Wide-Range Autonomous Ingress Tactical Hunter (WRAITH) Redesign Propulsion

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I. Executive Summary

This report discusses the propulsion of the WRAITH missile using a turbojet, an updated version of the JASSM 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. The mission shows that the WRAITH is being launched from a high-speed, high-altitude fighter jet or a B-1 Lancer, but flight is at low altitude to avoid radar detection using terrain masking. A model of the atmosphere was developed, and the dynamic pressure ranges that WRAITH 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 combination of two axial and a centrifugal compressor can achieve a maximum pressure ratio of 10. 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 3.93 and a fuel weight of 350 lbs. Finally, the analysis suggests that WRAITH has a maximum powered flight range of 350 miles, with a corresponding flight time of 11.33 minutes.

[1] Responsible for Fuel-to-Air Ratios, Thrust vs. Altitude and Velocity.
[2] Responsible for Compressor Design.
[3] Responsible for Specific Impulse and Remaining Range.
[4] Responsible for Standard Atmosphere.

II. Background

The Wide-Range Autonomous Ingress Tactical Hunter (WRAITH) 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 WRAITH Specifications

Baseline Design	Old Dimension	New Dimension
Body Diameter	21.61 in	21.61 in
Reference Area	369.25 in^2	369.25 in^2
Nose Length	40.68 in	40.68 in
Total Body Length	161.95 in	161.95 in
Elliptical Height	10.00 in	10.00 in
Elliptical Width	11.67 in	11.67 in
Roll Angles	0 deg	0 deg
Nose-tip Diameter	0 in	0 in
Center of Gravity Assumption	0.5 total length	0.5 total length
Effective Exhaust Area	23.76 in^2	23.76 in^2
Leading Edge Section Angles	0 deg	0 deg
Number of Surfaces	3	3
Sweep Angles	45 deg	45 deg
Wingspan	7.87 ft	9 ft
Root Chord Length	1.00 ft	1.00 ft
Nose Tip to Root Chord Leading Edge of Wing	60.6 in	60.6 in
Tail Area	0 ft^2	3.36 ft^2

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

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. The WRAITH is launched at high altitude of 40,000 ft at Mach 0.9 from either a fighter jet or a B-1 Lancer. The theory behind this is that the aircraft is in supersonic flight and slows to Mach 0.9 to launch the WRAITH without forcing the missile into an unusual Mach range. The WRAITH 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. The missile will aim to cruise at 2000 ft at 0.71 Mach to avoid radar detection to remain stealthy. When near the target, the WRAITH will decelerate a little to 0.6 Mach and will climb to 3,000 ft and will finally accelerate when descending to hit its target, reaching a speed of around Mach 0.9.

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: 30 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.

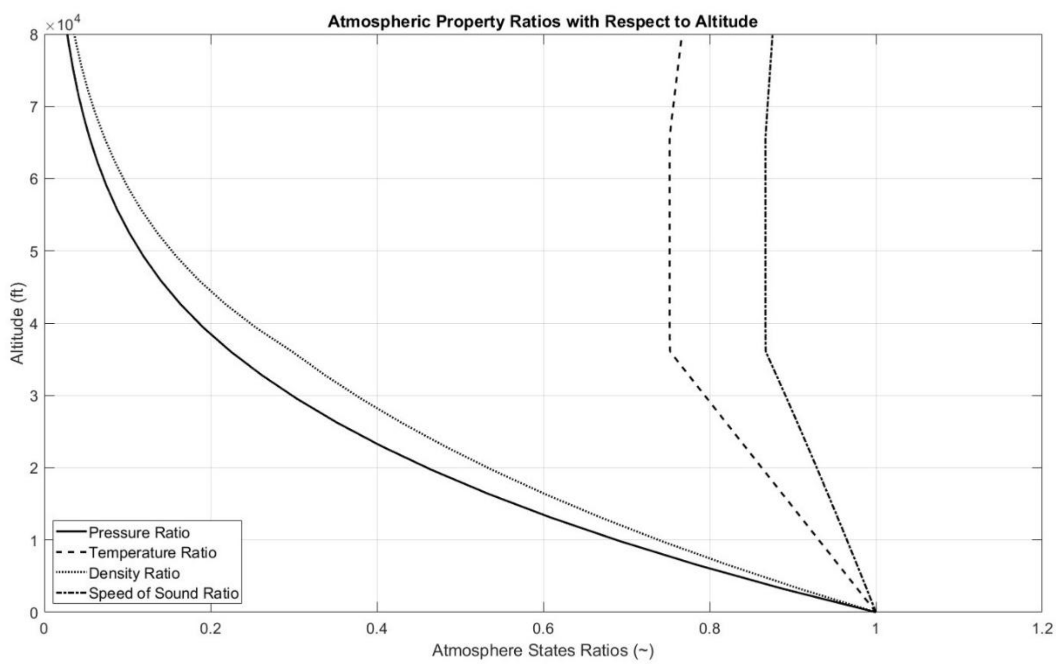

Figure 1 Standard Atmospheric State Ratios for 80,000 ft

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.

