

Creation of an Original Design for a Light Attack Aircraft

A Technical Report submitted to the Department of Mechanical and Aerospace Engineering

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Bachelor of Science, School of Engineering

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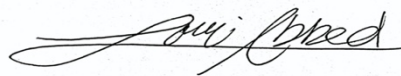
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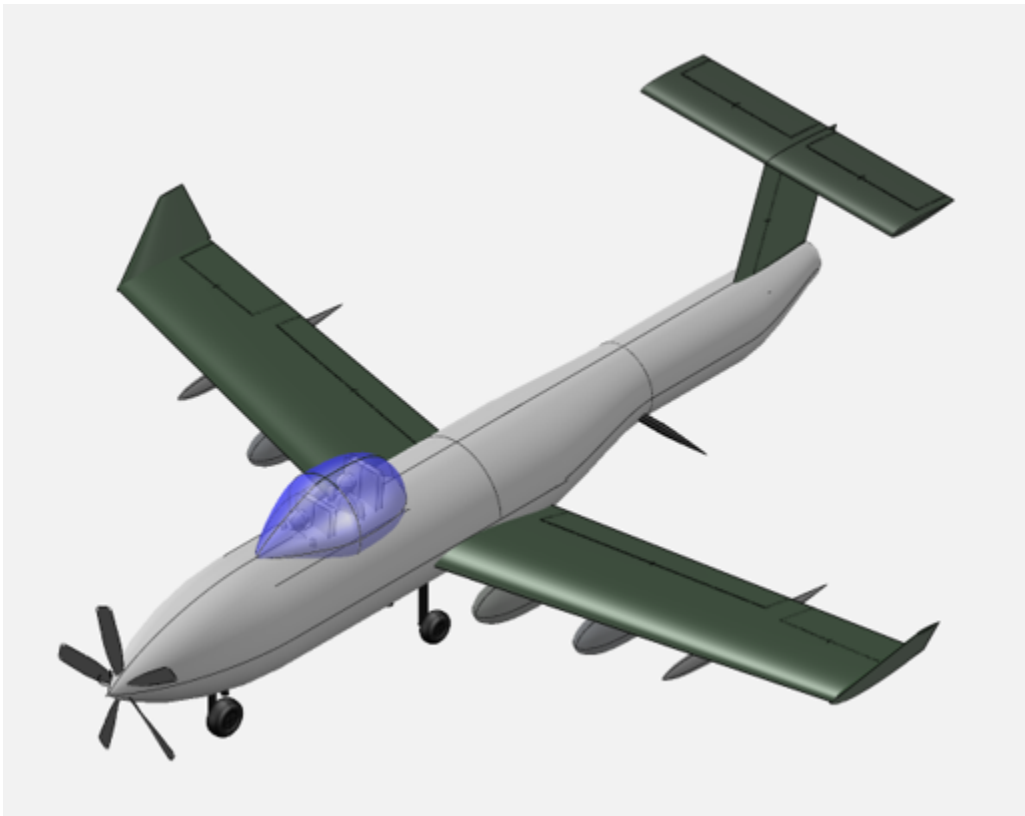
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The Hummingbird

AIAA 2020-2021 Undergraduate Design Proposal



MAE 4660

Mechanical and Aerospace Engineering

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May 14, 2021

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II Executive Summary

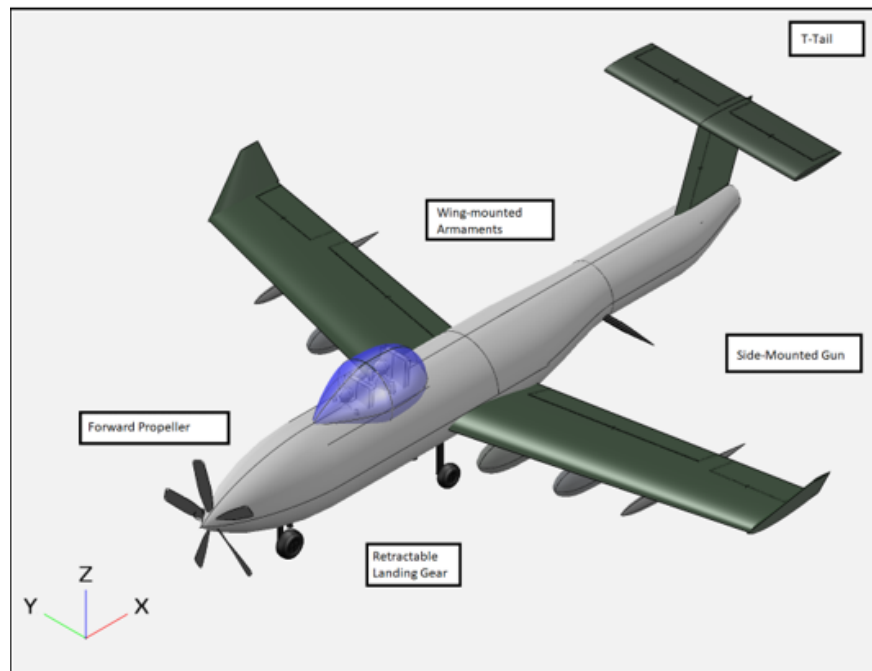


Figure 1: Highlighted Features

Advancements in military aircraft have pushed the boundaries of an aircraft's capabilities, but have also greatly increased cost. This creates a niche for a low cost aircraft that can be stationed in remote and underdeveloped locations to provide close air support that would be too expensive for other aircraft to provide. This report provides a preliminary design for a Light Attack Aircraft as requested in the RFP by the AIAA Student Design Competitions [9]. The RFP set out requirements for a light fighter plane capable of carrying two pilots, armaments, and an integrated gun on a loiter design mission. Additionally, the plane must be able to take off and land from short austere runways, and meet other performance and operational requirements while minimizing cost. This report presents the design of "the Hummingbird" as a proposal to this RFP. The Hummingbird's notable features include a single forward-mounted turboprop engine, a T-tail empennage, wing-mounted payload, and a deployable side-mounted gun; these features are visible" in Figure 1. The Hummingbird, with a full payload, has a maximum range of 1740 nmi and costs \$29 million to acquire and \$1743 per flight hour to operate. The Hummingbird has a take off gross weight (TOGW) of 14433 lb and an empty weight of 8002 lb. This design is able to meet all requirements set out in the RFP.

III Vehicle Summary

Table 1: Key Vehicle Characteristics

| | |
|---|------------------------------|
| TOGW (OEW) | 14433 (8002) lb |
| Design Mission Block Fuel Burn | 3126.9 lb |
| TSFC at Cruise | 0.54 lb/(h·lbf) |
| Takeoff/Landing Field Length | 3313 ft / 2555 ft |
| Takeoff/Landing Field Length at 6000 ft | 3931 ft / 2981 ft |
| Service Ceiling | 34000 ft |
| CL_{cruise} / CL_{max} | 0.81 / 2.75 |
| Aircraft Length | 42.7 ft |
| Wing Area (AR) | 42.9 ft |
| Wing Span | 278.85 ft ² (6.6) |
| Acquisition Cost | \$29,322,000 |
| Operating Cost per Flight Hour | \$1742.86 |

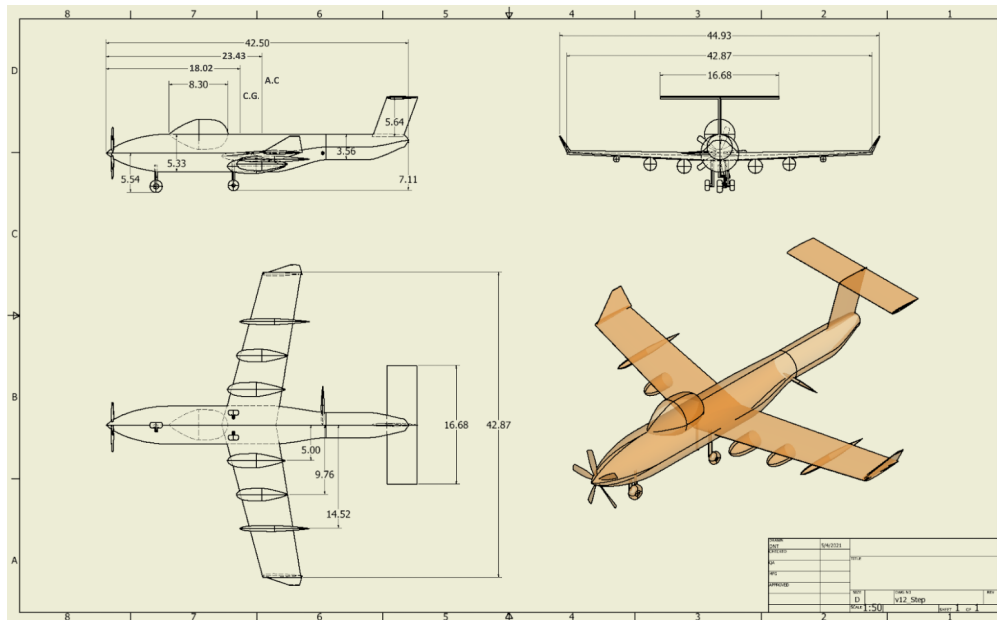


Figure 2: Dimensioned Diagram of the Hummingbird

IV Introduction

The AIAA Undergraduate RFP calls for a low cost light attack aircraft capable of fulfilling the close air support role traditionally fulfilled by a helicopter. Furthermore, the plane must be able to operate out of an austere field. These explicit goals imply several notional design objectives and associated measures for performance. Figure 3 outlines the

initial objective tree created after analyzing the RFP objectives. The intermediate levels represent implied objectives, with the last level indicating possible metrics to measure performance.

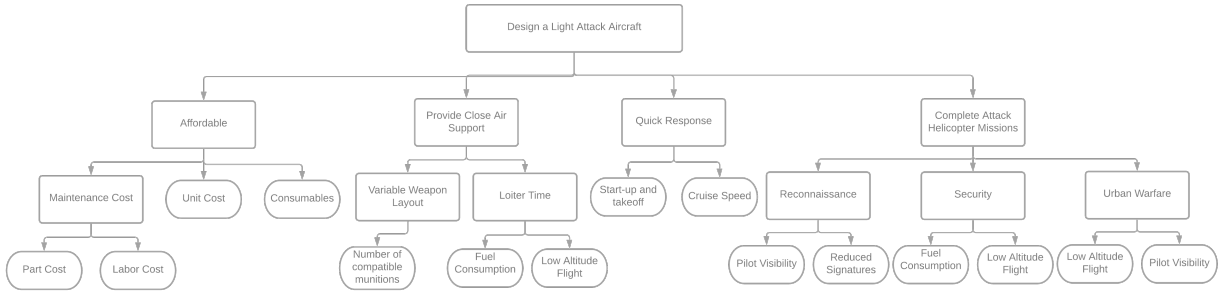


Figure 3: Initial Objective Tree

The Hummingbird was designed with these goals in mind with an added emphasis on affordability throughout the design process. This goal influenced design choices to minimize complexity and utilize existing components. For example, the engine selected was the Pratt & Whitney Canada PT6A-6 Variant, a commonly used and well-known engine.

V Design Process

To design the aircraft both empirical and model based methods were used. Empirical methods, such as the ones provided in Nikolai and Roskam set the starting point for further design work [3, 10, 11]. Initial values were assumed to be similar to those of existing aircraft such as the Fairchild Republic A-10 Thunderbolt II or the Embraer EMB 314 Super Tucano as these and other aircraft had similar design objectives and purposes. From there, equations were used to further the design until all major values had been propagated through an initial model. Those values served as the inputs for model based design. Table 2 provides the tools and what regime of analysis for which they were utilized.

Table 2: Tools Used

| Program | Applied Area |
|---|--|
| AAA [12] | Cost Analysis |
| The Flight Optimization System (FLOPS) [13] | Mission Analysis and Weight Estimation |
| GasTurb [14] | Engine Modeling |
| OpenVSP [15] | Modeling |
| Solidworks [16] | Structural Analysis |
| VSP Aero [15] | Aerodynamics |
| XROTOR [17] | Propeller Analysis |

From there the output was compared to the empirical results as a cross reference. Trade studies were then conducted

like the one shown below in Figure 26. Results were also compared to other aspects of the aircraft as a check for agreement. This process was iterated until a closed loop was formed with a satisfactory output achieved, and then continually updated as further design progress was made.

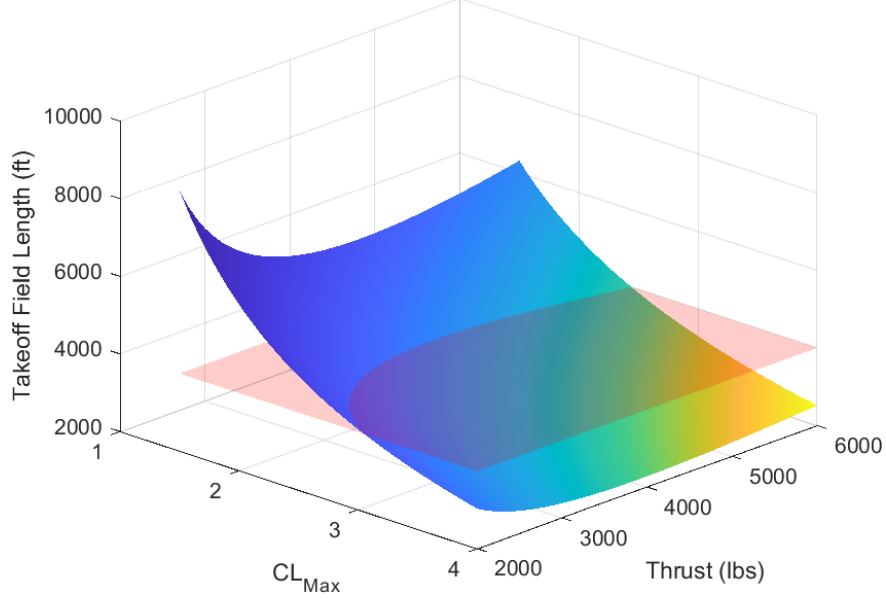


Figure 4: Sensitivity Analysis of the Takeoff Field Length

VI Sizing

VI.1 Initial TOGW

The initial takeoff gross weight served as the first variable defined and formed the basis for further initial design. To get the initial TOGW a weight fraction method was used. From the design mission, weight ratios were estimated for each phase of the flight. By taking these weight fractions, shown in Table 3, and the initial TOGW estimation of 8000 lb, the fuel weight was calculated. The fixed weight was estimated at 3600 lb. Using equations 1 and 2, the empty weight and required empty weight were calculated [3].

$$W_{empty\ available} = W_{TO} - W_{fuel} - W_{fixed} \quad (1)$$

$$W_{empty\ required} = 0.911 \cdot W_{TO}^{0.947} \cdot 0.84 \quad (2)$$

From there a new TOGW was estimated and the process was repeated until the empty weight and required empty weight converged. Appendix I includes the values that were tabulated and Figure 5 plots the empty weight and empty weight required calculated from this method.

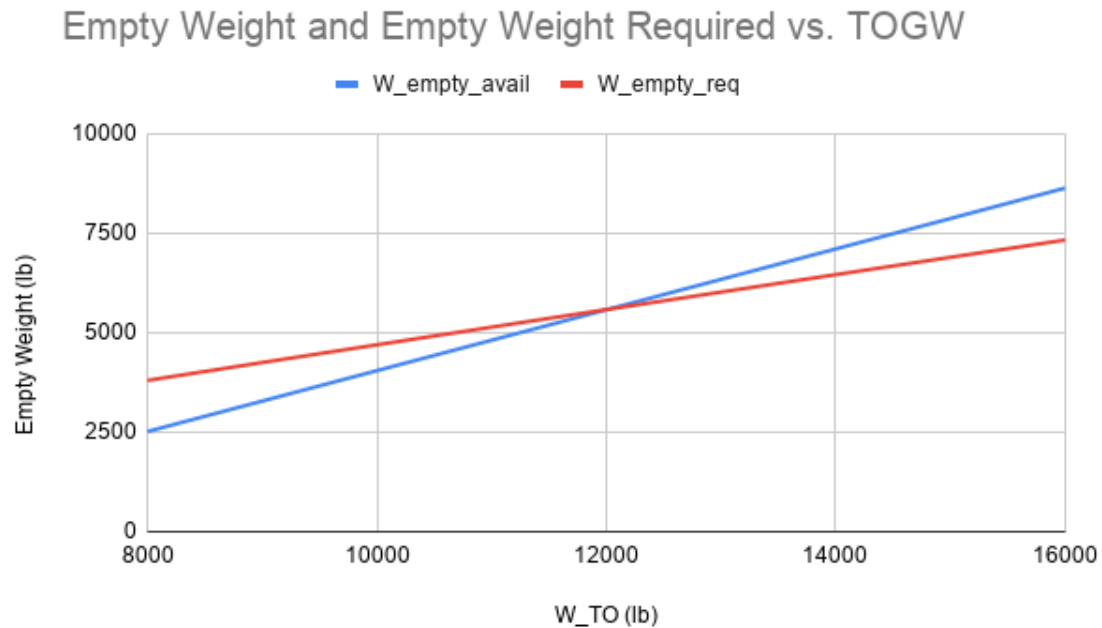


Figure 5: Empty Weight Calculations as a Function of TOGW

Table 3: Weight Fraction Estimates by Flight Segment

| Mission Profile | Weight Ratio (formula) | Weight Ratio |
|--------------------------|------------------------|--------------|
| Start/take off | w2/w1 | 0.975 |
| Climb out to cruise | w3/w2 | 0.975 |
| Cruise out | w4/w3 | 0.985 |
| Descent | w5/w4 | 0.980 |
| Loiter | w6/w5 | 0.898 |
| Climb out | w7/w6 | 0.980 |
| Cruise back | w8/w7 | 0.985 |
| Descent/land/taxi | w9/w8 | 0.990 |
| Reserve (climb to 3000) | w10/w9 | 0.990 |
| Reserve (Loiter 45 mins) | w11/w10 | 0.980 |

$$Payload\ Growth\ Factor = \frac{dW_{TO}}{dW_{fixed}} \quad (3)$$

From these calculations a payload growth factor of 3.13 was calculated from equation 3. A resultant initial TOGW of 12018 lb was found. This number seemed reasonable as it is similar to the weight of other aircraft in its class with a shared design objective.

VI.2 Initial Fuselage

Initial fuselage sizing was based around the general aviation sizing procedures put forth by Nicolai, where the aircraft is approximated as having a cylinder fuselage section (termed Section A in this analysis) and a rear-facing cone of the same diameter as a tail section (termed section B) [3]. The analysis was performed using parameters such as maximum fuel weight, cruise velocity and altitude, and estimated empennage dimensions and weights from a previous take off gross weight analysis. Several basic assumptions and approximations were laid out by Nicolai, which include: assuming the wings (and payload on wings) will be located at center of gravity (will not be included in calculations), and the fuselage's exterior material and structure is centered around the aircraft's center of gravity (not considered during this analysis). Analysis began using the Embraer EMB 314 Super Tucano length and estimated width of 37.34 and 4.66 feet respectively (derived from ideal thickness to length ratio for subsonic aircraft) [18].

Section A, Main Fuselage

The cylindrical section was developed to house the PT-6A Turboprop engine, the two passengers (from RFP), their cockpit furnishings, fuel tanks (including mechanical equipment), electronics, and structural internal wing sections [9]. The engine was approximated as a cylinder with dimensions and weights resembling that of a PT-6A turboprop engine, similar to that of the Super Tucano. Exact specifications are shown in Table 4.

Table 4: Initial Fuselage Sizing Complete Table of Weights and Dimensions

| Component | Total Weight (lb) | Length (ft) | Diameter/ Width (ft) | Approximated Shape |
|----------------------------|-------------------|-------------|----------------------|--------------------|
| Two Pilots | 360 | N/A | N/A | N/A |
| Cockpit Furnishings | 895.97 | 8.33 | 4.67 | Square Prism |
| JP-4 Fuel (Maximum Volume) | 2828.12 | 9.07 | 4.67 | Half - Cylinder |
| JP-4 Fuel (Minimum Volume) | 141.41 | N/A | N/A | N/A |
| Flight Electronics | 1633.5 | 3.42 | 3.42 | Cube |
| PT-6A Turboprop Engine | 598 | 6.00 | 3 | Cylinder |
| Initial Empennage | 78.23 | N/A | N/A | Point |

The estimated passenger weight (including gear) was reported as 180 lbs (given by Nicolai). Furnishing weight was determined as a function of passenger quantity and flight dynamic pressure, as described by Nicolai, this weight includes ejection seats for all passengers and assumes the aircraft is subsonic. The following equation was used [3]:

$$W_{Furnishings} = 34.5 \cdot pq^{0.25} + 22.8 \cdot \left(\frac{pq}{100}\right)^{0.743} \quad (4)$$

where p is the number of passengers and q is the dynamic pressure. The dynamic pressure was calculated at cruise velocity and altitude, as calculated in the TOGW analysis, which yielded 2438.495 lb/(ft²s²), and the number of passengers was input as 2. The results are shown in Table 4. Nicolai approximated for 2 passengers a cockpit length

of 100 inches, or roughly 8.33 feet.

Next, using a known required maximum fuel weight from TOGW analysis, the team calculated a fuel volume using the density of JP-4 jet fuel. The fuel tank volume, which includes the required fuel volume, was calculated as well; calculations assumed an internally mounted bladder tank with a packing factor of 6.5 (that took into account the mechanical equipment required by the fuel system). This fuel tank volume could be represented in an initial model as a half-cylinder, with the same diameter as the fuselage, which would be mathematically used to determine the required length of the fuel tank.

A general electronics setup table for all types of aircraft was given by Nicolai, this table was likely outdated and electronics in modern times are much lighter, so this estimate is likely conservative. After performing brief research into the nature of the electronics included in his documentation, slight modifications were made to the final list of selected electronics to reduce weight, mainly by limiting the selected components to only those necessary for the flight and survival of a small attack aircraft (specifically larger radar and reconnaissance components were omitted). The electronics from Nicolai were often missing volumes or weights, so an average electronics density was calculated from the known electronics and applied to the full table of components. This allowed a general volume of electronics to be created. Packaging and optimal functionality and maintenance placement was not considered in the initial sizing and all electronics were assumed to fit within a box. A generalized electronics weight and volume is depicted in Table 4.

For this section, the total length of the section was determined by summing the lengths of the relevant components (the cockpit, fuel tanks, electronics, and engine). Obviously, this would lead to an over estimation of total length, as the full fuselage volume is not utilized but it leaves room for components like the structural internal wing, wiring, and fuselage structure. Further, a slight fuselage taper and nose cone were added in OpenVSP to make the plane look more appropriate and generate a working aerodynamic model, this nose cone was sized manually and was ~7 feet in length. A rough landing gear model was also added in OpenVSP and allowed for height measurements as well, landing gear was assumed to contribute minimally to center of gravity and was not included in mathematical analysis. This information is included in Table 6.

Section B, Tail

The tail section was only constrained by the 15° tail strike military criteria for landing gear. This constraint became the slope of the cone, assuming the landing gear would be located somewhere near the end of the main fuselage. The cone was further constrained to have a diameter equal to that of the main fuselage. The only weighted component in this section's calculations was that of the empennage, which was approximated in the TOGW analysis. The empennage was validated using an empennage area to wing area ratio of ~0.3 as described by Nicolai. The Hummingbird's ratio was slightly higher than ideal, but was still usable. Approximations for empennage weight are based around an empirical

relationship between weight and the area of the horizontal and vertical tails [11]. The relevant empennage parameters are given in Table 5:

| Parameter | Value |
|-----------------------------------|-----------------------|
| Vertical Tail Area | 21.24 ft ² |
| Horizontal Tail Area | 49.88 ft ² |
| Total Empennage Area | 71.11 ft ² |
| Total Wing Area | 218.5 ft ² |
| Ratio of Empennage and Wing Areas | 0.325 |
| Estimated Empennage Weight | 78.23 ft ² |

Table 5: Initial Section C Component Parameters

Table 6: Initial Fuselage Sizing Summary of Relevant Parameters

| Section | Component | Location (From Nose to Front of Component) (ft) | Length (ft) |
|---------|--|--|-------------|
| A | Nose Cone | 0 | 6.98 |
| | Engine | 6.98 | 6.00 |
| | Cockpit | 12.98 | 8.33 |
| | Fuel Tanks | 21.31 | 9.07 |
| | Electronics | 30.38 | 3.42 |
| | Entire Section | 0 | 33.8 |
| B | Entire Section | 33.8 | 8.71 |
| A, B | Entire Fuselage | 0 | 42.51 |
| A, B | Optimal Fuselage (Ideal L/D Ratio of 8) | 0 | 37.34 |

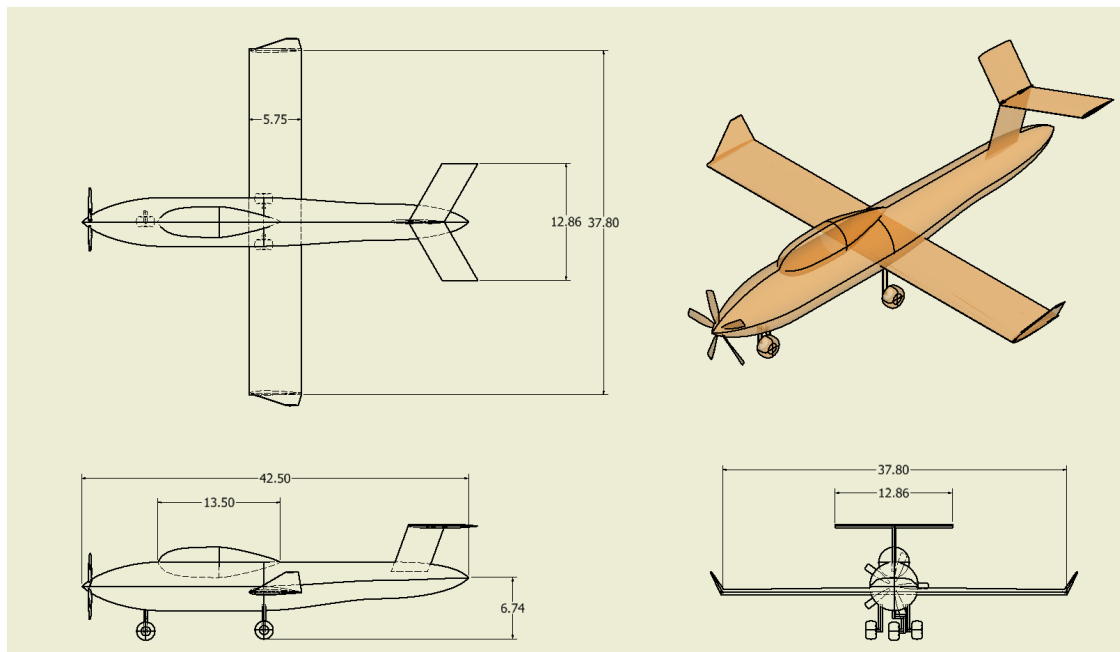


Figure 6: Isometric View of Initial Aircraft In Solidworks

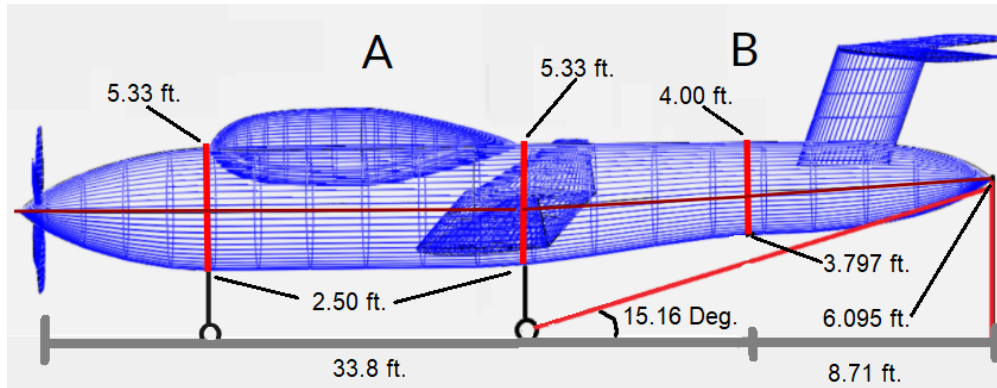


Figure 7: Dimensioned Side View of Initial Aircraft In OpenVSP, Sections A and B Shown

VII Configuration

VII.1 Concept Down Select

VII.2 Configuration Iteration

VII.3 Initial Configuration

VII.4 Fuselage

Since the initial model was slightly longer than the optimal fuselage length, variations in diameter and length were performed to quantify their impact on aircraft stability and weight distribution. With the initially proposed components, the team found the absolute smallest aircraft that could be feasibly designed would be 34.112 feet long. However it was determined that reducing the length of the aircraft for the selected empennage design was not advantageous for aircraft control, so the total fuselage length of 42.51 feet was maintained throughout the final design iterations. Rather, it was deemed more feasible to increase the diameter of the plane by about 1 ft. This allowed the team to achieve a length closer to optimal and gain additional internal volume. This further allowed the team to contour and shape the fuselage to increase aesthetic appeal, reduce weight, and increase aerodynamics. This increase in diameter also allowed for shorter, wider fuel tanks and significantly decreased the variation in center of gravity seen throughout the aircraft's flight. The next major change to the initial model was that of a titanium tub surrounding the pilots and cockpit, this was a consideration for aircraft and passenger survivability. The tank weight is shown in Table 7. This significant weight addition moved the center of gravity forward, drastically impacting the aircraft's weight distribution. To counteract this, the main weapon, the .50 caliber machine gun, which is discussed in the armament section, was added to the tail

section of the fuselage. This machine gun, the associated mount, and ammunition would be used to offset the weight of the titanium tub. This was deemed feasible with use of a gun camera in the electronic components. Ultimately these additions allowed the center of gravity to remain central with great improvements to aircraft survivability and function.

