Joint Air-to-Surface Standoff Missile Baseline Propulsion

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I. Executive Summary

This report discusses the propulsion of the JASSM missile using a turbojet, developed by Lockheed Martin. First, an operational range of speeds and altitudes were researched and postulated to develop the range of conditions the missile will experience. The missile was found to cruise at an altitude of 2,000-30,000 ft at Mach 0.71. There are two specified missions, one being launched from a crate suspended by a parachute, which starts the missile at a relatively low altitude and Mach number, and the other mission being launched from a high-speed, high-altitude fighter jet. A model of the atmosphere was developed, and the dynamic pressure ranges that the JASSM will experience were calculated. Additionally, the ideal turbojet analysis was conducted with a maximum turbine inlet temperature of 4000 °R and a practical compressor pressure ratio cap out at 10. The compressor analysis shows that the engine can achieve a maximum pressure ratio of 7.61. Performance calculations show that the engine thrust decreases with increasing altitude and Mach number, while specific impulse decreases as Mach number increases. Using the Breguet range equation with a lift-to-drag ratio of 3.59 and a fuel weight of 75 lbs. Finally, the analysis suggests that the JASSM has a maximum powered flight range of 250 miles, with a corresponding flight time of 3.8 minutes

[1] Responsible for Turbojet 4, Turbojet 5.
[2] Responsible for Turbojet 2, Turbojet 3.
[3] Responsible for Turbojet 6, Turbojet 7.
[4] Responsible for Standard Atmosphere, Turbojet 1.

II.Background

The Joint Air-to-Surface Standoff Missile (JASSM) is a stealthy, precision-guided cruise weapon designed to be launched from stand-off ranges, fly at low observable altitude, with terrain following profile to penetrate enemy air defenses, and deliver a high-explosive warhead onto a fixed target with guided terminal homing. In this design we retain the Lockheed Martin JASSM's baseline mission and guidance suite but replace the conventional horizontal tail surfaces with a streamlined boattail rear fuselage and optimized aft control fins if needed. The boattail reduces radar and aerodynamic drag while shifting stability and trim requirements rearward. This configuration aims to preserve low observable characteristics and range while trading some longitudinal stability for improved cruise efficiency and simpler external signatures.

Table 1 The JASSM Specifications

Baseline Design	Dimension
Body Diameter	21.61 in
Reference Area	369.25 in^2
Nose Length	40.68 in
Total Body Length	161.95 in
Elliptical Height	10.00 in
Elliptical Width	11.67 in
Roll Angles	0 deg
Nose-tip Diameter	0 in
Inlet Area	28.76 in^2
Effective Exhaust Area	23.76 in^2
Leading Edge Section Angles	0 deg
Number of Surfaces	3
Sweep Angles	45 deg
Wingspan	7.87 ft
Root Chord Length	1.00 ft
Nose Tip to Root Chord Leading Edge of Wing	60.6 in

Much of this geometry was found using a three-view of the JASSM rocket and using the total length as a base dimension [1]. The missile body was modeled in NX to measure the other derived geometry [2].

A. Mission Description

The JASSM can fly a variety of missions, including extremely different initial conditions. This report will observe these initial conditions and flight regime, specifically when discussing the dynamic pressure and Mach plots, as these conditions force the missile into vastly different circumstances. For both missions, the missile will aim to cruise at 2000 ft at 0.71 Mach to avoid radar detection to remain stealthy and will accelerate when descending to hit its target, reaching a speed of around Mach 0.9. The first mission is defined as being launched from a C-130, which deploys a crate of the missiles that slowly descend to 10,000ft using a parachute. The JASSM is then deployed from the crate at this altitude, and once it reaches Mach 0.1 begins to have a controlled descent to 5,000 ft. The JASSM then begins powered flight to increase its Mach number from 0.3 to its cruising speed of Mach 0.71, and descends to 2,000 ft, where it executes the rest of the mission as usual. The other defined mission is the JASSM being launched at high altitude of 40,000 ft from a fighter jet at Mach 0.9. The theory behind this is that the fighter is in super cruise, possibly being an F-22, and slows to Mach 0.9 to launch the JASSM without forcing the missile into an unusual Mach range. The JASSM then coasts down to 30,000 ft with its wings deployed to save fuel and increase range and then begins powered flight while descending to 2,000 ft at Mach 0.71, where it will carry out the rest of its mission.

Operational Cruise Mach number: 0.71
Maximum Mach number: 0.9
Cruise altitude: 2000 ft – 30,000ft [3] and [4]
Operational AoA: less than 10 deg
Stall Effective AoA: 10 deg

III. Standard Atmosphere

The standard atmosphere was found using an online resource which provided the ratios of temperature, pressure, and density for increasing altitudes in kilometers [5]. A function was then created to interpolate this data, which was previously extracted to a .mat file for code efficiency. Atmospheric data was found for up to 80,000 feet in altitude, and the ratios of pressure, temperature, density, and the speed of sound are shown in Figure . Note that this data is the temperature states static values, or the stream values if observed at zero velocity.