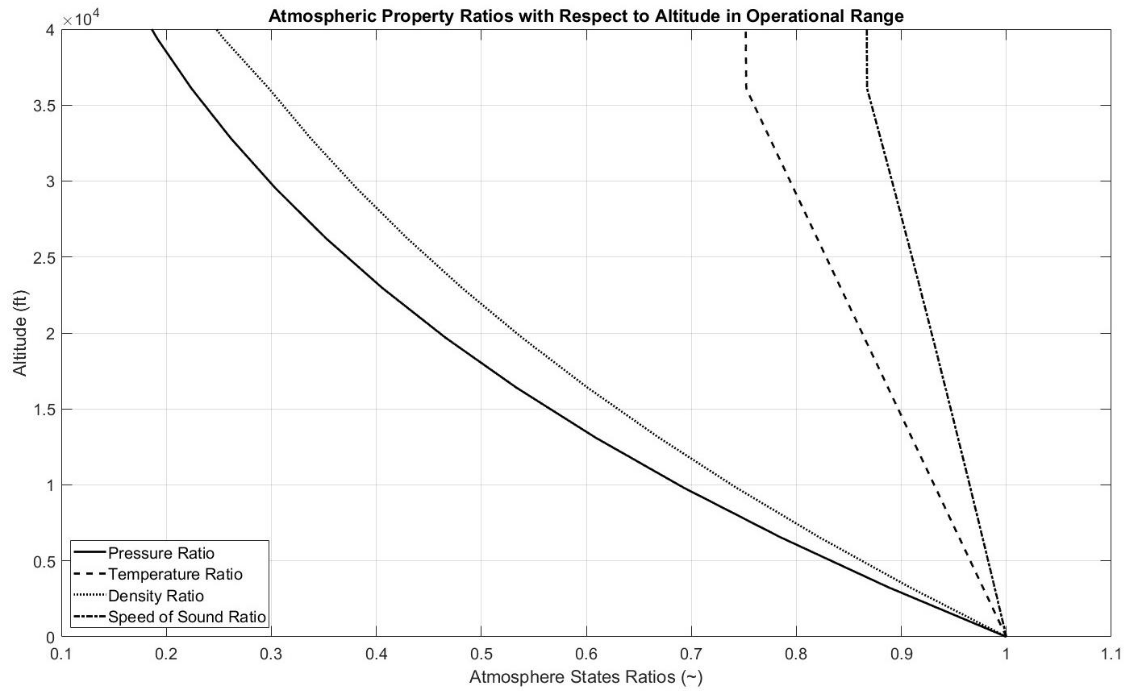

Figure 2 Standard Atmospheric State Ratios for Operational Range

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 .

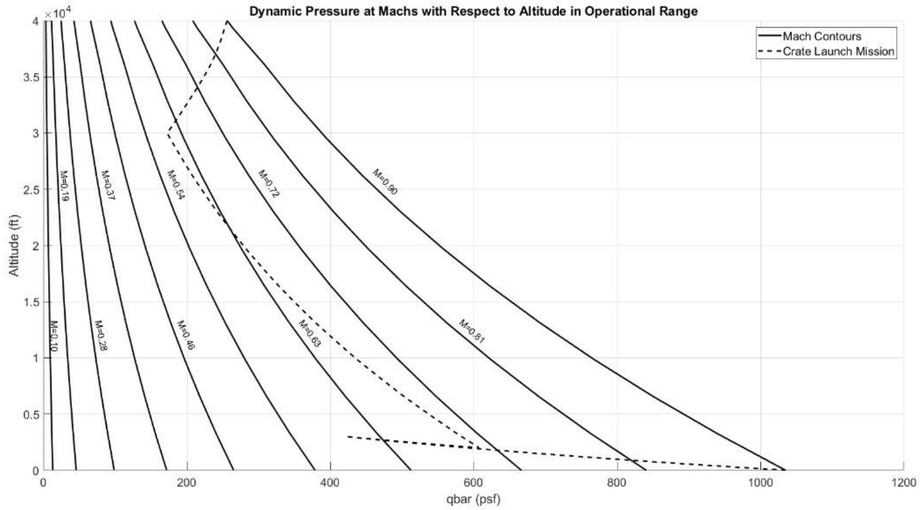

Figure 3 Dynamic Pressures at Various Mach Numbers and Altitudes

This plot also shows the two previously described missions flown in terms of altitude and dynamic pressure. Note that for different missions, the WRAITH 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 WRAITH 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 WRAITH 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 WRAITH 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 WRAITH could experience. This entire range describes how the WRAITH 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. Figure 4 combines the pressure and temperature ratios for the operational atmosphere, as well as the dynamic pressure normalized to the maximum dynamic pressure of 1034.9 psf.