Table 7: Final Fuselage Sizing Complete Table of Weights and Dimensions

| Component | Total Weight (lb) | Component Length (ft) | Component Diameter/ Width (ft) | Approximated Shape |
|--|-------------------|-----------------------|--------------------------------|--------------------|
| Pilot (2) | 360 | N/A | N/A | N/A |
| Titanium Shielding | 1102.31 | N/A | N/A | N/A |
| Cockpit Furnishings (Including Ejection Seat) | 895.97 | 8.33 | 5.33 | Square Prism |
| JP-4 Fuel (Maximum Volume) | 3402 | 9.07 | 5.33 | Half - Cylinder |
| JP-4 Fuel (Minimum Volume) | 170.1 | N/A | N/A | N/A |
| Flight and Survivability Electronics | 1633.5 | 10.76 | 1x 2.11 1x 1.82 | Two Square Prisms |
| PT-6A Turboprop Engine | 598 | 6.00 | 3 | Cylinder |
| Empennage | 78.23 | N/A | N/A | Point Mass |
| Rear Mounted Machine Gun (Including Mount) | 140.50 | 5.508 | 0.583 | Square Prism |
| Ammunition (200) | 771.618 | N/A | N/A | N/A |

Next a final list of electronic components was compiled. The total weight remained the same as the approximate weight generated in the initial design, however the individual electronics were packaged in a manner that allowed for ease of access for maintenance, described as “chest height and one deep” [3]. The electronics were approximated as cubes, laid out along both sides to reduce consumed fuselage length and placed relative to each other to ensure an even weight distribution. Electronics are shown in Table 8. One later adjustment that was made was the addition of more fuel. Similar to the initial fuselage sizing assessment, maximum fuel weight was given, this time by mission profile analysis in FLOPS, then fuel volume, required fuel tank volume, and the required length of the half cylinder fuel tank were calculated using the same methods as discussed previously. There was a 600 lb increase in fuel capacity, but due to the added diameter of the fuselage there was not a significant increase in fuel tank length nor was there a large variation in center of gravity. Lastly, the empennage underwent significant modifications throughout the iterative design process. Empennage design, empennage size, seemed dependent on the aircraft’s center of gravity, which varied minimally throughout the fuselage design process. The most notable change in empennage design was that the horizontal tail design was modified by removing the initial sweep. Ultimately, the total area of the empennage did not change significantly, and the area-based weight assessment was relatively constant.

Table 8: Final Fuselage Sizing Electronics

| Side | Component | Weight (lb) | Side Length (ft) | Location (Measured From Nose To Front of Component) (ft) |
|-------|------------------------------|---------------|------------------|---|
| Left | Lead Computing Optical Sight | 5 | 0.496 | 0 |
| | UHF DF Horning | 5 | 0.496 | 0.496 |
| | Range Only Radar | 25 | 0.848 | 0.992 |
| | Autopilot System | 168.5 | 1.603 | 1.84 |
| | Inertial Navigation System | 207 | 1.717 | 3.443 |
| | Terrain Following Radar | 249 | 1.826 | 5.16 |
| | TACAN | 61 | 1.142 | 6.986 |
| | UHF Communications | 51 | 1.076 | 8.128 |
| | Head-up Display | 37 | 0.967 | 9.204 |
| | Gyrocompass | 8.4 | 0.590 | 10.171 |
| Right | Gun Camera | 2 | 0.365 | 0 |
| | Air Data Computer | 14 | 0.699 | 0.365 |
| | Flight Data Recorder | 15.6 | 0.725 | 1.064 |
| | Intercom System | 19.2 | 0.777 | 1.789 |
| | Attack Radar | 387.2 | 2.11 | 2.566 |
| | Radar Warning and Horning | 182 | 1.645 | 4.676 |
| | High Frequency Radio | 78.4 | 1.242 | 6.321 |
| | Air-To-Ground IFF | 53 | 1.090 | 7.563 |
| | Radar Altimeter | 38.2 | 0.977 | 8.653 |
| | ILS-VOR | 27 | 0.870 | 9.63 |
| Both | All Electronics | 1633.5 | N/A | 10.766 |

Table 9: Fuselage Sizing Summary of Relevant Parameters

| Section | Component | Location (From Nose to Front of Component) (ft) | Length (ft) |
|-------------|--|--|--------------|
| A | Engine | 3.002 | 6.00 |
| | Cockpit | 10.170 | 8.33 |
| | Fuel Tanks | 18.516 | 8.358 |
| | Entire Section | 0 | 22.695 |
| B | Electronics | 28.596 | 10.767 |
| | Gun and Ammunition | 25.449 | 5.508 |
| | Entire Section | 22.695 | 19.955 |
| A, B | Entire Fuselage | 0 | 42.51 |
| A, B | Optimal Fuselage (Ideal L/D Ratio of 8) | 0 | 42.64 |

Table 10: Initial Fuselage Sizing Additional Design Information

| Parameter | Value |
|---|------------------------|
| Fuselage Diameter | 5.331 ft |
| Fuselage length | 42.51 ft |
| Total Internal Component Weight | 8982.124 lb |
| Center of Gravity (Measured From Nose) | 18.024 ft |
| Variation In Center of Gravity (Maximum Fuel to Minimum Fuel) | +0.276 ft (Forward) |

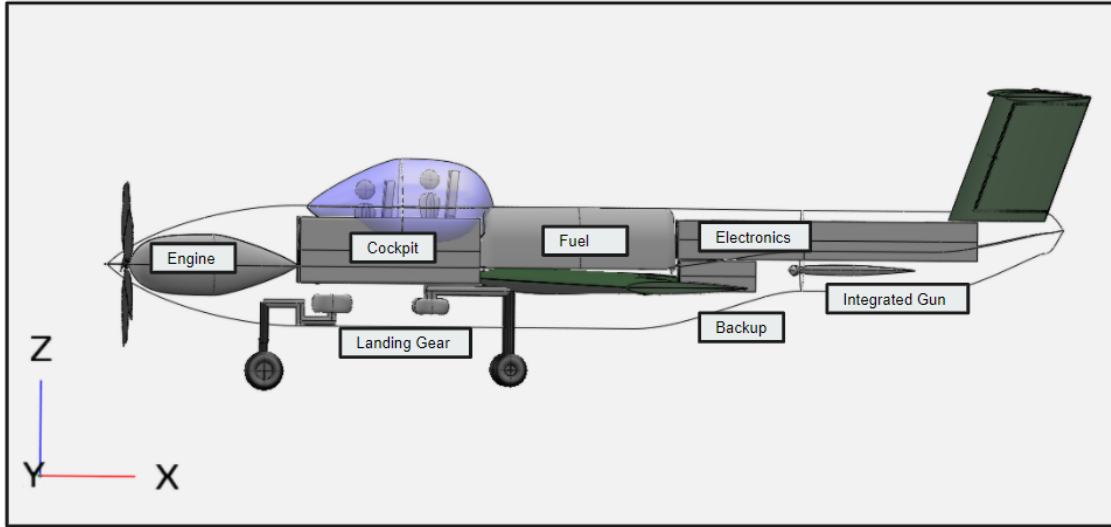


Figure 8: Cross-Section Side View of Final Aircraft In OpenVSP, Major Components Shown

Payload

Requirements:

- The aircraft payload has been designed to satisfy the following objectives: 3000 lbs of payload and armament
- Provisions for carrying/deploying a variety of weapons, including rail-launched missiles, rockets, and 500 lb (maximum) bombs
- Integrated gun for ground targets

Wing Mounted Armaments

To meet the payload requirements, team Hummingbird's design utilizes six wing hardpoints that will carry the following armament tabulated in Table 11:

Table 11: Default Armaments List

| Name | Weight | Description |
|----------------------|-----------------|--|
| Mark 82 Bomb | 2 x 500 lbs | Unguided, general purpose bomb |
| AGM-65K Maverick | 2 x 790 lbs | Air-to-ground, high penetration missile |
| AIM-9X Sidewinder | 2 x 190 lbs | Air-to-air missile |
| Total Payload | 2960 lbs | Not including side-mounted .50-cal cannon and ammunition |

The Mark 82 bomb is a general-purpose, unguided aircraft bomb that contains 89kg of explosives. The Mark 82 was developed in 1940 and remains in service today [19]. Including one of these bombs would satisfy the RFP. However, the team's design incorporates two in order to balance each other on the aircraft wings and provide more bombing capability. In addition to the pair of Mark 82 bombs, two AGM-65K Maverick missiles provide additional

ground-targeting capabilities. The Maverick family of missiles are a proven series of air-to-ground guided weapons that are valuable against a variety of targets, including air defences, ships, heavy armor, and ground equipment. These missiles are still being used by aircraft such as the A-10 and F-15E [20]. The team chose to use the K variant for two reasons. First, the K variant is a modern version of the AGM-65 that utilizes a heavyweight penetrator warhead, which will provide the aircraft with more offensive capability than using the standard H variant [21]. Second, the increased weight-per-missile allows the aircraft to carry fewer missiles to meet payload specification, which reduces drag and improves aerodynamic performance. Rounding out the default payload is a pair of AIM-9X Sidewinder air-to-air missiles. The AIM-9X is the latest variant of the popular Sidewinder missile platform, currently produced by Raytheon and utilized by aircraft such as the F/A-18, F-22, and the F-35 [22]. These missiles help the proposed design combat aerial threats to the aircraft or the friendly forces on the ground, such as attack jets and helicopters. The team considered other configurations that could be substituted with the default configuration, depending on the needs for a given mission. Aside from using other variants or combinations of the armaments from the default payload, such as using the AGM-65H or mounting more bombs instead of missiles, this design would also support the M134D-H, which is a self-contained mini-gun pod designed by Dillon Aero that weighs 350 lbs each [23]. Using these mini-gun pods, in addition to the side-mounted cannon, would increase the sustained damage output of the aircraft at the expense of powerful, one-time-use weaponry.

Internal Gun

In order to incorporate an internal gun, the team chose a side-mounted FN MP3 .50-caliber cannon in the rear of the plane. The FN MP3 is an 80.5lb machine gun that is designed for aircraft and advertises offensive and defensive power at nearly 1,850 meters [24]. Since the aircraft fuselage is relatively small and cramped in the nose, the team decided not to fit a forward-facing cannon, such as those seen in other attack aircraft like the A10 or A-29. Instead, the team drew inspiration from the AC130's side-mounted arsenal of 25mm, 40mm, and 105mm munitions [25]. A side-mounted cannon allows the Hummingbird to strafe the combat site and provide sustained coverage, while also allowing the aircraft to stow away the cannon when not in use to improve aerodynamics.

VIII Aerodynamics

VIII.1 About VSPAERO

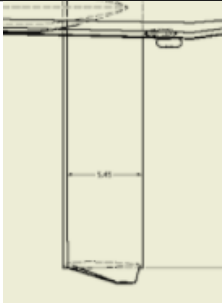


VSPAERO is a vortice lattice solver developed by David Kinney at NASA that is integrated with OpenVSP. VSPAERO applies discrete vortices to the degenerate geometry generated by OpenVSP in order to obtain a pressure distribution that can be used to find lift, drag, slip, and the (x,y,z) forces and moments. In addition to freestream and flight

conditions, the solver also includes capabilities to analyze propellers using actuator disks [26]. For the design process, the team initially utilized VSPAERO propeller analysis to determine aerodynamic parameters before transitioning to free stream conditions due to the improved run-time performance and ease of use.

VIII.2 Initial Wing Design

The Hummingbird's initial wing was rectangular in shape with an area of 218.5 ft² and a wingspan of 38 ft. For the airfoil, the NACA 63A-415 was chosen [27]. These starting values were developed based on comparisons made against other state-of-the-art attack aircraft, such as the A-10 and A-29, and were also utilized in the derivation of the take off gross weight. Both of these aircraft have straightforward, and rectangular wing designs. Table 12 contains comparisons between their dimensions and that of the initial wing. The team's initial design sizing is roughly between that of the A-10 and A-29, which is reflected by the design parameter comparison.

Table 12: Comparison of Wing Design Parameters Between Initial Design and State-of-the-Art

| | Initial Design | A-29 [28] | A-10 [29] |
|------------------------------|---|--|---|
| Wing Area (ft ²) | 218.5 | 209 | 506 |
| Wingspan (ft) | 38 | 37.17 | 57.5 |
| Airfoil | NACA 64A-415 | NACA 64A-415 | NACA 6716 |
| Fuselage Length | 43.74 | 37.47 | 53.25 |
| Wing Shape |  |  |  |

After implementing the initial design in OpenVSP, a basic analysis of the aerodynamic performance of the design was conducted using VSPAERO. Using propeller analysis, VSPAERO was run on angles of attack between 3° and 5° at a Mach of 0.3 in order to determine the peak L/D. A study of the wing design was done to see how the airfoil, sweep angle, taper ratio, and inclusion of winglets impacted L/D max. Figures 9 and 10 summarize the results of these propeller-based simulations.



Wing Analysis



| Config | Best AOA | Mach | L/D | CL | CD _o |
|-------------|----------|------|--------|-------|-----------------|
| NACA 64A415 | 4 | 0.3 | 12.189 | 0.499 | 0.0254 |
| NACA 64A215 | 5 | 0.3 | 11.728 | 0.534 | 0.0262 |
| NACA 2412 | 4 | 0.3 | 11.964 | 0.461 | 0.0252 |
| NACA 2415 | 4 | 0.3 | 11.848 | 0.461 | 0.0257 |
| 10° Sweep | 4 | 0.3 | 12.222 | 0.497 | 0.0260 |
| 20° Sweep | 5 | 0.3 | 12.337 | 0.573 | 0.0265 |
| 0.8 Taper | 4 | 0.3 | 11.834 | 0.506 | 0.0252 |
| 0.6 Taper | 5 | 0.3 | 11.501 | 0.635 | 0.0268 |

Figure 9: Comparison of L/D by Configuration of the Initial Wing Design

| Config | AOA | Mach | L/D | CL | CD _o |
|-------------|-----|------|--------|-------|-----------------|
| Winglets | 4 | 0.3 | 12.189 | 0.499 | 0.0254 |
| No Winglets | 4 | 0.3 | 11.629 | 0.474 | 0.0250 |

Figure 10: L/D Comparison With and Without Winglets

As shown in the chart, the straight-winged design with the NACA 64A415 had a higher L/D than designs with no winglets, other airfoils, and taper ratios less than one. However, increasing the sweep angle led to a minor increase in L/D over the default design. Ultimately, the team chose to use the rectangular wing with no sweep for the initial design, since the wings would be simpler to manufacture and lighter, which would improve the cost of the aircraft without majorly impacting L/D.

VIII.3 Increasing L/D

A significant increase in the fidelity of the initial model's fuselage and armament caused a large increase in the total drag of the aircraft, resulting in a need to increase the L/D produced by the wing design. In order to achieve this, a constant aspect ratio of 6.6 and introduced a sweep of 15° was maintained and increased the area and wingspan to 278.85 ft² and 42.9 ft, respectively. A taper ratio of 0.7 was introduced to reduce the weight and improve structural

stability of the larger wing. Finally, an incidence of 5° was added to the wing, which helped reduce the angle of max L/D that had increased due to the previous changes. Figure 11 shows the wing design with these modifications.

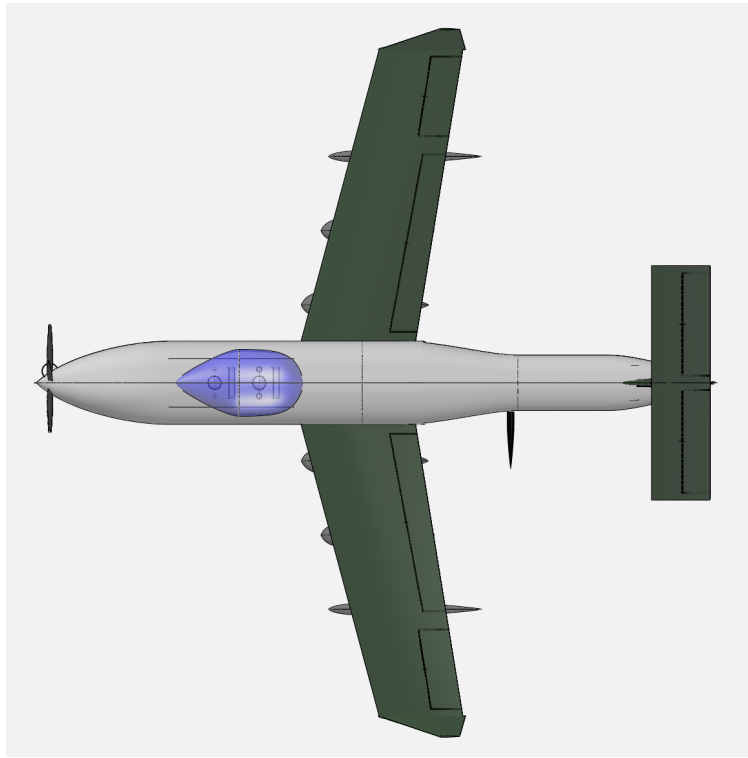


Figure 11: Revised Wing Design Including: Area Increase, Sweep, Taper, and Incidence

In order to perform a more in-depth aerodynamic analysis using VSPAERO, the decided to use a free stream analysis. This allowed the simulation to automatically iterate between a wide range of angles of attack without having to restart the simulation for each one, which was a limitation of the propeller analysis. The simulation was conducted between -6° to 20° angle of attack (AoA). This resulted in a max L/D of 11.7 at an AoA of 6° , which was in line with what was necessary based on the performance analysis. The plots of L/D, CL, and $CD_{0, i, tot}$ are shown in Figure 12.

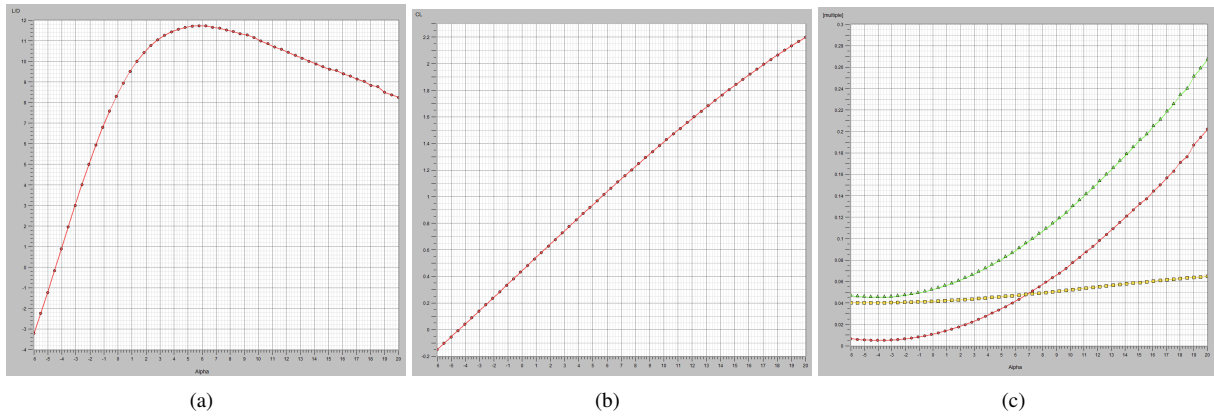


Figure 12: For -6° to 20° AoA: (a) L/D (b) C_L (c) C_D

Increasing Stability The free stream VSPAERO analysis revealed a pitch moment CM_y less than 0 for zero angle of attack, which is a sign of vertical instability. In order to combat the instability and increase CM_y to be positive at a zero-angle of attack, a dihedral angle of 2° was introduced. The empennage sizing was also increased (see Empennage section for more details).

VIII.4 Final Wing Aerodynamics

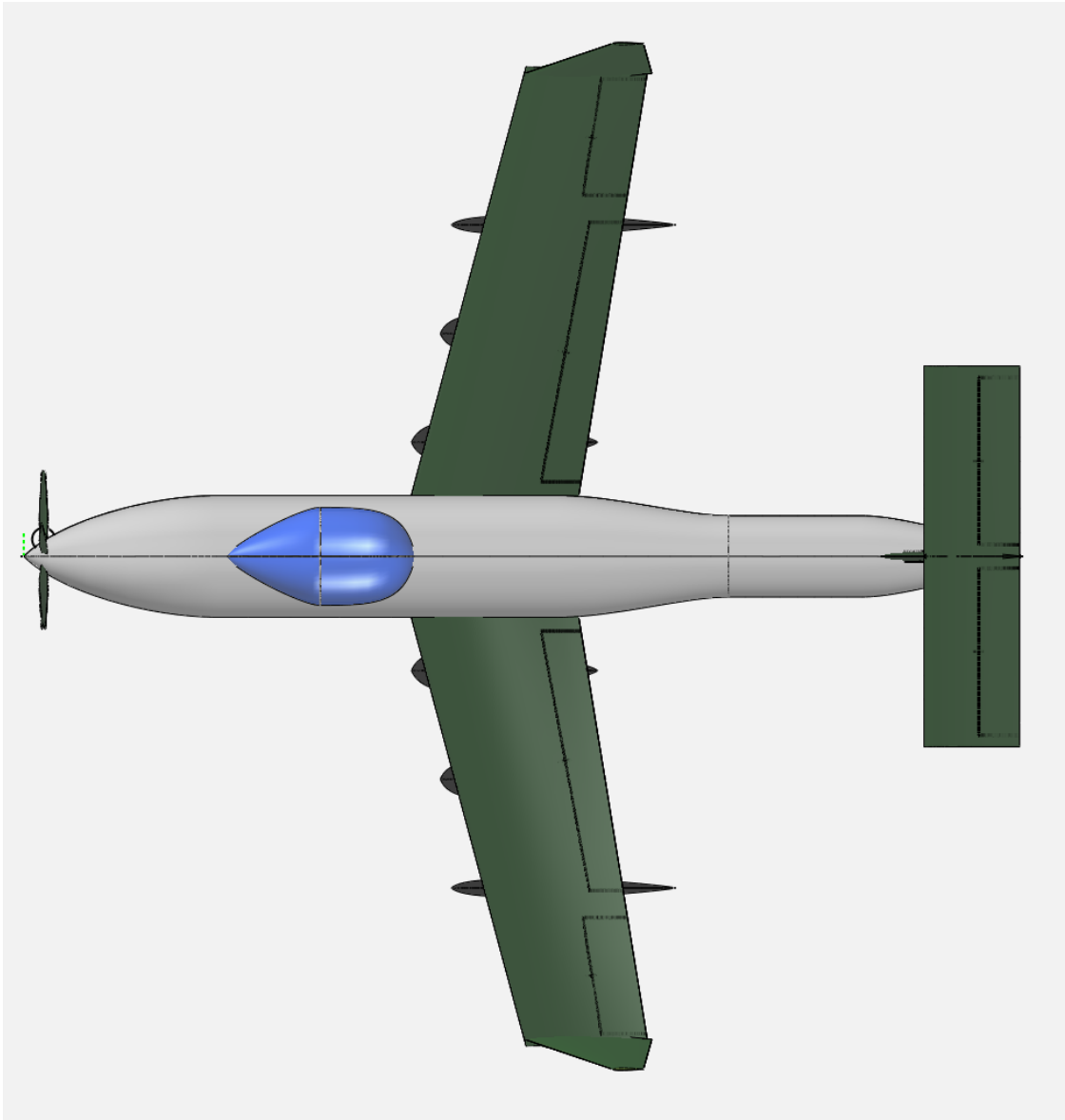


Figure 13: Final Wing Design

Figure 13 depicts the final wing design, including the resized empennage and slight dihedral angle. With this design, a detailed VSPAERO simulation was run from -6° to 20° with half-degree precision. The plots resemble those of Figure 12 from the previous simulation, with a final L/D Max of 11.3 at 6° and a positive pitch moment for zero angle of attack. Additionally, a stability analysis was performed using the built-in methods in VSPAERO, which produced an approximate summary of stability derivatives and moments as depicted in Figure 14. The team is confident that this wing design provides the aircraft with the means to affordably and effectively control the skies.

```

=====
# Name      Value      Units
Sref_      278.8500000  Lunit^2
Cref_      6.5000000    Lunit
Bref_      42.9000000    Lunit
Xcg_       20.0712290    Lunit
Ycg_       -0.0000000    Lunit
Zcg_       0.4014530    Lunit
Mach_      0.3000000    no_unit
AoA_       0.0000000    deg
Beta_      0.0000000    deg
Rho_       0.0023770    Munit/Lunit^3
Vinf_      293.8000000  Lunit/Tunit
Roll_Rate  0.0000000    rad/Tunit
Pitch_Rate 0.0000000    rad/Tunit
Yaw__Rate  0.0000000    rad/Tunit

#
# Case      Delta      Units      CFx      CFy      CFz      Cfx      Cfy      CPz      CL      CD      CS      CMl      CMn      Cpn
#
Base_Aero  +0.000  n/a      0.0041607 -0.0000000 0.3906736 0.0000000 0.0232089 -0.0000000 0.3906736 0.0085212 -0.0000000 -0.0000000 0.0232089 0.0000000
Alpha      +1.000  deg      -0.0029705 -0.0000001 0.4823283 -0.0000000 -0.0080625 0.0000000 0.4823066 0.0111994 -0.0000001 0.0000000 -0.0080625 -0.0000000
Beta       +1.000  deg      0.0035901 -0.0081431 0.3905413 0.0033719 0.0224971 -0.0012227 0.3905413 0.0085895 -0.0080792 -0.0033719 0.0224971 0.0012227
Roll_Rate  +1.000  rad/Tunit -0.0019584 -0.0184115 0.3870835 0.0440463 0.0224864 -0.0003263 0.3870835 0.0102495 -0.0184115 -0.0440463 0.0238864 0.0003263
Pitch_Rate +1.000  rad/Tunit 0.0025701 0.0000001 0.5246223 0.0000000 -0.2082294 -0.0000000 0.5246223 0.0123512 0.0000001 -0.0000000 -0.2082294 0.0000000
Yaw__Rate +1.000  rad/Tunit -0.0022255 0.0370359 0.3908150 -0.0117059 0.0216540 0.0155451 0.3908150 0.0093549 0.0370359 0.0117059 0.0216540 -0.0155451
Mach       +0.100  no_unit 0.0042600 -0.0000000 0.4013810 0.0000000 0.0245141 -0.0000000 0.4013810 0.0089456 -0.0000000 -0.0000000 0.0245141 0.0000000
#
#
# Base      Derivative:
# Aero      wrt      wrt      wrt      wrt      wrt      wrt      wrt
# Coef      Total      Alpha Beta      p      q      r      Mach      U
#           -      per      per      per      per      per      per      per
#           -      rad      rad      rad      rad      rad      M      u
#
CFx  0.0041607 -0.4085862 -0.0326910 -0.0838120 -0.1437822 -0.0874704 0.0009937 0.0002981
CFy -0.0000000 -0.0000044 -0.4665649 -0.2521809 0.0000132 0.5072801 -0.0000001 -0.0000000
CFz 0.3906736 5.2514266 -0.0075819 -0.0491741 12.1089643 0.0019474 0.1070740 0.0321222
Cfx 0.0000000 -0.0000004 0.1931931 0.6033008 -0.0000000 -0.1503360 -0.0000000 -0.0000000
Cfy 0.0232089 -1.7917241 -0.0407831 0.0092796 -20.9220248 -0.0212980 0.0130519 0.0039156
CPz -0.0000000 0.0000013 -0.0700567 -0.0044699 -0.0000013 0.2125210 -0.0000000 -0.0000000
CL 0.3906736 5.2501880 -0.0075819 -0.0491741 12.1089643 0.0019474 0.1070740 0.0321222
CD 0.0085212 0.1534466 0.0039140 0.0236724 0.3462320 0.0114185 0.0042439 0.0012732
CS -0.0000000 -0.0000044 -0.4629039 -0.2521809 0.0000132 0.5072801 -0.0000001 -0.0000000
CMl -0.0000000 0.0000004 -0.1931931 -0.6033008 0.0000000 0.1503360 0.0000000 0.0000000
CMn 0.0232089 -1.7917241 -0.0407831 0.0092796 -20.9220248 -0.0212980 0.0130519 0.0039156
Cpn 0.0000000 -0.0000013 0.0700567 0.0044699 0.0000013 -0.2125210 0.0000000 0.0000000
#
#
# Result      Value      Units
SM            0.3412686 no_unit
X_np         22.2894747 Lunit
#
#

```

Figure 14: Output of the VSPAERO Stability Analysis

IX Stability and Control

IX.1 Tail Sizing

The tail of the aircraft was sized according to Nicolai's method [3]. The moment arms of the vertical and horizontal tail were sized in three different iterations. The first, and very rough, iteration utilized center of gravity estimates from the A-29 calculated by taking relative measurements from a photograph of the aircraft. It was assumed that the center of gravity of the aircraft was near the back middle of the wing. This resulted in an lvt of 19.2 ft and an lvt of 18.9 ft. The second iteration utilized the calculated center of gravity of the aircraft and assumed the tail to be at the end of the fuselage, resulting in moment arms of lvt = 18.8 ft and lvt = 19.2 ft. Finally, Raymer's method of equating lvt and lvt to 55% of the total fuselage length was used [30]. This created significantly longer moment arms of 23.5 ft.