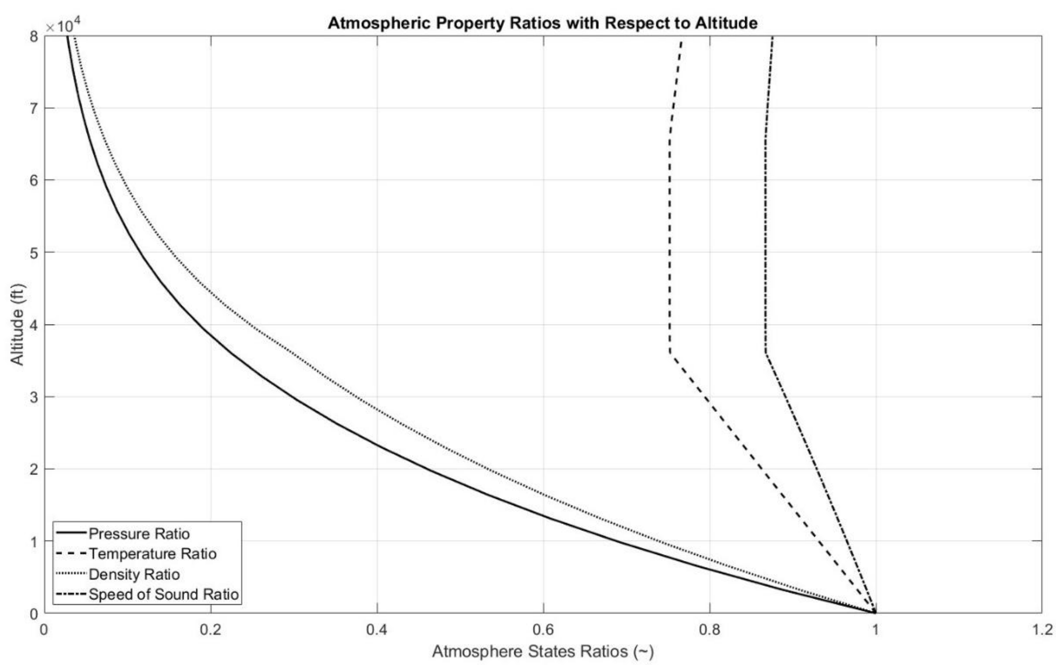

Figure 1 Standard Atmospheric State Ratios for 80,000 ft

These ratios are a multiplier of the local sea level absolute states. The standard sea level states were researched and found to be pressure = 14.7 psi, temperature = 518.7˚R, density = 0.002377 lb/ft^3, and the speed of sound = 1,117 ft/s. In general, pressure and density decrease with altitude and approach zero as they reach the boundary of space. Temperature decreases until around 40,000 ft, where it stays constant until around 65,000 ft where it starts increasing due to less protection from the sun. It was assumed for this that the gas constants did not change with altitude, so the speed of sound ratio varied to the square root of the temperature. While this is an inaccurate assumption, for our true operational altitude, these values will minimally change, meaning that the assumption is not inaccurate for our operational ranges. With a defined ceiling of 40,000 ft, the plot of the states can be narrowed to the operational range in Figure 2.

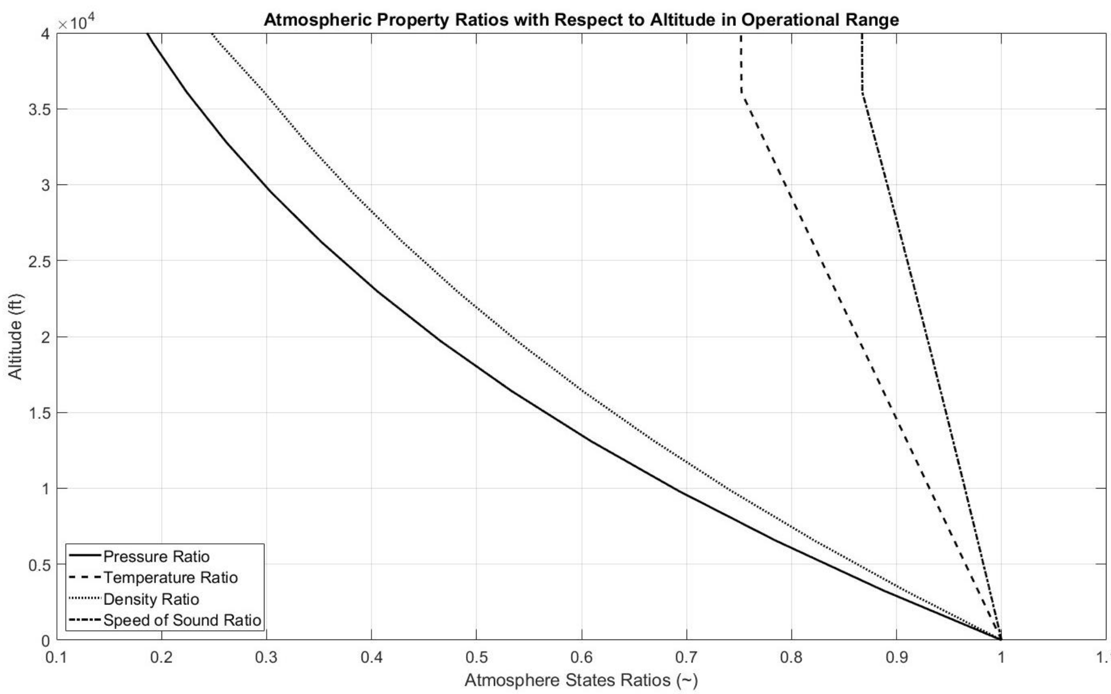

Figure 2 Standard Atmospheric State Ratios for Operational Range

The speed of sound ratio was then used to find the average Mach value for the operational range, which was found by averaging the ratio of the speed of sound and multiplying it by the ground speed of sound velocity. This yielded an average speed of sound of 1037 ft/s. The dynamic pressure can then be calculated for the operational range of Mach numbers using this average speed of sound and the change in density through altitude. The contours for these are shown for a range of Mach numbers in Figure .