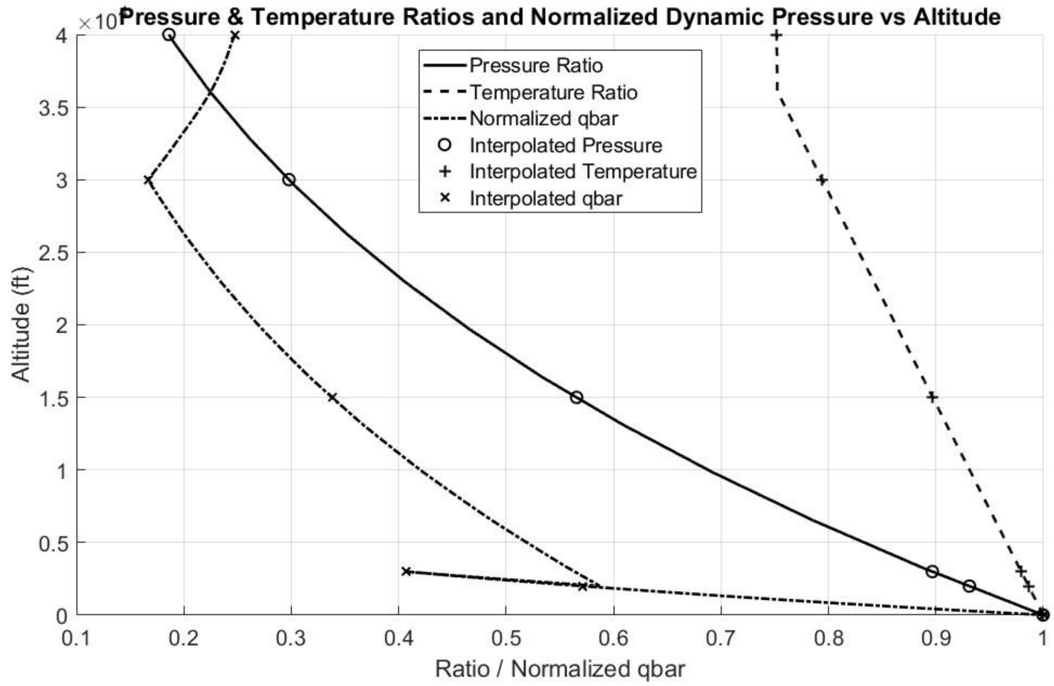

Figure 4 Combined Atmosphere and Normalized Dynamic Pressure Plot

It is seen that the minimum dynamic pressure is at when the aircraft stops coasting to a lower altitude and begins the burn to its cruising altitude.

The ranges of the Mach numbers with their respective altitudes are 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 5 can be found using Equation (1). The turbojet was only assumed to start burning at 30,000 feet, so an additional point was used in between the 30,000ft and 2,000ft segments.

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

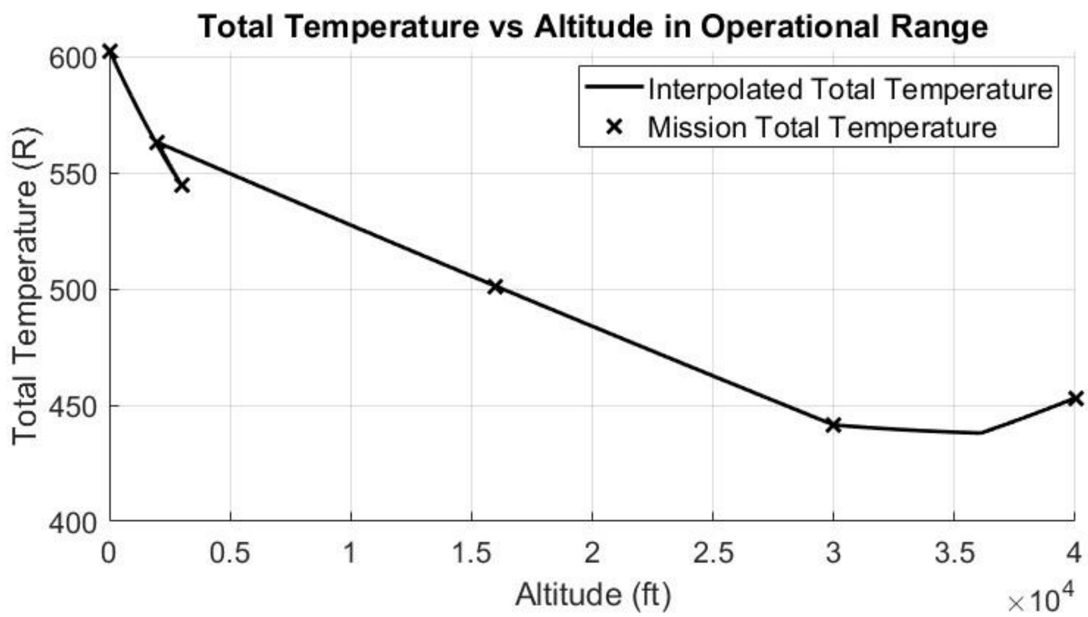

Figure 5 Varying Stagnation Temperature for Flight Regime

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 441.5 °R to 602.7°R. Below is a table with the altitude, corresponding Mach number, and the Stagnation temperature for the range where the engine is on.

Altitude (ft)	30,000	15,000	2,000	3,000	10
Mach (~)	0.6	0.658	0.71	0.6	0.9
T_{0t} (°R)	441.5	505.6	563.1	544.6	602.7

IV.Compressor Design

The baseline design of the compressor stages was comprised of three axial compressors. This yielded a maximum compression ratio of 7.61, which is under the given limit of 10. It is generally thought that increasing the pressure ratio to this maximum will allow for more efficient combustion, and thus a more powerful and efficient engine. By increasing the pressure ratio, the thrust for the given fuel rate can be maximized, so that the engine requires less fuel for the same thrust. To obtain this maximum pressure ratio, the final axial compressor was replaced by a centrifugal compressor. It was also generally thought that to reduce the complexity of the engine and thus the engine weight, the three compressors could be run on the same shaft, which would minimize the need for several gearboxes. To do this however, since the temperature increases through the compressor stages, and thus the speed of sound increases, to generate maximum compression, the diameter of each compression stage must increase so that the tip velocity matches the speed of sound. The general process to find these diameters was to find the maximum pressure ratios of the three compressors for the entire range of initial temperature, and to use the minimum diameters. As the final compressor is given a maximum diameter, using the minimum diameter ensures that the compression can be optimized for every initial temperature. After diameters for each compressor have been found, the maximum angular velocity can be found in rpm by ensuring that the Mach at the compressor tip is under 1, and that the total pressure ratio does not exceed the maximum allowable.