To continue Nicolai's tail sizing method, tail volume coefficients were selected from the Beechcraft T-34 single turboprop military trainer aircraft: 0.048 for vertical and 0.76 for horizontal. This led to calculations of the area of the vertical and horizontal tails. With an earlier fuselage length of 33.7 ft, the areas were 21 ft² and 50 ft², respectively. These increased to 23 ft² and 57 ft² with a 44.3 ft fuselage. With a final fuselage length of 43 ft, the longitudinal stability was low, so the horizontal tail volume coefficient was increased to 0.9 (which will be discussed further in this

section), resulting in a vertical tail area of 24.5 ft² and horizontal tail area of 69.5 ft².

The T-tail design was chosen during an early trade study on this aircraft with information from Raymer's textbook. This orientation was chosen to increase the effectiveness and efficiency of the horizontal tail, as well as allowing it to be clear of wing wake and propwash. While the T-tail is not suitable for all flight regimes, the cruise and maximum Mach values were expected to be well within the suitable regime for the T-tail design.

Aspect ratios, taper, and sweep for the vertical and horizontal tails were selected by literature from the Indian Institute of Technology [31]. Lower aspect ratios for the tail than the wing were selected in order to reduce structural weight. Professor Tulapurkara details that minimizing drag from the tail should not be prioritized above reduction of structural weight, and thus cost, so a lower aspect ratio is preferable. The aspect ratios for the vertical and horizontal tails selected are, respectively, 1.3 and 4. Taper ratio was selected based on minimizing cost, so both tails used a value of 1. Finally, the sweep of the vertical tail was selected as 20° in order to increase the moment arms of the vertical and horizontal (via the T-tail design) tails for a low/medium Mach number. The sweep of the horizontal tail was left at 0° as an increased sweep to decrease drag divergence Mach was unnecessary at the expected low/medium Mach flight.

IX.2 Static Margin, Stability Derivatives, Control Surfaces

The static margin was calculated using Nicolai's equations. A plot of coefficient of lift over angle of attack was first created of just the wing and tail separately. This resulted in a static margin of around 6% of the chord length, calculated via Nicolai's method [3]. VSPAero was then used to run aerodynamics analyses of the entire aircraft. After one failed attempt that resulted in a negative margin, success was found when analyzing the updated aircraft design including pods. With an aircraft center of gravity at around 16 ft, the static margin was quite small at only 3%, again using Nicolai's method. The center of gravity was then pushed further back to around 18 ft, which resulted in a suitable static margin of just under 5%. This value changed one final time with a change in horizontal tail size, resulting in a static margin of 4.2%. This value was in the range that was expected to provide stability for the aircraft.

Directional stability was calculated at the same time as the static margin using VSPAero. Between the two subsequent analyses, the change in coefficient of pitching moment with respect to angle of attack, $C_{m,\alpha}$ moved more negative, thus showing a greater restoring moment and greater stability. However, both of these trials struggled as the coefficient of pitching moment at zero lift, $C_{m,0}$ was a negative value, meaning that there was no positive trim condition for the aircraft, which was unacceptable to the aircraft [32]. This instability was corrected by increasing the horizontal tail volume coefficient from 0.76 to 0.9. This succeeded in pushing the $C_{m,0}$ value positive to 0.035. This also resulted in a new C_m , value of $-0.03 / ^\circ$ ($-1.8 / \text{rad}$), the negative value of which signals stability. Therefore, the positive $C_{m,0}$ and negative C_m , show directional stability of the plane.

Lateral stability was calculated using VSPAero simulations. Unfortunately, $C_{l,\beta}$ the derivative of the rolling moment

coefficient with respect to side-slip angle β was found to be positive for the aircraft with a value of $0.003 / ^\circ$ ($0.19 / \text{rad}$). Lateral stability is achieved with a negative $C_{l,\beta}$. While the limited time did not permit a correction this stability concern, this would be a primary focus of future work. This lateral instability could be solved by increasing the sweep of the wings; the current wings are swept only 15° , so there is room to increase this sweep in future improvement [33]. Another option would be to utilize a greater dihedral angle than our current 2° in our wings to increase lateral stability. The current $C_{l,\beta}$, of $0.003 / ^\circ$ is a small enough value that the team is confident that this could be moved negative with only a few alterations to the design.

Directional stability was also calculated using VSPAero simulations. $C_{n,\beta}$ the derivative of the yawing moment coefficient with respect to side-slip angle, was calculated as $-0.0012 / ^\circ$ ($-0.07 / \text{rad}$). This number is negative, yet should be positive for directional stability. This value could be moved positive by increasing the size of the vertical tail, just as the horizontal tail area was increased to create a positive $C_{m,0}$ for longitudinal stability. This could also be improved by increasing the sweep of the wings. A larger sweep angle could improve both the directional and lateral stability, so would be a first step in future work on this design. The $C_{n,\beta}$ of $-0.0012 / ^\circ$ has a very small magnitude far smaller than that of the beneficial longitudinal stability derivative, so the team is again confident that directional stability could be achieved with only minimal alterations this design.

The three control surfaces were sized using Roskam's method of area ratios[11]. The Embraer EMB-312 Tucano was selected as a similar aircraft driven by a single propeller. The ailerons were sized to be 10% of the area of the wing, resulting in an aileron area of 28 ft^2 . The elevator was 44% of the horizontal tail area, resulting in an elevator area of around 31 ft^2 . Finally, the rudder was a significantly large 70% of the vertical tail area, resulting in a rudder area of 17 ft^2 .

An additional control surface was added via a trim tab at the aft of the elevator on the horizontal tail. This was sized as 5% of the elevator area, resulting in a trim tab area of 1.5 ft^2 . This will be a pilot-controlled variable trim tab to allow for changed settings during take off, cruise, and landing.

X Propulsion

X.1 Engine Selection Process

The first objective for the propulsion development portion of the project was to select a usable engine platform for the aircraft. The mission requirements and weight/size descriptions of the aircraft in the RFP were the main criteria used in initially selecting an engine type and class. Given the constraint of only allowing engine technologies that are able to be realistically developed by 2025, initial research was done on both engine design and potential future improvements.

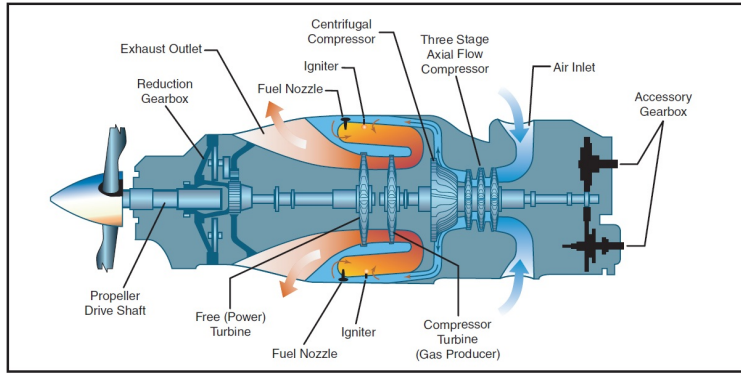


Figure 15: Reverse Flow Turboprop [1]

First, for military aircraft, a common method of propulsion is a turboprop engine. Turboprop engines have an inlet, compressor, burner, and turbine which work to provide shaft power to the propeller in order to produce thrust. Thrust is created both from the propellers, which converts the mechanical shaft power supplied by the turbine to usable thrust, and the engine core, which is a function of the freestream and exit velocities as well as the air and fuel mass flow rates through the engine. For a turboprop specifically, figures of merit exist to quantify the engine's efficiency such as the propeller efficiency and the gearbox efficiency. More generally, turboprop engines also have thermal efficiency η_{th} and propulsive efficiency η_p , which quantify the thermodynamic losses in the engine. For a turboprop, the attractiveness of the engine lies in its high propulsive efficiency [2]. Turboprop engines are found in a variety of both military and civilian aircraft. For military use, both the EMB-314/A-29 Super Tucano and the AT-6 Wolverine are powered by turboprop engines, two aircraft studied as capable existing options for the design mission.

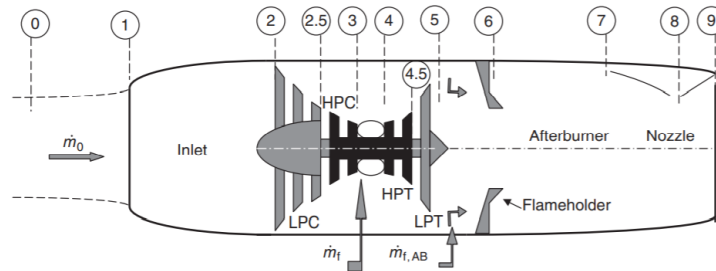


Figure 16: Afterburning Turbojet Engine Schematic [2]

Another usable engine type is the turbojet, shown above. Looking at modern military aircraft, one can see that turbojets have been steadily phased out of use in modern aircraft design, being replaced almost entirely by either turbofan engines (discussed later) or lower-speed turboprop engines where appropriate. While turbojets do still see use as missile propulsion systems, it cannot be overlooked how by most modern standards for aircraft, there is no satisfactory justification to run a turbojet engine over a comparable turbofan, with the lone exception perhaps being the

relative simplicity of a turbojet relative to a turbofan.

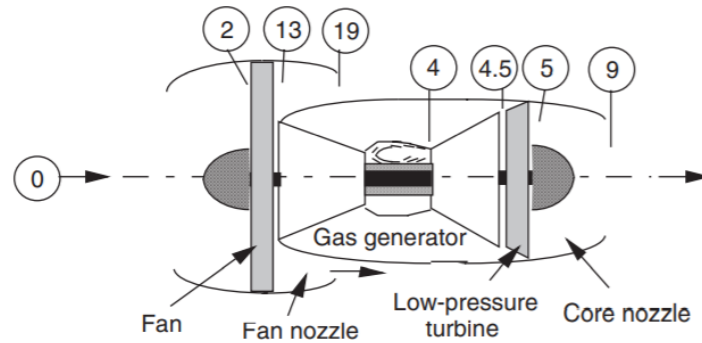


Figure 17: Separate Exhaust Turbofan Engine Schematic [2]

Another type of aircraft engine commonly employed today, both in civilian and military applications, is the turbofan engine. As the name implies, the turbofan was developed to reach a middle ground between the previous two engine types, combining the higher efficiencies of turboprop engines with the superior thrust capabilities of turbojet engines. A turbofan is broadly divided into two sections: the core and the fan. The core of a turbofan is functionally identical to a standard turbojet engine. Free stream air enters the engine through an inlet, passing through a compressor or series of compressors before mixing with fuel and igniting in the combustion chamber. This heated air then passes through the turbine, which extracts power to run the compressor and fan, before being expelled out a nozzle to propel the aircraft forward. Where a turbofan differs from a turbojet centers on the fan at the front of the engine. As shown below, not all the air that passes through the fan enters the core of the engine, and some of it bypasses the engine core entirely. For the air that does not enter the core, the engine functions similarly to a standard propeller. This air is slightly accelerated as it passes through the fan, and generates additional thrust alongside the core turbojet component. This improved thrust output comes with nearly no significant additional fuel flow rate relative to the core alone, making a turbofan more fuel efficient than a traditional turbojet [1].

As this high-level breakdown has demonstrated, each classification of propulsion system brings its own advantages and downsides to the discussion. One must consider a variety of factors, including thrust output, weight, fuel efficiency, and flight envelope, in order to make an informed decision on what general propulsion system is the most ideal for a given aircraft design. From the RFP, it is known that the light attack aircraft must have a service ceiling of at least 30,000 feet, while in preliminary analysis, it was derived that the aircraft should have a cruise capability of about 300 knots at a minimum. Plotting these two parameters on the graph below, it is clear that a piston-prop engine does not meet these minimum requirements. Therefore, the best starting point for a propulsion system is likely a turboprop engine of some variety. While a turbofan may also be worth investigating, there is valid concern that most turbofans may be overqualified for the given mission requirements. There can be such a thing as having too much thrust, and

when looking at cost-effectiveness and overall weight gains from using a turbofan over a turboprop, the trade-offs for integrating a turbofan propulsion system begin to become concerning relative to the gains.

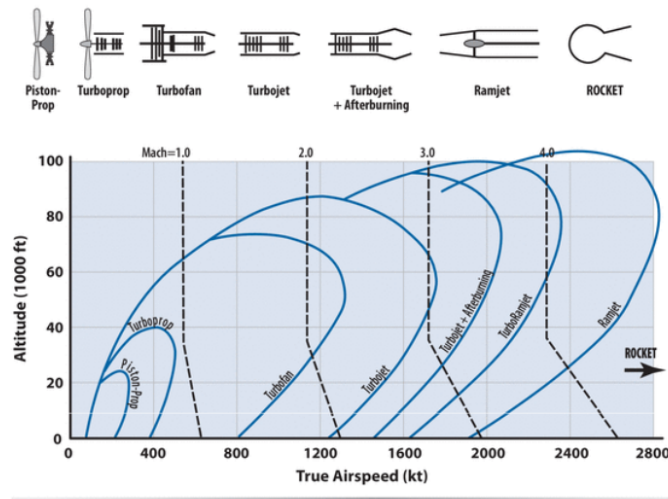


Figure 4.6 Level-flight propulsion options.

Figure 18: Service Regions for Various Propulsion Systems [3]

Given the requirements of the RFP, and the nature of the austere light attack aircraft being designed, the conclusion of the initial research is that a turboprop engine is the most appropriate form of propulsion. A turbofan could also be applicable, but the added weight is a major drawback as well as, given the low flight speeds of the attack aircraft's intended usage, the increased power may be unnecessary.

X.2 About GasTurb

GasTurb is a gas turbine cycle analysis software that can allow users to create gas turbine engine designs and off-designs, calculate engine's fluid dynamic parameters and performance, run parametric and Monte Carlo simulations for design optimization, model engine deterioration over time, do detailed stress analysis, and model engine performance(s) for various flight envelopes. The tool was designed by career aerospace performance engineer Dr. Joachim Kurzke with the company GasTurb GmbH in Aachen, Germany. GasTurb 13 was used to help determine and optimize engine performance using known flight envelopes and engine size/weight estimations. The objective of GasTurb 13 analysis for this project was to obtain engine performance and efficiency values, predict fuel consumption, and get basic engine cycle information (such as temperature and pressure) to base future improvements on. The model was initially developed using known flight regimes, mainly altitude and Mach number in combination with estimated engine parameters, such as size, burner temperatures, and spool speed.

X.3 Establishing an Initial Model

Since the Embraer EMB 314 Super Tucano, a comparable aircraft to The Hummingbird, is powered using the Pratt Whitney PT-6A-68C turboprop engine, that was the engine used as a baseline for the analysis. The first objective of the GasTurb modelling was to recreate this engine design digitally. The Pratt Whitney PT-6A engine manual had a large quantity of data regarding engine performance, design, and operating conditions. The relevant data used in creating the PT-6A model will be given in successive paragraphs. The most significant information found in the engine manual was a total temperature and total pressure by engine stage graph. It should be noted that the Pratt Whitney graph shown below identifies engine stages at different numbers than the GasTurb program output. The following table was created to clarify this issue.

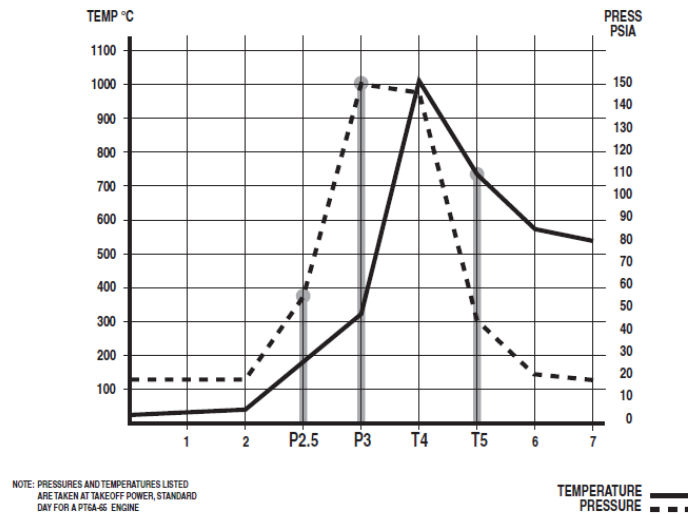


Figure 19: PT-6A Total Temperatures and Pressures By Engine Stage [4]

Table 13: Approximated Temperature & Pressure Values For PT-6A

| # | Stage | Temperature (R) | Pressure (Psi) |
|---|------------------|-----------------|----------------|
| 0 | Ambient | 529.67 | 15 |
| 1 | Air intake | 539.67 | 15 |
| 2 | Compressor inlet | 581.67 | 15 |
| 3 | Compressor exit | 1059.67 | 150 |
| 4 | Turbine inlet | 2289.67 | 145 |
| 5 | Interturbine | 1839.67 | 40 |
| 6 | Turbine exit | 1526.67 | 20 |
| 7 | Exhaust | 1009.67 | 15 |

Some additional minor information was also extracted from the text. First, compressor compressor ratios were found, they were seen to vary by engine model and would range from 9.0 to 12.1, this value was extremely relevant to GasTurb as it is one of the main inputs. Tip-clearance of compressor and turbine blades was also given explicitly

as 0.010” which will be needed for more complex modelling. Propeller and spool rotational speeds were also given, 1500 RPM and 39000 RPM respectively. Lastly, a parametric graph relating air temperature, pressure, and engine performance was given which is an additional resource on which to base the initial engine design. Ultimately, this document, the flight mission profile (mainly the aircraft speed and maximum/minimum altitude(s)), and knowledge of thermodynamic engine cycles was used to develop the first PT-6A GasTurb model [34].

X.4 Current PT6-A GasTurb 13 Model

GasTurb Analysis Inputs

The GasTurb model utilized to model the PT-6A was the “1-spool Turboprop” with the following properties shown in Table 14.

Table 14: PT-6A GasTurb Additional Input Values

| Property: | Units: | Value: | Comments: |
|--|--------|---------|--|
| Ambient Conditions: Total Temperature | R | 518.67 | Default, sea level conditions |
| Ambient Conditions: Total Pressure | Psia | 14.6959 | Default, sea level conditions |
| Ambient Conditions: Ambient Pressure | Psia | 14.6959 | Default, sea level conditions |
| Ambient Conditions: Relative Humidity [%] | N/A | 0 | Default |
| Compressor Efficiency: Polytropic Efficiency | N/A | 0.9 | Within reasonable value |
| Compressor Design: Nominal Spool Speed | RPM | 39000 | Given from PT-6A manual |
| Turbine Efficiency: Polytropic Efficiency | N/A | 0.9 | Within reasonable values |
| Heat Exchanger (Method 1): Heat exchanger Design Eff. | N/A | 0.85 | Within reasonable values, adjusted to return correct outputs |
| Heat Exchanger (Method 1): Heat exchanger Design P35/P3 | N/A | 0.975 | Within reasonable values, adjusted to return correct outputs |
| Heat Exchanger (Method 1): Heat exchanger Design P7/P6 | N/A | 0.75 | Within reasonable values, adjusted to return correct outputs |

Since the propeller design was being completed simultaneously and separately in XROTOR, estimates as to the propeller dimensions are given as follows in Table 15

Table 15: PT-6A Turboprop Propeller Inputs

| Property: | Unit: | Value: | Comments: |
|-----------------------------|-------|---------|---|
| Propeller Diameter | ft | 6.06955 | From initial analysis (see Propeller Design section) |
| Propeller Design RPM | N/A | 1500 | Given from PT-6A manual |
| Propeller Efficiency | N/A | 0.875 | Within reasonable values |
| Static Propeller Efficiency | N/A | 0.875 | Within reasonable values |

GasTurb Analysis Outputs

The PT-6A GasTurb model was validated by comparing the outputs to the known Pratt Whitney PT-6A engine manual recorded values. The output engine temperatures were all within 20% of the manual values. The recorded power output of the engine was 850 hp, the GasTurb model was 846.5 hp. The turboprop outputs are given in the table below, the additional thrust output is included and was estimated to be 3000 lbs for the actual PT-6A.

Table 16: Current Turboprop Model Flow Rates, Total Temperature, and Total Pressure

| Station | Mass Flow Rate (lb/s) | Total Temperature (R) | Total Pressure (Psia) |
|---------|-----------------------|-----------------------|-----------------------|
| 0 | N/A | 518.67 | 14.696 |
| 1 | 25 | 518.67 | 14.696 |
| 2 | 25 | 518.67 | 14.696 |
| 3 | 25 | 1082.26 | 154.307 |
| 3.5 | 25 | 1624.86 | 150.450 |
| 4 | 25.277 | 2291.67 | 145.440 |
| 5 | 25.277 | 1719.55 | 39.189 |
| 6 | 25.277 | 1719.55 | 19.595 |
| 7 | 25.277 | 1197.84 | 14.696 |

Table 17: PT-6A GasTurb Turboprop Model Relevant Outputs

| Parameter | Value |
|--------------------|------------------|
| Thrust | 2836.76 lb |
| Power Generated | 846.5 hp |
| Fuel Flow | 0.27738 lb/s |
| PSFC | 1.1796 lb/(hp*h) |
| TSFC | 0.3520 lb/(lb*h) |
| Thermal Efficiency | 0.1158 % |

GasTurb Thermodynamic Cycle Graphs

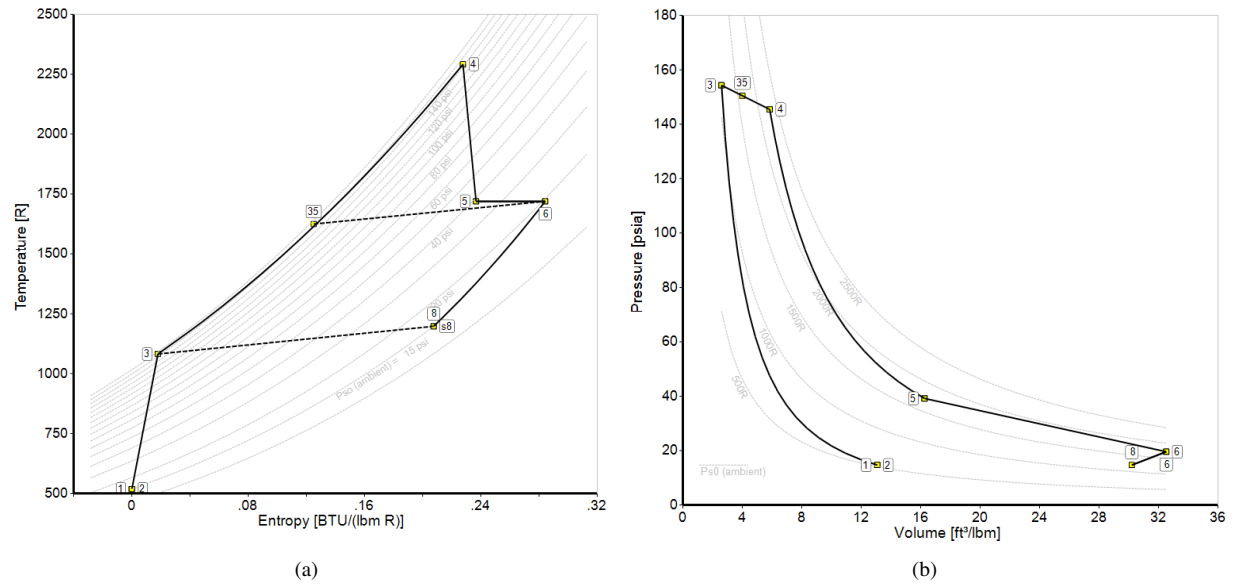


Figure 20: T-S and P-V Diagrams

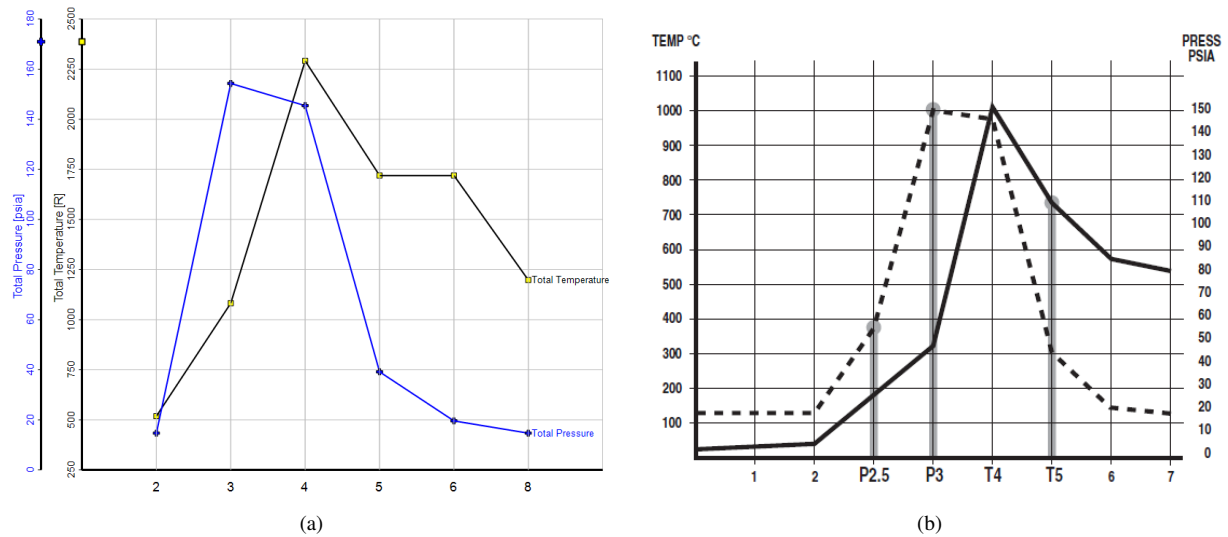


Figure 21: Total Temperature and Pressure Graphs By Engine Stage
(Left: GasTurb, Right: PT-6A manual)