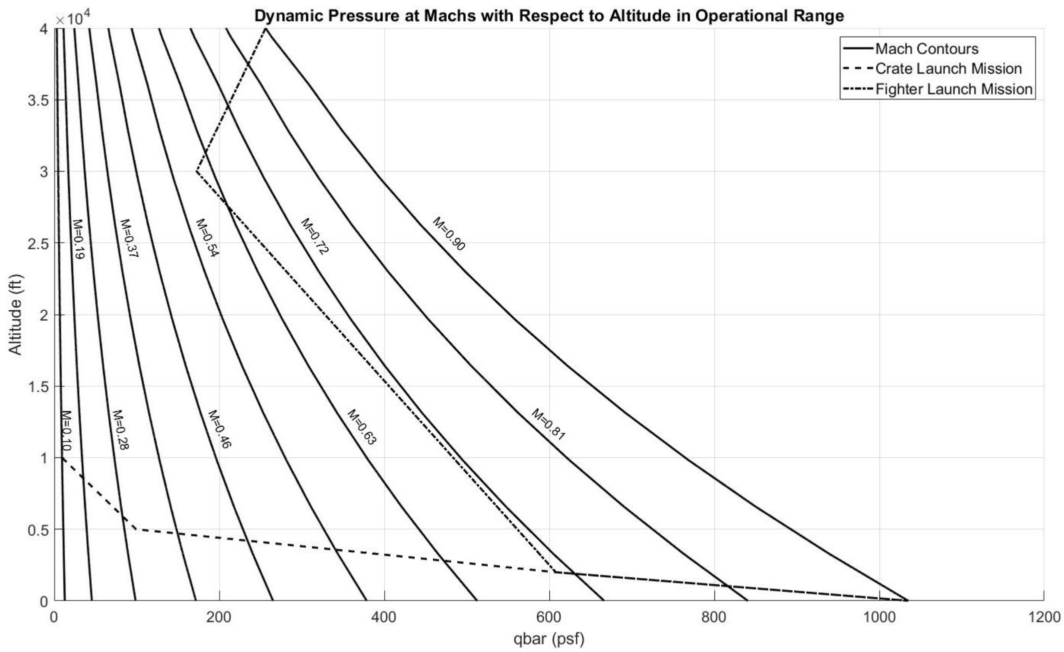

Figure 3 Dynamic Pressures at Various Mach Numbers and Altitudes

This plot also shows the two previously described missions flown in terms of altitude and dynamic pressure. Note that for different missions, the JASSM must be within a dramatically different range of dynamic pressures as its altitude and Mach speed vary significantly. This shows how the controller for the JASSM must consistently adapt to its current conditions, and the aircraft itself must be robust enough to fly at these varieties of conditions. As previously mentioned, in mission 1 the JASSM is launched at slow speed at low altitude from a parachute-suspended crate and descends to the cruise altitude of 2,000 ft at Mach 0.71. Mission 2 is when the JASSM is launched from a high altitude at a fast speed, such as from an F-22 at 40,000 ft, where it coasts to a lower speed, and then cruises to its point of Mach 7.1 at 2,000 ft. After this, it is estimated that on its descent to its target, it could reach a speed of Mach 0.9, so this is done to demonstrate the farthest extent of the dynamic pressure that the JASSM could experience. This entire range describes how the JASSM must be able to have a powerful enough control surface to maneuver at low dynamic pressures, while having an accurate enough surface to make minute adjustments at high dynamic pressures.

The ranges of the Mach numbers with their respective altitudes is then used to find the stagnation temperature of the free stream. Using the static temperature ratio from the standard atmosphere calculator shown with the plots above, the stagnation temperature can be found for the entire range of altitudes at each Mach number. Using the Mach number as a contour, Figure 4 can be found using Equation (1). The turbojet was only assumed to start burning at 30,000 feet for Mission 2.

$$T_{0t} = T_0\left(1 + \frac{\gamma_0 - 1}{2} M^2\right) \tag{1}$$

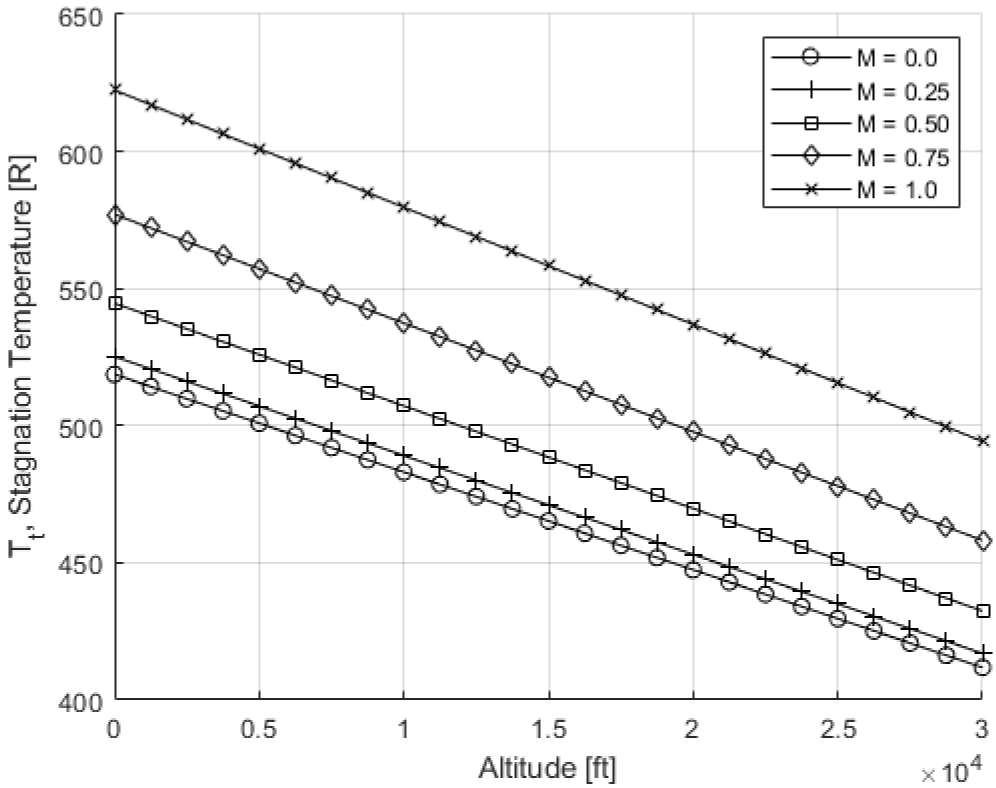

Figure 4 Varying Stagnation Temperature for Flight Regime

Using this plot, the range of stagnation temperatures can be found by finding the minimum and maximum stagnation temperature. This range was found to be from 411.8 °R to 622.4 °R.

IV. Turbojet (1)

In this part we are recreating Figure 3.9 which shows different variations with compression ratio ($\frac{P_3}{P_2}$) with Mach number for a Turbine Inlet temperature of Turbine inlet temperature of $T_4 =$ $4000\ °R$. Each curve represents different freestream static temperature, Showing the missile's atmospheric flight profile starting at 30,000 ft above sea level. The free stream five different temperatures used in this case are $T_0 = 519\ °R,\ 512\ °R,\ 501\ °R,\ 483\ °R,$ and $412\ °R.$ The results were calculated using Ideal Turbojet relationship Assuming isentropic flow through the inlet and compressor this is shown in Equation (2).

$$\left(\frac{P_3}{P_2}\right)_{T_{Max}} = \left(\frac{\sqrt{\frac{T_4}{T_0}}}{1+\frac{(\gamma_0-1)}{2}M_0^2}\right)^{\frac{\gamma_4}{\gamma_4-1}} \tag{2}$$