The background behind the types of compressors is that axial compressors force flow to move in the direction it was previously going in, but by adding energy it compresses the air downstream. A centrifugal compressor turns the air to flow away from the center shaft into a diffuser, which by slowing down the air takes advantage of the mass flow relationship which adds pressure to the air. The axial compressor maximum ratios are first found using Equation (2) assuming that the rotor Mach is equal to the maximum of 1.4 from the maximum tip Mach of 1. The relationship between tip and rotor Mach is shown in Equation (3).

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

$$M_{rotor} = 2M_{tip} - 0.6 \tag{3}$$

After this has been found, the temperature ratios are found for the compressors using the isentropic relation shown in Equation (4), where the initial temperature T_2 is first found from bringing the free stream air to zero velocity, shown in Equation (5).

$$T_3 = T_2\left(\frac{p_3}{p_2}\right)^{\frac{\gamma_3 - 1}{\gamma_3}} \tag{4}$$

$$T_2 = T_0\left(1 + \frac{\gamma_0 - 1}{2}M^2\right) \tag{5}$$

Once the two pressure ratios of the axial compressors are found, the required pressure ratio of the centrifugal compressor is found by dividing the maximum allowable pressure ratio by the product of the two axial compression ratios. Using this pressure ratio, the temperature ratio of the centrifugal compressor is found using Equation (4) above. By then also assuming that the tip Mach of the centrifugal compressor is equal to 1, the temperature at the blade tip can then be found using Equation (6).

$$T_{3tip} = T_3/\left(1 + \frac{\gamma_3 - 1}{2}M^2\right) \tag{6}$$

To then achieve this pressure ratio, the angular rotation can then be found for the compressor using the maximum diameter of the compressor and Equations (7) and (8).

$$a_{tip} = \sqrt{\gamma R T_{3tip}} \tag{7}$$

$$\omega_{tip} = 2\frac{a_{tip}}{d_{max}} \tag{8}$$

Once the angular speed of the centrifugal compressor is found, the other compressors use Equation (9) to find their diameter. Note that currently this angular speed is in rad/s.

$$d_i = 2\frac{a_{tip}}{\omega_{tip}} \tag{9}$$

Once the compressor has been designed, the maximum angular speed of the compressors must be found, and the temperatures and pressures of the compressor must be proven and shown. To do this, all viable angular speeds are input to find the compression ratio, and the optimal angular speed is selected. First, for each axial compressor, the tip velocity is first found using Equation (10), and the Mach speed of the tip is then found using Equation (11).

$$u_{tip} = \frac{1}{2}d_i\omega \tag{10}$$

$$M_{tip} = \frac{u_{tip}}{\sqrt{\gamma R T_2}} \tag{11}$$

The pressure ratio is then recalculated using the previously used axial equations, and the temperature ratio is then found. After this has been found, the temperature after the centrifugal compressor is found by setting the tip Mach to be 1, which is assumed to be a product of optimal centrifugal compressor blade design, and reusing a flipped Equation # to find the temperature at the blade tip. This is then used to find the temperature after the compressor tip by flipping Equation #. This temperature is then used to find the temperature ratio of the centrifugal compressor, which is in turn used to find the pressure ratio of the centrifugal compressor using a flipped Equation #. The rpm of the shaft can be found by multiplying the angular velocity in rad/s by 60sec/min and dividing by 2πrad/rotation. Note that sometimes loops are used for situations where a gamma calculated from a certain temperature is used to calculate that certain temperature. It was also found that some errors occurred in the process, leading to pressure ratios just under 10 in the compressor. To fix this, an adjustment value was given to the axial compressor diameters, and by further decreasing the diameter by a small amount, the error could be corrected to nominal values. This correction was usually around half an inch.

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. Using a maximum compressor diameter of 16 inches to allow for structure around the engine, the following diameters were found for each compressor, shown in Table 2.

Table 2 Compressor Section Diameters

Axial Compressor 1 (C1) Diameter (ft)	1.0118
Axial Compressor 2 (C2) Diameter (ft)	1.1187
Centrifugal Compressor (C3) Diameter (ft)	1.3300

Using these diameters, the angular speeds and pressure ratios (PR) for each section can be found as well as the total pressure ratio. These speeds and pressure ratios are shown in Table 3. It is seen that as the initial temperature increases, it is more difficult for the axial compressors to function, so the centrifugal compressor must pick up the lost compression. Through this design, the pressure ratio of 10 was essentially achieved, and the angular speeds and pressure ratios should be noted as a maximum for the engine.