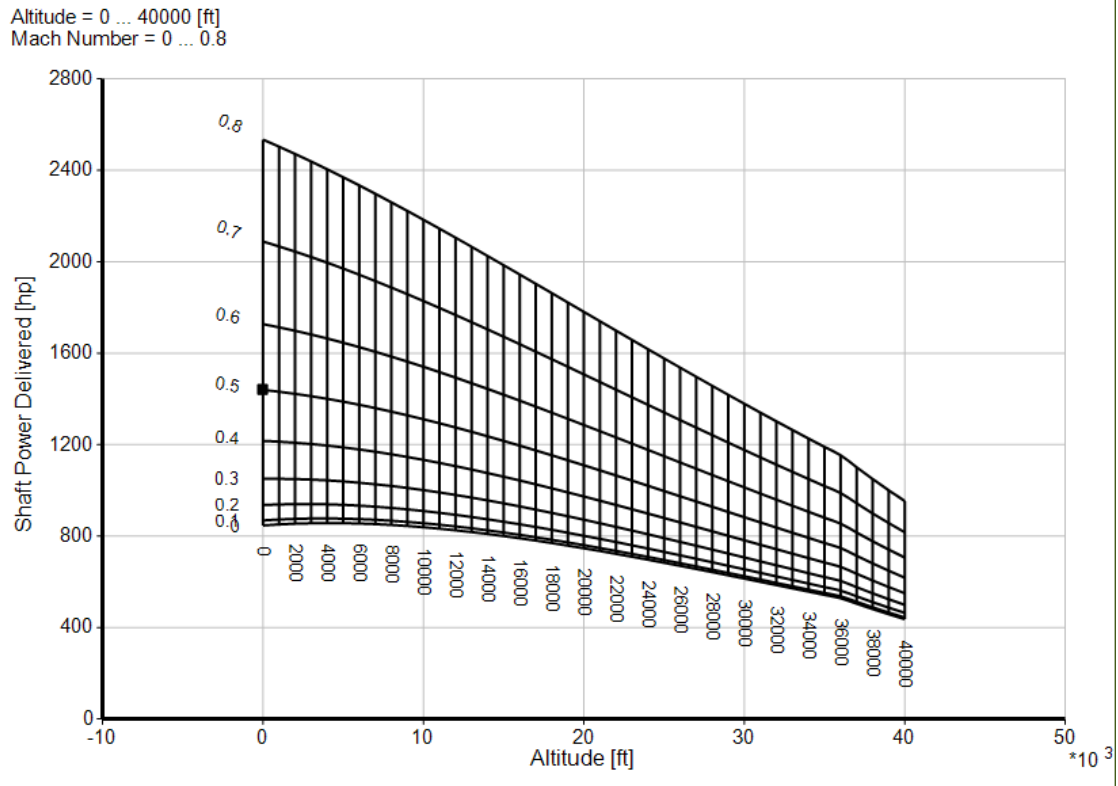


Figure 22: Parametric Analysis, Altitude and Mach Number vs. Shaft Power Delivered

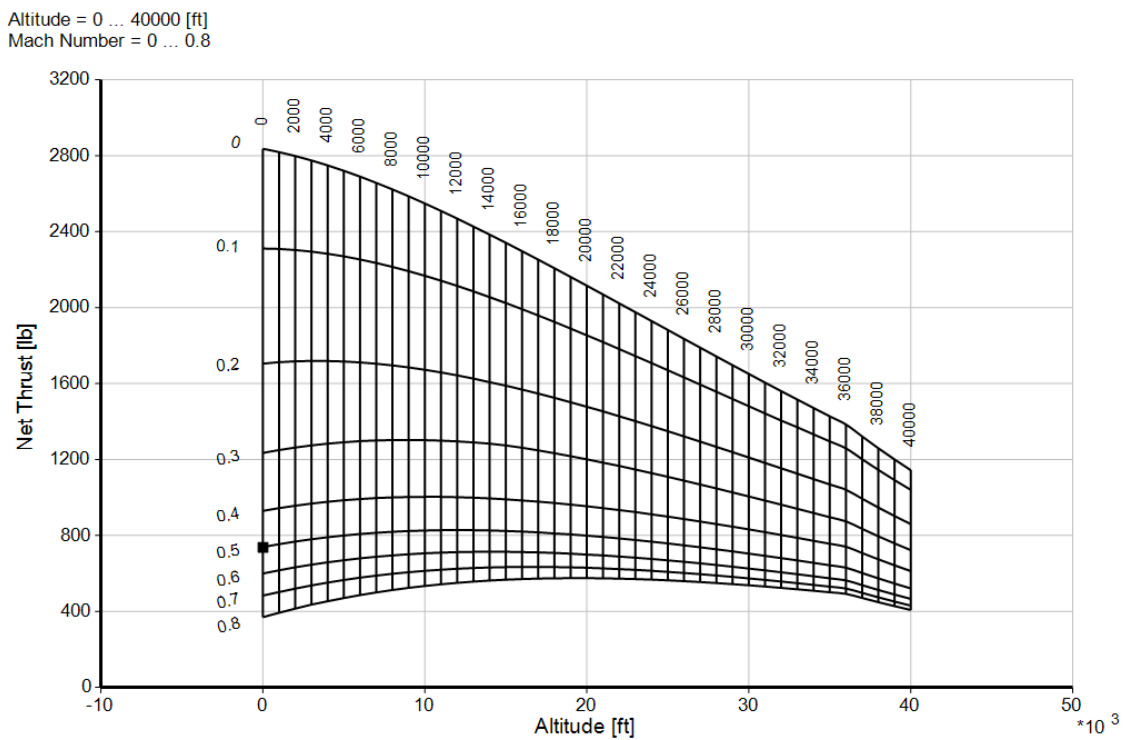


Figure 23: Parametric Analysis, Altitude and Mach Number vs. Net Thrust

Altitude = 0 ... 40000 [ft]
Mach Number = 0 ... 0.8

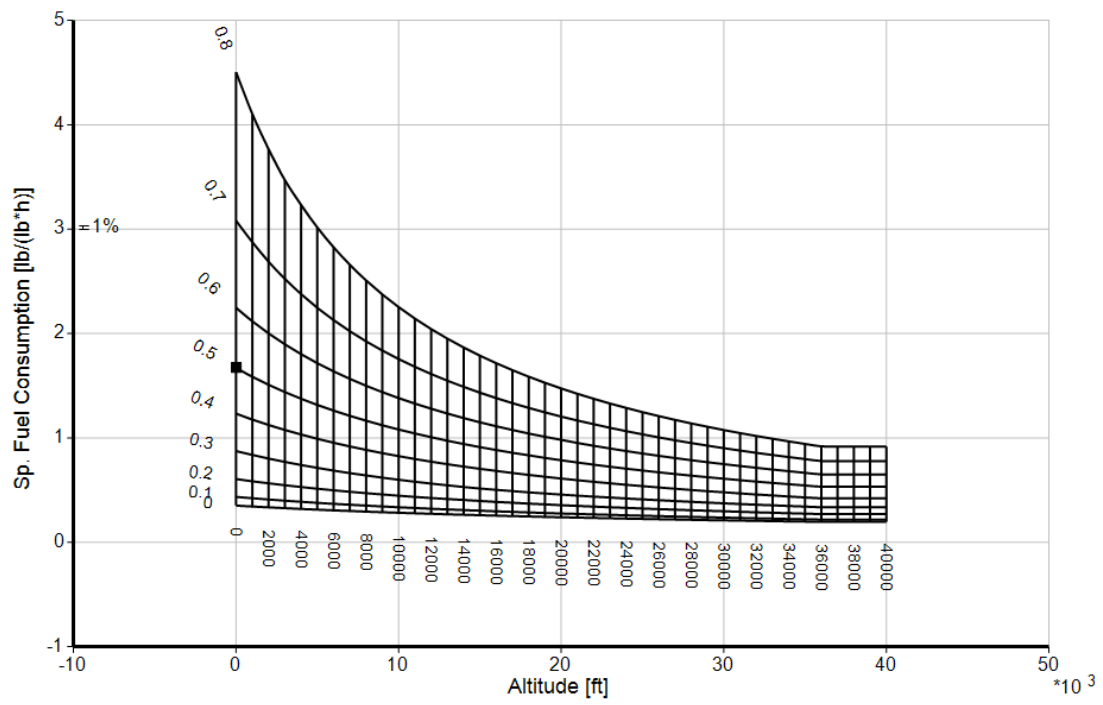


Figure 24: Parametric Analysis, Altitude and Mach Number vs. Thrust Specific Fuel Consumption

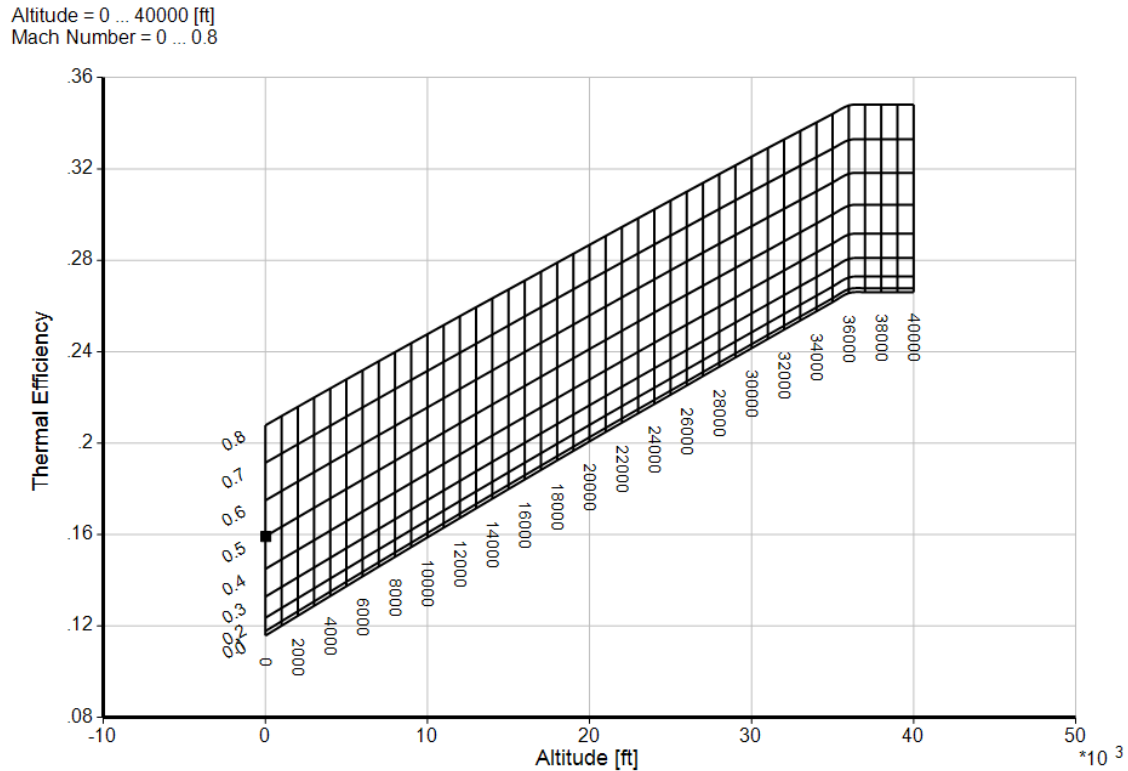


Figure 25: Parametric Analysis, Altitude and Mach Number vs. Thermal Efficiency

2025 Improved PT6-A GasTurb Model

The RFP calls for a propulsion system that is optimized for 2025 engine technology. The likelihood of a new light attack aircraft engine being developed in that time is low, so it was decided to perform this analysis by using an existing engine platform and modifying it to include better technology, which will be discussed below. The engine outputs for this engine will be used in the aircraft performance discussion.

Proposed Improvements and Expected Gains

Engine cycles are limited by several bounds: the maximum stored energy in the fuel, the maximum service temperatures of engine materials and the maximum strengths of components in the system. The maximum performance of any gas turbine cycle will occur when the kinetic energy output by the system is equivalent to the total energy of the fuel input to the system, this of course is never truly possible as there are always extreme losses to entropy, escaping heat, acoustics, and many other inefficiencies. The best engine designers can do is to modify and improve engine cycles to remove or limit these factors. For most modern engines, the Brayton cycle, consisting of (1) isentropic compression (compressor), (2) isobaric heat addition (combustion), (3) isentropic expansion (turbine), and (4) isobaric heat rejection (exhaust), is the basis upon which engine performance is based.

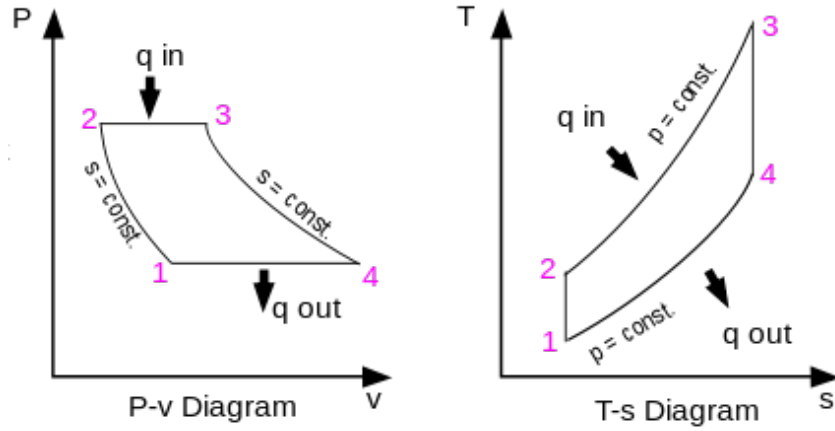


Figure 26: The Brayton Cycle [5]

Most importantly, the Brayton efficiency (%), shown below, gives a good indication of which engine components to improve to best reduce engine losses and increase performance [35, 5]:

$$\eta_T = 1 - (T_4 - T_1)/(T_3 - T_2) \quad (5)$$

Equation 5 suggests the following: when improving cycle efficiency it is beneficial to have a lower exhaust temperature, T_4 , a higher engine inlet temperature, T_1 , a higher turbine temperature, T_3 , and a lower combustion inlet temperature. In the case of a gas turbine system, these correlate to the following objectives: (1) obtaining the highest possible compression prior to combustion, (2) increasing the temperature of the fluid as much as possible during combustion, and (3) removing as much energy from the fluid as possible in the turbine section of the engine.



Figure 27: An additively manufactured component's complex cross section [6]

Within the foreseeable future, the main potential improvement to increase the strength of centrifugal compressors is the addition of additive manufacturing (AM). AM has already had a massive impact on the aerospace industry, from

rocket fuel injectors to aircraft turbines, companies like General Electric have made huge strides to both increase the quality and affordability of additively manufactured engine components, a market predicted to continue growing into the near future [6]. The benefits of using AM compressor stages include: the mechanical properties of the material can be tailored to best manage the axial forces as well as allow tighter tolerances to reduce spillage internally (for shrouded impellers), again increasing the performance of the system. Additively manufactured IN-738 nickel superalloy test coupons created via selective laser melting (SLM) were shown to have significant anisotropy in their material properties, which when optimized, can result in up to 25% increases in strength [36]. This can be directly related to increases in spool speeds and angular velocities of compressor components. Unfortunately, AM does not come without its drawbacks—mainly quality assurance and cost. Due to their complexity, it is very difficult to assess the created microstructures and geometry of AM components. Some methods require destructive testing of simultaneously printed components or advanced x-ray techniques, both of which are expensive. It is important to analyze the microstructure, as AM parts are very sensitive to unmelted particle/powder inclusions and porosity, as shown below [37]. Though it is presumed for the sake of analysis that these problems can be overcome by 2025 and direct improvements to compression ratio, spool speed, and compressor efficiency will be added.

Another practical means of increasing the temperature gain of the engine is to incorporate AM fuel injectors. As described earlier, AM has the opportunity to build very precise and complex components. In the case of the fuel injection system, AM could allow for designs that better particulate the injected fuel, increasing the exposed surface area of fuel particles and allowing closer to stoichiometric fuel combustion to occur. This would lead to greater efficiencies and slightly increased engine temperatures. General Electric predicts that not only are AM fuel injectors 25% lighter, but can improve fuel efficiency by upwards of 5% [6]. An AM fuel injection system will be included and can be seen in the increased burner efficiency parameter.

Another potential solution to increasing the pressure at the combustion chamber inlet is to utilize a water mist or coolant before the fluid enters the compressor stage of the aircraft. The justification for this is that the coolant/mist will cool the air, reducing its volume and increase the mass flow rate into the compressor. This system presents the practical questions of how to cool the water and where the water would come from. It may be practical in this case to utilize a hydrogen fuel cell auxiliary power unit, whose byproducts of functioning are electricity and water. These systems in CFD simulations have been shown to increase takeoff and landing performance as well as generally improve aircraft efficiency (as predicted by the Brayton efficiency) [38]. Lastly, these systems will expose the engine to additional moisture so internal corrosion and oxidation may be accelerated, especially in unprotected metals and aft end ceramics. Ultimately, in a trade study analysis, it was found that these systems were expensive, reduced survivability, and could lead to unforeseen damage to the aircraft and engine, the team did decide not to implement these changes in the engine platform.

A consideration for engine improvements was the use of electric aircraft propulsion. Aside from the obvious

reason of cleaner energy, the major benefit of electric propulsion would be decreasing the noise profile of the engine. Given that stealth is a concern and the aircraft is expected to loiter for 45 minutes in what could be a dangerous area, decreasing the noise profile in this manner could be invaluable. Additionally, there is some evidence that EAP could improve certain performance metrics of the aircraft. The main drawback of EAP is that it is a relatively new field compared to the other forms of propulsion explored before, especially for military aircraft. Currently, the foremost electric aircraft in the field is the X-57 Maxwell. The X-57 uses Lithium-Ion batteries to power its 14 electric motors which are distributed throughout the wing. Due to the capacity of the batteries, the plane may only be used for relatively short flights, which is a deterrent given the RFP's longer ferry mission [34].

The next major objective for engine designers is to increase the temperature gain during combustion, this is typically limited by the melting temperature of the materials used in the combustion chamber and high pressure turbine sections afterwards. This is achieved through use of high service temperature materials or by applying thermal barrier coatings (TBC's) (with high melting and sintering temperatures) to lower exposure temperatures of structural materials (usually metals). In the case of high temperature engine materials, metals typically have optimal strengths but lack in high melting temperatures, while ceramics, in this application, are very desirable, as they have extremely high melting temperatures (though for some, high temperature phase transformations may present a concern) and high strengths (though brittle failure is more probable with ceramic materials). This class of materials is classified as ultra-high temperature ceramics, and research is currently focused on borides, carbides, and nitride materials. The application of TBC's onto the structural metals in the engine will similarly allow higher operation temperatures. These materials will be technologically feasible by 2025 and will be worth the additional expenses, they will be implemented to the improved engine.

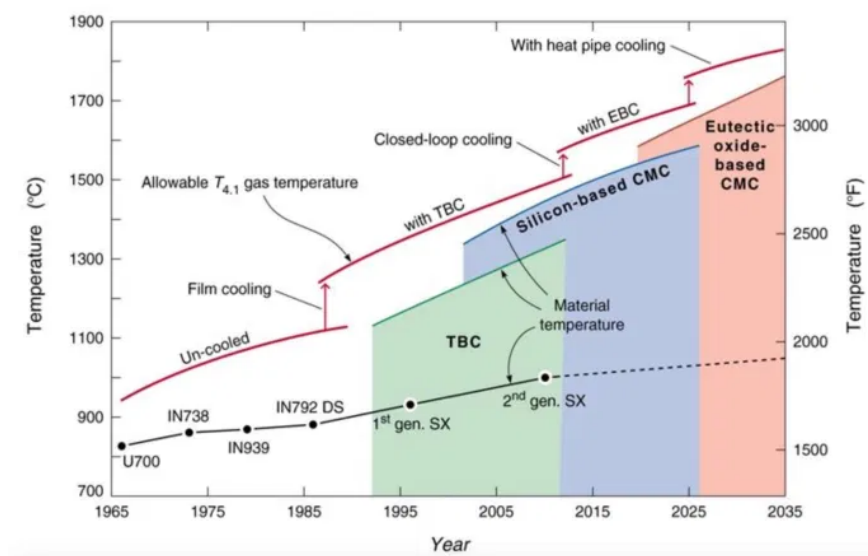


Figure 28: Expected Future Engine Temperature Capabilities [7]

Lastly, some other considerations for improving the combustion process include use of alternative fuels and ignition methods. One future technology that may be implemented is that of hydrogen fuel. Though impractical for this application as it can be volatile, has a low density, and would require cryogenic storage conditions, hydrogen fuel is enticing because it has a much higher energy density than traditional hydrocarbon jet fuels and could result in greater thrust outputs and efficiencies, as well as potentially mitigating environmental concerns (for general aviation) [39]. Additionally, to further justify improvements to the selected turboprop engine, more general aviation trends were used. These trends were based on the justification that computational fluid dynamics and improvements in engine materials will make future engines capable of producing more thrust, less aerodynamic drag, and having less weight. It was predicted that an improvement in thermodynamic efficiencies by 0.4% per year until 2025, resulting in a total 2% increase in thermodynamic efficiency, predominantly shown in compressor and turbine sections of the engine, this claim is substantiated from implementation of previously mentioned use of AM components, stronger materials, and TBC coatings. Similarly, the propulsive efficiencies showed a 0.3% increase per year and a 1.2% increase by 2025. The same source was also able to predict an improvement of 6% in regards to engine cooling losses, again enabled by AM, specifically losses from oversized air-cooling channels in turbine blades (prevalent in modern engines) could be removed by 2025. The same source suggests that future combustion efficiencies can be as high as 99% by 2025, converting almost all of the energy from the aircraft's fuel source into usable energy [8].

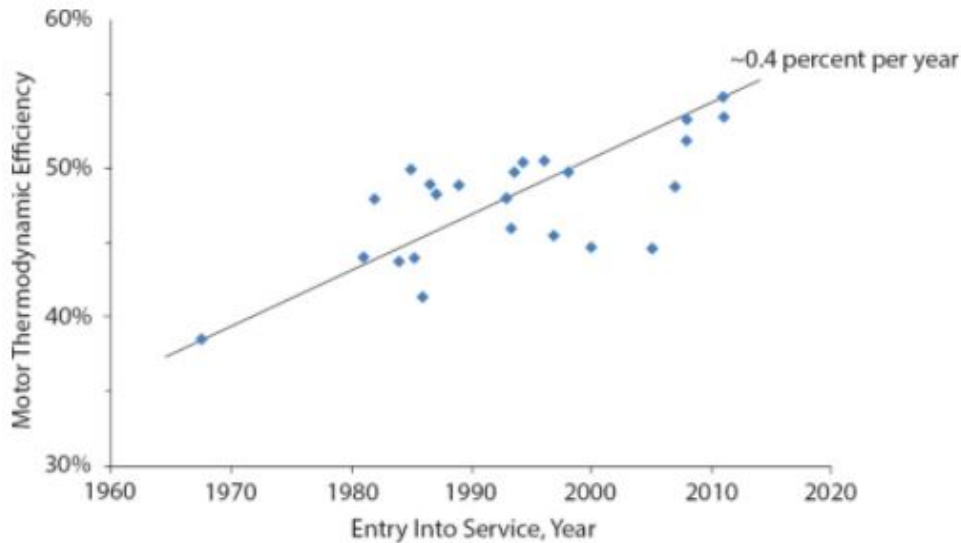


Figure 29: A Trend of Engine Thermodynamic Efficiency Improvements By Year [8]

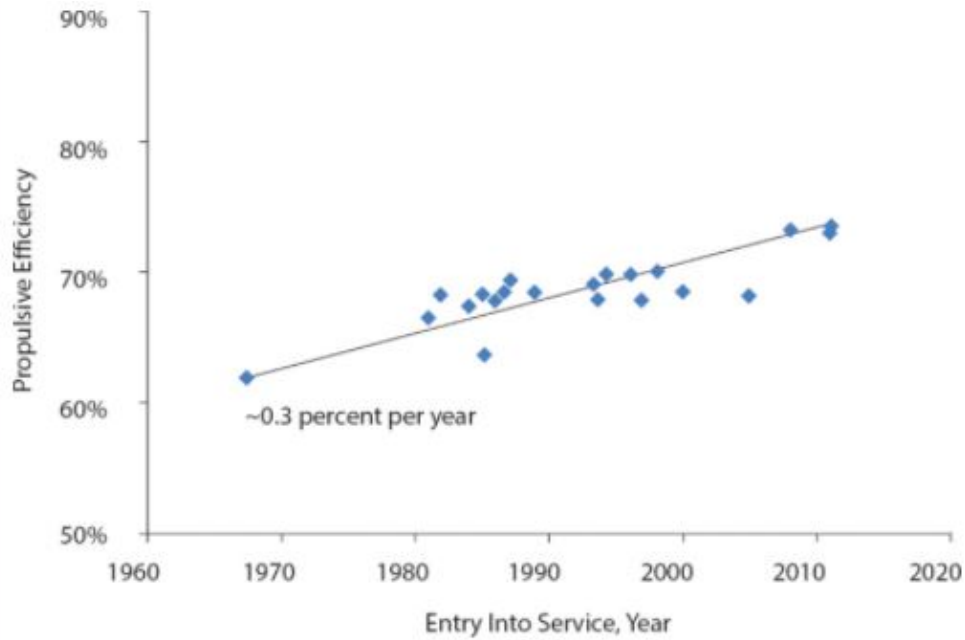


Figure 30: A Trend of Engine Propulsive Efficiency Improvements By Year [8]

A summary of engine improvements is given as follows:

- Burner exit temperature increased to 2405 R, an increase of about 5% from previous engine model [7].
- Burner efficiency increased to 99%, an increase of 5% from previous engine model [8].
- Heat exchanger efficiency increase of 6% from previous engine model [8].
- Combustor and turbine efficiency increase of 1.5% from previous engine model [8].
- Spool speed and propeller RPM up 2% account for improvements in shape and strength of materials [36].

