500])
grid on
hold off
```

13.9 Part B Matlab Code

```
%% Part B

data = importdata('ICAO.txt') ;

ppt = data(:,3)' ;
ttt = data(:,2)' ;
z = data(:,1)' ;

for i = 1:length(z)

    if z(i) <= 7958

        tts(i) = Ts*(1-((g12-1)/g12)*(z(i)/zs)) ; % K
        pps(i) = Ps*(1-((g12-1)/g12)*(z(i)/zs))^(g12/(g12-1)) ; % kpa

    else

        tts(i) = 210 ; % K
        pps(i) = 33.6*exp(-(z(i)-7958)/6605) ; % kpa

    end

end

figure ;
plot(pps,z)
hold on
plot(ppt,z)
%title('Altitude (m) vs Pressure ratio (P/P_s)','fontsize',12) ;
xlabel('Pressure Ratio (P/P_s)') ;
ylabel('Altitude (m)')
legend('Theoretical', 'Tabulated','location','best')
ylim([0,22000])
grid on
hold off

figure ;
plot(tts,z)
hold on
plot(ttt,z)
```

```
%title('Altitude (m) vs Temperature ratio (T/T_s)','fontsize',12) ;
xlabel('Temperature Ratio (T/T_s)')
ylabel('Altitude (m)')
legend('Theoretical', 'Tabulated','location','best')
xlim([0,300])
ylim([0,22000])
grid on
hold off
```

13.10 Part C Matlab Code

```
%% Part C: Parameter Results With Varying M1

M1 = linspace(.8,5,100) ;

for i = 1:length(M1)

    z = 4300 ; % m
    eta_d = .92 ;
    M2 = .15 ;
    Tt3_max = 2400 ; % K
    qf = 43.2 ; % MJ/kg
    eta_n = .94 ;
    Ae = .015 ; % m^2
    g12 = 1.4 ;
    g34 = 1.3 ;
    R = 286.9 ; % J/(kg-K)
    a = 986 ;
    b = .179 ;

    % Atmospheric Variables
    zs = 8404 ; % m
    Ps = 101.3 ; % kpa
    Ts = 288 ; % K

    [T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z,M1(i),g12,R,zs,Ps,Ts) ;
    [Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1(i),M2,Tt1,P1,Pt1,Cp1,eta_d) ;
    [Tt3, M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2,Tt2,P2,Pt2,a,b,ds12) ;
    [Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n,Tt3,P1,Pt3,a,b,ds12,ds23) ;
    [M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e) ;
    [m_dotf, m_doti, f, J_T, P_T, T_T(i), Veq, TSFC(i), Isp, eta_th, eta_p, eta_o(i), P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1) ;

end

figure ;
plot(M1,eta_o,'k','linewidth',2)
xlabel('Free Stream Mach Number (M_1)')
```

```
ylabel('Overal Efficiency (\eta_o)')
grid on

figure ;
plot(M1,T_T,'k','linewidth',2)
xlabel('Free Stream Mach Number (M_1)')
ylabel('Total Thrust (T)')
grid on

figure ;
plot(M1,TSFC,'k','linewidth',2)
xlabel('Free Stream Mach Number (M_1)')
ylabel('Thrust Specific Fuel Consumption (TSFC)')
grid on
```

13.11 Part D Matlab Code

```
%% Part D

z = linspace(2000,30000,100) ; % m

for i = 1:length(z)

    M1 = 2.4 ;
    eta_d = .92 ;
    M2 = .15 ;
    Tt3_max = 2400 ; % K
    qf = 43.2 ; % MJ/kg
    eta_n = .94 ;
    Ae = .015 ; % m^2
    g12 = 1.4 ;
    g34 = 1.3 ;
    R = 286.9 ; % J/(kg-K)
    a = 986 ;
    b = .179 ;

    % Atmospheric Variables
    zs = 8404 ; % m
    Ps = 101.3 ; % kpa
    Ts = 288 ; % K