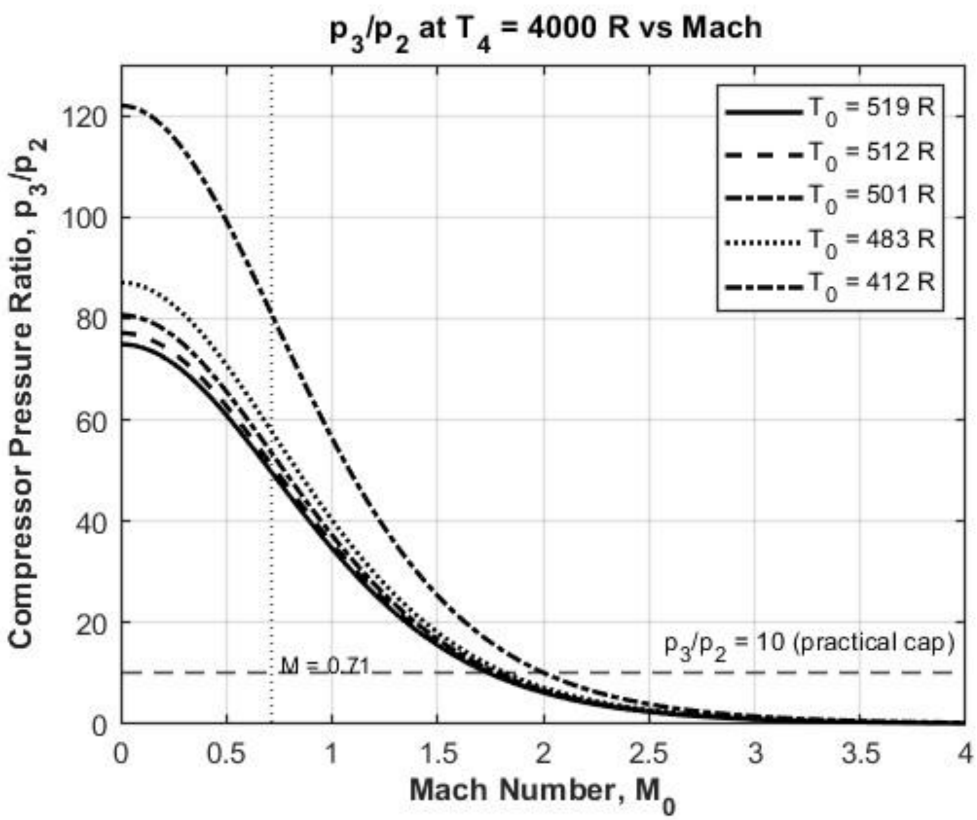

Figure 5 Varying Maximum Pressure Ratio for Static Temperatures

As shown in figure 5 with the increasing Mach number, the compressor pressure ratio decrees rapidly. The JASSM flight speed will be Mach 0.71, Theoretically the required compressor ratio is 60. Hower ever the max compression ratio for this baseline design chosen is 10 because a JASSM will not be able to go for higher compression ratios.

V. Turbojet (2)

Using the maximum pressure ratios found for each static temperature, the maximum and minimum compressor exit temperatures can be found for varying Mach numbers at each static temperature and its corresponding maximum pressure ratio. The following equation shows the relationship between the Mach number, static temperature, pressure ratio, and the compressor exit temperature. In this equation, the relative velocity of the air at the end of the compressor is very low compared to the free stream velocity, so it is estimated that the total temperature and static temperature at these stages are equal, leading to $T_{3t} = T_3$, which yields the equation. Note that this also comes from the fact that the static temperature is increased by the free stream Mach number to yield the total temperature before the inlet, which due to isentropic and adiabatic assumptions is constant through the inlet. The equations for temperature are shown below in Equation (3) and (4).

$$T_{3t} = T_2 \left(\frac{p_3}{p_2}\right)^{\frac{\gamma_3 - 1}{\gamma_3}} \tag{3}$$

$$T_3 = T_0 \left(1 + \frac{\gamma_0 - 1}{2} M^2\right) \left(\frac{P_3}{P_2}\right)^{\frac{\gamma_3 - 1}{\gamma_3}} \tag{4}$$

It is seen that the temperature at station 3 is a function of the heat capacity ratio at station 3 which is in turn a function of that temperature. Equation (5) is then used for iteration to find the compressor exit temperature for any given free stream static temperature and speed, and the maximum pressure ratio of 10 for each static temperature, found above.

$$\gamma_3 = 1.29 + 0.16e^{-0.0007 * T_3} \tag{5}$$

To create a modified figure for the compressor exit temperature, the Mach was varied across the flight regime from 0.1 to 0.9, and the previously stated chosen static temperatures were used as contour lines. For each static temperature selected, the maximum pressure ratio found above was used, which is 10 for all the temperatures. Below is Figure 6 generated using this data.

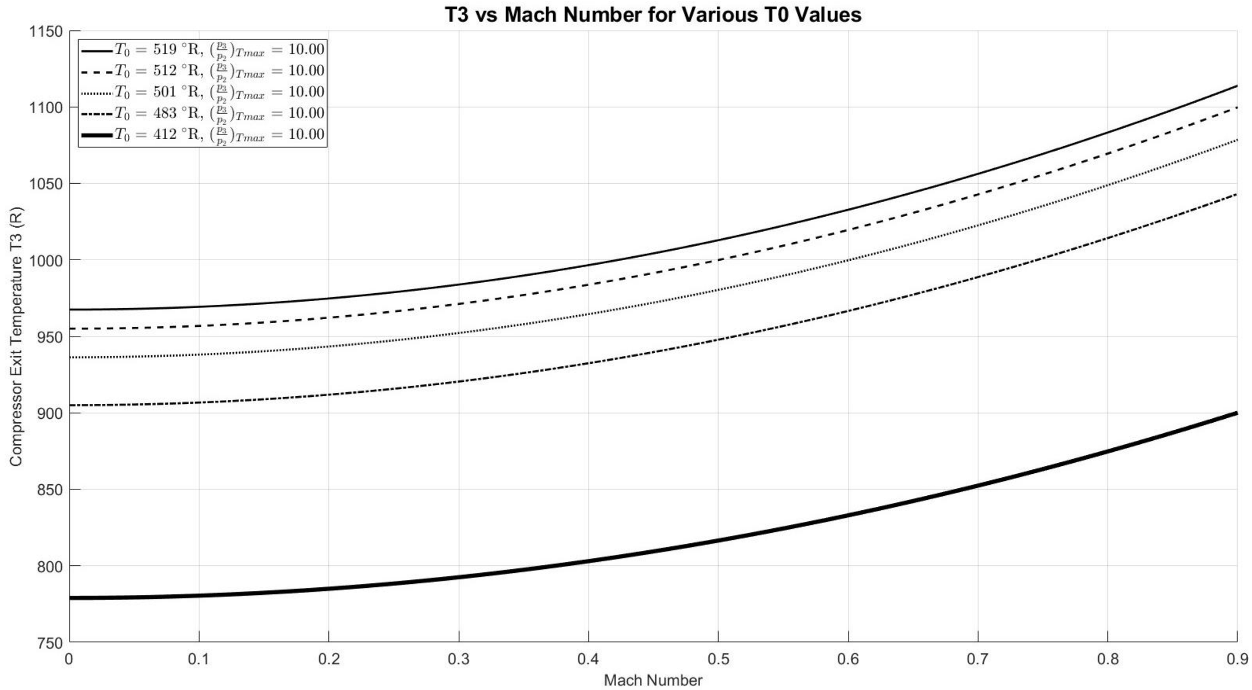

Figure 6 Temperature After Compressor for Different Static Temperatures

This plot shows that for constant static temperatures and pressure ratios, as the free stream Mach number increases, the compressor exit temperature increases. It is important to note that this temperature is the temperature at the maximum thrust for each Mach number and static temperature, meaning that this is the range for the maximum compressor exit temperatures.

