# *Harpy* Design Report

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**Nomenclature**

| | |
|---|---|
| VTOL | Vertical takeoff and landing |
| V/STOL | Vertical/Short takeoff and landing |
| b | Wingspan, tip-to-tip |
| $S_w$ | Reference wing area |
| $S_v$ | Reference vertical tail area |
| $S_{fs}$ | Reference fuselage area |
| $AR_w$ | Wing aspect ratio |
| $AR_{ht}$ | Horizontal tail aspect ratio |
| $AR_{vt}$ | Vertical tail aspect ratio |
| c | Chord |
| $C_r$ | Chord at wing root |
| $C_t$ | Chord at wing tip |
| $C_m$ | Mean chord |
| e | Oswald Efficiency Factor |
| $f$ | Equivalent flat plate area |
| $\lambda$ | Taper ratio |
| $\Lambda LE$ | Leading edge sweep angle |
| $\Lambda_C/4$ | Quarter chord sweep angle |
| $X_{ac}$ | Aerodynamic center |
| $X_{cg}$ | Center of gravity |
| Cl | Coefficient of lift |
| $C_{lw}$ | Coefficient of lift with respect to angle of attack of the wing |
| $C_{lt}$ | Coefficient of lift with respect to angle of attack of the horizontal tail |
| $C_{lvt}$ | Coefficient of lift with respect to angle of attack of the vertical tail |
| $k_n$ | Wing Body interferences coefficient |
| $k_{rl}$ | Correction factor |
| L | Lift force |
| $l_f$ | Length of fuselage |
| $\rho$ | Density |
| t | Horizontal tail efficiency |
| v | Vertical tail efficiency |
| $\varepsilon$ | Induced angle of attack on the tail by the wing |
| V | Velocity |
| D | Drag force |
| p | Roll rate |
| $V_{ht}$ | Horizontal tail volume ratio |
| $V_{vt}$ | Vertical tail volume ratio |
| $C_L$ | Static roll stability coefficient |
| $C_n$ | Static yaw stability coefficient |
| $C_{core}$ | Work output coefficient, turbojet core |
| $C_{fan}$ | Work output coefficient, ducted fan |
| $M_0$ | Propulsion system input mach number |
| $M_9$ | Propulsion system output mach number |
| $T_0$ | Propulsion system input total temperature |
| $T_9$ | Propulsion system output total temperature |
| $f_f$ | Fuel fraction |
| $\eta_{fan}$ | Ducted fan efficiency |

| | |
|---|---|
| $\lambda$ | Heat capacity ratio for air |
| $\tau_r$ | Ram effect temperature ratio |
| $\tau_c$ | Compressor temperature ratio |
| $\tau_\lambda$ | Burner temperature ratio |
| $\tau_f$ | Fuel temperature ratio |
| $\tau_t$ | Turbine temperature ratio |
| $\pi_c$ | Compressor pressure ratio |

## I. Benchmarking, Design Requirements, & Stakeholders

AMONG military aircraft, the niche of close air support has not seen a proper update in upwards of four decades. Considering the recent decommissioning of the *Fairchild Republic A-10*, the United States and its allies have a military need for an aircraft that can replace the A-10, while having multirole capabilities. We seek to design such an aircraft, primarily by fulfilling the following design requirements: close air support capabilities, anti-infantry/anti-vehicle capabilities, V/STOL capabilities, reconnaissance capabilities, low attack speed, and high maneuverability.

### A. Benchmarking

Existing aircraft were analyzed first to establish more specific design requirements. Low altitude attack aircraft are relatively common, and thus many different design concepts were observed. Specifications for aircraft that closest fill the design requirements can be found in Table 1 below:

**Table 1: Benchmark Aircraft [1-6]**

| **Aircraft** | **F-35 *Lightning*** | **AV-8B *Harrier II*** | **A-10 *Thunderbolt II*** | **A-37 *Dragonfly*** | **Su-25 *Frogfoot*** | **EMB 314 *Super Tucano*** |
|---|---|---|---|---|---|---|
| **Role** | Multi-role combat, VTOL | Ground attack, VTOL | Close air support | Attack/counter insurgency | Close air support | Attack/counter insurgency |
| **Year of Service** | 2006 | 1985 | 1972 | 1964 | 1978 | 2003 |
| **Wingspan (m)** | 10.7 | 9.25 | 17.5 | 10.9 | 14.4 | 11.1 |
| **Wing Area ($m^2$)** | 42.7 | 22.6 | 47.0 | 17.1 | 33.7 | 19.4 |
| **Wing Shape** | Trapezoidal | Trapezoidal | Elliptical | Trapezoidal | Trapezoidal | Trapezoidal |
| **Length (m)** | 15.7 | 14.1 | 16.3 | 8.62 | 15.5 | 11.4 |
| **Max Takeoff Weight (kg)** | 29,900 | 9,415 vertical 14,100 rolling | 20,865 | 6,350 | 19,300 | 5,400 |
| **Powerplant** | Afterburning turbofan | Single Turbofan | Twin turbofan | Twin turbojet | Twin turbojet | Single turboprop |

| | | | | | | |
|---|---|---|---|---|---|---|
| **Combat Range (km)** | 1,239 | 556 | 400 | 740 | 750 | 550 |
| **Payload (kg)** | 8,200 | 4,200 | 7,260 | 1,800 | 4,400 | 1,550 |

## B. Chosen Technologies

The first and most critical design requirement to analyze is V/STOL capabilities. It was determined to be necessary as this aircraft would fill the role of close air support as well as counter insurgency. In these scenarios, versatility is a requirement, and VTOL gives an aircraft far greater ability to take off and land from non-standard environments. For this aircraft, it was determined that it needed to be able to take off and land on WASP class amphibious assault ships, similar to how the Harrier II has been used. This also restricts the aircraft's size, as it should be possible to fit many on a single ship[7]. For the purposes of this aircraft, it was designed to take up as much or less space than the *Harrier*, and the wingspan was limited to 12 meters at a maximum.

Similar to the Harrier, the modern F-35 implements a VTOL system. More critically, however, the F-35 can function as a reconnaissance aircraft. These capabilities are important, as in a combat situation, high accuracy of intelligence is critical to precision strikes[8].

Both aircraft can provide close ground support, but the aircraft purpose built for this is the A-10 Thunderbolt. With high survivability and good low altitude maneuverability, the A-10 is commonly used in situations where ground-based fire is an issue[9]. Similarly, the Frogfoot is a Soviet built aircraft that fills this same niche, at a slightly reduced scale[10]. From these close air support craft, our aircraft will improve upon the maneuverability to improve attack speed, improve attack precision, and improve climb speed and top speed in order to heighten evasiveness.

Lastly, the A-37 Dragonfly and the Super Tucano are both classified as "counter insurgency" aircraft, or COIN aircraft. They excel in irregular warfare conditions, and are usually light, rugged, and in smaller military operations. They are maneuverable, survivable, can take off and land on unconventional runways, and carry relatively light payloads[10]. From these aircraft, we seek to implement their low weight, high maneuverability, and low maintenance costs.

## C. Stakeholders

Given the design requirements, anticipated stakeholders are as follows:

- United States Department of Defense
- International militaries experiencing irregular conflict
- Military pilots
- Ground based troops

## D. Design Requirements and Tradeoffs

Using the above chosen technologies and benchmarked aircraft, the following design goals were established:

**Table 2: Initial Design Requirements [1-6]**

| Parameter | Target Value |
|---|---|
| Max take-off weight, rolling (kg) | 8,000 kg |
| Max take-off weight, VTOL (kg) | 7,000 kg |
| Take-off distance, rolling (m) | 1,000 m |
| Landing distance, rolling (m) | 1,000 m |
| Net Range (km) | 2,500 km |
| Combat Range (km) | 500 km |
| Payload (kg) | 1,200 kg |
| Cruise Speed | 0.8 Mach |
| Combat Speed | 0.3 Mach |
| Operational Ceiling | 10,000 m |
| Climb Speed | > 100 m/s |

V/STOL aircraft are often greatly limited in their capabilities due to the implementation of a vertical takeoff and landing system. These systems often require the aircraft to have reduced weight and payload when VTOL is necessary. This aircraft will likely require a reduced payload for vertical takeoffs.