GasTurb Analysis Inputs

The improved GasTurb model utilized the “1-spool Turboprop” with the following properties:

Table 18: Improved PT-6A GasTurb “Basic Data” Input Values

| Property | Units | Value | Comments |
|----------------------------------|--------|--------|--|
| Inlet Corr. Flow W2Rstd | lb/s | 25 | Recommended value (5-25 lb/s) |
| Inlet Pressure Ratio | N/A | 1 | Default |
| Pressure Ratio | N/A | 10.5 | Within the researched range (9-12.1), adjusted to return correct outputs |
| Burner Exit Temperature | R | 2405 | Improved parameter |
| Burner Design Efficiency | N/A | 0.99 | Improved parameter |
| Burner Part Load Constant | N/A | 1.6 | Default, not used in analysis |
| Fuel Heating Value | BTU/lb | 18638 | Given value for JP-4 jet fuel |
| Overboard Bleed | lb/s | 0 | Default, used to reduce mass flow in combustor, not needed |
| Mechanical Efficiency | N/A | 0.99 | Within reasonable values |
| Burner Pressure Ratio | N/A | 0.9667 | Within reasonable values, adjusted to return correct outputs |
| Turbine Exit Duct Pressure Ratio | N/A | 0.5 | Within reasonable values, adjusted to return correct outputs |
| Exhaust Pressure Ratio P8/Patm | N/A | 1 | Within reasonable values, adjusted to return correct outputs |

The “Secondary Air System” values were left as default and did not contribute to the analysis. “Cooling with Steam” and “Propeller Map” options were not utilized in this analysis. The remaining inputs are documented in the following table:

Table 19: PT-6A GasTurb Additional Input Values

| Property | Units | Value | Comments |
|--|-------|---------|---|
| Ambient Conditions: Total Temperature | R | 518.67 | Default, sea level conditions |
| Ambient Conditions: Total Pressure | Psia | 14.6959 | Default, sea level conditions |
| Ambient Conditions: Ambient Pressure | Psia | 14.6959 | Default, sea level conditions |
| Ambient Conditions: Relative Humidity [%] | N/A | 0 | Default |
| Compressor Efficiency: Polytropic Efficiency | N/A | 0.915 | Improved parameter |
| Compressor Design: Nominal Spool Speed | RPM | 39780 | Improved parameter |
| Turbine Efficiency: Polytropic Efficiency | N/A | 0.915 | Improved parameter |
| Heat Exchanger (Method 1): Heat exchanger Design Eff. | N/A | 0.79 | Improved parameter |
| Heat Exchanger (Method 1): Heat exchanger Design P35/P3 | N/A | 0.975 | Within reasonable values, adjusted to return correct outputs |
| Heat Exchanger (Method 1): Heat exchanger Design P7/P6 | N/A | 0.75 | Within reasonable values, adjusted to return correct outputs |
| Propeller Inputs: Propeller Diameter | ft | 6.06955 | From initial analysis (see Propeller Design section) |
| Propeller Inputs: Propeller Design RPM | N/A | 1530 | Improved parameter |
| Propeller Inputs: Propeller Efficiency | N/A | 0.875 | Within reasonable values |
| Propeller Inputs: Static Propeller Efficiency | N/A | 0.875 | Within reasonable values |

GasTurb Analysis Output

The improved model was validated by the researched improvement percentages, specifically the thermal efficiency: the models thermal efficiency improved by 3.19% from the original model, which is inline with the predicted 2% increase in efficiency that was researched. The model was able to show a 28% increase in thrust and a 45% improvement in shaft power delivered. Further, GasTurb predicted a slightly higher mass flow rate through the engine (increased by 15%) and fuel flow rate (14% improvement). The results are as follows:

Table 20: Improved Turboprop Model Flow Rates, Total Temperature, and Total Pressure

| Station | Mass Flow Rate (lb/s) | Total Temperature (R) | Total Pressure (Psia) |
|---------|-----------------------|-----------------------|-----------------------|
| 0 | N/A | 518.67 | 14.696 |
| 1 | 28.940 | 518.67 | 14.696 |
| 2 | 28.940 | 518.67 | 17.433 |
| 3 | 28.940 | 1082.26 | 154.307 |
| 3.5 | 28.940 | 1651.69 | 150.450 |
| 4 | 29.306 | 2405.00 | 145.440 |
| 5 | 29.306 | 1801.75 | 39.189 |
| 6 | 29.306 | 1801.75 | 19.595 |
| 7 | 29.306 | 1259.91 | 14.696 |

Table 21: Improved PT-6A GasTurb Turboshaft Model Relevant Outputs

| Parameter | Value |
|--------------------|-------------------|
| Thrust | 3645.52 lb |
| Power Generated | 1233.3 hp |
| Fuel Flow | 0.31694 lb/s |
| PSFC | 0.92516 lb/(hp*h) |
| TSFC | 0.3130 lb/(hp*h) |
| Thermal Efficiency | 0.1477% |

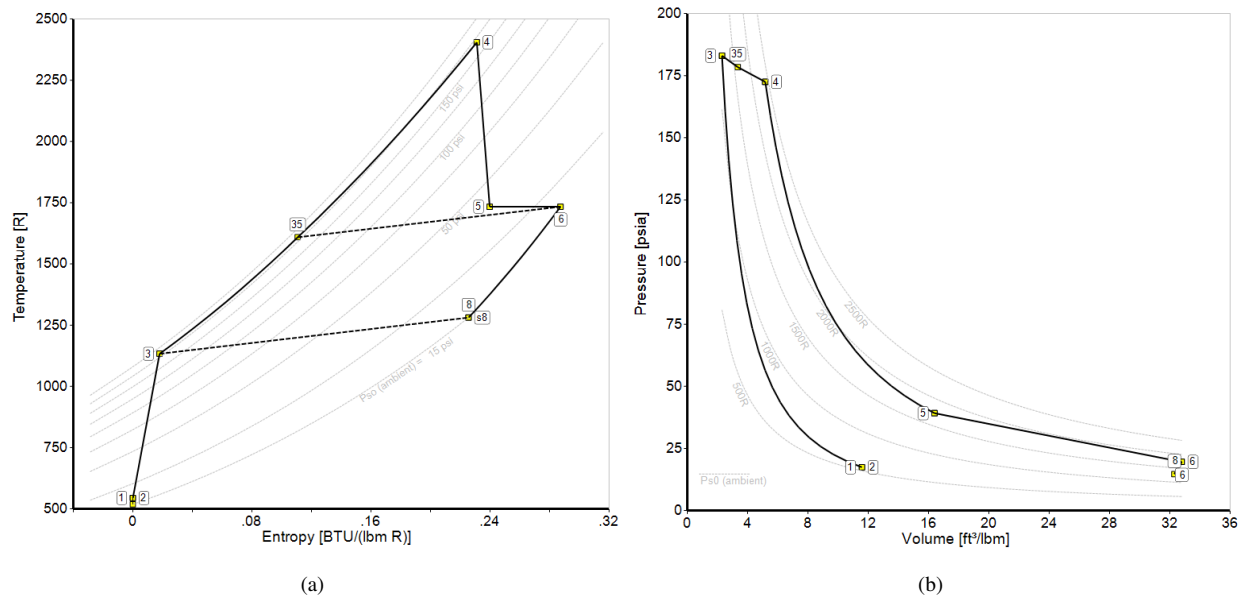


Figure 31: T-S and P-V Diagrams

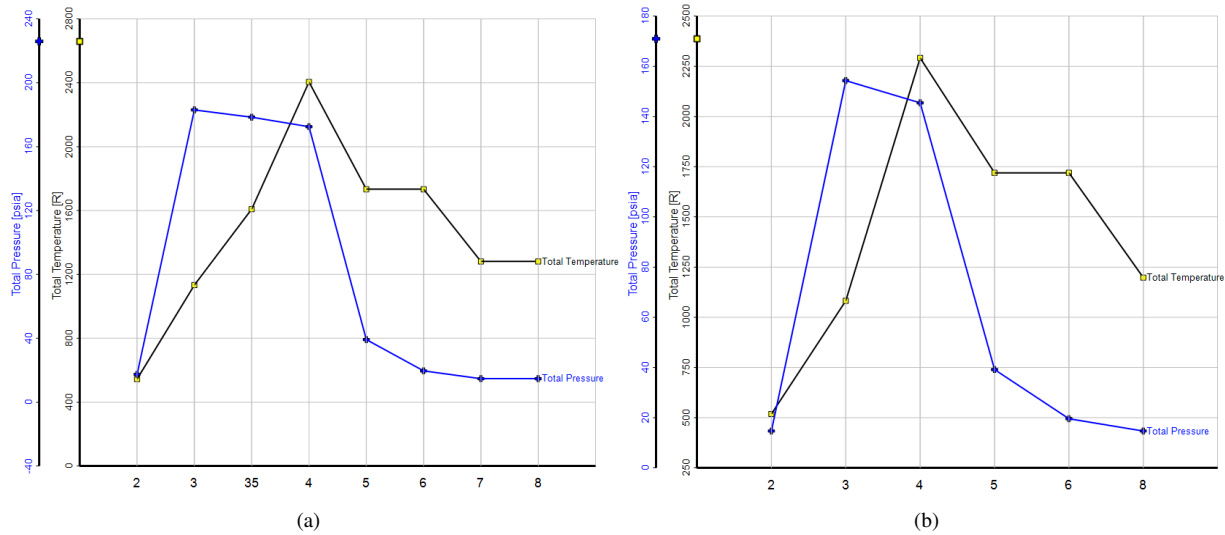


Figure 32: Total Temperature and Pressure Graphs By Engine Stage
(Left: 2025 PT-6A GasTurb, Right: Current PT-6A GasTurb)

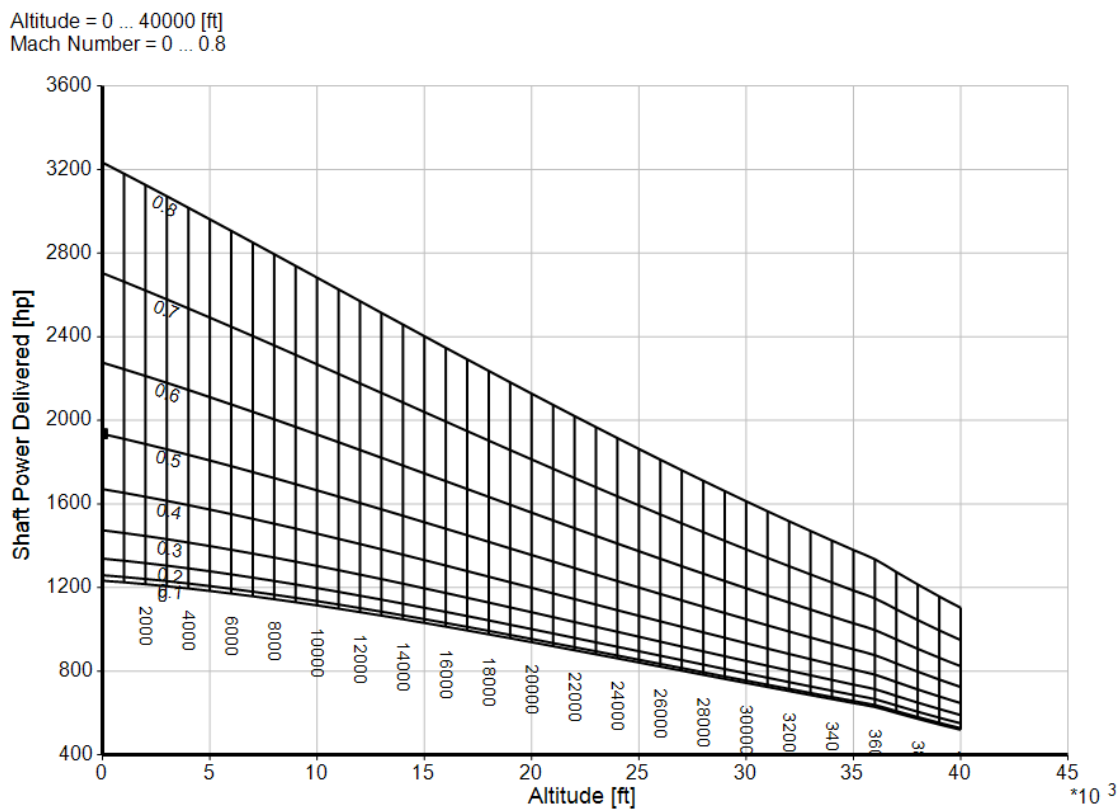


Figure 33: Parametric Analysis, Altitude and Mach Number vs. Shaft Power Delivered

Altitude = 0 ... 40000 [ft]
Mach Number = 0 ... 0.8

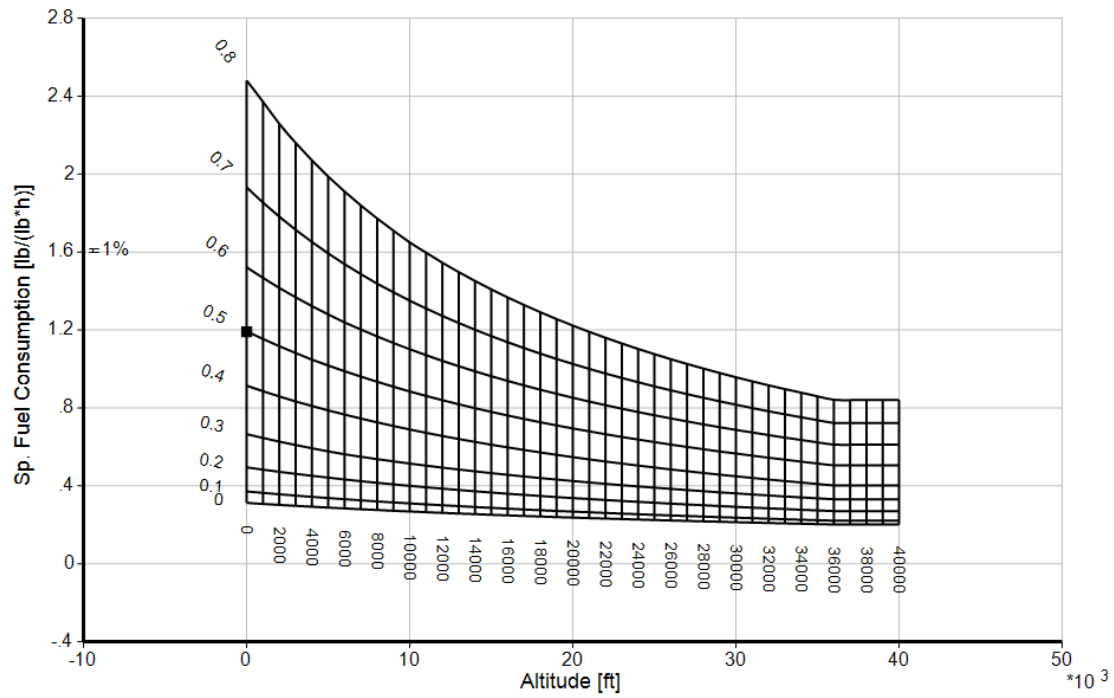


Figure 34: Parametric Analysis, Altitude and Mach Number vs. Net Thrust

Altitude = 0 ... 40000 [ft]
Mach Number = 0 ... 0.8

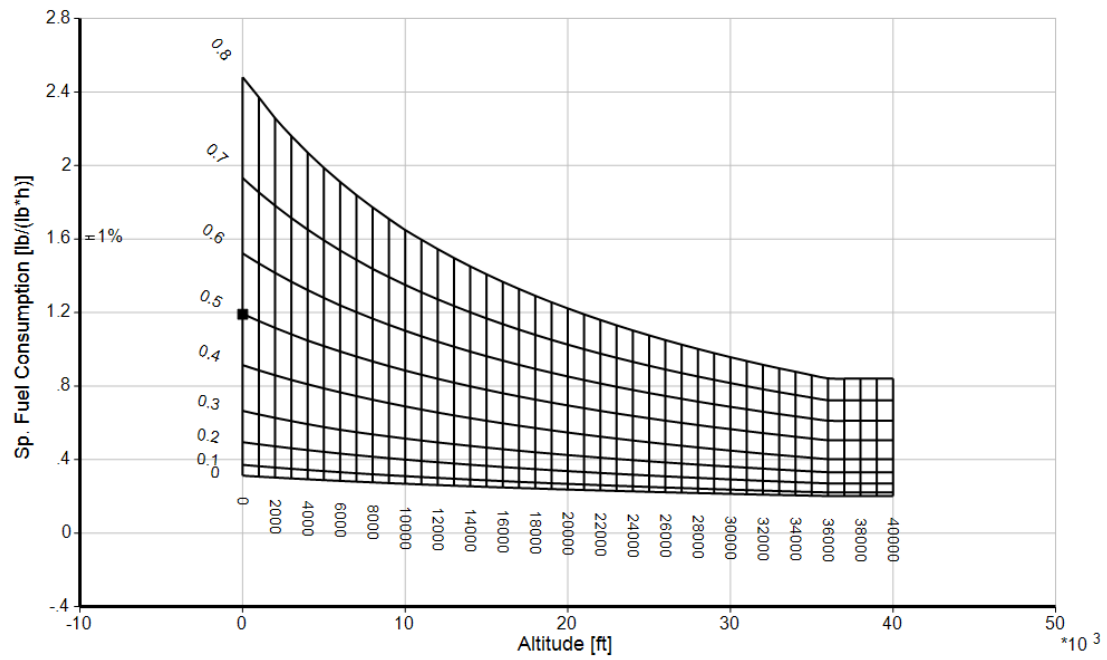


Figure 35: Parametric Analysis, Altitude and Mach Number vs. Thrust Specific Fuel Consumption

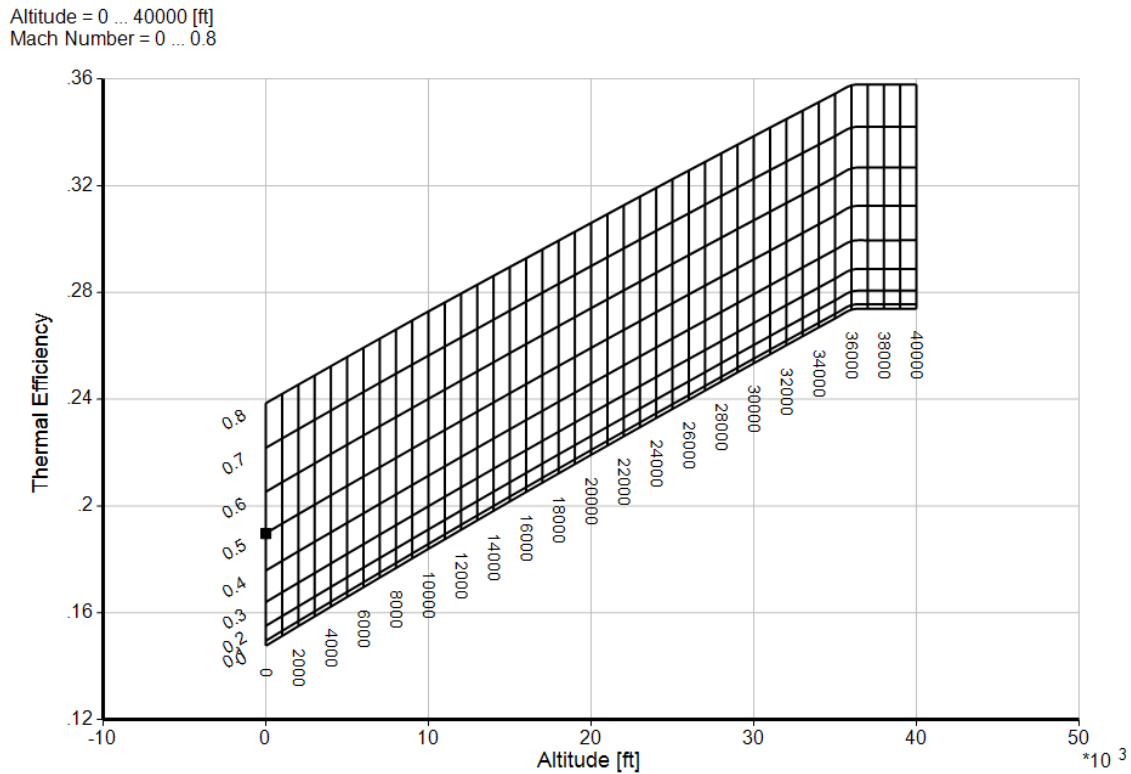


Figure 36: Parametric Analysis, Altitude and Mach Number vs. Thermal Efficiency

X.5 Rotor

Based on RFP requirements and trade studies performed by the team, it was determined that an aircraft design powered by a single turboprop would be optimal. XROTOR was chosen as the design tool based on the team's previous experience with the program and it being an extensively used tool in the professional aerospace community. The following sections will detail XROTOR's capabilities, the initial rotor design, the iterative refinement process, and the final rotor design.

About XROTOR

XROTOR is a design and analysis software for ducted propellers, free-tip propellers, and windmills [17]. The program has many capabilities available to the user including: automated design of minimum induced loss rotor, interactive modification of a rotor geometry, twist optimization, operating parameter envelope analysis, structural analysis, acoustic analysis, plotting of geometry, aerodynamic parameters, and performance maps, and many more. The program offers three main formulation techniques for analysis: graded momentum, potential, and vortex. All case studies for the final model were run using the most accurate and computationally expensive method, vortex formulation. This program was created by Dr. Mark Drela and Harold Youngren in collaboration with MIT and offered in both Unix and Windows 32

environments. The objective of XROTOR design and analysis for this project is to initially design the rotor, modify the rotor, obtain thrust and efficiency values, and analyze the acoustics of the rotor.

X.6 Initial Rotor Model

The initial rotor model was developed in XROTOR using a combination of published PT6A values and research on typical single turboprop aircraft rotors. The inputs seen below in Table 22 were used to design the initial rotor in the DESI sub menu. Additionally, this model was generated using the XROTOR default graded momentum formulation.

Table 22: XROTOR Inputs for Initial Rotor Model

| Property | Units | Value | Comments |
|-----------------------|-------|-------|-----------------------------------|
| Blades | N/A | 5 | Based on Super Tucano |
| Tip Radius | ft | 3 | Estimated, nominal value |
| Hub Radius | ft | 0.41 | Estimated, nominal value |
| Hub Wake Displacement | ft | 0.164 | Estimated, nominal value |
| Velocity | ft/s | 196.9 | From Initial Takeoff Calculations |
| RPM | RPM | 2200 | From PT6A-6 data |
| Power | hp | 1691 | From PT6A-68C data |
| Lift Coefficient | N/A | 0.5 | Estimated, nominal value |

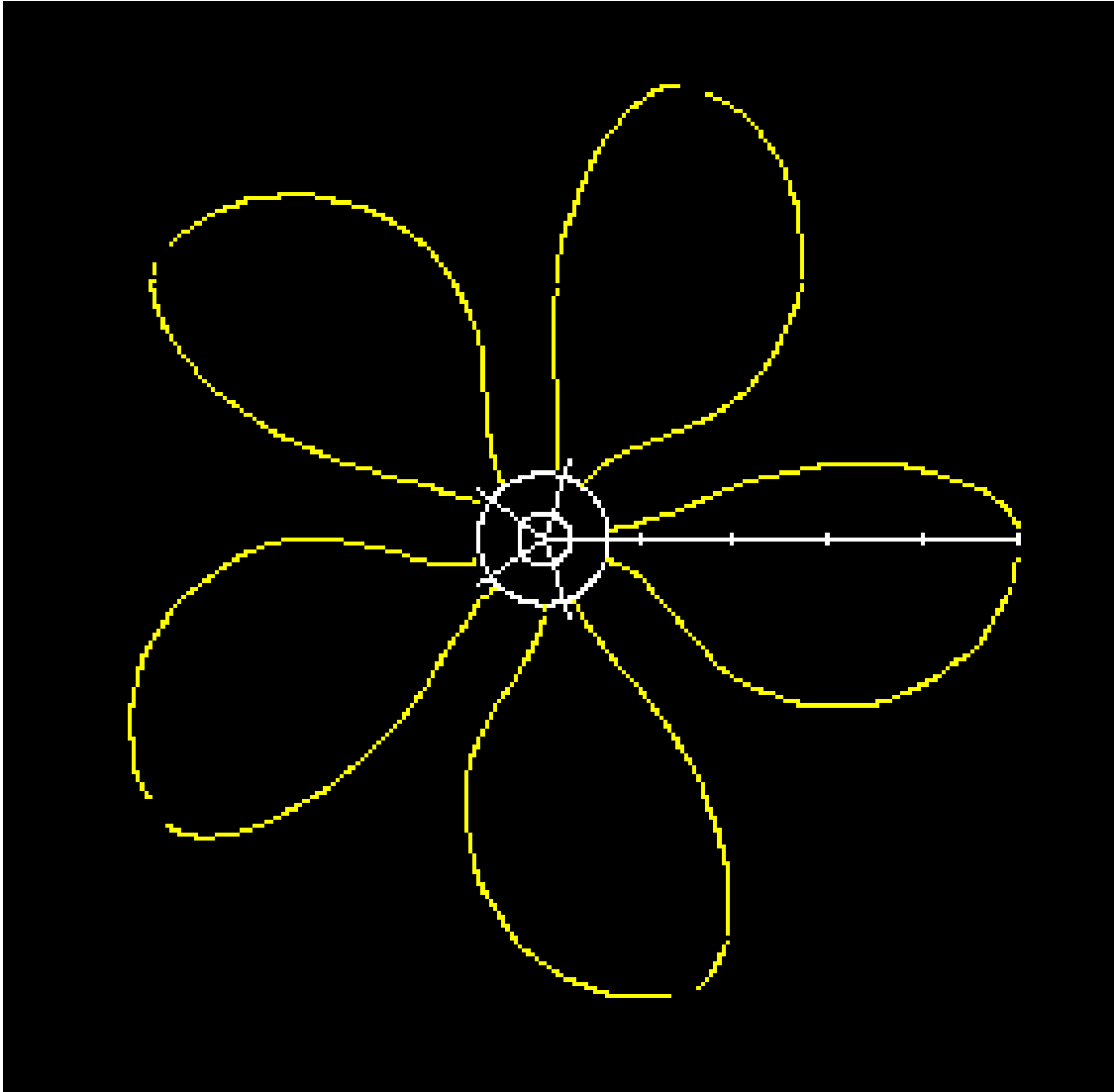


Figure 37: Initial Rotor Model Front Geometry

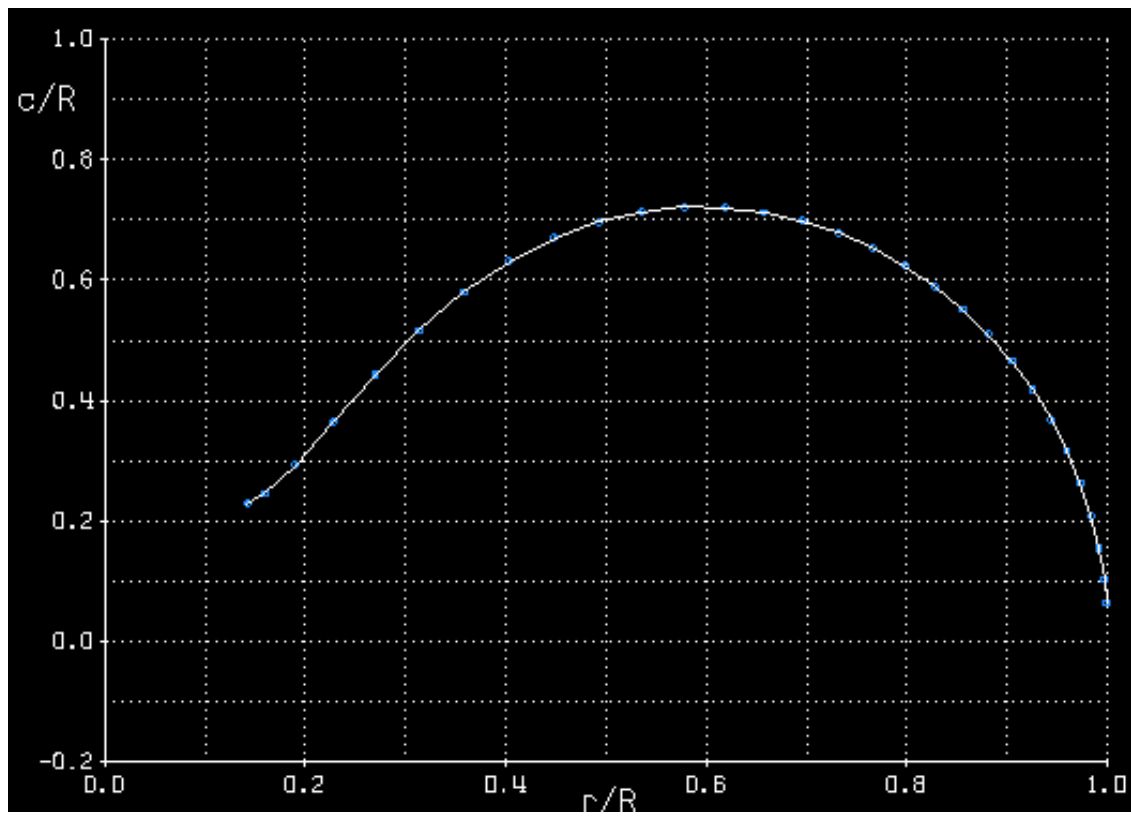


Figure 38: Initial Rotor Model Chord Distribution (c/R vs r/R)

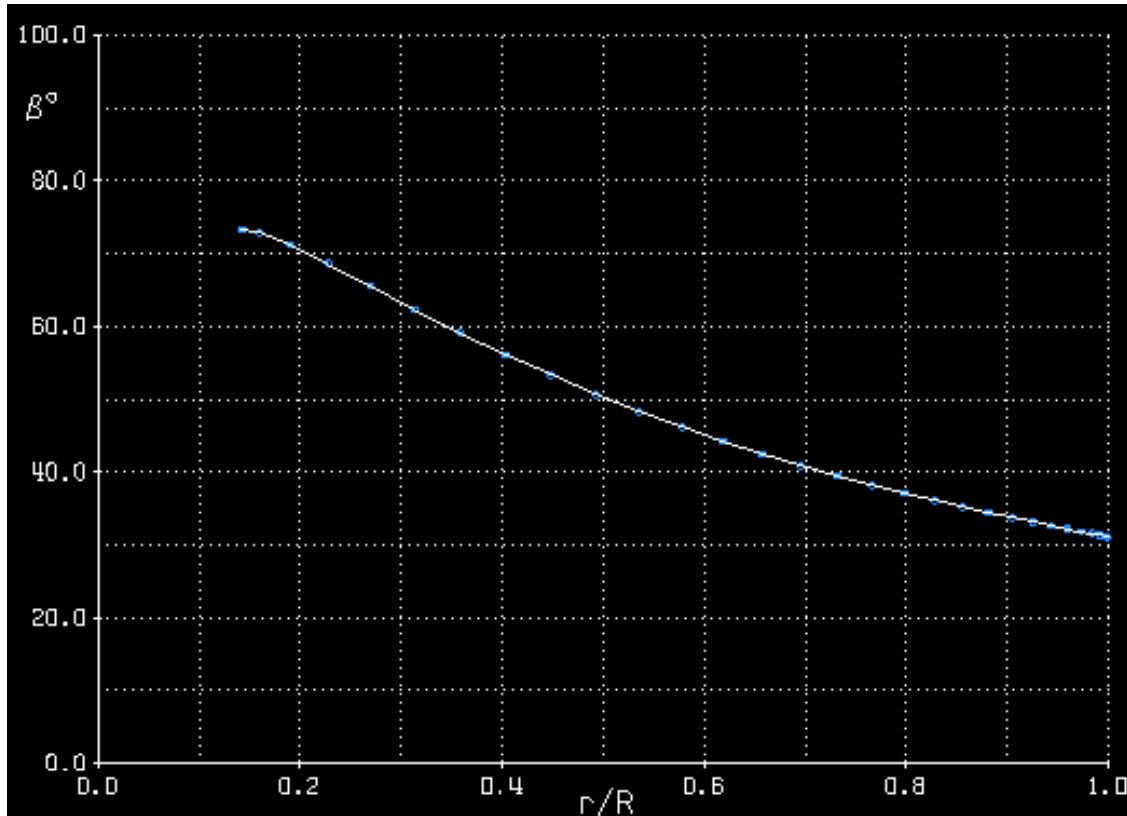


Figure 39: Initial Rotor Model Blade Twist Angle Distribution (β° vs r/R)

The front geometry, chord distribution, and blade twist distribution of the initial rotor produced are shown above in Figures 37 to 39. This design acted as the base model for all future modifications and optimizations. However, because this rotor was created based on early estimations for the takeoff case and generalized research, it was incapable of supporting steady level cruise. The next section will describe the methods used to improve this initial design and the final rotor model.