VI. Turbojet (3)

The turbojet is assumed to have three axial compressor stages. For maximum efficiency, the turbojet is made to be more complicated, allowing for different angular velocities for each stage. To then find the angular velocity for each stage, it is first checked whether the maximum pressure ratio of 10 can be met using the maximum pressure ratios of each stage. The limit of each compressor stage is the tangential speed of the tip of the compressor blades which cannot exceed Mach 1. The equation for finding the maximum angular speed is shown below, where first the tangential speed is found using the maximum Mach number, and then related to the angular speed using geometry. Note that "a" is used instead of the minimum diameter, as if the engine was larger than this diameter, it would extend through the fuselage. Note that this diameter is conservative, as the engine will need to be a smaller diameter than the outer fuselage, but this assumption is made to give a preliminary estimation. The algebra is then performed in Equations (6), (7), (8), and (9).

$$V_{tip} = M_{tip} * a_{tip} \tag{6}$$

$$V_{tip,max} = a_{tip} = \sqrt{\gamma RT} \tag{7}$$

$$\omega_{max} = \frac{2V_{tip,max}}{a} = \frac{2\sqrt{\gamma RT}}{a} \tag{8}$$

$$\omega_{max,i} = \frac{2\sqrt{\gamma_{i-1}RT_{i-1}}}{a} \tag{9}$$

The maximum angular speed for each compressor, defined by the exit station "i" is found using the temperature and heat capacity ratio at the previous station, and the gas constant which is found as 1716 ft*lbf/(slug°R). a is in feet, the temperature is in degrees Rankine, and the heat capacity ratio is unitless. The range of Mach numbers for the tip of the axial rotor is from 0.8 to 1, and the Mach number for the fluid flowing over the rotor airfoil has a range of 1 to 1.4. Relating these two ranges yields a relationship between the tip Mach and rotor Mach in Equation (10).

$$M_{rotor} = 2M_{tip} - 0.6 \tag{10}$$

To find the maximum pressure ratio possible, the maximum rotor Mach of 1.4 is used, which is the corresponding Mach for the maximum tip Mach. This is used in Equation (11) below with the given assumption that the C_p for this engine is 0.7.

$$\left(\frac{p_3}{p_2}\right)_{Tmax} = 1 + 0.5 * \gamma * C_p * M_{rotor}^2 \tag{11}$$

Using this equation with the heat capacity ratio given from the temperature of air entering the compressor, the maximum compression ratio is found for each station. The temperature is then found by using (12) and (13) from the previous section but used to calculate the temperature for each individual compressor. Not that T_2 is then equal to the free stream stagnation temperature which is found using the other equation below.

$$T_3 = T_2\left(\frac{p_3}{p_2}\right)^{\frac{\gamma_3-1}{\gamma_3}} \tag{12}$$

$$T_2 = T_0\left(1 + \frac{\gamma_0 - 1}{2}M^2\right) \tag{13}$$

The first equation must be iterated to find the exit temperature due to the previously stated cyclical loop of gamma being affected by temperature. This entire process is repeated for each compressor. The stations then defined are 0 being the air right before it enters the intake, 2 being the states inside the intake, 2.3 being the station right after the first compressor, 2.6 being the station right after the second compressor, and 3 being the station right after the third and final compressor. The order of using these equations is first finding the temperature and then heat capacity ratio (γ). Using the heat capacity ratio at station 2, the maximum pressure ratio is then found using the heat capacity ratio (HCR) from station 2, which is then used to find the temperature at station 2.3 and the HCR at station 2.3. The HCR is then used to find the pressure ratio at station 2.6 which is used to find the temperature and HCR at 2.6. This is finally repeated to find the pressure ratio for the final compressor which is used to find the temperature at station 3. Using this order of equations, the maximum possible total pressure ratio for the entire compressor section is found by multiplying the ratios of each station together. The table below lists the maximum pressure ratios (PR) found using the minimum stagnation temperature. The maximum angular speeds for each compressor are also shown Table 2.

Table 2 Compressor Pressure Ratio and Max Angular Speed

Compressor #	1	2	3	Total
Pressure Ratio	1.9672	1.9672	1.9672	7.6128
Angular Speed (rpm)	27,960	30,870	33,855	~

This shows that the maximum possible pressure ratio for this combination of axial compressors for the flight regime is 7.6128, which is less than the allowed limit of 10. Table 3 is the table of the minimum and maximum temperature after each compressor stage for the maximum thrust for the range of stagnation temperatures.

Table 3 Range of Temperatures After Compressors for Max Thrust

Station	2.3	2.6	3
Minimum T (°R)	500.1	605.7	731.3
Maximum T (°R)	750.0	901.9	1080.8

Figure 7 below shows the trend of maximum pressure ratios per the stagnation temperature range.