Close air support aircraft often have a great deal of armor required, as flying in a combat zone can attract enemy fire which could disable the aircraft. Aircraft like the *A-10* and the *Su-25* have a great deal of armor around the cockpit to allow for safety in combat environments. However, the tradeoff here is higher weight. If the aircraft is required to have armor, that weight is lost in payload or fuel supply.

Similarly, close air support fighters need redundancy in their systems in case of malfunction or disability. The *A-10*, for instance, has redundant propulsion systems, hydraulic systems, navigational systems, and support systems. This is required for this aircraft as well.

For highly maneuverable aircraft, static stability should be neglected in favor of agility. Modern fighters are often designed outside of the regimes of static stability in order to improve their evasiveness. For this aircraft, we seek to design for near neutral stability in order to improve the maneuverability. This means, however, that an integrated flight controller is necessary to keep the aircraft controlled in steady flight.

### E. Technology Readiness Level

**Table 3: Technology Readiness Level [11-14]**

| System | TRL | Development Needed |
|---|---|---|
| Propulsion System | 5 | Without question, the least mature technology to be used in our aircraft is the propulsion and VTOL system. The engine selected for this project, a turbofan modified J-97, was never put into full production, and therefore it will naturally be a less mature technology. However, V/STOL systems have been becoming more common is fixed wing aircraft, and the technology is highly feasible.<br>Development needed: Major modifications, research, implementation |
| Aerodynamic Control Systems | 8 | Aerodynamic control systems of this nature have been proven to be effective in environments similar to that which this aircraft will experience. This includes autopilot features, trimming features, pilot feedback and responsiveness, and electrically or hydraulically controlled control surfaces.<br>Development needed: Specific adjustment and implementation |
| Landing Gear | 9 | Landing gear for aircraft of this size is thoroughly proven technology. Rough-operation landing gear are used on the A-10, the A-37, the EMB 314, and the SU-25.<br>Development needed: Implementation |
| Avionics/Instrumentation | 9 | Modern avionics are reliable systems, proven effective in both civil and military aircraft.<br>Development needed: Specific adjustment and implementation |
| Survivability Systems | 8 | Survivability systems, such as flares, fuselage armor, and redundant systems are used in most close air support craft, and have proven their effectiveness on the A-10 and SU-25.<br>Development needed: Adjustments, implementation |

| | | |
|---|---|---|
| Weapons Systems | 7 | Low weight precision bombs are currently used by aircraft such as the F-35, F-22, and F-15. For example, the GBU-39B Small Diameter Bomb is used on the F-15 and uses a global positioning system to precisely direct toward a target. Similarly, air to ground guns are used in many ground-attack aircraft. The guns used on the A-10 and the SU-25 are slightly too large for this design. A gun build specific to this aircraft would be preferred, but options currently exist that would serve this aircraft well.<br>Development needed: Specific adjustments, implementation |

## II. Sizing and Performance Estimation

### A. Geometric Parameters for Improved Aerodynamic Estimations

For the wing, the NACA 64A204 was selected. This is due to its performance on the F-16. Our wing shapes and our wingspans are both extremely similar and, while our two planes have different mission profiles, there were still some similarities that we wanted to use from that airfoil. Those similarities being that we wanted to have the maximum lift at the angle of attack (AoA) of that airfoil. While the F-16 is by far the faster airplane of the two, we still wanted to keep the ability to fly out of situations quickly as this airplane is not suited for air-to-air combat. Thus, allowing for a quick escape and a thinner airfoil was deemed necessary. Another good thing about the airfoil would be that it has a relatively low drag coefficient as well due to the 6% thickness ratio. This allows for a smaller wetted area reducing the pressure drag of the airfoil.

For the tail, a symmetrical airfoil profile was believed to be most advantageous because the net force acting on both the tails will change direction during flight depending on which direction the elevators and rudders are deflected. Symmetric airfoils also offer better stability at higher speeds and provide more predictable control characteristics. Furthermore, symmetric airfoils are less susceptible to stalling at high angles of attack than their cambered counterparts, which can be critical in certain combat situations. In addition, symmetric airfoils are typically easier to manufacture and maintain than cambered airfoils. The NACA 0010 and NACA 0009 airfoil profiles were chosen for the horizontal tails and vertical tail because they are both thinner designs that have higher performance at higher speeds. The NACA 0009 specifically has a high critical Mach number which makes it suitable for maintaining maneuverability at high speeds, which is important for vertical stabilizers. In addition, both airfoils provide a good lift-drag ratio.

For these calculations, the following equation as used to convert between airfoil lift coefficient and wing lift coefficient:

$$C_{L,\alpha} = \frac{C_{l,\alpha}}{1+\frac{C_{l,\alpha}}{\pi * AR}} * \frac{1}{\sqrt{1-M^2}} \quad (1)^{14}$$

Similarly, the following equation was used to estimate drag coefficient of each component:

$$C_D = \frac{f}{S_w} * \frac{C_L{}^2}{\pi * e * AR} \quad (2)^{15}$$

At cruise speed, the following aerodynamic values were calculated using the above equations:

**Table 4: Aerodynamic Performance Parameters**

| | Wing | Horizontal Tail | Vertical Tail |
|---|---|---|---|
| **Airfoil Type** | NACA 64A204 | NACA 0010 | NACA 0009 |
| **Zero Lift Coefficient** | 0.053 | 0 | 0 |
| **Max Lift Coefficient** | 1.9 | 1.1 | 1 |
| **Lift Slope Coefficient** | 2.34 | 3.53 | 1.67 |
| **Drag Coefficient** | 0.005 | 0.01 | 0.01 |
| **Moment Coefficient** | -0.042 | 0 | 0 |
| **Thickness Ratio (t/c)** | 6% | 10% | 9% |

Initially, three different tail configurations, the H-Tail, the single vertical tail and the twin vertical tail, were weighed and considered during the selection process. At first it was best decided to go with the twin vertical tail design, with a trapezoidal shape (same as the wings), for various reasons such as but not limited to how the twin vertical tails are known for reducing parasitic drag when compared to the single vertical tail [11]. Parasitic drag when traveling at low speeds is fine but at higher speeds can prove to be a major concern. In addition to also having improved rudder authority at low speeds due to a shorter rudder offset moment, the twin tails also reduce the shear force at the root, which are emphasized while performing combat maneuvers, which in turn makes the tail feel "lighter" on the structure even when being physically heavier, thus saving weight. Furthermore, the redundancy applications prevalent in the twin tail design can be an added benefit as this is a jet fighter that will be under fire when in use so if one tail were to get damaged, the aircraft can maintain flight. Also, the twin vertical tail design has a lower vertical wingspan, which decreases total height and can help combat radar detection, and overall are generally considered to be much more efficient than their single tail counterparts.

Since this is a small lightweight aircraft by design, using space and weight accordingly is extremely important. These design requirements limit the size and weight the tails can be, and because of this constraint the H-tail cannot be used because it takes up an abundance of space that can be better allocated and is quite heavy when compared to the single tail configuration.

However, after further consideration it was decided to go with the single vertical tail over the twin tails because structurally there was not enough space to support the twin tails properly. The single tail design itself has plenty of benefits, such as a lower physical weight, simpler design to construct and a lower maintenance cost.