Table 3 Pressure Ratios and Rotation Speed

T2 (°R)	N (rpm)	PR_{C1} (~)	PR_{C2} (-)	PR_{C3} (-)	PR_{Tot} (-)
441.49	18667	1.8522	1.8853	2.8619	9.9934
505.62	19886	1.8424	1.8795	2.8863	9.995
563.11	20905	1.8342	1.8746	2.9068	9.9946
544.55	20583	1.8369	1.8761	2.9004	9.9952
602.65	21574	1.8291	1.8715	2.9213	9.9997

Lastly, the temperature after each section is found and is shown in Table 4, where 2.3 is the station after C1, 2.6 is the station after C2, and station 3 is the station after C3 and the compressor outlet temperature.

Table 4 Compressor Temperatures

T_2 (°R)	$T_{2.3}$ (°R)	$T_{2.6}$ (°R)	T_3 (°R)
441.49	526.62	629.76	847.23
505.62	601.14	716.84	963.46
563.11	667.65	794.23	1066.5
544.55	646.2	769.3	1033.4
602.65	713.26	847.18	1137.1

V.Fuel-to-Air Ratios

Table 5 shows the 5 different compressor outlet temperatures calculated in MATLAB that are used for the fuel to air ratio plot.

Table 5 MATLAB Output for Compressor Outlet Temperature

Altitude (ft)	Tt2 (R)	Compressor Outlet Temperature (Tt3) in R
10	602.7	1,137
2,000	544.6	1,033
3,000	563.1	1,067
15,000	501.27	956
30,000	441.49	847

The fuel-to-air ratios are plotted in Figure 6 to raise the temperature from 5 different T_3 as shown in Table 5 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.

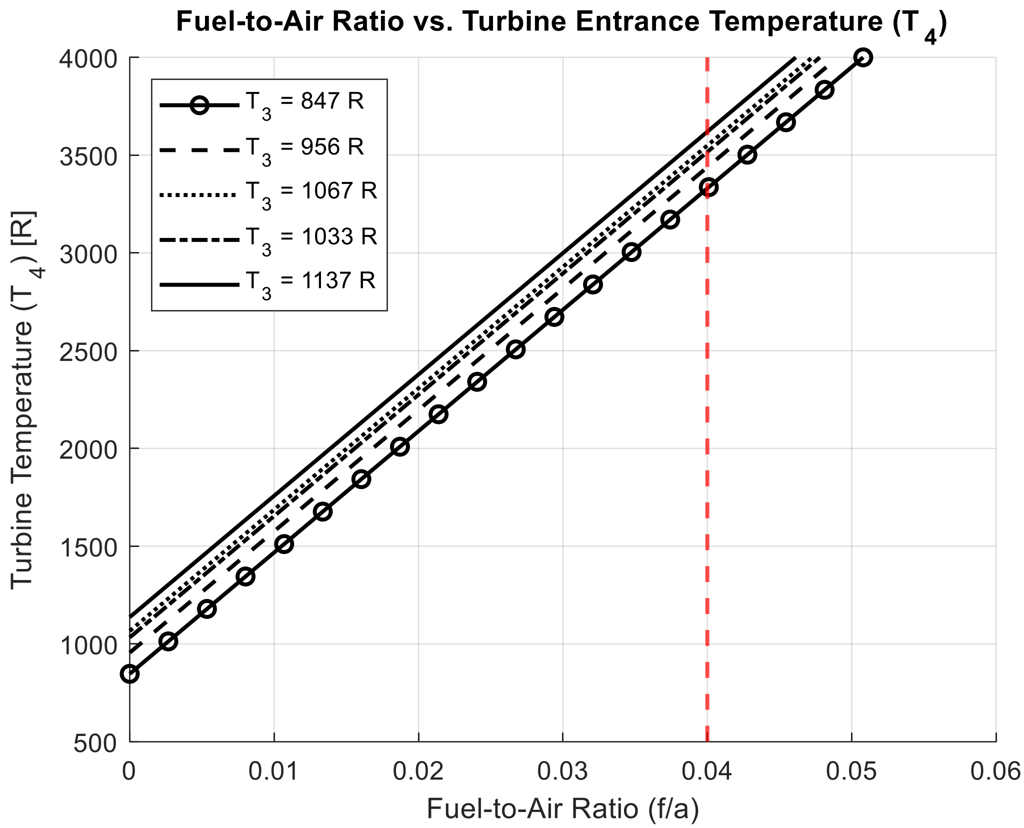

Figure 6 Fuel-to-air Ratio vs T4

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).

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

$$T_2 = T_0 \left\{1 + \frac{\gamma_0}{2} M_0^2\right\} \quad (13)$$

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 $T_4 = 3{,}812.8\ °R$.

VI. Thrust vs. Altitude and Velocity

The thrust created by the turbojet engine is presented in assuming a constant maximum compressor of 10. Moreover, 5 contour lines of standard atmosphere static pressure and temperature lines are presented in Figure 7. The standard atmospheric conditions are selected based on the WRAITH mission from 10 ft to 30,000 ft. The assumptions of ideal turbojet with perfect gas ($\gamma_0 = \gamma = 1.4$) are made to find the thrust as shown in Figure 7, 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 as show in Figure 8.