    [T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z(i),M1,g12,R,zs,Ps,Ts) ;
    [Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1,M2,Tt1,P1,Pt1,Cp1,eta_d) ;
    [Tt3, M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2,Tt2,P2,Pt2,a,b,ds12) ;
    [Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n,Tt3,P1,Pt3,a,b,ds12,ds23) ;
    [M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e) ;
    [m_dotf, m_doti, f, J_T, P_T, T_T(i), Veq, TSFC(i), Isp, eta_th, eta_p, eta_o(i), P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1) ;

end

figure ;
plot(z,eta_o,'k','linewidth',2)
xlabel('Flight Altitude (z)')
```

```
ylabel('Overal Efficiency (\eta_o)')
grid on

figure ;
plot(z,T_T,'k','linewidth',2)
xlabel('Flight Altitude (z)')
ylabel('Total Thrust (T)')
grid on

figure ;
plot(z,TSFC,'k','linewidth',2)
xlabel('Flight Altitude (z)')
ylabel('Thrust Specific Fuel Consumption (TSFC)')
grid on
```

13.12 Part E Matlab Code

```
%% Part E

z = (2000:500:20000) ; % m

for i = 1:length(z)

    M1 = linspace(.8,5,100) ;

    for j = 1:length(M1)

        eta_d = .92 ;
        M2 = .15 ;
        Tt3_max = 2400 ; % K
        qf = 43.2 ; % MJ/kg
        eta_n = .94 ;
        Ae = .015 ; % m^2
        g12 = 1.4 ;
        g34 = 1.3 ;
        R = 286.9 ; % J/(kg-K)
        a = 986 ;
        b = .179 ;

        % Atmospheric Variables
        zs = 8404 ; % m
        Ps = 101.3 ; % kpa
        Ts = 288 ; % K

        [T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z(i),M1(j),g12,R,zs,Ps,Ts) ;
        [Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1(j),M2,Tt1,P1,Pt1,Cp1,eta_d) ;
        [Tt3, M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2,Tt2,P2,Pt2,a,b,ds12) ;
        [Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n,Tt3,P1,Pt3,a,b,ds12,ds23) ;
        [M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e) ;
        [m_dotf, m_doti, f, J_T, P_T, T_T, Veq, TSFC(j), Isp, eta_th, eta_p, eta_o(j), P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1) ;

    end
```

```
    [maxvaly,idx] = max(eta_o) ;
    M1_eta(i) = M1(idx) ;
    [minvaly,idx] = min(TSFC) ;
    M1_TSFC(i) = M1(idx) ;

end

figure ;
plot(z,M1_eta,'k','linewidth',2)
xlabel('Flight Altitude (z)')
ylabel('Free Stream Mach Number (M_1)')
grid on
ylim([2,4])

figure ;
plot(z,M1_TSFC,'k','linewidth',2)
xlabel('Flight Altitude (z)')
ylabel('Free Stream Mach Number (M_1)')
grid on
ylim([0,5.5])
```

13.13 Part F Matlab Code

```
%% Part F

eta_d = (.5:.05:1) ;

for i = 1:length(eta_d)

    M1 = 2.4 ;
    z = 4300 ; % m
    M2 = .15 ;
    Tt3_max = 2400 ; % K
    qf = 43.2 ; % MJ/kg
    eta_n = .94 ;
    Ae = .015 ; % m^2
    g12 = 1.4 ;
    g34 = 1.3 ;
    R = 286.9 ; % J/(kg-K)
    a = 986 ;
    b = .179 ;

    % Atmospheric Variables
    zs = 8404 ; % m
    Ps = 101.3 ; % kpa
    Ts = 288 ; % K

    [T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z,M1,g12,R,zs,Ps,Ts) ;
    [Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1,M2,Tt1,P1,Pt1,Cp1,eta_d(i)) ;
    [Tt3, M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2,Tt2,P2,Pt2,a,b,ds12) ;
    [Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n,Tt3,P1,Pt3,a,b,ds12,ds23) ;
    [M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e) ;
    [m_dotf, m_doti, f, J_T, P_T, T_T(i), Veq, TSFC(i), Isp, eta_th, eta_p, eta_o(i), P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1) ;

end

figure ;
plot(eta_d,eta_o,'k','linewidth',2)
xlabel('Inlet/Diffuser Efficiency (\eta_d)')
```