Specific structural constraints were put on the tails to better fit the current fuselage/wing designs, such as the length of both the vertical ($L_{VT}$) and horizontal ($L_{HT}$) moment arms that were set at 8 meters in length away from the tip of the aircraft and about 2 meters behind the center of gravity, so they do not compromise the wings.

**Table 5: Typical Tail Volume Coefficients [16]**

| | Typical Values | |
|---|---|---|
| | Horizontal $c_{HT}$ | Vertical $c_{VT}$ |
| Sailplane | 0.50 | 0.02 |
| Homebuilt | 0.50 | 0.04 |
| General aviation—single engine | 0.70 | 0.04 |
| General aviation—twin engine | 0.80 | 0.07 |
| Agricultural | 0.50 | 0.04 |
| Twin turboprop | 0.90 | 0.08 |
| Flying boat | 0.70 | 0.06 |
| Jet trainer | 0.70 | 0.06 |
| Jet fighter | 0.40 | 0.07–0.12* |
| Military cargo/bomber | 1.00 | 0.08 |
| Jet transport | 1.00 | 0.09 |

Table 5 above shows the different volume coefficients for each tail for different types of aircrafts. It was used more as a reference to acquire and refer to the tail volume coefficients for the vertical ($c_{VT}$) and horizontal ($c_{HT}$) tails for a Jet Fighter Aircraft. The horizontal tail coefficient was used as a starting point meanwhile the vertical tail coefficient was used more as a reference and a guideline. This was the case because the correlation between wingspan to directional characteristics is generally weaker than the correlation of the chord to pitch characteristics.

**Table 6: Tail Aspect Ratio and Taper Ratio**[16]

| | Horizontal Tail | | Vertical Tail | |
|---|---|---|---|---|
| | A | λ | A | λ |
| Fighter | 3–4 | 0.2–0.4 | 0.6–1.4 | 0.2–0.4 |
| Sailplane | 6–10 | 0.3–0.5 | 1.5–2.0 | 0.4–0.6 |
| Others | 3–5 | 0.3–0.6 | 1.3–2.0 | 0.3–0.6 |
| T-tail | – | – | 0.7–1.2 | 0.6–1.0 |

Table 6 shows the different aspect ratios and taper ratios for each tail for different types of aircrafts. It was used more as a reference to refer to the aspect ratios (A) and taper ratios (λ) for the vertical and horizontal tails for a Jet Fighter Aircraft. The aspect and taper ratios were calculated and then compared to the table to see if said values fell within the acceptable range. The geometric parameters and specifications for the wing and tails are tabled below. Some values were decided on based on background research. As you can see since our aircraft has one vertical tail, it is not allowed to have any dihedral angle on the vertical tail.

**Table 7: Wing Geometry**

| | Notation | Formula | Dimension |
|---|---|---|---|
| **Span** | $b_w$ | Desired | 10m |
| **Area** | $S_w$ | $b(C_r + C_t)/2$ | 27.5m$^2$ |
| **Aspect Ratio** | $AR_w$ | $b^2/S_w$ | 3.64 |
| **Root Chord** | $C_{r_w}$ | Desired | 4.25m |
| **Tip Chord** | $C_{t_w}$ | Desired | 1.25m |
| **Mean Chord** | $\bar{C}_w$ | $(\frac{2}{3})C_r\frac{1+\lambda+\lambda^2}{1+\lambda}$ | 3.02m |
| **Taper Ratio** | $\lambda_w$ | $C_t/C_r$ | 0.294 |
| **Leading Edge Sweep Angle** | $\Lambda_{LE}$ | Desired | 31º |
| **Quarter Chord Line Sweep Angle** | $\Lambda_{C/4}$ | Desired | 24.2 º |
| **Aerodynamic Center** | $x_{ac}$ | $\frac{1+2\lambda}{1+\lambda} * \frac{b_w}{6} * \tan(\Lambda_{c/4})$ | 4.4m |
| **Geometric Twist** | $\alpha_{geo}$ | $a_{Tot} - (a_{L0_{root}} - a_{L0_{tip}})$ | -3.35º |
| **Total Twist** | $\alpha_{Tot}$ | $\frac{k_1 * C_{m_{root}} + k_2 * C_{m_{tip}} - C_L * S_t}{(1.4 * 10^{-5}) * AR^{1.43} * \Lambda_{c/4}}$ | -2.16º |
| **Downwash** | $\varepsilon$ | $\frac{2 * C_{L_w}}{\pi * AR_w}$ | 0.533º |

**Table 8: Tail Geometry**

| | Notation | | Formula | | Dimension | |
|---|---|---|---|---|---|---|
| | Horizontal Tail | Vertical Tail | Horizontal Tail | Vertical Tail | Horizontal Tail | Vertical Tail |
| **Span** | $b_{HT}$ | $b_{VT}$ | $\frac{2 * (C_{HT} * S_w * \bar{C}_w)}{L * (C_r + C_t)}$ | $\frac{2 * (C_{VT} * S_w * b_w)}{L * (C_r + C_t)}$ | 3.69m | 2.17m |

| | | | | | | |
|---|---|---|---|---|---|---|
| **Area** | $S_{HT}$ | $S_{VT}$ | $b(C_r + C_t)/2$ | $b(C_r + C_t)/2$ | 4.156m$^2$ | 4.125m$^2$ |
| **Aspect Ratio** | $AR_{HT}$ | $AR_{VT}$ | $b^2/S_{HT}$ | $b^2/S_{VT}$ | 3.28 | 1.14 |
| **Root Chord** | $C_{r_{HT}}$ | $C_{r_{VT}}$ | Desired | Desired | 1.75m | 1.5m |
| **Tip Chord** | $C_{t_{HT}}$ | $C_{t_{VT}}$ | Desired | Desired | 0.5m | 0.4m |
| **Mean Chord** | $\bar{C}_{HT}$ | $\bar{C}_{VT}$ | $(\frac{2}{3})C_r\frac{1+\lambda+\lambda^2}{1+\lambda}$ | $(\frac{2}{3})C_r\frac{1+\lambda+\lambda^2}{1+\lambda}$ | 1.24m | 1.06m |
| **Taper Ratio** | $\lambda_{HT}$ | $\lambda_{VT}$ | $C_t/C_r$ | $C_t/C_r$ | 0.286 | 0.267 |
| **Leading Edge Sweep Angle** | $\Lambda_{LE_{HT}}$ | $\Lambda_{LE_{VT}}$ | $\Lambda_{c/4} + 5^o$ | Desired | 36$^o$ | 35$^o$ |
| **Aerodynamic Center** | $x_{ac_{HT}}$ | $x_{ac_{VT}}$ | $\frac{1+2\lambda}{1+\lambda} * \frac{b}{6} * \tan(\Lambda_{c/4})$ | $\frac{1+2\lambda}{1+\lambda} * \frac{b}{6} * \tan(\Lambda_{c/4})$ | 8.62m | 8.36m |
| **Tail Dihedral Angle** | $\Gamma_h$ | $\Gamma_v$ | $tan^{-1}(\sqrt{\frac{S_{VT}}{S_{HT}}})$ | ---------------- | 44.89$^o$ | -------- |

### B. Drag Estimations

Ideally, we would like to have this aircraft be as fast as possible, and to do that we have to minimize the drag as much as possible. To do this, the following estimation was used:

$$(C_{D_0})_{\text{subsonic}} = \frac{\Sigma(C_{f_c} FF_c Q_c S_{\text{wet}_c})}{S_{\text{ref}}} + C_{D_{\text{misc}}} + C_{D_{\text{L\&P}}} \tag{3}$$