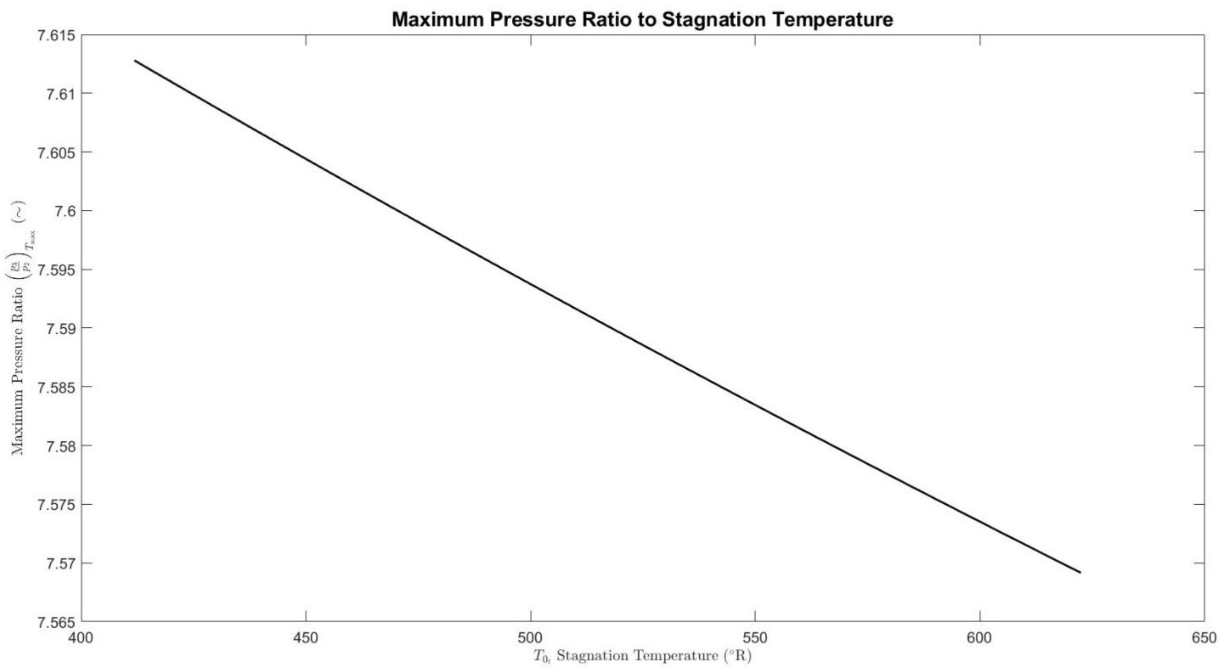

Figure 7 Maximum Pressure Ratios for Varying Static Temperatures

This plot makes sense, as starting at a higher stagnation temperature will cause the air to have more energy. Since the compressor is essentially adding energy to the air, it is more of a summation of energy instead of a multiplication of energy. Due to this, the more energy the air starts with, the less the energy the compressor can add with respect to the total energy, meaning that the ratio of compression will end up being less.

As the maximum possible pressure ratio is less than the theoretical maximum pressure ratio for a turbojet, the maximum thrust will be less than the thrust output at the theoretical maximum pressure ratio, meaning that expectations must be limited for the flight regime.

VII. Turbojet (4)

The fuel-to-air ratios are plotted in Figure 8 to raise the temperature from T_3 to T_4 =4000 °R. Because of the high temperature of stoichiometric combustion, most missiles must operate at a low value of fuel to air ratio (f/a <0.04) [6]. Therefore, a vertical line at 0.04 for fuel-to-air ratio is made to determine the maximum combustor exit temperature T_4. C_p value is calculated using T_4 =4000 °R and JP-10 fuel for the turbo jet engine.

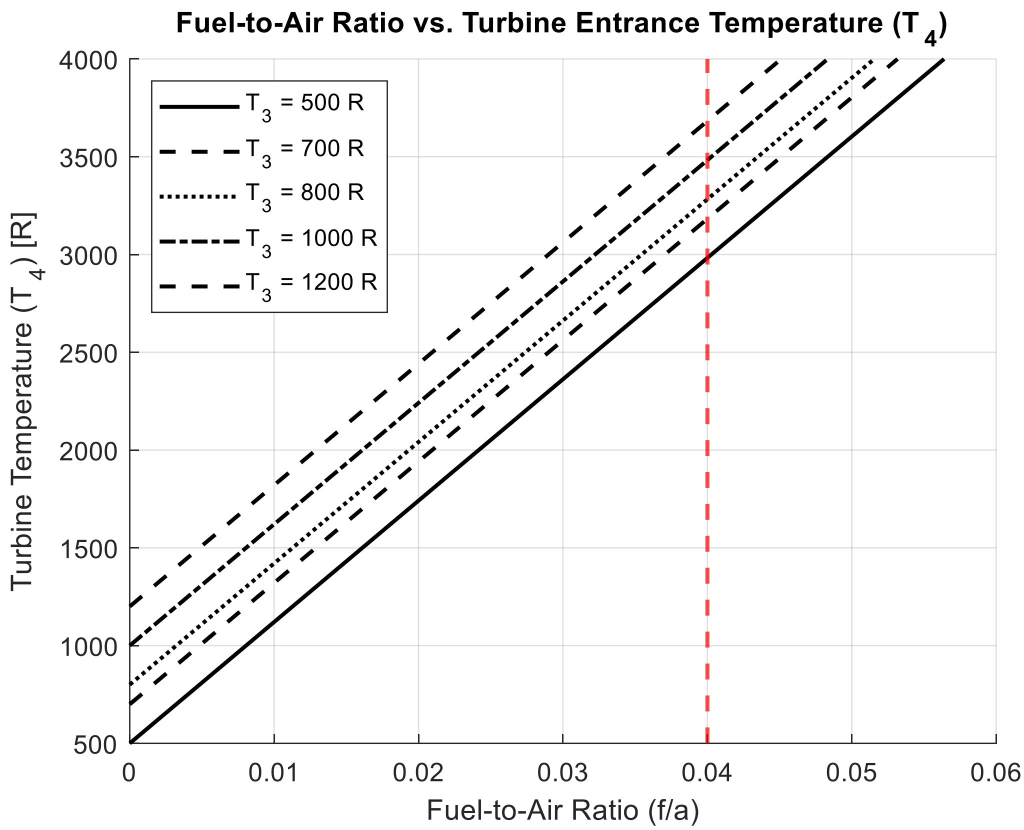

Figure 8 Fuel-to-air Ratio vs T4

The compressor outlet temperature T_3 is 1,284 °R once calculated after 8 iterations by using the previous compressor ratios of 7.6128, static temperature, and cruise Mach number at 0.71. For the ideal turbojet, T_3 is found using Equation (14) and T_2 is found using Equation (15).

$$T_3 = T_2\left(\frac{p_3}{p_2}\right)^{\frac{\gamma_3-1}{\gamma_3}} \tag{14}$$

$$T_2 = T_0\left\{1 + \frac{\gamma_0}{2}M_0^2\right\} \tag{15}$$

Using MATLAB, T_3 and γ_3 are solved simultaneously to get the final value of $T_3 = 1{,}200.39\,°R$. Similar method is used to find $T_4 = 3{,}812.8\,°R$.