```
ylabel('Overal Efficiency (\eta_o)')
grid on

figure ;
plot(eta_d,T_T,'k','linewidth',2)
xlabel('Inlet/Diffuser Efficiency (\eta_d)')
ylabel('Total Thrust (T)')
grid on

figure ;
plot(eta_d,TSFC,'k','linewidth',2)
xlabel('Inlet/Diffuser Efficiency (\eta_d)')
ylabel('Thrust Specific Fuel Consumption (TSFC)')
grid on
```

13.14 Part G Matlab Code

```
%% Part G

eta_n = (.5:.05:1) ;

for i = 1:length(eta_n)

    M1 = 2.4 ;
    z = 4300 ; % m
    M2 = .15 ;
    eta_d = .92 ;
    Tt3_max = 2400 ; % K
    qf = 43.2 ; % MJ/kg
    Ae = .015 ; % m^2
    g12 = 1.4 ;
    g34 = 1.3 ;
    R = 286.9 ; % J/(kg-K)
    a = 986 ;
    b = .179 ;

    % Atmospheric Variables
    zs = 8404 ; % m
    Ps = 101.3 ; % kpa
    Ts = 288 ; % K

    [T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z,M1,g12,R,zs,Ps,Ts) ;
    [Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1,M2,Tt1,P1,Pt1,Cp1,eta_d) ;
    [Tt3, M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2,Tt2,P2,Pt2,a,b,ds12) ;
    [Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n(i),Tt3,P1,Pt3,a,b,ds12,ds23) ;
    [M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e) ;
    [m_dotf, m_doti, f, J_T, P_T, T_T(i), Veq, TSFC(i), Isp, eta_th, eta_p, eta_o(i), P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1) ;

end

figure ;
plot(eta_n,eta_o,'k','linewidth',2)
xlabel('Nozzle Efficiency (\eta_n)')
```

```
ylabel('Overal Efficiency (\eta_o)')
grid on

figure ;
plot(eta_n,T_T,'k','linewidth',2)
xlabel('Nozzle Efficiency (\eta_n)')
ylabel('Total Thrust (T)')
grid on

figure ;
plot(eta_n,TSFC,'k','linewidth',2)
xlabel('Nozzle Efficiency (\eta_n)')
ylabel('Thrust Specific Fuel Consumption (TSFC)')
grid on
```

13.15 Part H Matlab Code

```
%% Part H

M2 = (.1:.1:2.5) ;

for i = 1:length(M2)

    z = 4300 ; % m
    eta_d = .92 ;
    M1 = 2.4 ;
    Tt3_max = 2400 ; % K
    qf = 43.2 ; % MJ/kg
    eta_n = .94 ;
    Ae = .015 ; % m^2
    g12 = 1.4 ;
    g34 = 1.3 ;
    R = 286.9 ; % J/(kg-K)
    a = 986 ;
    b = .179 ;

    % Atmospheric Variables
    zs = 8404 ; % m
    Ps = 101.3 ; % kpa
    Ts = 288 ; % K

    [T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z,M1,g12,R,zs,Ps,Ts) ;
    [Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1,M2(i),Tt1,P1,Pt1,Cp1,eta_d) ;
    [Tt3, M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2(i),Tt2,P2,Pt2,a,b,ds12) ;
    [Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n,Tt3,P1,Pt3,a,b,ds12,ds23) ;
    [M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e) ;
    [m_dotf, m_doti, f, J_T, P_T, T_T(i), Veq, TSFC(i), Isp, eta_th, eta_p, eta_o(i), P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1) ;

end

figure ;
plot(M2,eta_o,'k','linewidth',2)
xlabel('Mach Number Entering Combuster (M_2)')
```