However, there are many changes in form factors and friction coefficients from fuselage to wing to engine housing as well. On the other hand, there were a few variables that stayed the same. For instance, Q stayed mostly the same at 1. Most of the airplane didn't have too much interference drag, and much of the time, that variable stayed the same. For engine housing, FF was described as:

$$FF = (.9 + \frac{5}{f^{1.5}} + \frac{f}{400}) \tag{4}$$

Where *f* is described as a fineness ratio by Raymer and in this situation for fighters, he said 6 was an appropriate number to use. For wings and tails, the FF was shown to be:

$$FF = (1 + \frac{.6}{\frac{x}{c}} * \frac{t}{c} + 100 * \left(\frac{t}{c}\right)^4) * (1.34 * M^{.18} * \cos(\Lambda)^{.28} \tag{5}$$

The drag coefficient, $C_f$ was shown to be:

$$C_f = \frac{.0594}{Re^{.2}} \tag{6}$$

Using these it was estimated that the wings had a drag coefficient of .00671, the fuselage had a drag coefficient of.002, the engines had a combined drag coefficient of 0.0084, the vertical tail had a drag coefficient of .0046 and the horizontal tails had a combined drag coefficient of 0.00958.

### C. Powerplant Estimations

The thrust requirements are determined at two operation stages that require the most thrust. The first is during the vertical takeoff stage where all the weight of the aircraft needs to be lifted and controlled by engine thrust alone without aid of aerodynamic forces. The second stage is during a high-speed cruise where the aircraft reaches a max velocity of 1.5 Mach. The maximum take-off weight of the aircraft is to be determined at 6,500kg, the gravitational force will be 63.74kN. The thrust requirement for the vertical take-off stage is dependent on the gravitational force, adding 20% extra thrust force and rounding up that number, a 80kN thrust is required to achieve vertical takeoff and landing. The desired top speed is Mach 0.95 or 323.3 m/s. From the equation below a rough estimation can be made for thrust.

$$F_{Thrust} = F_{Drag} + F_{lift}. \text{ And } F_{Lift} = \text{Weight of Aircraft} \tag{7}$$

Using the lift coefficient of 0.053 from wing design 2 and a drag coefficient of 0.005. Cruising at 323.2755m/s with maximum weight of 63.74kN at 11000m with air density of 0.2978 kg/m^2, the wing area is 27.5 m$^2$. The required thrust is 79.183kN and adding 20% extra thrust force considering different weather conditions the final required thrust is 86kN.Within the above two stages, the high-speed cruising requires the most thrust, thus the thrust requirement will be based on the calculated max thrust of 86kN.

### D. Engine Selection

Two engines were researched in detail: RB-162-86 and J97/LCF459. RB-162-86 is a turboprop engine designed by Rolls Royce specifically for lightweight VTOL aircraft weighing in only 126 kg and providing a thrust of 23kN. J97/LCF459 is a turbojet engine developed by GE and can provide a max thrust of 75kN and weighing in 386 kg [12].

The J97/LCF459 engine seems to be the best fit for the design requirements, however, this engine has a large fan diameter coming in at 1.5m which will be difficult to be integrated in a small airframe. The RB-162-86 is much smaller with a fan diameter of 0.635m, but the low thrust output means multiple engines are required, so the design requirements need to be modified. For this report, the J97/LCF engine will be selected for future development of the Baby F35. Future designs with modified airframe or tri-mounted RB-162-86 versions will be benchmarked and tested [12].

With the J97/LCF engine selected, a dual engine setup will provide a max thrust of 150kN. With the thrust defined, another calculation is done to estimate the lowest altitude at which the top speed can be achieved.

The result demonstrates that the aircraft can travel at Mach 1.5 when air density is below 0.26169kg/m^2 around 11750m. Due to the constraint of Turbofan engine, Mach 1.5 is a speed that is only theoretical, and the real top speed of the aircraft will be capped at Mach 0.952. In real mission profiles, the engine will be operated close to sea level and cruises with 70% throttle. The turbofans will be kept off max throttle to avoid extra fuel consumption.

### E. Fuel Consumption and Mission Profile

To determine how much fuel this aircraft would consume in flight, it was necessary to create a specific thrust profile of this engine. Given that this engine uses a turbojet to power an external ducted fan, it was modeled as a turboprop system. A diagram of the engines used in this aircraft is given below:

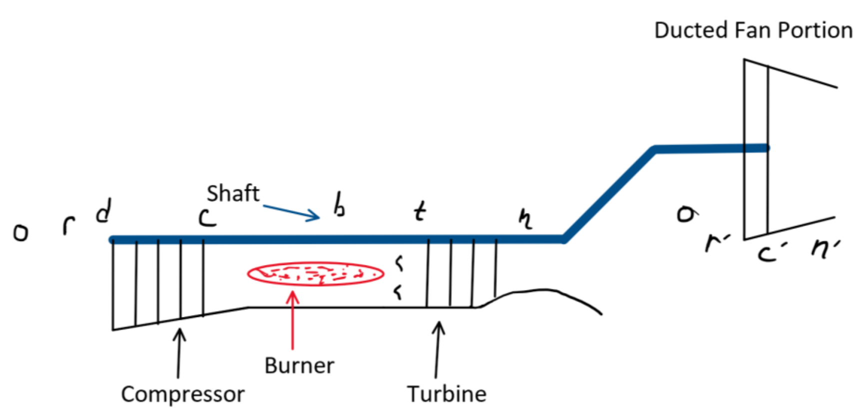



**Figure 1: Cross Section of Propulsion System**

The equations for work output coefficients of this engine are given below:

$$C_{core} = (\gamma - 1)M_0{}^2\{(1-f)\frac{M_9}{M_0}\sqrt{\frac{T_9}{T_0}} - 1\} \tag{8}$$

$$C_{fan} = \eta_{fan}\{(1-f_f)\tau_\lambda(1-\tau_t) - \tau_r(\tau_c - 1)] \tag{9}$$

where:

$$\frac{M_9}{M_0} = \sqrt{\frac{\tau_r \tau_c \tau_t - 1}{\tau_r - 1}} \tag{10}$$

$$\tau_\lambda = \frac{T_{t,4}}{T_0} \tag{11}$$

$$\tau_c = \pi_c^{\frac{\gamma-1}{\gamma}} \tag{12}$$

$$\frac{T_9}{T_0} = \frac{\tau_\lambda}{\tau_r \tau_c} \tag{13}$$

$$\tau_r = 1 + \frac{\gamma - 1}{2} {M_0}^2 \tag{14}$$

$$\tau_t = \frac{1}{\tau_c \tau_r} + \frac{\tau_r - 1}{{\eta_{fan}}^2 \tau_\lambda} \tag{15}$$

$$f_f = \frac{\tau_\lambda - \tau_r \tau_c}{\tau_f - \tau_\lambda} \tag{16}$$

$$\tau_f = \frac{h_f}{c_p T_0} \tag{17}$$

Using the drag estimations and the above equations, the total range at cruise was determined to be 2,900 km. However, because this is a ground attack vehicle primarily, the attack range and loiter time are more important parameters than the maximum possible range. Therefore, assembling a general mission profile and calculating fuel consumption percentages to each section is necessary.

An outline of the mission profile is shown below in both *Figure x* and *Table x*.