The performance curves for the WRAITH turbojet engine demonstrate the primary trends within the subsonic flight envelope in Figure 7. The thrust increases with Mach number at all chosen altitudes. Additionally, as pressure and temperature decrease, thrust also decreases. The most thrust of almost 1600 ft are created when closing in the target at 10ft

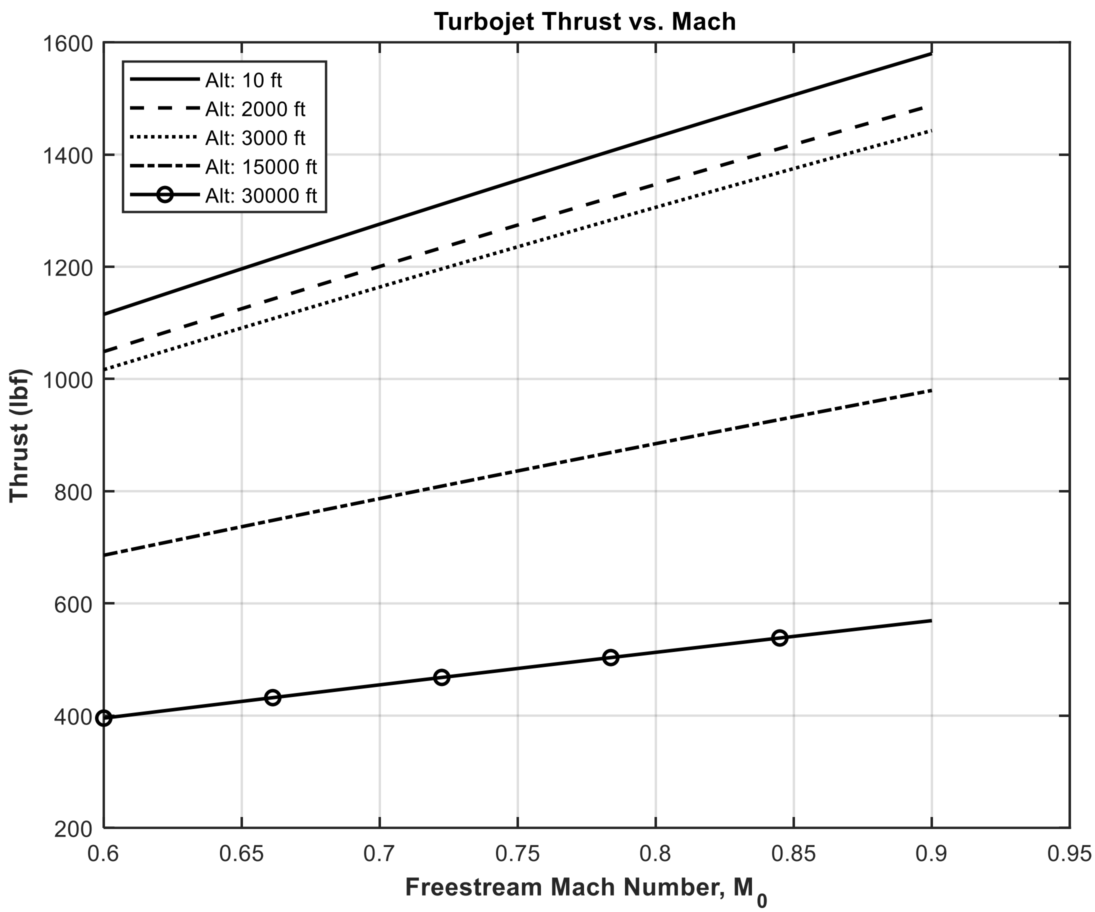

Figure 7 Updated Thrust Vs. Mach Number at Different Altitude at Pressure Ratio of 10

Figure 8 shows that with a new pressure ratio of 20, assuming that it is possible. That results in only a small increase in thrust at different altitudes.