```
ylabel('Overal Efficiency (\eta_o)')
grid on

figure ;
plot(M2,T_T,'k','linewidth',2)
xlabel('Mach Number Entering Combuster (M_2)')
ylabel('Total Thrust (T)')
grid on

figure ;
plot(M2,TSFC,'k','linewidth',2)
xlabel('Mach Number Entering Combuster (M_2)')
ylabel('Thrust Specific Fuel Consumption (TSFC)')
grid on
```

13.16 Part I Matlab Code

```
clear all
close all
clc

M2 = (.1:.01:2.5) ;

for i = 1:length(M2)

    z = 27400 ; % m
    eta_d = .92 ;
    Tt3_max = 2400 ; % K
    M1 = 5 ;
    qf = 43.2 ; % MJ/kg
    eta_n = .94 ;
    Ae = .015 ; % m^2
    g12 = 1.4 ;
    g34 = 1.3 ;
    R = 286.9 ; % J/(kg-K)
    a = 986 ;
    b = .179 ;

    % Atmospheric Variables
    zs = 8404 ; % m
    Ps = 101.3 ; % kpa
    Ts = 288 ; % K

    [T1, P1, Tt1, Pt1, Cp1, a1, V1] = Module_1(z,M1,g12,R,zs,Ps,Ts) ;
    [Tt2, T2, Pt2, P2, Cp2, a2, V2, ds12] = Module_2(g12,R,M1,M2(i),Tt1,P1,Pt1,Cp1,eta_d) ;
    [Tt3(i), M3, q23, T3, P3, Pt3, Cp3, a3, V3, ds23, ds13] =
Module_3(g34,R,Tt3_max,M2(i),Tt2,P2,Pt2,a,b,ds12) ;
    [Me, Pte, Pe, Tte, Te, Cpe, ae, Ve, m_dote, ds3e, ds1e, M_prime] =
Module_4(g34,R,Ae,eta_n,Tt3(i),P1,Pt3,a,b,ds12,ds23) ;
    [M4, Pt4, P4, Tt4, T4, Cp4, a4, V4, dse4, ds14] =
Module_5(g34,R,M_prime,Tte,P1,Pte,a,b,ds12,ds23,ds3e) ;
    [m_dotf, m_doti, f, J_T, P_T, T_T(i), Veq, TSFC, Isp, eta_th, eta_p, eta_o(i), P] =
Module_6(qf,q23,m_dote,Pe,P1,Ae,Ve,V1) ;

end

figure ;
```

```
plot3(M2,Tt3,eta_o,'k','linewidth',2)
ylabel('Total Temperature In Combuster (T_{t3})','FontSize',9)
xlabel('Mach Number Entering Combuster (M_2)','FontSize',9)
zlabel('Overal Efficiency (\eta_o)','FontSize',9)
zlim([-4500,1])
grid on

fprintf('The Maximum Value of Overall Efficiency = %f \n',max(eta_o))
[maxvaly,idx] = max(eta_o) ;
M2_eta = M2(idx) ;
fprintf('The Maximum Value of Overall Efficiency Corresponding M2 = %f \n',M2_eta)
[maxvaly,idx] = max(eta_o) ;
Tt3_eta = Tt3(idx) ;
fprintf('The Maximum Value of Overall Efficiency Corresponding Tt3 = %f \n',Tt3_eta)

figure ;
plot3(M2,Tt3,T_T,'k','linewidth',2)
ylabel('Total Temperature In Combuster (T_{t3})','FontSize',9)
xlabel('Mach Number Entering Combuster (M_2)','FontSize',9)
zlabel('Total Thrust (T)','FontSize',9)
grid on

fprintf('The Maximum Value of Thrust = %f \n',max(T_T))
[maxvaly,idx] = max(T_T) ;
M2_TT = M2(idx) ;
fprintf('The Maximum Value of Thrust Corresponding M2 = %f \n',M2_TT)
[maxvaly,idx] = max(T_T) ;
Tt3_TT = Tt3(idx) ;
fprintf('The Maximum Value of Thrust Corresponding Tt3 = %f \n',Tt3_TT)
```