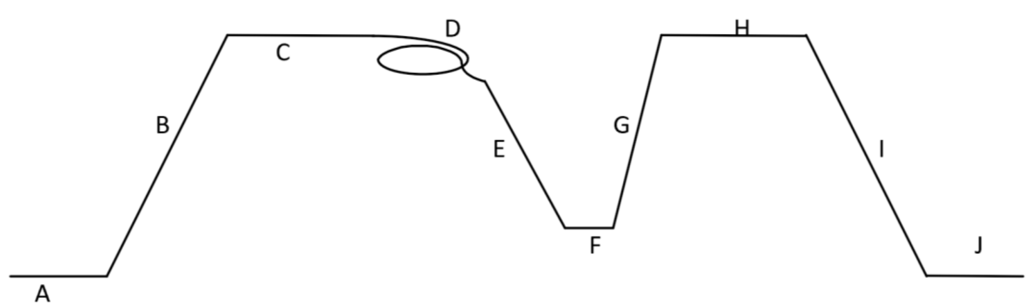



**Figure 2: Mission Profile**

**Table 9: Mission Profile Outline**

| Section | Purpose | Altitude (m) | Travel (km) | % Fuel Consumed |
|---|---|---|---|---|
| A | Take-off | 0 | 0.5 | 1.7 |
| B | Initail Climb | 50-10000 | 1 | 6.7 |
| C-D | Cruise-Loiter | 10,000 | 500 | 17.2 |
| E | Descent | 10,000-100 | 5 | ~0 |
| F | Ground-attack | 100 | 350 | 35.6 |

| | | | | |
|---|---|---|---|---|
| G | Fast Climb | 100-10,000 | 0.5 | 11.1 |
| H | Cruise II | 10,000 | 500 | 17.2 |
| I | Descent | 10,000-0 | 5 | ~0 |
| J | Landing | 0 | 0.5 | ~0 |

## F. VTOL System

Modern VTOL system vertical takeoff systems such as the one on the F35 consist of one lift fan and vector thrust capability from the engine. Older systems such as the one on the Harrier use 4 dedicated thrust vectoring nozzles on the side of the aircraft without a dedicated main nozzle.

The VTOL system with a lift fan is simpler and more efficient, however, it is difficult to fit in a small airframe. The VTOL system on the harrier is more efficient, but more complicated and cannot achieve the equivalent performance as the aircraft with a main engine nozzle. The VTOL system for the Baby F35 will be a combination of both the harrier and F35. The lift fan will be removed due to the size constraint, two dedicated thrust vector nozzles will be added under the wing on the fuselage and the two main nozzles for the J97 engine can vector thrust like the one on the F-35 [13].

The thrust ratio between the front and rear is set to around 35 to 65. This was calculated using the center of mass of the aircraft.

## G. Subsystem Estimations

1. **Aerodynamic Control Surfaces**
    a. Elevator
    b. Aileron
    c. Rudder

This subsystem includes the three most common aerodynamic control surfaces, elevator, ailerons, and rudder. The elevator is positioned on the trailing edge of the horizontal tail and can be moved to alter the air flow, resulting in the generation of a pitch moment on the aircraft. A pair of movable flaps known as ailerons are situated on the trailing edge of each wing, with differential operation causing one flap to deflect up while the other moves down, or vice versa, thus generating a roll moment by modifying the air flow over the wings. The rudder is a movable flap situated on the trailing edge of the vertical tail and can be deflected to alter the air flow over the vertical tail, resulting in the generation of a yaw moment on the aircraft [14].

2. **Aircraft Propulsion Systems**
    a. Engine
        i. (J97/LCF459) Max thrust 75kN
    b. VTOL thrust vector
        i. For takeoff/land on *WASP* class amphibious assault ships
    c. Fuel storage and supply
    d. Nozzle

The components that make up the aircraft engines, along with the related fuel tanks and hardware, are known as the propulsion systems. These systems are responsible for generating a thrust force that propels the aircraft while it is in flight. Specifically, for Baby F35, VTOL system and main nozzle will also be included since these two components will be integrated into the selected turbojet engine (J97/LCF459).

3. **Aircraft Structural Systems**
    a. Wing (wing tips)
    b. Fuselage
    c. Horizontal/Vertical tail
    d. Engine compartment
    e. Landing
    f. Weapon attachment
    g. GAU-19 gun

   h. Mark 81 Bomb w/ Enhanced Paveway II Enhanced Computer Control Group (ECCG)
   i. AGM-176 air to surface missile

This subsystem includes multiple structural systems of a fixed-wing aircraft, including the wings, fuselage, horizontal and vertical tails, and aerodynamic control surfaces, as well as additional subsystems for engine and weapon attachment and landing gear. The design of each of these subsystems must account for the various forces and moments that result from gravity, propulsion, and aerodynamic loadings, with aerodynamic forces and moments being the most significant [14].

4. **Air Data and Flight Instrumentation**
   a. Pressure & Airspeed sensor
   b. Engine monitoring system
   c. Gyroscopic instrumentation

The air data and flight instrumentation subsystem include flight instruments which incorporate sophisticated electronics and are collectively known as avionics. They serve a crucial role in measuring key flight properties and performance metrics. And redundant mechanical systems are ready to provide information if the electronic system fails. Through the provision of valuable data, pilots are able to effectively operate the aircraft with greater precision and control.

5. **Computer systems**
   a. Guidance and positioning (GPS)
      i. Flight monitoring computer
   b. Control
      i. Flight control computer
   c. Defense avionics
      i. AN/AAR-57 Ultraviolet Based Missile Warning System

The flight computer system is an indispensable and commonplace feature. These systems perform a range of crucial functions, including guidance, navigation, and control, through the use of embedded computer systems as part of an integrated avionics package. A dedicated flight control computer is often available to perform high-level planning computations, such as routing, automatic pilots, and flight management tasks. Another flight control computer converts analog input from the pilot to digital signals, with the aid of software, the digital signal is then used for control [14].

6. **Aircraft Pilots**
   a. Survival
   b. Ergonomics
   c. Pilot

This subsystem includes the physical size requirements for the human pilot, safety and ergonomics. Features include an augmented reality system that projects information directly onto pilot's line of sight, custom fitted helmet, and ejection seat.

7. **Integration of Flight Systems**
   a. Hydraulic control system (Aerodynamic control surfaces)
   b. Electronic system (Propulsion, Flight instrument, and Avionics)

The complete aircraft is a composite of various interdependent flight subsystems. This is where the integration of the flight subsystems comes in; it controls the effective functioning of all the other subsystems ensuring optimal aircraft performance. The hydraulic control system controls the aerodynamic surfaces and VTOL system; it is linked to the flight control computer, including a redundant system that gives the pilot full manual control. Electronic systems control fuel injection and throttle response for the propulsion system, as well as flight instruments and flight computers. Backup integration systems are in place to prevent loss of control if an electronic system fails.

8. **Emergency Power**

Emergency power is integrated into the overall propulsion system. Each turbofan can provide 75kN of thrust which on its own is enough to keep the aircraft airborne. In the event when the two Turbofans outside of the aircraft are damaged or fail. The two J97 jet engines located inside the airframe can disconnect the turbofans and provide a total of 42kN thrust to help the aircraft glide towards safety. The weight of the emergency power is also considered inside the overall propulsion system and each J97 engine weighs 315kg, for a total weight of 630kg.