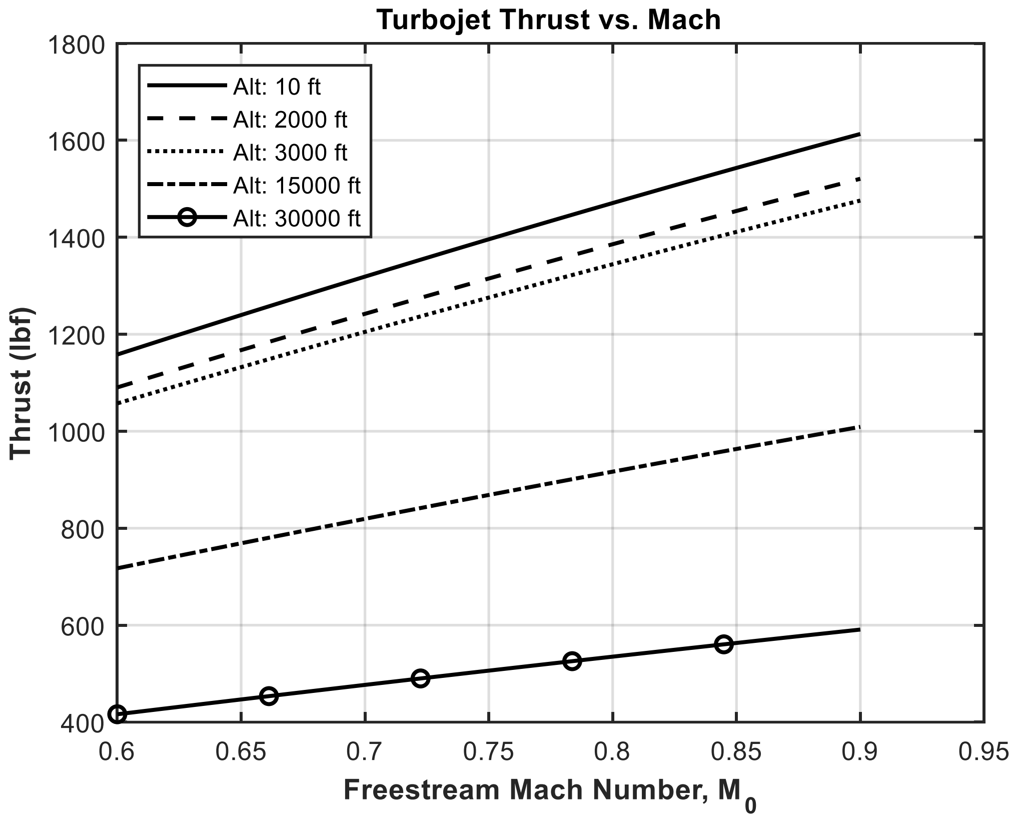

Figure 8 Updated Thrust Vs. Mach Number at Different Altitude at Pressure Ratio of 20

Figure 9 shows the previous thrust Vs Mach at different altitude plot. The incorrect equation shows that the thrust is extremely high when compared to the correct thrust equation. The correct equation shows realistic thrust that common turbojets can generate.

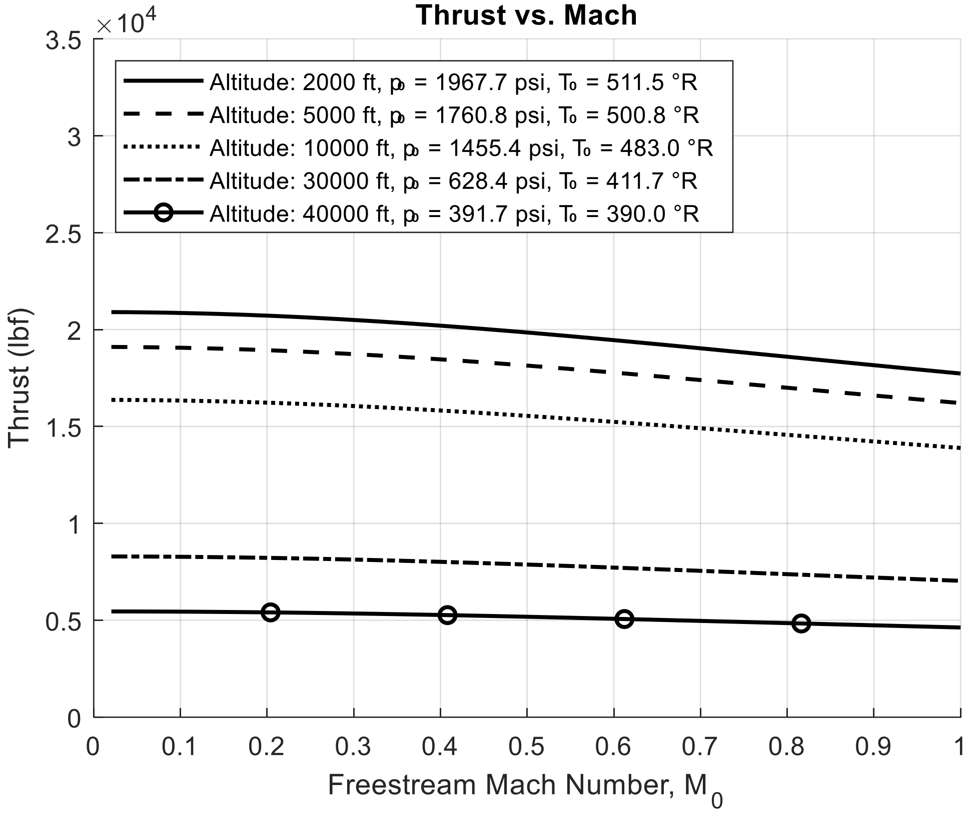

Figure 9 Previous Thrust Vs. Mach Number at Different Altitude at Pressure Ratio of 10

VII. Specific Impulse

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 (21).

$$\mathrm{I_{sp}} \;=\; \mathrm{T} \,/\, \dot{\mathrm{m}}\,\mathrm{g_0} \;\; [\mathrm{sec}] \tag{14}$$

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 (21).

$$\left[g_0 c_p T_0/\left(a_0 H_f\right)\right](I_{SP})_{Tmax} = T_{max} T_0/[(p_0 A_0 \gamma_0 M_0)(T_4 - T_0)] \tag{15}$$

Rearranging the equation to instead use the known combustion temperature ratio and the compressor ratio, Equation (23) is found. The turbojet redesign increases the specific impulse by increasing the compressor pressure ratio (p_3/p_2).

$$[g_0 c_p T_0/(a_0 H_f)](I_{SP})_{Tmax} = T_{max}/(p_0 A_0)/\left[\gamma_0 M_0\left[(T_4/T_0) - (p_3/p_2)^{(\gamma_0-1)/\gamma_0}\right]\right] \tag{16}$$

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.