VIII. Turbojet (5)

The thrust created by the turbojet engine is presented in assuming a constant maximum compressor of 10. 5 contour lines of standard atmosphere static pressure and temperature lines are presented in Figure 9. The standard atmospheric conditions are selected based on the JASSM mission from 2,000 ft to 40,000 ft. The assumptions of ideal turbojet with perfect gas ($\gamma_0 = \gamma = 1.4$) are made to find the thrust as shown in Figure 9, comparing the thrust as low compressor pressure ratio vs a relatively high compressor ratio provides only a small increase in thrust but large increase in specific impulse.

The performance curves for the JASSM turbojet engine demonstrate two primary trends within the subsonic flight envelope in Figure 9. First, at a constant Mach number, thrust output decreases significantly as altitude increases due to lower air pressure. Second, at any given altitude, the thrust decline as the Mach number increases to 1.0 due to the negative effect of increasing ram drag

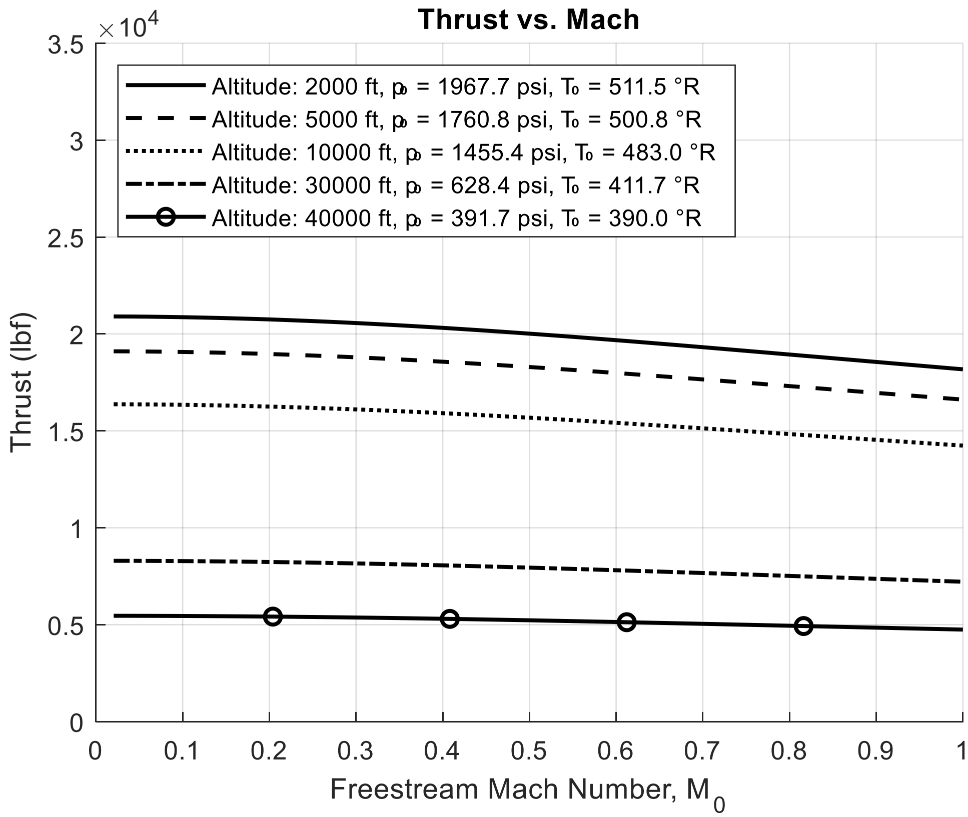

Figure 9 Thrust Vs. Mach Number at Different Altitude

IX. Turbojet (6)

Specific impulse is the measure of the efficiency of the engine, with the thrust of the engine normalized by the mass flow rate. The definition of specific impulse is defined using Equation (16).

$$I_{sp} = T / \dot{m}\, g_0 \ \text{[sec]} \tag{16}$$

By applying known thermodynamic relationships for turbojet engines and incorporating results from the previously analyzed compressor and combustion sections, the governing expressions for mass flow rate are obtained using Equation (17).

$$[g_0 c_p T_0/(a_0 H_f)](I_{SP})_{Tmax} = T_{max} T_0/[(p_0 A_0 \gamma_0 M_0)(T_4 - T_0)] \tag{17}$$

Rearranging the equation to instead use the known combustion temperature ratio and the compressor ratio, Equation (18) is found.

$$[g_0 c_p T_0/(a_0 H_f)](I_{SP})_{Tmax} = T_{max}/(p_0 A_0)/\left[\gamma_0 M_0[(T_4/T_0) - (p_3/p_2)^{(\gamma_0 - 1)/\gamma_0}]\right] \tag{18}$$

The equation above that then allows for the combustion and pressure ratio, calculated earlier, to be used in this equation and graphed over the flight regime at a variety of altitudes, shown in Figure 10.

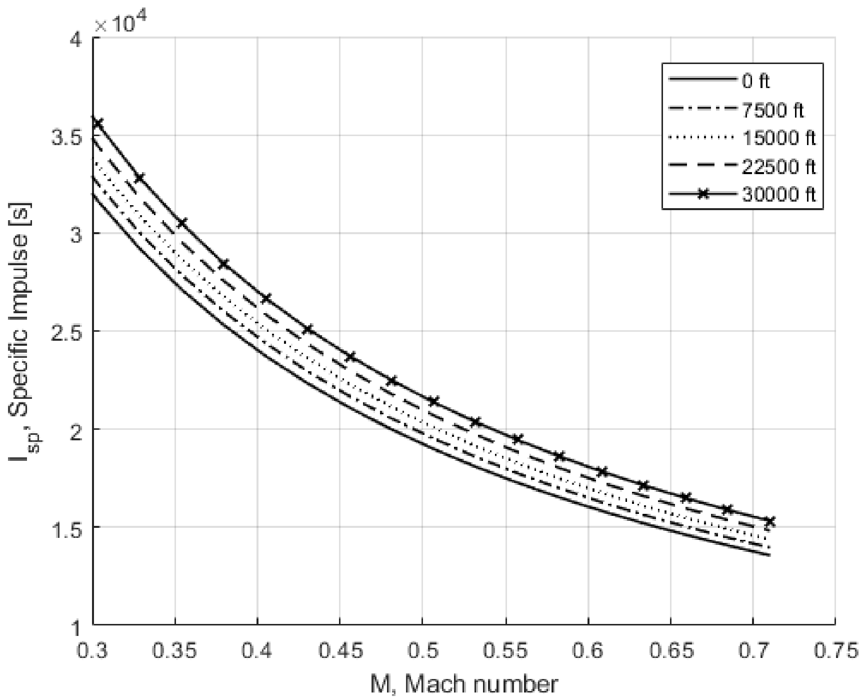

Figure 10 Specific Impulse

As shown in the figure above, specific impulse exhibits a decreasing trend with increasing Mach number and shows a minor decrease at higher altitudes.