Final Rotor Model

The RFP states multiple specific objectives and requirements that relate to and constrain the rotor design and propulsion in general: takeoff field length of under 4000 ft, maximum efficiency for extended periods of cruise, and a service ceiling above 30000 ft. Creating the initial rotor model based on takeoff conditions emphasized the priority of maximizing low velocity thrust; however, in order to continue producing thrust at higher velocities and altitudes, the rotor needed significant modifications. The MODI sub menu of XROTOR offered multiple functions and tools to improve the initial design. The first series of modification iterations involved scaling down the chords to increase rotor thrust and efficiencies for the cruise and service ceiling cases. Using the SCAL function in XROTOR, changes can be applied radially to all blades according to the following function:

$$(Chord)_{new} = (Chord)_{old} \cdot [A + B \cdot (r/R)] \quad (6)$$

where A is the Constant Chord Scaling Factor and B is the Linear Chord Scaling Factor. The final chord distribution seen below in Figure 40 was developed after multiple iterations of scaling the chord using both scaling factors. By comparing Figures 38 and 40, the resulting chord distribution modifications can be described as both constantly decreased, as well as linearly relaxed as the radius increases.

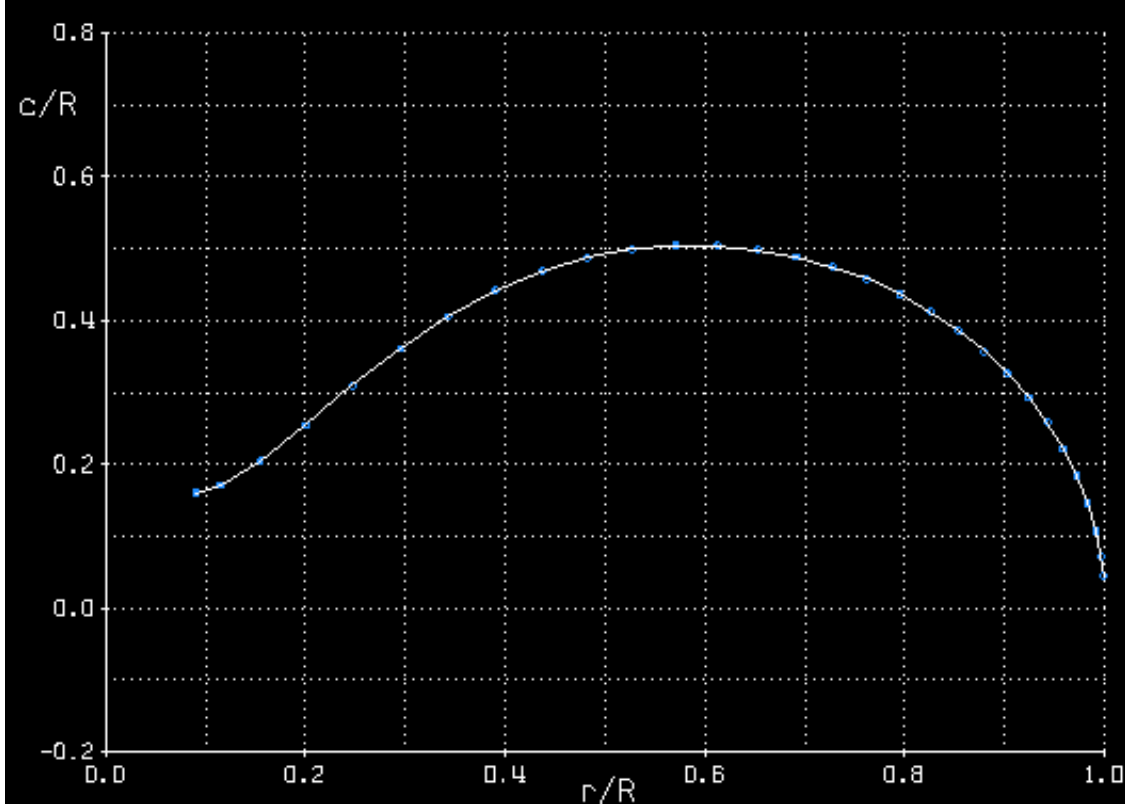


Figure 40: Final Rotor Model Chord Distribution (c/R vs r/R)

The second major modification made to the initial rotor design dealt with the blade twist angle distribution. XROTOR offers the OPTI function which finds the most efficient twist angle distribution for a given rotor design. Using this function produced the twist angle distribution seen below in Figure 41. It is worth noting that the β_0 spike seen at the maximum radius was caused by an unknown $\alpha(\text{CL})$ function inversion failure and has no perceivable impact on rotor performance.

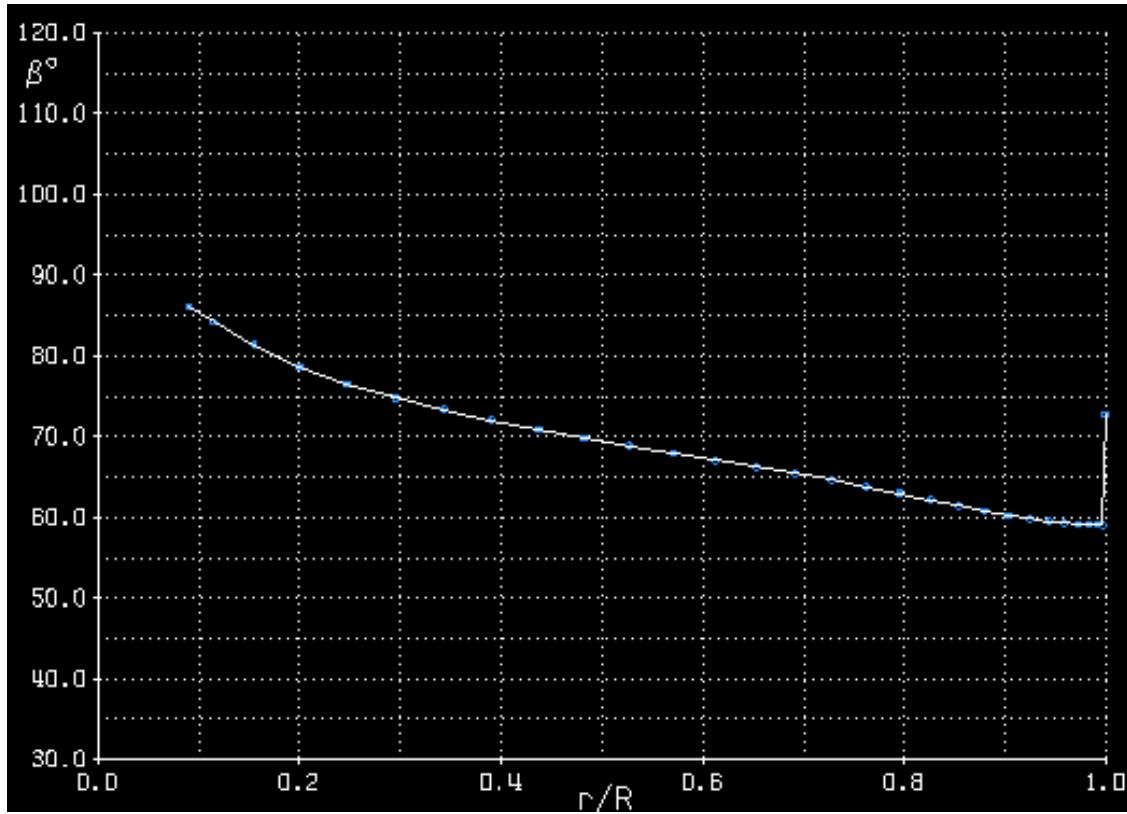


Figure 41: Final Rotor Model Blade Twist Angle Distribution (β° vs r/R)

The final modification to the rotor design consisted of modifying both the central hub radius and hub wake displacement radius to be more consistent with previous design studies conducted using XROTOR [40]. This change manifested in both values being set to 0.246 ft, a decrease in hub radius and an increase in hub wake displacement radius. The resulting final rotor model front geometry is presented below in Figure 6. The performance outputs and case studies conducted with this final propeller design can be found in the Performance section of this work.

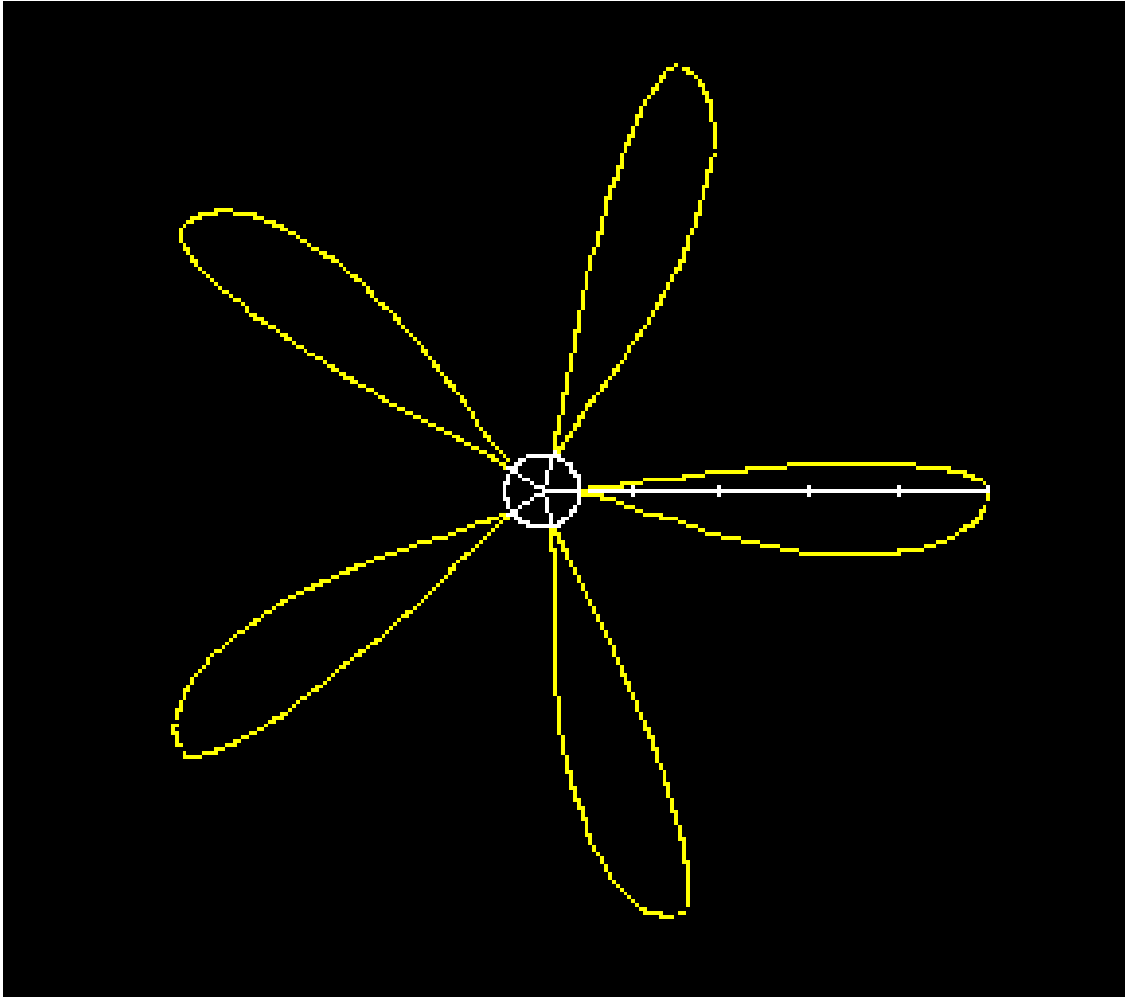


Figure 42: Final Rotor Model Front Geometry

XI Structural Design

XI.1 Wing Stress

For the wing structure of the design a multi-rib layout with two I-beam spars along the wing was chosen and selected the aluminum alloy 7075-T6 as the material due to its relatively low density and high yield strength. The wing configuration can be seen in figure 43.

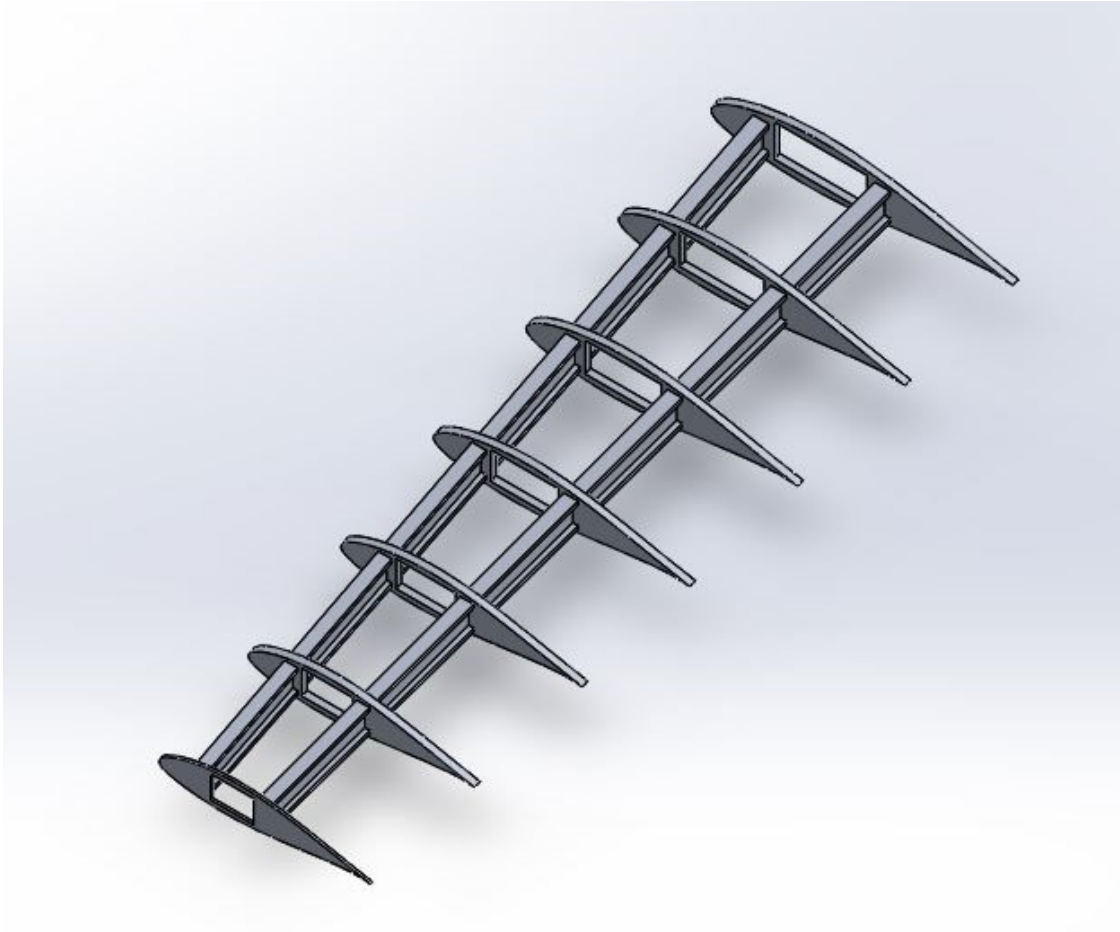


Figure 43: Wing Configuration

The ribs are tapered from the root chord of the wing to the tip chord and are swept at the angle specified in the wing design. The cutouts in the center of each rib serve to not only reduce the weight of the wing, but can also allow for storage inside of the wing. The initial model of the wing was well above the estimated weight from the equations and had a very high margin of safety, so the size of the spars were reduced in order to obtain a result closer to the weight estimation. A study on the structure in Solidworks to analyze the stress on the wing when the aircraft is experiencing an ultimate load factor of 3.75, and the results can be seen in Figure 44.

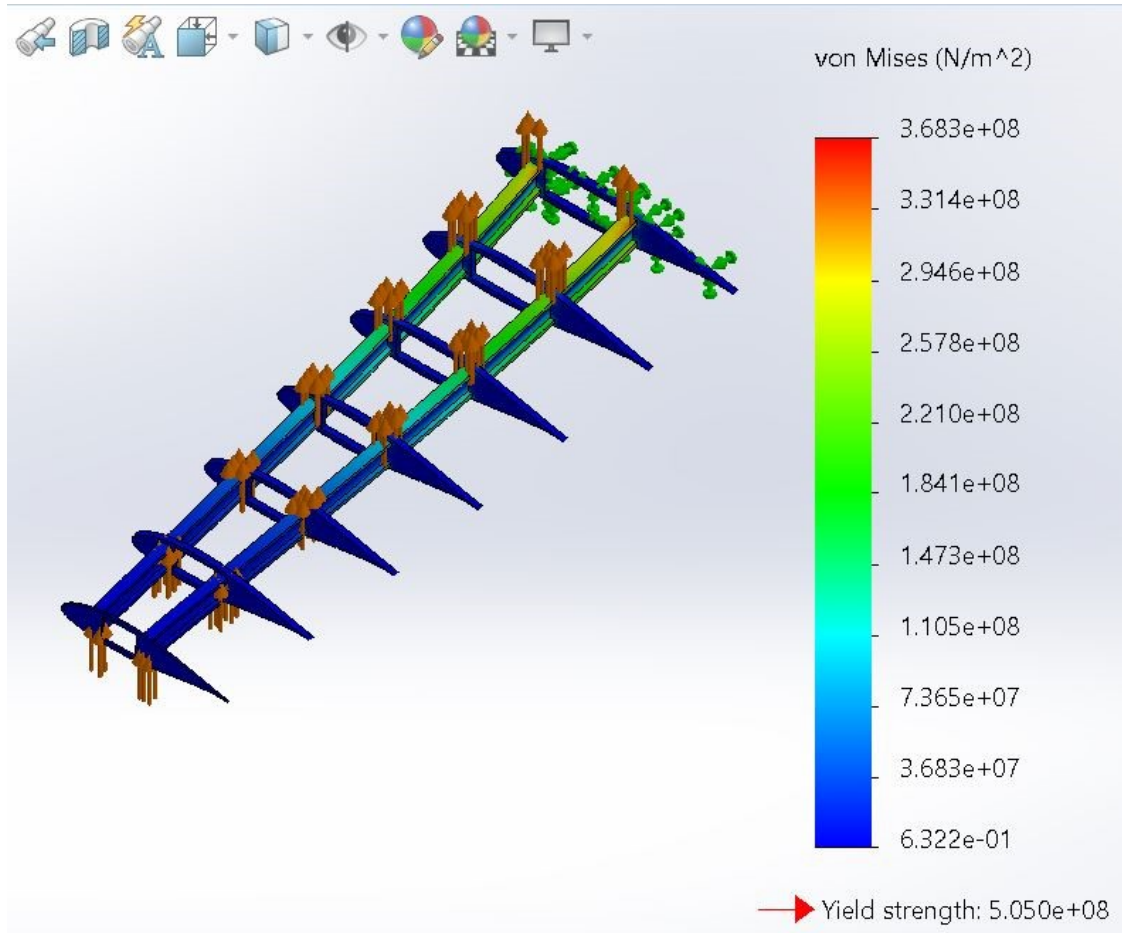


Figure 44: FEA of the Wing

In this case, the factor of safety is 1.37 when the aircraft is under the ultimate load.

XI.2 V-n diagram

To determine the flight envelope of the aircraft a V-n plot was made. Using a Matlab plotter, Figure 45 was made [41]. This plot shows the envelope for flaps down (outer line) flaps down (innermost lines), and at ultimate load factors (ULF) of 3.75 and 6. 3.75 was chosen since it is the assumed value in FLOPS and satisfies the FAR requirement that the ULF be greater than equation 7 but less than 3.8. For the Hummingbird equation 7 is equal to 3.08.

$$n = 2.1 + \frac{24000}{MTOW + 10000} \quad (7)$$

6 was also chosen since that is the expected load from a rolling pull out maneuver from MIL-A-8861 B [42]. Table 23 lists all the important envelope speeds.

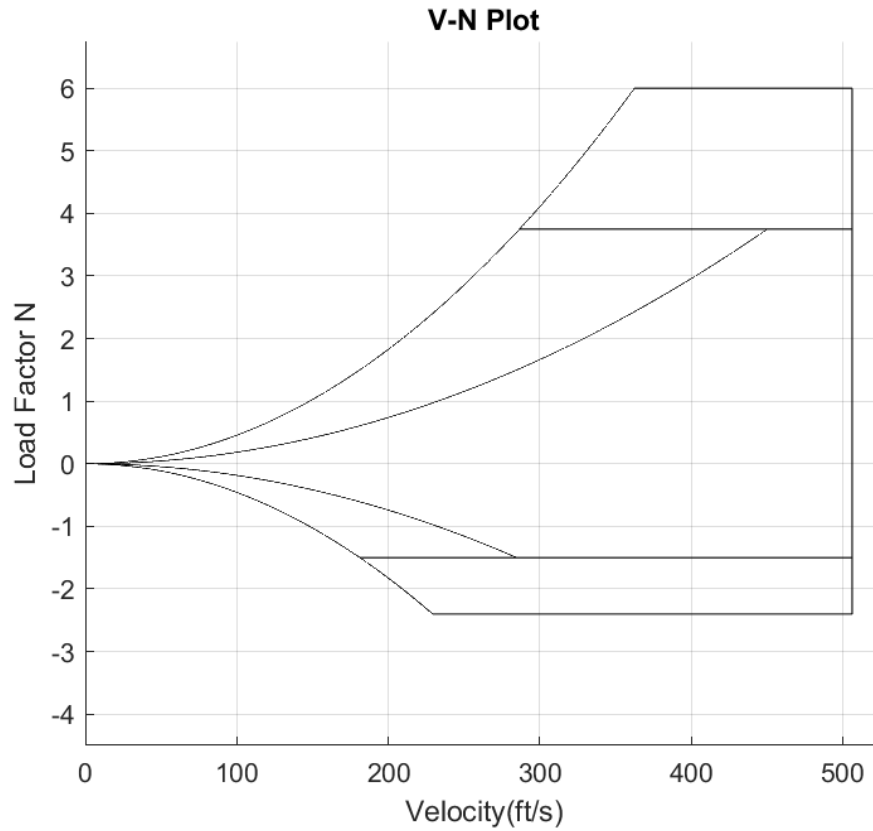


Figure 45: V-n Diagram

Table 23: Design Speeds from the V-n Diagram

| Design Speeds | Velocity (ft/s) |
|---|-----------------|
| Cruise | 337 |
| Stall (Flaps Up) | 145 |
| Stall (Flaps Down) | 231 |
| Maneuvering Speed (Flaps Up) | 450 |
| Maneuvering Speed (Flaps Down) | 287 |
| Negative Maneuvering Speed (Flaps Up) | 279 |
| Negative Maneuvering Speed (Flaps Down) | 179 |
| Dive Speed | 506 |

To calculate and plot a V-n Plot with gust lines a different Matlab plotter was used. This plotter only calculates in metric, however, are identical to values found in Table 23 [43].

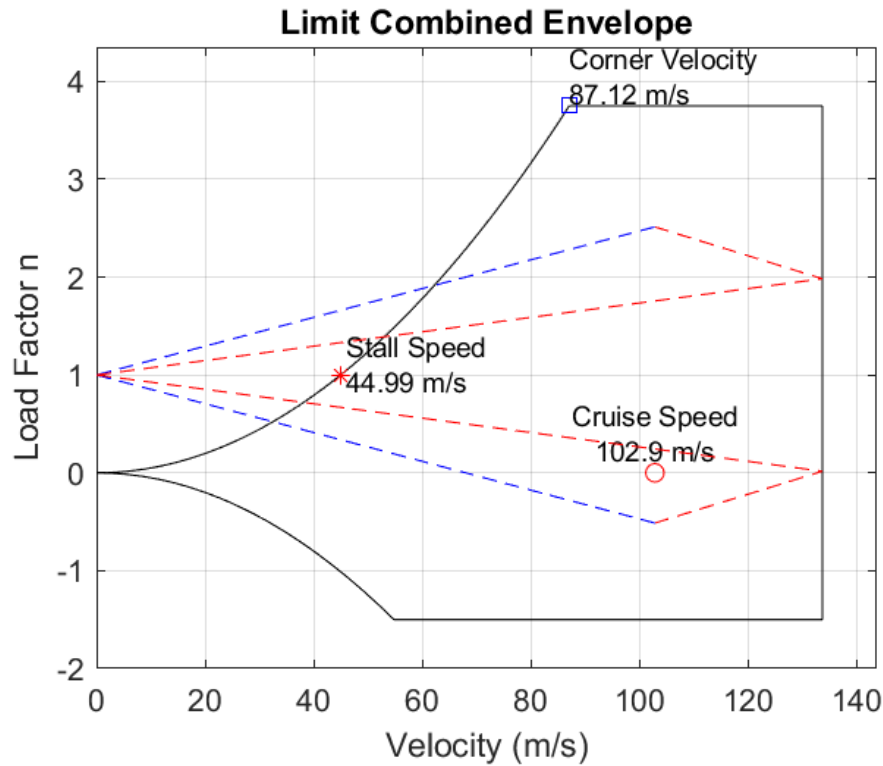


Figure 46: V-n Diagram with Gust Lines With ULF = 3.75

XII Weight Statement

XII.1 Initial Estimates

For the initial weight estimates of the structures of the aircraft, the equations provided in Nicolai and Carichner's Fundamentals of Aircraft and Airship Design were used [3]. These equations use design aspects of the aircraft to estimate the weight of the wings, fuselage, horizontal and vertical tail, and landing gear. These initial estimates are displayed in the table below.