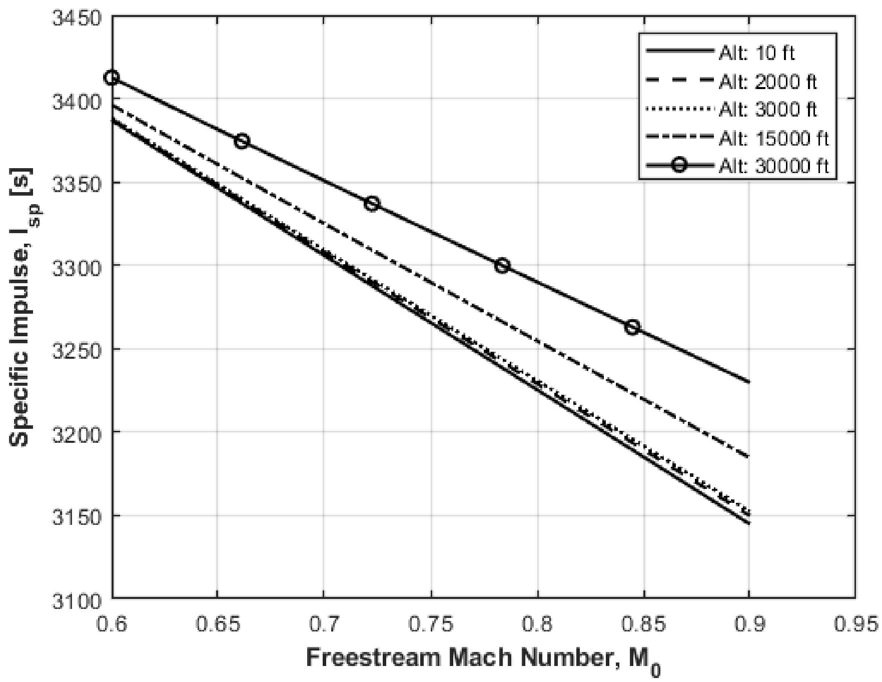

Figure 10 Specific Impulse

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

VIII. Remaining Range

Using the lift-to-drag characteristics established in the preceding aerodynamic analysis, the missile's range was evaluated. The range of the WRAITH could then be determined using the Breguet range equation, shown in Equation (24).

$$R = \left(\frac{L}{D}\right) I_{SP} V_{avg} \ln[W_L/(W_L - W_P)] \tag{17}$$

With a launch weight (W_L) of 2250 lbs. and a lift to drag determined at the average velocity. The redesigned aerodynamics produced a lift-to-drag of 3.93 and the weight of the fuel was determined to be 350 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 11. The mass rates of air and fuels are shown in Equations (25) and (26).

$$\dot{m_{air}} = \rho_0 V_{avg} A_0 \tag{18}$$
$$\dot{m_{fuel}} = \dot{m_{air}} * f/a \tag{19}$$

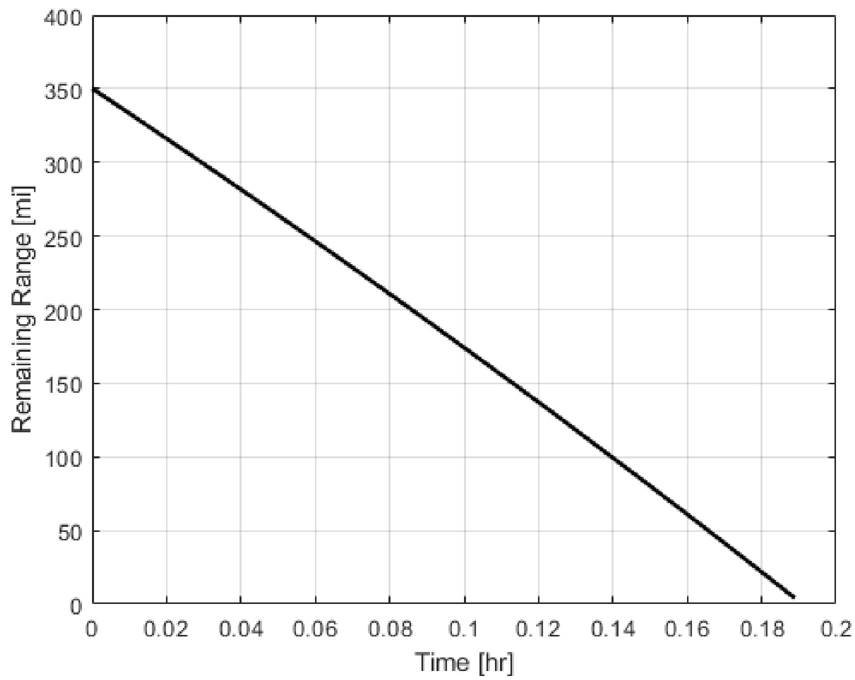

Figure 11 Remaining Flight Time

Calculating the mass flow rates at the average velocity results in a $\dot{m_{air}}$ of 0.35 lbs/s, $\dot{m_{fuel}}$ of 0.016 lbs/s and applying the Breguet range equation with an available fuel weight yielded a 40% increase in range from the baseline from 250 miles to 350 miles, with a flight time 11.33 minutes.

IX.References

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X. Appendix A

Standard Atmosphere – Mohamed Almazrouei
Compressor Design – Nathaniel Hollman
Fuel-to-Air Ratios & Thrust vs. Altitude and Velocity – Tri Phan
Specific Impulse & Remaining Range – Tiger Sievers