X. Turbojet (7)

Using the lift-to-drag characteristics established in the preceding aerodynamic analysis, the missile's range was evaluated. Figure 11 shows the lift-to-drag and trimmed angle of attack from Mach 0.6 to 0.71, allowing for an appropriate average cruise velocity to be chosen for the cruise altitude.

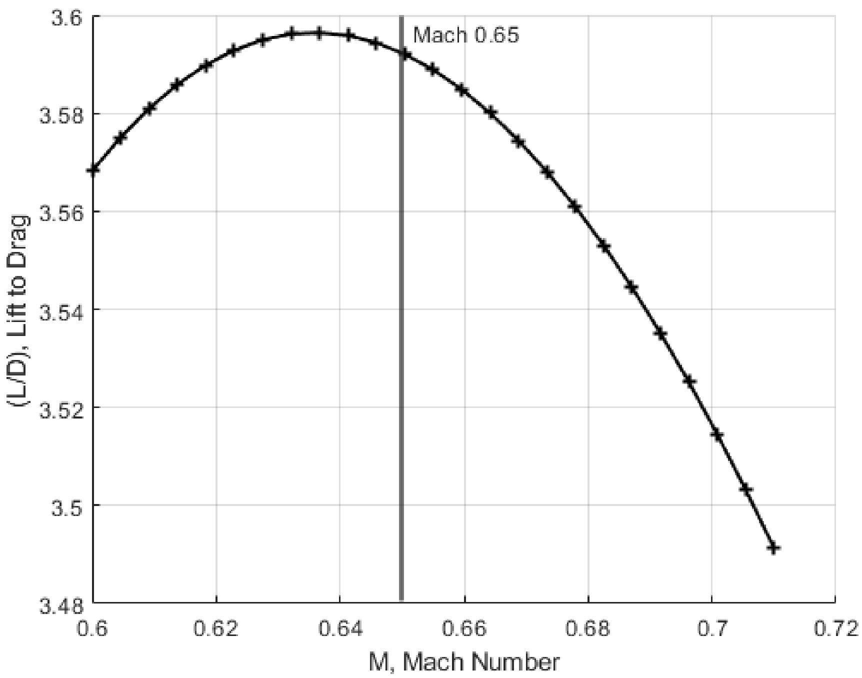

Figure 11 Cruise Lift-to-Drag

The selected cruise conditions are summarized in Table 4, corresponding to an average Mach number of 0.65. The associated lift-to-drag ratio and trimmed angle of attack were determined from the aerodynamic data at this condition.

Table 4 Average Cruise Velocity at 10,000 ft

Mach	Velocity (ft/s)	L/D	α (deg)
0.65	700.43	3.59	8.27

The range of the JASSM could then be determined using the Breguet range equation, shown in Equation (19).

$$R = \left(\frac{L}{D}\right) I_{SP} V_{avg} \ln[W_L/(W_L - W_P)] \tag{19}$$

With a launch weight (W_L) of 2250 lbs. and a lift to drag determined at the average velocity. The weight of the fuel was determined to be 75 lbs. for the purpose of this analysis. Using the determined fuel-to-air ratio of 0.04, and the equations shown below. The remaining flight time was graphed in Figure 12. The mass rates of air and fuels are shown in Equations (20) and (21).

$$\dot{m_{air}} = \rho_0 V_{avg} A_0 \tag{20}$$

$$\dot{m_{fuel}} = \dot{m_{air}} * f/a \tag{21}$$

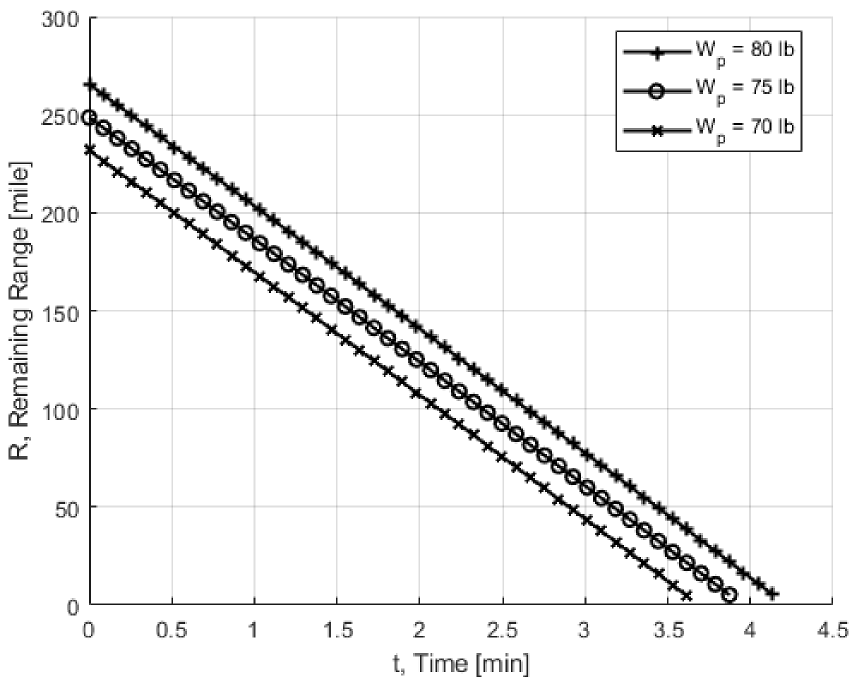

Figure 12 Remaining Flight Time

Calculating the mass flow rates at the average velocity results in a $\dot{m_{air}}$ of 7.9 lbs/s, $\dot{m_{fuel}}$ of 0.32 lbs/s and applying the Breguet range equation with an available fuel weight yielded a maximum range of roughly 250 miles. The corresponding powered flight time was calculated to be approximately 3.8 minutes, as illustrated in the figure above. Notably, the remaining-range curve is essentially linear over the plotted flight, because the propellant weight is very small compared with the vehicle's total weight, each unit of fuel burned produces nearly the same incremental range, so the logarithmic curvature predicted by the Breguet relation is largely masked.

XI.References

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XII. Appendix A

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