Table 24: Empirical Weight Estimations

| Structure | Weight (lb) |
|-----------------|-------------|
| Wing | 1237.719 |
| Fuselage | 1954.496 |
| Horizontal Tail | 87.578 |
| Vertical Tail | 39.213 |
| Landing Gear | 585.763 |

After designing the wing structure, Solidworks evaluated the weight of the wing to be 1379.0 lbs, making it fairly similar to the weights that were previously calculated. The consistency with weight estimation through the equations,

and Solidworks increased confidence in the weight statement calculated. These estimations were relatively close to the final weight estimations done in FLOPS.

XII.2 FLOPS Weight Analysis

The design parameters from previous sections formed the input for the FLOPS analysis both for the weight estimations and the mission analysis presented in the next section. This analysis mainly used the default values with the exception of the pilot weights to include the ejection seats, overriding the wing weight with the one outputted by the Solidworks model, the engine weight, and provisions for external payload. Table 25 shows the finalized weight estimations.

Table 25: Weight Statement

| Component | Percent Weight | Weight (lb) |
|-----------------------------------|---------------------|---------------------|
| Wing | 8.57 | 1237 |
| Horizontal Tail | 0.74 | 107 |
| Vertical Tail | 0.99 | 142 |
| Landing Gear | 3.50 | 505 |
| Nacelle | 0.42 | 60 |
| <i>Structure Total</i> | <i>(27.80)</i> | <i>(4013)</i> |
| Engine | 4.14 | 598 |
| Miscellaneous Systems | 0.30 | 44 |
| Fuel System Tanks and Plumbing | 3.36 | 484 |
| <i>Propulsion Total</i> | <i>(7.80)</i> | <i>(1126)</i> |
| Surface Controls | 4.71 | 680 |
| Auxiliary Power | 1.83 | 264 |
| Instruments | 0.76 | 110 |
| Hydraulics | 0.97 | 140 |
| Electrical | 3.23 | 466 |
| Avionics | 0.83 | 119 |
| Armament | 0.55 | 80 |
| Furnishings and Equipment | 1.79 | 259 |
| <i>System and Equipment Total</i> | <i>(14.92)</i> | <i>(2154)</i> |
| <i>Weight Empty</i> | <i>50.52</i> | <i>7292</i> |
| Crew | 2.98 | 430 |
| Unusable Fuel | 0.50 | 72 |
| Engine Oil | 0.12 | 18 |
| Ammunition | 1.31 | 200 |
| Miscellaneous | 0.01 | 1 |
| <i>Operating Weight</i> | <i>55.44</i> | <i>8002</i> |
| External Store | 20.79 | 3000 |
| <i>Zero Fuel Weight</i> | <i>76.23</i> | <i>11002</i> |
| Mission Fuel | 23.77 | 3431 |
| <i>Gross Weight</i> | <i>100</i> | <i>14433</i> |

XIII Performance

XIII.1 Takeoff and Landing Field Length

To calculate the Takeoff and Landing Performance, the empirical steps were used [3]. Initially, using the preliminary sizing data, a trade study on the sensitivity of Thrust and CL_{max} was done to estimate a target for the thrust and CL_{max} required.

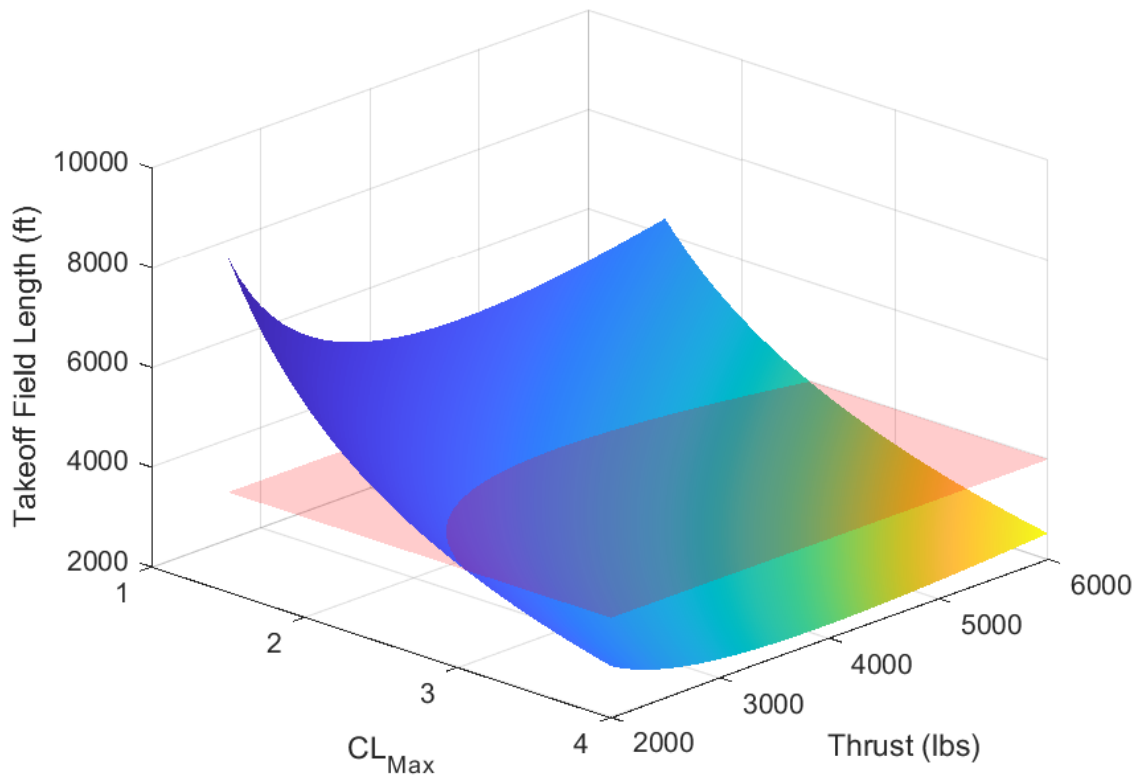


Figure 47: Sensitivity Analysis of the Takeoff Field Length

In Figure 26 one can see that the maximum coefficient of lift needs to be greater than 2 and the thrust needs to be greater than 3,000 lb. Thus a central target of 3 and 4,000 lb were chosen to give a margin if future design adjustments negatively impact either parameter. Note: the red intersecting plane is the 4,000 field length requirement.

Later when the propulsion and wings were sized and the team confirmed that a thrust of 4,000 lbs and a $CL_{2.75}$ were achievable, the team ran the equations again using those values. Table 26 and ?? tabulates the results. Appendix II has tabulated values of the assumptions made.

Table 26: Takeoff Field Lengths

| Segment | 0 ft Density Alt. (ft) | 6000 ft Density Alt. (ft) |
|--------------|---------------------------|------------------------------|
| Ground | 1708.21 | 2043.51 |
| Rotation | 311.34 | 340.53 |
| Transition | 1293.68 | 1547.61 |
| Total | 3313.23 | 3931.65 |

Table 27: Landing Field Lengths

| Segment | 0 ft Density Alt. (ft) | 6000 ft Density Alt. (ft) |
|--------------|---------------------------|------------------------------|
| Approach | 435.18 | 481.19 |
| Free Roll | 348.80 | 381.50 |
| Braking | 1770.88 | 2118.49 |
| Total | 2554.86 | 2981.18 |

XIII.2 Mission Performance

To analyze the mission performance of the Hummingbird the team used FLOPS. Using the data from the previous sections a FLOPS model of the plane can be built. FLOPS is a program designed to evaluate and optimize initial aircraft concepts. In this case, design parameters such as the lengths and weights of components are inputted along with an engine deck from the GasTurb engine model. FLOPS takes the aircraft model and simulates it through the desired design mission.

Figure 48 shows the design mission translated into a FLOPS mission.

```
# MISSION SEQUENCE DEFINITION

START      MACH NUMBER = 0.2000      ALTITUDE =      0. FT

1. CLIMB    USE CLIMB SCHEDULE 1

2. CRUISE   USE CRUISE SCHEDULE 3      SEGMENT DISTANCE = 100.0 N MI

3. HOLD     USE CRUISE SCHEDULE 2      HOLD TIME = 240.0 MIN

4. CLIMB    USE CLIMB SCHEDULE 1

5. CRUISE   USE CRUISE SCHEDULE 1      SEGMENT DISTANCE = 100.0 N MI

6. DESCENT

7. END      MACH NUMBER = 0.3000      ALTITUDE =      0. FT
```

Figure 48: Design Mission Sequence

For the climb segments the minimum speed is 0.2 Mach. For Cruise Schedule 1, FLOPS sets an optimal altitude and Mach number for the desired range. Cruise Schedule 3, however, a fixed Mach, 0.533, is given while an optimal altitude for the set range is used. This difference in cruise definition enforced the 20 minute requirement to go from climb to the hold segments. The results of the FLOPS analysis are found in Table 28. The plane is carrying its full payload for this mission.

Table 28: Design Mission Results. 3126.9 lb fuel burn, Total Range of 836.1 nmi.

| Segment | Initial WT (lb) | Fuel (lb) | | Time (min) | | Dist (nmi) | | Mach | | Alt. (ft) | |
|----------|-----------------|-----------|-------|------------|-------|------------|-------|-------|-------|-----------|-------|
| | | Segmt | Total | Segmt | Total | Segmt | Total | Start | End | Start | End |
| Taxi Out | 14433 | 7 | 7 | 5 | 5 | | | 0 | 0.2 | | |
| Climb | 14426 | 37 | 44 | 2.2 | 7.2 | 6.8 | 6.8 | 0.2 | 0.3 | 0 | 10000 |
| Cruise | 14389 | 204 | 249 | 17.6 | 24.8 | 100 | 106.8 | 0.300 | 0.533 | 10000 | 10000 |
| Hold | 14184 | 1635 | 1884 | 240 | 268.4 | 0 | 106.8 | 0.533 | 0.289 | 3000 | 3000 |
| Climb | 12549 | 19 | 1903 | 1.2 | 269.6 | 3.5 | 110.3 | 0.289 | 0.3 | 3000 | 10000 |
| Cruise | 12530 | 1213 | 3115 | 224 | 493.6 | 714.8 | 825.1 | 0.3 | 0.3 | 10000 | 10000 |
| Descent | 11318 | 5 | 3120 | 3.5 | 497.1 | 11.0 | 836.1 | 0.3 | 0.3 | 10000 | 0 |
| Reserves | 11313 | 311 | 4331 | | | | | | | | |
| Taxi In | | 7 | | 5 | 502.1 | | | | | | |

For the Ferry mission a similar run is made only at 60% payload capacity and for the cruise segment, the plane flies at the optimal altitude velocity. The results are shown below in Table 29.

Table 29: Ferry Mission Results. 1325.3 lb fuel burn, Total Range of 950 n mi.

| Segment | Initial WT (lb) | Fuel (lb) | | Time (min) | | Dist (nmi) | | Mach | | Alt. (ft) | |
|----------|-----------------|-----------|-------|------------|-------|------------|-------|-------|-------|-----------|-------|
| | | Segmt | Total | Segmt | Total | Segmt | Total | Start | End | Start | End |
| Taxi Out | 10999 | 7 | 7 | 5 | 5 | | | 0 | 0.320 | | |
| Climb | 10992 | 38 | 45 | 2.6 | 7.6 | 7.6 | 7.6 | 0.320 | 0.300 | 0 | 18000 |
| Cruise | 10954 | 1266 | 1311 | 298.1 | 305.7 | 923.0 | 930.6 | 0.300 | 0.300 | 18000 | 18000 |
| Descent | 9688 | 8 | 1318 | 6.3 | 312.1 | 19.4 | 950.0 | 0.300 | 0.300 | 18000 | 0 |
| Reserves | 9680 | 131 | 1449 | | | | | | | | |
| Taxi In | | 7 | | 5 | 317.1 | | | | | | |

XIII.3 Payload-Range

Using FLOPS the maximums of the aircraft can be explored. Different ranges can be found by varying the payload capacity and expending reserves. Table 30 provides the results while Figure 49 graphically shows that same information.

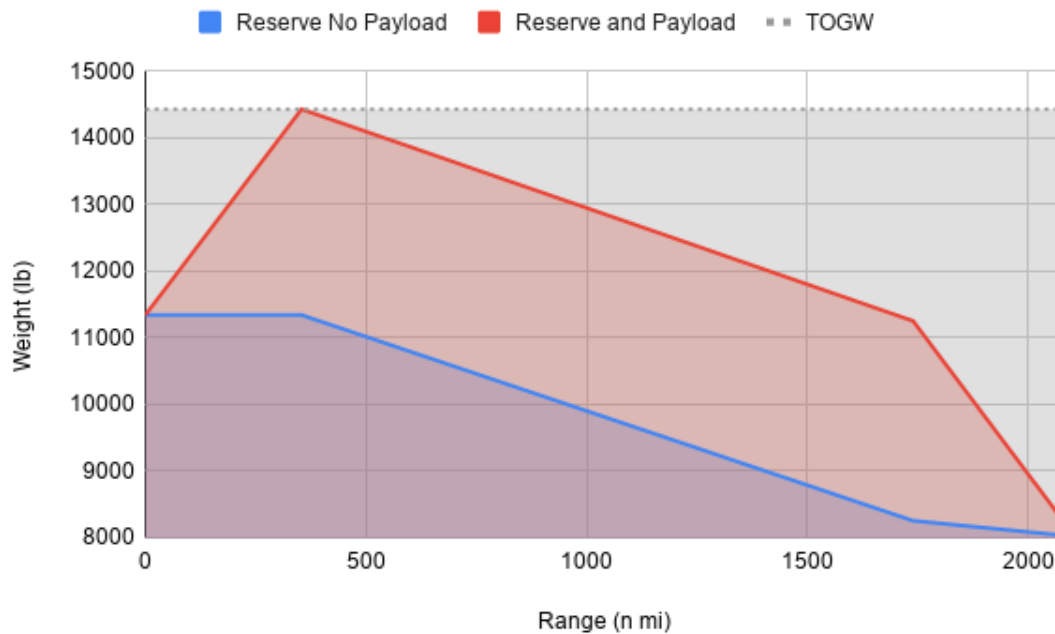


Figure 49: Payload Range Diagram. Note that the bottom of the chart is set at the OEW

Table 30: Possible Configurations to Achieve More Range

| Range (n mi) | Fuel Weight (lb) | Description |
|--------------|------------------|--|
| 0 | 0 | Empty |
| 354 | 3089 | Max Payload and Fuel with Reserve |
| 1740 | 6089 | Theoretical Max possible Fuel Weight with Reserve Held |
| 2080 | 6400 | Theoretical Max possible Fuel Weight and Spent Reserve |

Thus, the Hummingbird could in theory achieve a range of over 2000 n mi if all possible variable weight is converted into fuel. I.e. The theoretical max is achieved if the internal fuel cells are full and the payload was all external fuel stores.

XIII.4 Survivability

This section of the report details important performance metrics outputted by XROTOR for takeoff, cruise, and service ceiling cases. The values of importance for this project are total thrust produced and the ideal efficiency of the rotor. Full XROTOR case outputs can be found in the Appendix of this work. Additionally, acoustic analysis performed in XROTOR for each of these cases are shown. The calculations and method for determining the service ceiling can also be found in this section. Survivability considerations and implementations are discussed with reference to their importance in the RFP.

Takeoff

Below are the inputs and outputs, as well as the acoustic analysis plot, for the XROTOR takeoff case.

Table 31: XROTOR Inputs for Initial Rotor Model

| Property | Units | Value | Comments |
|----------|-------|-------|---|
| Power | hp | 1300 | Based on improved PT6A model |
| Velocity | ft/s | 223.1 | From takeoff calculations |
| RPM | RPM | 1500 | Based on improved PT6A model |
| Altitude | ft | 500 | Sea level, 500 used to produce usable noise signature |

Table 32: XROTOR Outputs for Takeoff Case

| Property | Units | Value |
|------------------|-------|--------|
| Thrust | lbf | 1645.7 |
| Ideal Efficiency | N/A | 0.8293 |
| Max Ground Noise | dB | 69 |

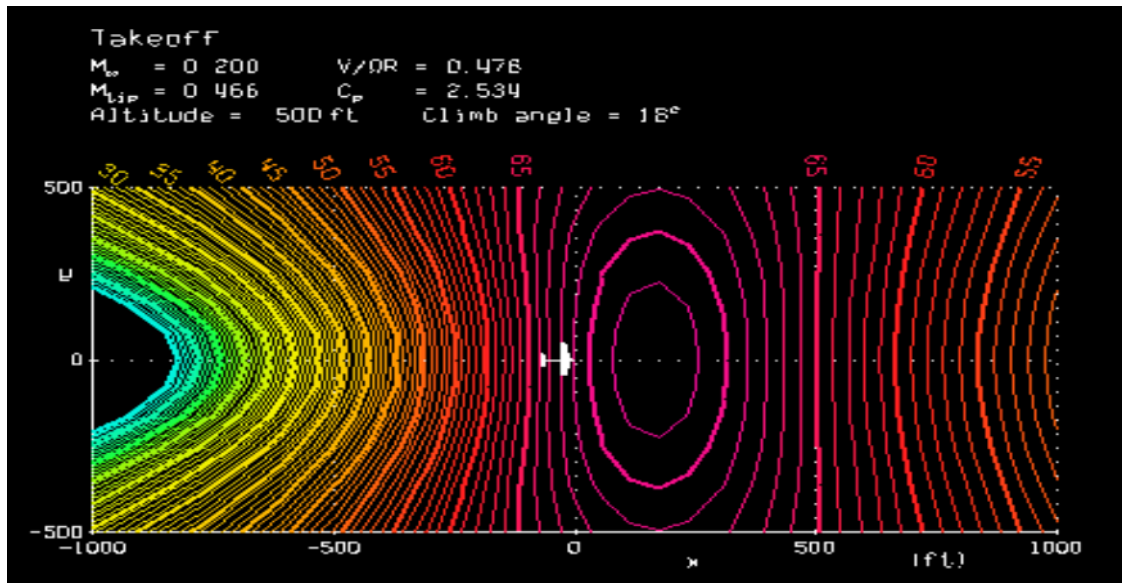


Figure 50: XROTOR Acoustic Analysis for Takeoff Case

Cruise

Below are the inputs and outputs, as well as the acoustic analysis plot, for the XROTOR cruise case.

Table 33: XROTOR Inputs for Cruise Case
 *XROTOR crash, lower power than available used.

| Property | Units | Value | Comment |
|----------|-------|-------|-------------------------------|
| Power | hp | 983 | Based on improved PT6A model* |
| Velocity | ft/s | 321.5 | From cruise calculations |
| RPM | RPM | 1500 | Based on improved PT6A model |
| Altitude | ft | 10000 | Cruise height |

Table 34: XROTOR Outputs for Cruise Case

| Property | Units | Value |
|------------------|-------|--------|
| Thrust | lbf | 1052.2 |
| Ideal Efficiency | N/A | 0.9136 |
| Max Ground Noise | dB | 44 |

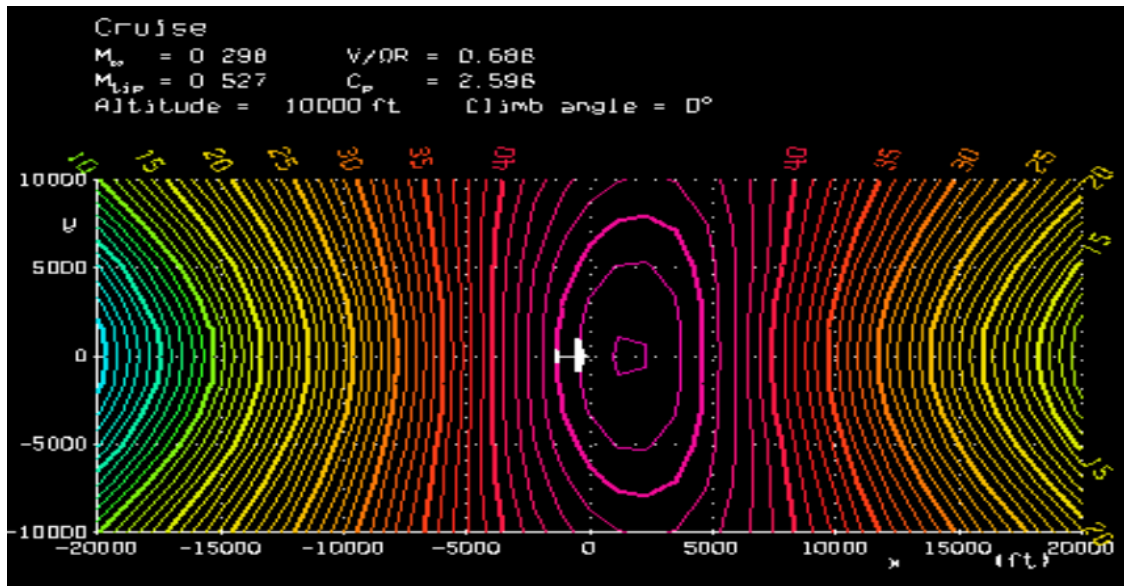


Figure 51: XROTOR Acoustic Analysis for Cruise Case

Service Ceiling

In order to calculate the service ceiling of 34000 ft for the aircraft, a MATLAB script was created to iterate the following equation at varying altitudes III.2:

$$P_s = \frac{V(T - D)}{W} \quad (8)$$

The script begins at 30000 ft and iterates up 100 ft per calculation loop until P_s falls under 100 fpm, the standard for aircraft [44]. In order to iterate the drag value at a new height, a built-in MATLAB density function updates every loop. However, iterating the thrust value at a new height required calculation of a thrust slope. This was done by

running an additional analysis case at 31000 ft and dividing the thrust difference by 10 to reach the decrease in thrust per 100 ft of gained altitude. The accuracy of the program's output was checked by comparing the calculated thrust at 34000 ft, 519.4 lbf, to an analysis case run in XROTOR at 34000 ft, 521.6 lbf. The full script for this application is located in Appendix III.

Below are the inputs and outputs, as well as the acoustic analysis plot, for the XROTOR service ceiling case.

Table 35: XROTOR Outputs for Service Ceiling Case

*XROTOR crash, lower power than available used.

| Property | Units | Value | Comment |
|----------|-------|-------|-------------------------------|
| Power | hp | 509.6 | Based on improved PT6A model* |
| Velocity | ft/s | 295.3 | From cruise calculations |
| RPM | RPM | 1500 | Based on improved PT6A model |
| Altitude | ft | 34000 | Service Ceiling |

Table 36: XROTOR Outputs for Service Ceiling Case

| Property | Units | Value |
|------------------|-------|--------|
| Thrust | lbf | 521.6 |
| Ideal Efficiency | N/A | 0.8896 |
| Max Ground Noise | dB | 31 |

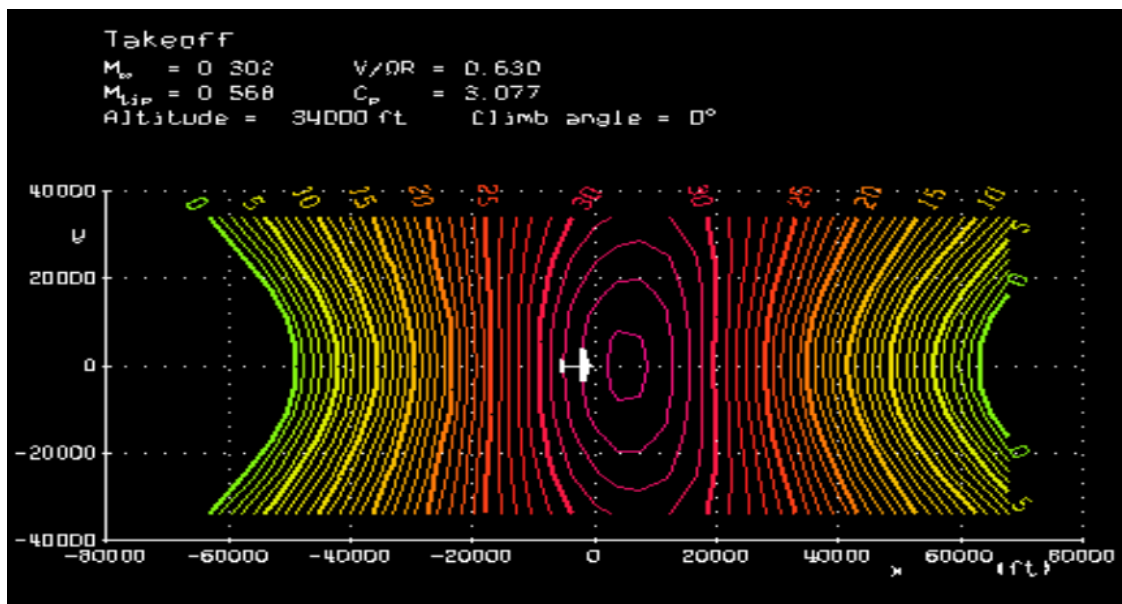


Figure 52: XROTOR Acoustic Analysis for Service Ceiling Case

Other Survivability Considerations

The RFP for this project lists survivability as a main objective and therefore multiple survivability considerations were made for this aircraft. Researching survivability for light attack aircraft illuminated two main methods of

achieving increased survivability: reduced susceptibility and reduced vulnerability. While these two strategies are not mutually exclusive, it was determined that resources should be invested in decreasing vulnerability rather than reducing susceptibility [3]. The main factor in this decision was the significant cost associated with creating a low-susceptibility or stealth aircraft. This section contains an annotated list of features and design decisions made to decrease the aircraft's vulnerability to threats.

Chaff

Chaff dispensing systems have been used on combat aircraft since WW2 and provide emergency protection from incoming missiles. The ALE-47 airborne countermeasures dispensing system is currently installed on over 4000 vehicles in 30 countries around the world including: F-18, F-16, and F-15 [45]. This system was chosen specifically because it offers system flexibility, platform customization, and continuous product support.

Titanium Tub

Cockpit and pilot protection are at the forefront of all vulnerability considerations and the RFP states so clearly. This aircraft will feature an 1100 lb titanium 'tub' similar to the one found in the A-10 Thunderbolt II [46]. While this feature adds significant weight to the aircraft, it is critical to the aircraft's survivability and overall mission success.

Gas Tank

The fuel tank is exceedingly important to protect for a combat aircraft as it can be a major cause of catastrophic system failures. In order to minimize the probability of a fire or explosion, two main considerations were made. The first consideration made is the location: fuel tank protection is maximized by positioning the fuel tank at the top of the central fuselage [3]. This decreases the probability that an incoming projectile will make impact with the highly explosive jet fuel stored inside. The second consideration involves the use of a nitrogen fuel-inerting system to further minimize the probability of a catastrophic explosion. These systems function by pumping inert nitrogen-enriched air into the fuel tank as it empties, keeping the oxygen concentration levels below 12% to prevent combustion [47].