### H. Stability Estimations

When looking at the stability of this airplane, we wanted to make a plane that was extremely precise in its encounters with good control of its movements. Thus, making everything really close to neutral as possible was a way to make this a reality. As the following table shows, the plane is stable but only by the slightest margins. This would allow for greater control authority over its pitch, roll, and yawing motions. When calculating the stability in pitch these equations were considered:

$$C_{m_\alpha} = C_{L_{\alpha_w}}\left(\frac{x_{cg}}{\bar{c}} - \frac{x_{ac}}{\bar{c}}\right) + C_{m_{\alpha_f}} - \eta V_H C_{L_{\alpha_t}}\left(1 - \frac{d\varepsilon}{d\alpha}\right) \tag{18}$$

$$C_{m_0} = C_{m_{0_w}} + C_{m_{0_f}} + \eta V_H C_{L_{\alpha_t}}(\varepsilon_0 + i_w - i_t)$$ (19)[14]

When calculating the stability in yaw, the following equation was considered:

$$C_{n_{\beta_t}} = V_v \eta_v C_{L_{\alpha_t}}\left(1 + \frac{d\sigma}{d\beta}\right)$$ (20)[14]

$$C_{n_{\beta_{wf}}} = -k_n k_{Rl} \frac{S_{fs} l_f}{S_w b}$$ (21)[14]

When calculating the stability in roll, the following equation was considered:

$$C_{\mathcal{L}_\beta} = \left(\frac{C_{\mathcal{L}_\beta}}{\Gamma_w}\right)\cdot \Gamma_w - \frac{C_L}{4}\cdot \tan\Lambda_{w c/4} + (C_{\mathcal{L}_\beta})_{position} - \eta_V \cdot \left(1 + \frac{d\sigma}{d\beta}\right)\cdot \frac{S_{VT}}{S_w}\cdot \frac{z_v}{b}\cdot C_{L_{\alpha VT}}$$ (22)[14]

A summary of stability estimations can be found below in Table 10

**Table 10: Summary of Stability Coefficients**

| Direction | Stability Coefficients | Analysis |
|---|---|---|
| **Pitch** | $C_{M,\alpha CG} = 1.49$ <br> $C_{M,0} = 7.16$ | Statically unstable in pitch due to $C_{M,\alpha CG}$ being greater than zero |
| **Roll** | $C_{\mathcal{L},\beta} = -0.869$ | Statically stable in roll, but very close to neutral stability. This will allow for greater control authority over roll. |
| **Yaw** | $C_{N,\beta} = 0.0834$ | Statically stable in yaw, but very close to neutral stability. This will allow for greater control authority over yaw. |

When considering the dynamic stability, the longitudinal and lateral dynamic stability were both analyzed. When performing the lateral dynamic stability analysis, the spiral mode was found to be essentially zero. This means that in its current state, the aircraft will have neutral responses to spiral mode. The Dutch roll was then analyzed and when looking at those characteristics, the frequency was .00157 Hz. From this calculation, it was found that the damping was 1.59.

### I. Flight Envelope

Close Air Support (CAS) and Anti-Infantry/Anti-Vehicle Capabilities: The primary aerodynamic coefficients that are relevant to these design requirements are lift (L) and drag (D). To provide close air support and anti-infantry/anti-vehicle capabilities, the aircraft should have a high lift-to-drag ratio (L/D). This means that the aircraft should generate a high lift force while also minimizing drag.

VTOL Capabilities: To achieve vertical and short takeoff and landing (V/STOL) capabilities, the aircraft should have a high thrust-to-weight ratio (T/W). The aircraft's engines should provide enough thrust to overcome the weight of the aircraft and generate lift.

Reconnaissance Capabilities: Reconnaissance capabilities require the aircraft to have a high range and endurance. This means that the aircraft should have a low fuel consumption rate (specific fuel consumption, SFC) and a high lift-to-drag ratio (L/D) to maximize its range.

Low Attack Speed and High Maneuverability: Low attack speed and high maneuverability require the aircraft to have a high lift coefficient (CL) and a high roll rate (p). The lift coefficient determines the amount of lift generated by the aircraft's wings at a given angle of attack, while the roll rate determines the aircraft's ability to roll and turn quickly.

For moment coefficients, military aircraft designed for close air support and anti-infantry/anti-vehicle capabilities require a high level of maneuverability and responsiveness, which may require a neutral or slightly negative moment coefficient. However, aircraft designed for reconnaissance and surveillance may require a more stable configuration, with a positive moment coefficient to ensure smooth and steady flight.

**Table 11: Important Velocities and Load Factors**

| | Notation | Formula | Value |
|---|---|---|---|
| **Stall Speed** | $V_s$ | $\sqrt{\frac{2\, W_{MTOM}}{\rho_0 * c_{Lmax} * S_w}}$ | 47.89 m/s |
| **Maneuvering Speed** | $V_A$ | $V_S * \sqrt{n_1}$ | 126.7 m/s |
| **Cruise Speed** | $V_C$ | $4.77\sqrt{\frac{W_{MTOM}}{S}}$ | 272 m/s |
| **Design Dive Speed** | $V_D$ | $1.4 * V_C$ | 380.8 m/s |
| **Demonstrated Dive Speed** | $V_{DF}$ | $1.55 * V_C$ | 421.6 m/s |
| **Never Exceed Speed** | $V_{NE}$ | $0.9 * V_D$ | 342 m/s |
| **Minimum Load Factor** | $N_2$ | Desired | -2.8G |
| **Maximum Load Factor** | $N_1$ | Desired | 7G |

Shown below in figure 3 is the V-n diagram that illustrates the range of airspeeds and normal load factors at which the aircraft can safely operate in.

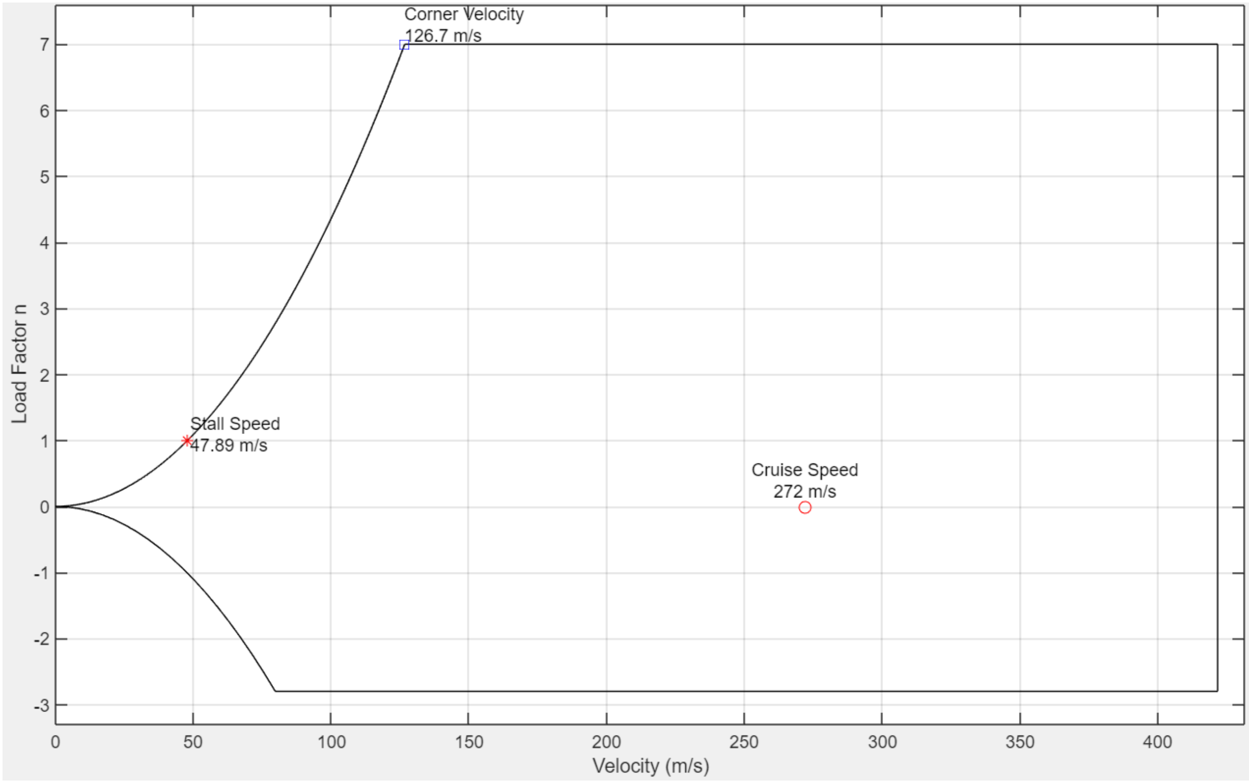



**Figure 3: V-n Diagram, stall speed, cruise speed, and corner velocity labeled[17]**

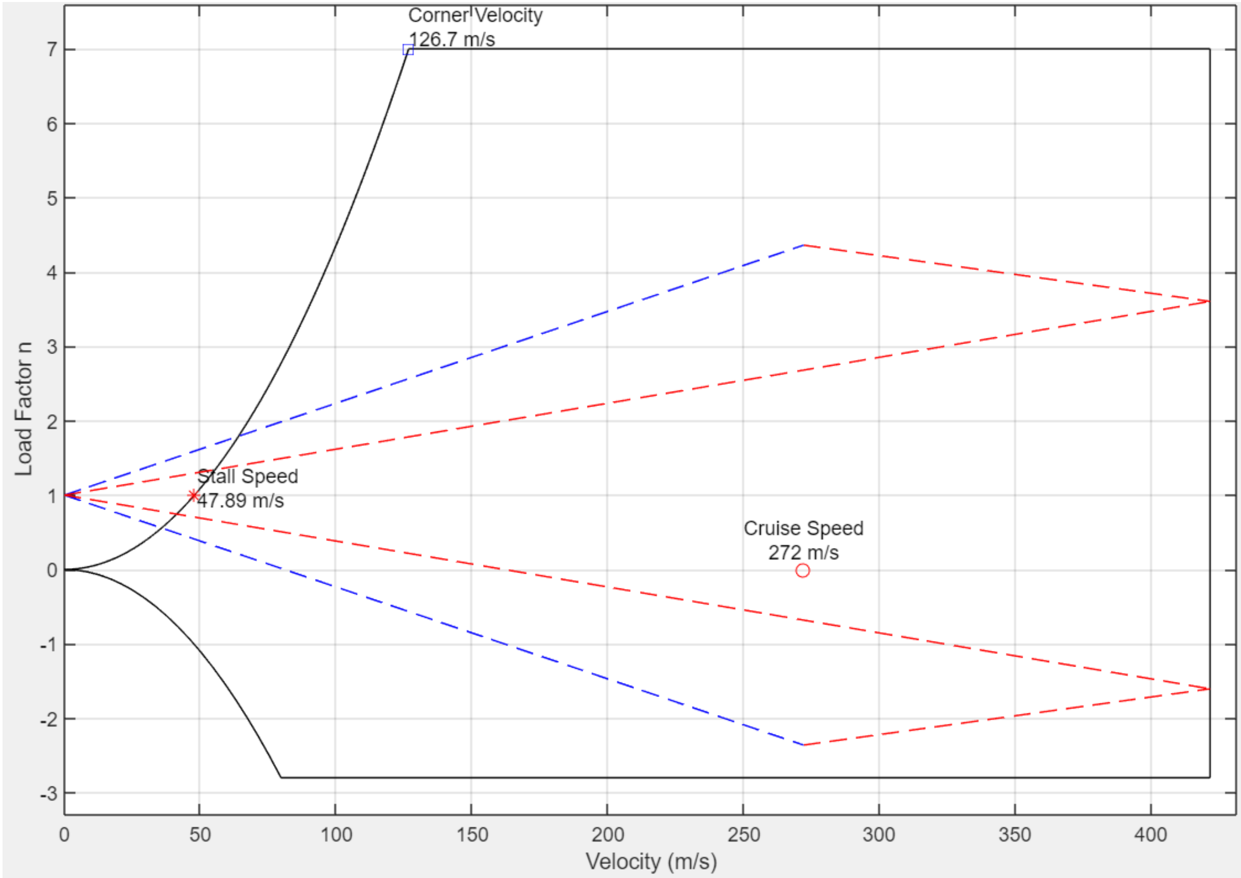


**Figure 4: Gust Envelope, stall speed, cruise speed, and corner velocity labeled**[17]

In Figure 4, the blue dashed line represents the limit of vertical gust during cruise and the red dashed line indicates the limit of vertical gust at dive speed.

### J. Wing Twist

When we look at this airplane, we want to create an airframe that has some give and can bend when the situation demands, but we also want to make a durable aircraft that When calculating total wing twist, there are three factors that must be looked at. There is the aerodynamic twist, there is the geometric twist and there's the induced AoA denoted by $\alpha_{L=0}(y)$, $\alpha(y)$, and $\alpha_i(y)$ respectively. These three angles give the total wing twist of the aircraft. The following equations are used when calculating the separate angles and the full wing twist as well:

$$\alpha_{total} = \alpha(y) - \alpha_i(y) - \alpha_{(L=0)} \tag{23}$$

$$\alpha_{(L=0)} = \int_0^{\pi} \frac{dz}{dx}(\cos\theta - 1)d\theta \tag{24}$$

$$\alpha_i = \frac{\Gamma_0}{2 * b * V_\infty} \tag{25}$$

$$\Gamma_0 = \frac{L}{\frac{\pi}{4} * \rho * V_\infty * b} \tag{26}$$

When using all these equations, the $\alpha_{total} = 2.16°$, $\alpha_{L=0} = 1.19°$, $\alpha_i = .00053°$ and $\alpha(y) = 3.35°$.

### K. Cost Estimations

We thought it to be beneficial to compare our aircraft's cost to some of our benchmarked aircrafts for comparison purposes. This aircraft will be relatively high tech compared to the *Dragonfly* or A-10, but its small size in comparison to the F-35 will likely put material costs lower.

**Table 12: Benchmark Aircraft Cost**

| Benchmark | Cost |
|---|---|
| A-10 Thunderbolt II | $18,800,000 |
| Cessna A-37 Dragonfly | $161,000 |
| F-35 Lightning | $75,000,000 |

**Table 13: RDT&E Cost**

| | Formula | Cost |
|---|---|---|
| **Devel Support Cost** | $67.4W_e^{0.630}W^{1.3}$ | $107,075,459 |
| **Flight Test Cost** | $1947W_e^{0.325}V^{0.822}FTA^{1.21}$ | $20,196,206 |
| **Manufacturing Materials Cost** | $31.2W_e^{0.921}V^{0.621}Q^{0.799}$ | $5,404,073 |
| **Engine Production Cost** | $3112[9.66T_{max} + 243.25M_{max} + 1.74T_{turbineinlet} - 2228]$ | $2,047,038 |
| **RDT&E + Flyaway** | Sum of Costs * Labor Rates | $540,204,844 |

After calculating the operations and maintenance costs, our aircraft came out to be roughly $22 million dollars, slightly more expensive than the A-10 however significantly cheaper than the F-35. Table 13 lists out all the values of importance regarding estimating the operations and maintenance costs of this aircraft. It is worth noting that a few assumptions were made. For military aircrafts, it is typical to assume that the maintenance cost will be roughly equal to the crew salaries and the fuel cost, meaning maintenance cost is about 50% of the total cost. Fuel cost tends to be about 15% and the crew's salaries are normally 35% of the total cost. About the fuel cost, the fuel selected was LA Energy's JAA Petroleum Oil which came at a price of $3.88 per gallon ($14.69 per kg). Fuel rate was then found to be 0.17 kg/s based on the fuel weight and time of flight. Also, it was assumed that the average yearly flight hours would be 400.