Control Surface redundancy

Dividing control surfaces into multiple sections allows for continued control and capability in the event that one section is damaged. This concept has been employed in the past, with great success, on aircraft such as the A-10 Thunderbolt II [3].

XIV Cost Estimation and Analysis

Venturing from the technical production of the aircraft, the team had to estimate a cost for the aircraft to develop a realistic and reasonably priced aircraft. During the cost analysis process the primary focus was calculating the cost of the aircraft and the price of the aircraft. The cost of an aircraft is the total amount of expenditures for resources, usually

measured in dollars, needed to manufacture an aircraft. The price of an aircraft is the amount of dollars paid for the aircraft by customers which includes the profit the manufacture earns from selling the aircraft, which is in addition to the cost to produce the aircraft [48]. Due to the nature of this aircraft's intended assignment as a military light attack aircraft the cost analysis must be adapted based on several assumptions. For example, military/government operations typically have a limited amount of budget. A military's operational goal is to achieve the required missions within a specific budget amount. In order to fulfill this strenuous requirement, the acquisition and operating cost for the aircraft needs to be as low as possible.

XIV.1 Overview of Cost Methodology

A complex aircraft analysis tool was utilized in order to create a well-constructed cost estimation. For this specific project the cost analysis tool used was embedded in the Advanced Aircraft Analysis 4.0 (AAA) program. The cost analysis method used to develop the AAA software was adapted from Dr. Jan Roskam's methodology in the "Airplane Design Part VIII: Airplane Cost Estimation: Design, Development, Manufacturing and Operating." AAA has a variety of categories for aircraft analysis which reflects the Roskam methodology including Weight, Aerodynamics, Performance, Geometry, Propulsion, Stability and Control, Dynamics, Loads, Structures, and Cost. Within each of these categories are various subsections to help with the intended analysis.

This project, in particular, utilized the Cost category which had six subsets: AMPR Weight Estimation, Research/Development/Test/Evaluation Cost, Acquisition Cost, Military Operating Cost, Life Cycle Cost and Disposal, and Airplane/Engine/Propeller Price Data. By completing all relevant subsets of the Cost Analysis tab, a fairly accurate cost estimation was produced. In order to operate the Cost tool there were a combination of inputs needed in each category that resulted in the calculation of the output values. The tool requires a combination of previously calculated values, researched numerical factors, historical values, and internally calculated values performed by the program to achieve the intended output values.

XIV.2 Assumptions

Making engineering educated assumptions was critical in completing the cost estimation and analysis. The cost methodology laid out in the Roskam textbook provided specific ranges for various cost factors with detailed guidelines for evaluating each variable. Some variables had to be chosen from outside research rather than the Roskam textbook ranges including: the service life, number of aircraft manufactured, and number of flights per year. The service life chosen was 25 years because during this is the average aircraft service life during peacetime, and the team assumed peacetime for the next 30 years. The number of aircraft manufactured was chosen because the A-29 has been around

since 2015 and just over 250 aircraft have been sold; using this number it was estimated that in the year 2025 the roll out would consist of approximately 25% of a five-year project would be produced in the first order. One cannot assume this project would find as much success as the A-29 Super Tucano therefore 60 aircraft seemed reasonable. Finally, the number of flights per year was a calculated output values based mostly on the average mission time of two hours. The variable nomenclature can be found in Appendix IV.

Table 37: Input Variables and Assumptions

| Variable | Assumption |
|----------------------------|--|
| $F_{\text{difficulty}}$ | 1.5 (moderate use of technology) |
| F_{CAD} | 1.0 (using manual drafting techniques) |
| N_{rdte} | 20 (military program range 6-20) |
| $R_{\text{rate 1989}}$ | \$65/hr (AAA provided table) |
| N_{flight} | 20 (military aircraft 6-20) |
| N_{static} | 0 |
| F_{mat} | 1.0 (aluminum alloy) |
| $C_{\text{eng rdte}}$ | \$1,000,000 (research supported) |
| $R_{\text{man rdte 1989}}$ | \$33/hr (AAA provided table) |
| $R_{\text{t rate 1989}}$ | \$44/hr (AAA provided table) |
| R_{rdte} | 0.33 (suggested in AAA) |
| F_{obs} | 1.1 (no stealth requirement airplanes) |
| F_{Tsf} | 0.2 (if test and simulation facilities are required) |
| F_{pro} | 0.1 (for suggested 10% profit) |
| $F_{\text{fin rate}}$ | 0.1 (for low interest on military deals) |
| N_{man} | 50 (given in RFP) |
| $R_{\text{man prog 1989}}$ | \$35/hr (AAA provided table) |
| $R_{\text{t prog 1989}}$ | \$45/hr (AAA provided table) |
| F_{int} | 0 (for military airplanes) |
| t_{ft} | 20 (average for military) |
| F_{fton} | 4 (average for military) |
| $U_{\text{ann flt}}$ | 1200 (given in RFP) |
| F_{misc} | 0.02 (suggested in AAA) |
| F_{OL} | 1.005 (factor that accounts for oil and lubricants) |
| N_{yr} | 25 yrs (peacetime) |
| R_{loss} | 5 (estimated from A-6 and common aircraft) |
| R_{crew} | 2 (given in RFP) |
| MHP_{flhr} | 2 (suggested in AAA) |
| F_{disp} | 0.01 (suggested in AAA) |
| Year | 2025 (given in RFP) |

XIV.3 Calibration

In order to calibrate the AAA tool to perform an accurate cost estimation for the Hummingbird, the program had to be tested and adjusted to find the known value of the A-29 Super Tucano. Although AAA calculates a variety of cost values the only way to calibrate the cost prototype is to use the acquisition cost because that is the only known value. Life cycle cost and operating cost are difficult to calibrate due to proprietary military dealings. Before the calibration, using the

A-29 a preliminary cost model produced an acquisition cost per Hummingbird airplane of \$31,990,000; this excessive evaluation made it clear adjustments had to be applied hence the A-29 calibration. By tweaking many variables that had acceptable ranges of large magnitude, the cost model began to take shape. After extensive adjustments, the calibration was complete once the AAA model produced a cost per airplane of \$25,697,000 (merely *sim*\$250,000 away from the known value). Once this acquisition cost was refined, many inputs stayed the same due to the similarities of the light attack aircraft categories. Major variable changes included the Takeoff Weight, Max Level Speed, and Number of Aircraft Produced. These relative inputs were altered to represent the values of the Hummingbird in order to produce a comprehensive cost estimate.

XIV.4 Results

The cost analysis evaluation calculated the values for a large amount of cost values. In regards to this project, four essential categories of aircraft cost analysis were identified to be used for a comprehensive cost estimation. The four categories included: Research/ Development/Test/Evaluation (RDTE) Cost, Life Cycle Cost, Total Cost per Plane, and Operating Cost.

The RDTE Cost is used solely as a preliminary evaluation consisting of values accumulated during beginning phases of the aircraft program involving activities which takes a new aircraft all the way from planning and conceptual design to certification. These beginning phases typically include design, construction, and ground and flight testing. The Hummingbird Program's preliminary RDTE Cost evaluation was \$912,131,000.

The Manufacturing and Acquisition Cost category allowed for the total cost per aircraft. This category includes all materials and labor to manufacture and acquire an aircraft. This cost estimation is in ending phases of the aircraft program. The total cost per aircraft is \$29,322,000.

The Operating Cost category is used as an evaluation only for aircraft in peace time. This method should only be used for preliminary cost estimating purposes. It is important to assume that because this is a military aircraft, political and budgetary constraints must be disregarded because these variables are impossible to translate into the cost methodology. After the completion of this cost module section the final Operating Cost evaluation was \$2,495,846,000. This breaks down to \$1742.86 per hour of operation for each individual aircraft. This number is higher than the \$1000 per hour in the press when involving light attack aircraft; this value is an output that uses over 15 variables. With this in mind, the human error factor of a less than complete understanding of the AAA tool would justify this reasonable operating cost for a new aircraft program. Two additional important output values include the maintenance cost for the program and fuel costs. The Program Cost for Fuel, Oil and Lubricants came out to \$152,694,000, while the Direct Maintenance Personnel Cost for the Program was \$865,383,000.

The final cost category was the Life Cycle Cost of a military aircraft. This methodology uses all previous cost

estimations to calculate the total life cycle cost of a program. This cost is a summation of the program's total cost for RDTE, Acquisition, Operation, and Disposal over the life time of a program. The program's Life Cycle Cost was \$2,561,744,000. This is the total cost that all governments and/or militaries must focus on in any cost discussions due to its comprehensive assessment; in this instance the comprehensive assessment of the entire Hummingbird Program cost

XV Conclusion

The Hummingbird is an ideal solution to the RFP because it is more than capable of meeting the desired performance requirements. The acquisition cost is higher, at \$29 million, than other aircraft though the operating cost, at \$1743 per flight hour, is far lower than the majority of aircraft in service. The plane is conventionally designed to minimize cost and maximize opportunities for "off the shelf" parts.

The aircraft is confidently able to carry out the design and ferry missions with room to spare. There is still a large margin available in terms of both space and weight for future iterations. Future work could include using that free space for expanding the plane's capabilities, such as increased survivability measures, or for a higher fidelity optimization and weight and cost reduction. Future work will also include slight increases in tail area and sweep to provide the small increase in lateral and directional stability needed. In its current form, however, Team Hummingbird fully recommends the Hummingbird as the initial conceptual design for comprehensive design and production work.

XVI Acknowledgements

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Appendices

I Initial TOGW Calculation

Table 38: Initial TOGW Calculation Varying TOGW by 1000 lb

| W_fixed | W_fuel | W_TO | W_empty_avail | W_empty_req |
|---------|-------------|-------|---------------|-------------|
| 3600 | 1882.590308 | 8000 | 2517.409692 | 3802.103025 |
| 3600 | 2117.914096 | 9000 | 3282.085904 | 4250.747612 |
| 3600 | 2353.237885 | 10000 | 4046.762115 | 4696.752368 |
| 3600 | 2588.561673 | 11000 | 4811.438327 | 5140.395514 |
| 3600 | 2823.885462 | 12000 | 5576.114538 | 5581.903229 |
| 3600 | 3059.20925 | 13000 | 6340.79075 | 6021.462939 |
| 3600 | 3294.533039 | 14000 | 7105.466961 | 6459.232434 |
| 3600 | 3529.856827 | 15000 | 7870.143173 | 6895.346348 |
| 3600 | 3765.180616 | 16000 | 8634.819384 | 7329.920876 |
| 3600 | 4000.504404 | 17000 | 9399.495596 | 7763.057297 |
| 3600 | 4235.828193 | 18000 | 10164.17181 | 8194.844656 |

Table 39: Finer Initial TOGW Calculation Varying TOGW by 200 lb

| W_fixed | W_fuel | W_TO | W_empty_avail | W_empty_req |
|---------|-------------|-------------|---------------|-------------|
| 2800 | 2242.691479 | 9530.237015 | 4487.545536 | 4487.545536 |
| 3000 | 2389.991233 | 10156.18203 | 4766.190801 | 4766.190801 |
| 3200 | 2536.625756 | 10779.30018 | 5042.674425 | 5042.674425 |
| 3400 | 2682.643675 | 11399.79809 | 5317.154417 | 5317.154417 |
| 3600 | 2828.087391 | 12017.85595 | 5589.768557 | 5589.768557 |
| 3800 | 2972.99418 | 12633.63215 | 5860.63797 | 5860.63797 |
| 4000 | 3117.397049 | 13247.26696 | 6129.869914 | 6129.869914 |
| 4200 | 3261.325416 | 13858.8854 | 6397.559983 | 6397.559983 |

II Takeoff and Landing Assumptions

Table 40: Assumed Values for Takeoff and Landing Calculations

| Parameter | Value | Comment |
|---------------------|-------|--------------------------------|
| Rolling Coefficient | 0.025 | MIL Standard |
| Ground Friction | 0.04 | Firm and dry dirt |
| Ground CL | 0.26 | Nikolai Equation |
| CD_0 | 0.022 | Averaged from Similar Aircraft |
| delta CD_Flaps | 0.01 | Nikolai |
| delta CD_gear | 0.035 | Nikolai |
| CD_brake | 0.43 | Nikolai |

III Noise Signatures

III.1 XROTOR Outputs

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=====
Free Tip Vortex Wake Formulation Solution: Takeoff
no. blades : 5          radius(m) : 0.9100      Wake adv. ratio: 0.91712
thrust(N)   : 0.732E+04 power(W)   : 0.969E+06   adv. ratio: 0.47572
Efficiency  : 0.5137    speed(m/s) : 68.000    torque(N-m): 0.617E+04
Eff induced: 0.5187    Eff ideal  : 0.8293    Tcoef      : 0.9932
Tnacel(N)   : -2.8373  hub rad.(m): 0.0750    disp. rad. : 0.0750
Tvisc(N)    : -45.7269 Pvisc(W)  : 0.335E+04
rho(kg/m3)  : 1.22600  Vsound(m/s): 340.000    mu(kg/m-s) : 0.1780E-04
-----
Sigma: 0.73871
          Ct: 0.87115   Cp: 2.53440   J: 1.49451
          Tc: 0.99320   Pc: 1.93341   adv: 0.47572

i  r/R  c/R  beta(deg)  CL   Cd   REX10^6  Mach  effi  effp  na.u/U
1  0.090 0.1606 86.09 0.240 0.0081 0.69 0.203 0.560 0.681 0.000
3  0.155 0.2049 81.38 0.538 0.0071 0.91 0.208 0.734 0.942 0.000
5  0.247 0.3097 76.48 0.765 0.0060 1.46 0.221 0.714 0.976 0.000
7  0.343 0.4062 73.37 0.957 0.0054 2.06 0.238 0.650 0.985 0.000
9  0.437 0.4690 70.92 1.113 0.0052 2.60 0.260 0.601 0.989 0.000
11 0.527 0.4994 68.89 1.235 0.0051 3.02 0.284 0.563 0.991 0.000
13 0.612 0.5041 67.10 1.323 0.0051 3.32 0.309 0.535 0.992 0.000
15 0.691 0.4889 65.44 1.381 0.0052 3.48 0.334 0.514 0.992 0.000
17 0.762 0.4576 63.82 1.416 0.0053 3.50 0.359 0.500 0.992 0.000
19 0.825 0.4130 62.22 1.435 0.0056 3.35 0.381 0.492 0.992 0.000
21 0.879 0.3575 60.80 1.446 0.0059 3.05 0.400 0.487 0.992 0.000
23 0.924 0.2929 59.82 1.458 0.0064 2.60 0.417 0.482 0.991 0.000
25 0.959 0.2218 59.31 1.473 0.0074 2.03 0.430 0.478 0.990 0.000
27 0.983 0.1464 59.13 1.497 0.0107 1.37 0.439 0.489 0.986 0.000
29 0.997 0.0722 59.10 1.518 s 0.0621 0.68 0.445 0.622 0.920 0.000
=====
Free Tip Vortex Wake Formulation Solution: Cruise
no. blades : 5          radius(m) : 0.9100      Wake adv. ratio: 1.08105
thrust(N)   : 0.468E+04 power(W)   : 0.733E+06   adv. ratio: 0.68559
Efficiency  : 0.6259    speed(m/s) : 98.000    torque(N-m): 0.467E+04
Eff induced: 0.6342    Eff ideal  : 0.9136    Tcoef      : 0.4141
Tnacel(N)   : -0.9176  hub rad.(m): 0.0750    disp. rad. : 0.0750
Tvisc(N)    : -45.9067 Pvisc(W)  : 0.253E+04
rho(kg/m3)  : 0.90455  Vsound(m/s): 328.800    mu(kg/m-s) : 0.1682E-04
-----
Sigma: 0.73871
          Ct: 0.75447   Cp: 2.59644   J: 2.15385
          Tc: 0.41415   Pc: 0.66172   adv: 0.68559

i  r/R  c/R  beta(deg)  CL   Cd   REX10^6  Mach  effi  effp  na.u/U
1  0.090 0.1606 86.09 0.117 0.0079 0.78 0.300 0.657 0.217 0.000
3  0.155 0.2049 81.38 0.253 0.0069 1.00 0.305 0.847 0.852 0.000
5  0.247 0.3097 76.48 0.383 0.0057 1.57 0.315 0.840 0.946 0.000
7  0.343 0.4062 73.37 0.526 0.0050 2.15 0.330 0.774 0.972 0.000
9  0.437 0.4690 70.92 0.660 0.0047 2.62 0.348 0.723 0.981 0.000
11 0.527 0.4994 68.89 0.777 0.0045 2.96 0.368 0.683 0.986 0.000
13 0.612 0.5041 67.10 0.872 0.0045 3.17 0.390 0.653 0.988 0.000
15 0.691 0.4889 65.44 0.945 0.0045 3.25 0.413 0.630 0.989 0.000
17 0.762 0.4576 63.82 0.997 0.0046 3.20 0.434 0.616 0.990 0.000
19 0.825 0.4130 62.22 1.033 0.0048 3.02 0.455 0.608 0.990 0.000
21 0.879 0.3575 60.80 1.059 0.0050 2.72 0.473 0.603 0.990 0.000
23 0.924 0.2929 59.82 1.084 0.0054 2.30 0.488 0.597 0.990 0.000
25 0.959 0.2218 59.31 1.113 0.0060 1.78 0.500 0.589 0.989 0.000
27 0.983 0.1464 59.13 1.156 0.0072 1.20 0.508 0.585 0.987 0.000
29 0.997 0.0722 59.10 1.230 0.0098 0.60 0.513 0.589 0.984 0.000

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=====
Free Tip Vortex Wake Formulation Solution: 30000ft
no. blades : 5          radius(m) : 0.9100      Wake adv. ratio: 1.12497
thrust(N) : 0.270E+04    power(W) : 0.440E+06    adv. ratio: 0.62962
Efficiency : 0.5518      speed(m/s) : 90.000     torque(N-m): 0.280E+04
Eff induced: 0.5597      Eff ideal : 0.8897      rpm : 1500.000
Tnacel(N) : -0.7114     hub rad.(m): 0.0750     Tcoef : 0.5576
Tvisc(N) : -28.5880     Pvisc(W) : 0.157E+04    disp. rad. : 0.0750
rho(kg/m3) : 0.45922     Vsound(m/s): 303.280    mu(kg/m-s) : 0.1477E-04
-----
Sigma: 0.73871
          Ct: 0.85670    Cp: 3.07080    J: 1.97802
          Tc: 0.55758    Pc: 1.01041    adv: 0.62962

i  r/R  c/R  beta(deg)  CL    Cd  REx10^6  Mach  effi  effp  na.u/U
1  0.090 0.1606 89.45 0.231 0.0099 0.41 0.299 0.312 0.035 0.000
3  0.155 0.2049 84.74 0.465 0.0088 0.53 0.304 0.667 0.882 0.000
5  0.247 0.3097 79.84 0.623 0.0074 0.84 0.315 0.710 0.954 0.000
7  0.343 0.4062 76.73 0.780 0.0066 1.15 0.331 0.668 0.974 0.000
9  0.437 0.4690 74.29 0.926 0.0063 1.41 0.351 0.630 0.982 0.000
11 0.527 0.4994 72.25 1.051 0.0062 1.60 0.374 0.600 0.985 0.000
13 0.612 0.5041 70.46 1.152 0.0062 1.72 0.399 0.575 0.987 0.000
15 0.691 0.4889 68.80 1.227 0.0063 1.78 0.423 0.556 0.988 0.000
17 0.762 0.4576 67.18 1.279 0.0065 1.76 0.447 0.544 0.989 0.000
19 0.825 0.4130 65.58 1.314 0.0067 1.66 0.469 0.537 0.989 0.000
21 0.879 0.3575 64.16 1.339 0.0071 1.50 0.489 0.533 0.989 0.000
23 0.924 0.2929 63.18 1.362 0.0077 1.27 0.506 0.528 0.988 0.000
25 0.959 0.2218 62.67 1.391 0.0086 0.99 0.519 0.522 0.987 0.000
27 0.983 0.1464 62.49 1.436 0.0105 0.66 0.529 0.522 0.985 0.000
29 0.997 0.0722 62.46 1.505 s 0.0206 0.33 0.534 0.574 0.973 0.000
=====
Free Tip Vortex Wake Formulation Solution: 31000ft
no. blades : 5          radius(m) : 0.9100      Wake adv. ratio: 1.10490
thrust(N) : 0.256E+04    power(W) : 0.410E+06    adv. ratio: 0.62962
Efficiency : 0.5618      speed(m/s) : 90.000     torque(N-m): 0.261E+04
Eff induced: 0.5698      Eff ideal : 0.8911      rpm : 1500.000
Tnacel(N) : -0.6360     hub rad.(m): 0.0750     Tcoef : 0.5484
Tvisc(N) : -27.1394     Pvisc(W) : 0.151E+04    disp. rad. : 0.0750
rho(kg/m3) : 0.44276     Vsound(m/s): 301.756    mu(kg/m-s) : 0.1466E-04
-----
Sigma: 0.73871
          Ct: 0.84253    Cp: 2.96649    J: 1.97802
          Tc: 0.54836    Pc: 0.97609    adv: 0.62962

i  r/R  c/R  beta(deg)  CL    Cd  REx10^6  Mach  effi  effp  na.u/U
1  0.090 0.1606 88.62 0.209 0.0100 0.41 0.301 0.391 0.145 0.000
3  0.155 0.2049 83.91 0.428 0.0088 0.53 0.306 0.704 0.878 0.000
5  0.247 0.3097 79.01 0.584 0.0073 0.84 0.317 0.734 0.953 0.000
7  0.343 0.4062 75.90 0.741 0.0066 1.16 0.333 0.686 0.973 0.000
9  0.437 0.4690 73.45 0.886 0.0062 1.42 0.354 0.645 0.981 0.000
11 0.527 0.4994 71.41 1.011 0.0061 1.61 0.377 0.612 0.985 0.000
13 0.612 0.5041 69.63 1.111 0.0061 1.73 0.401 0.586 0.987 0.000
15 0.691 0.4889 67.97 1.186 0.0062 1.78 0.426 0.567 0.988 0.000
17 0.762 0.4576 66.34 1.238 0.0064 1.76 0.450 0.553 0.989 0.000
19 0.825 0.4130 64.74 1.273 0.0066 1.67 0.473 0.546 0.989 0.000
21 0.879 0.3575 63.33 1.299 0.0069 1.50 0.493 0.542 0.989 0.000
23 0.924 0.2929 62.35 1.323 0.0075 1.28 0.510 0.536 0.988 0.000
25 0.959 0.2218 61.84 1.352 0.0084 0.99 0.523 0.530 0.987 0.000
27 0.983 0.1464 61.65 1.398 0.0101 0.67 0.533 0.528 0.985 0.000
29 0.997 0.0722 61.63 1.478 0.0151 0.33 0.538 0.548 0.979 0.000
=====

```

```

=====
Free Tip Vortex Wake Formulation Solution: Service Ceiling
Wake adv. ratio: 1.12620
no. blades : 5 radius(m) : 0.9100 adv. ratio: 0.62962
thrust(N) : 0.232E+04 power(W) : 0.380E+06 torque(N-m): 0.242E+04
Efficiency : 0.5508 speed(m/s) : 90.000 rpm : 1500.000
Eff induced: 0.5591 Eff ideal : 0.8896 Tcoef : 0.5578
Tnacel(N) : -0.6142 hub rad.(m): 0.0750 disp. rad. : 0.0750
Tvisc(N) : -25.8872 Pvisc(W) : 0.142E+04
rho(kg/m3) : 0.39536 Vsound(m/s): 297.560 mu(kg/m-s) : 0.1437E-04
=====
Sigma: 0.73871
Ct: 0.85703 Cp: 3.07755 J: 1.97802
Tc: 0.55780 Pc: 1.01263 adv: 0.62962

i r/R c/R beta(deg) CL Cd REx10^6 Mach effi effp na.u/U
1 0.090 0.1606 89.46 0.231 0.0104 0.37 0.305 0.312-0.014 0.000
3 0.155 0.2049 84.75 0.466 0.0092 0.47 0.310 0.666 0.876 0.000
5 0.247 0.3097 79.85 0.624 0.0077 0.74 0.321 0.709 0.952 0.000
7 0.343 0.4062 76.74 0.781 0.0069 1.02 0.338 0.668 0.972 0.000
9 0.437 0.4690 74.29 0.927 0.0066 1.25 0.358 0.630 0.981 0.000
11 0.527 0.4994 72.25 1.053 0.0065 1.42 0.381 0.599 0.985 0.000
13 0.612 0.5041 70.46 1.154 0.0065 1.53 0.406 0.574 0.987 0.000
15 0.691 0.4889 68.80 1.229 0.0066 1.57 0.431 0.556 0.988 0.000
17 0.762 0.4576 67.18 1.282 0.0068 1.55 0.456 0.543 0.988 0.000
19 0.825 0.4130 65.58 1.317 0.0071 1.47 0.478 0.536 0.989 0.000
21 0.879 0.3575 64.16 1.342 0.0074 1.33 0.498 0.532 0.988 0.000
23 0.924 0.2929 63.18 1.366 0.0080 1.13 0.516 0.527 0.988 0.000
25 0.959 0.2218 62.67 1.394 0.0091 0.87 0.529 0.521 0.987 0.000
27 0.983 0.1464 62.49 1.439 0.0113 0.59 0.539 0.521 0.984 0.000
29 0.997 0.0722 62.46 1.506 s 0.0232 0.29 0.544 0.578 0.969 0.000

```

III.2 Matlab Script for Service Ceiling

```

Ps = 200; %fpm Thrust = 607; %lbf alt = 30000; %feet while Ps > 101 alt2 = alt *.3048; Cd = .05; [T, a, P, rho] =
atmosisa(alt2); %SI units rho = rho / 515.378; V = 295; %ft/s A = 278.85; %ft^2 Drag = Cd * .5 * rho * V^2 * A; %lbf
W = 10000; %lbs Ps = (V/W *(Thrust-Drag))*60 %fpm alt Thrust alt = alt + 100; Thrust = Thrust - 9.5/4.448; end

```

IV Cost Variables

Table 41: Variables Nomenclature

| Variable | Definition | Variable | Assumption |
|----------------------------|---|----------------------------|---|
| $F_{\text{difficulty}}$ | Difficulty Judgement Factor | N_{man} | # of Airplanes Produced to Production Standard in Manufacturing Phase |
| F_{CAD} | CAD Judgement Factor | $R_{\text{man prog 1989}}$ | Manufacturing Labor Rate for the Entire Program in 1989 |
| N_{rdte} | # of Airplanes Produced for the RDTE Phase | $R_{\text{t prog 1989}}$ | Tooling Labor Rate for the Entire Program in 1989 |
| $R_{\text{rate 1989}}$ | | F_{int} | Interior Cost Factor |
| N_{flight} | # of Flight Test Airplanes | t_{ft} | # of Flight Test Hours |
| N_{static} | # of Static Test Airplanes | F_{ftoh} | Overhead Factor for Production Flight Tests |
| F_{mat} | Materials Correction Factor | $U_{\text{ann flt}}$ | Annual Utilization in Flight Hours |
| $C_{\text{eng rdte}}$ | Cost per Engine during RDTE phase | F_{misc} | Cost of Miscellaneous Items Factor |
| $R_{\text{man rdte 1989}}$ | Manufacturing Labor Rate in RDTE Phase in 1989 | F_{OL} | Factor Which Accounts for the Cost of Oil and Lubricants |
| $R_{\text{t rate 1989}}$ | Tooling Labor Rate in RDTE Phase in 1989 | N_{yr} | Operating Years of the Airplane |
| R_{rdte} | RDTE Production Rate in Units per Month | R_{loss} | Annual Loss Rate |
| F_{obs} | Low Observable Factor | R_{crew} | Crew Ratio Per Airplane |
| F_{Tsf} | Cost Adjustment Factor for Test and Simulation Facilities | MHP_{flhr} | # of Maintenance Man hours per Flight Hour |
| F_{pro} | RDTE Profit Fraction | F_{disp} | Disposal Cost Factor |
| $F_{\text{fin rate}}$ | RDTE Finance Cost Fraction | Year | The Year |