To calculate the crew salaries, it was assumed to be a 3-man crew. For every block hour, each crew member's wage was $600. Then the following adjustments were made to convert from passenger to military aircrafts. The manned maintenance hours per flight hour was assumed to be 12.5. The pilot to crew ratio was taken at 1.1. Also, the block hours per flight hours were estimated at 500.

**Table 14: Operation and Maintenance Cost**

| | Formula | Cost (per year) |
|---|---|---|
| **Fuel Cost** | Fuel Burned per year * Fuel Price from Vender | $3,595,475.18 |
| **Crew Salaries** | $100 * (V_C * \frac{W_0}{10^5})^{.3} + 237.2$ | $7,422,747.33 |
| **Maintenance Cost** | Fuel + Crew | $11,018,222.51 |
| **Total O&M Cost** | Fuel + Crew + Maintenance | $22,036,445.02 |

## L. Serviceability and Maintenance

Taking our benchmark aircrafts into consideration, the Harpy will undergo depot-level maintenance intermittently. Depot-level maintenance can be costly and time-consuming, and aircraft are unavailable to operators while the maintenance occurs. Nevertheless, adequate depot-level maintenance is essential to ensuring an aircraft's safe operation and capability to perform missions. The extent of such maintenance can also influence whether an aircraft's life can be extended cost-effectively.

# III. Final Design and Engineering Drawings

## A. Analyzed Conceptual Designs

<u>Fuselage Design</u>

Initially, due to its role on WASP class carriers, the size of the aircraft was constrained to a 10 meter by 10-meter box when seen from a Top View. Therefore, the length of the fuselage was constrained to a length of 10 meters. From the constrained fuselage length other fuselage dimensions were estimated based on benchmarked ratios. The vertical max fuselage width was set at 2m, and the horizontal max fuselage width was set at 2.5 meters.

However, after further evaluation it was decided that the initial constraint box that was set was not as important as it first appeared. Therefore, the wingspan was later modified on the chosen design so that the same aerodynamic calculations done on the reference wing would not be affected by the surface area covered by the fuselage.

In collaboration with the Propulsion team the sizing of the Engine and VTOL mechanism housing was decided. The Engine Housing was designed to have a 7.25-meter length and a 1.25m cross sectional diameter at its max width.

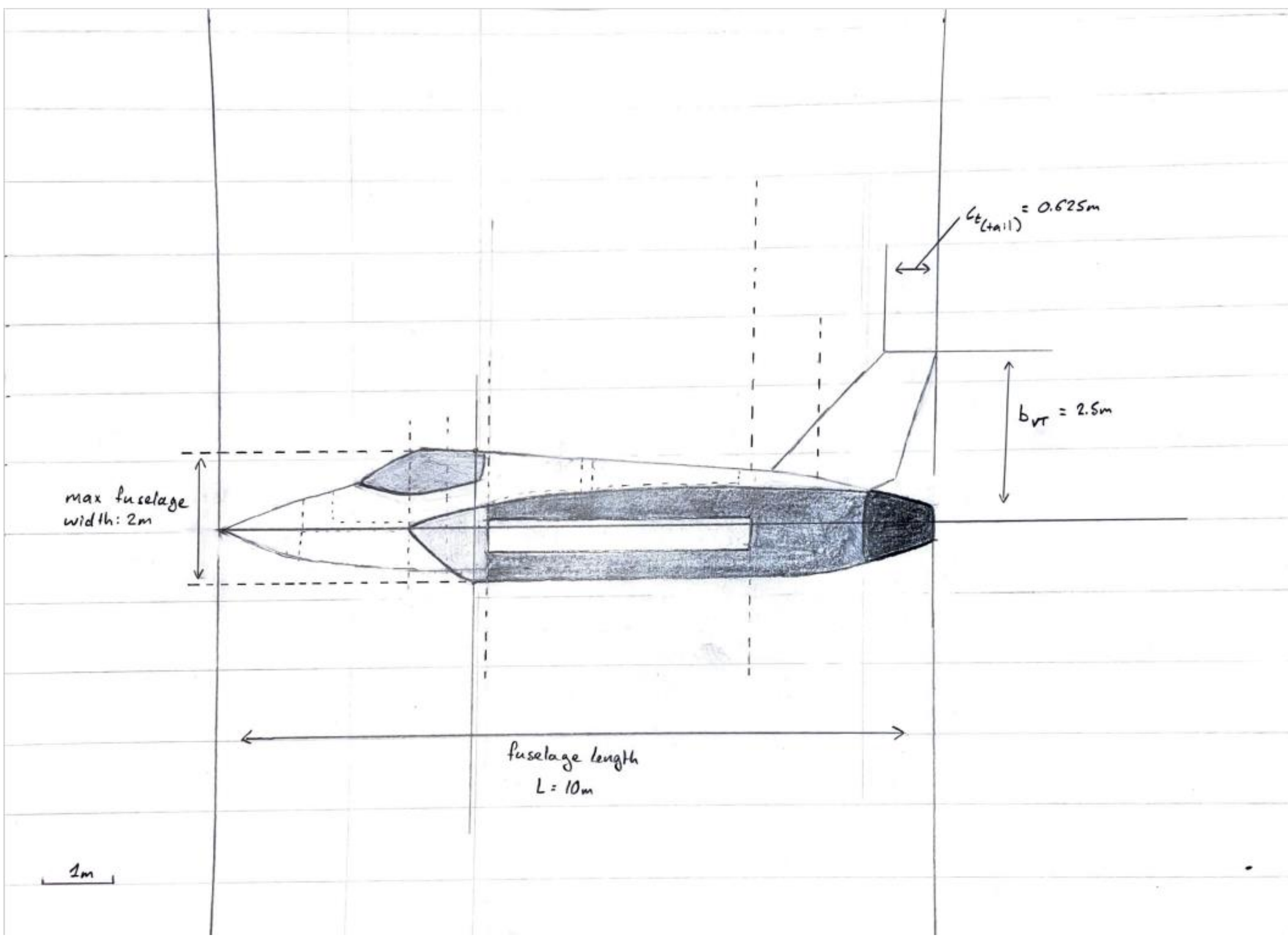



**Figure 5: Side Profile of Conceptual Fuselage Design**

Wing Tail/Design

Different Aircraft within this class were benchmarked to design a wing/tail configuration. Two Wing/Tail configurations were chosen and analyzed in order to compare and choose which best fits our aircraft design. The design requirements set for the aircraft require it to have the ability to fly at low speeds, while also having the ability to fly at higher subsonic speeds.

The Wing area was maximized for each wing design within the 10 meter by 10 meter reference sizing constraint box. In addition, the exact area that will be taken by the fuselage from the center of the reference wing, and how this will affect the Coefficient of Lift of the wing was still unknown, therefore adding more wing surface area to the initial design was the safer option. The two designs that were looked at were a High Aspect Ratio Design and a Low Aspect Ratio Design.

The first wing / tail configuration that was analyzed used a wing that had a backward swept Leading Edge and Trailing Edge. The reference wingspan was maximized at 10 meters in order to maximize the wing area. Because of the relatively small average chord compared to the second design, the wing had a larger Aspect Ratio. This would give the wing a better performance during low speeds compared to the low aspect ratio design. The distance between the fuselage tip and the Aerodynamic center of the wing was set at 4 meters, and the distance between the fuselage tip and the Aerodynamic center of the Horizontal Tail was set at 8m. The second wing/ tail configuration was similar to the first design however does not have a swept trailing edge. The reasoning for this design was to analyze a scenario where we would have a larger wing surface area. The increased surface area of the wing results in a smaller aspect ratio, which in turn makes the aircraft more maneuverable. Wings that have a lower aspect ratio tend to be thicker than wings with a high aspect ratio which also provides the ability to store more fuel inside the wings. Having more fuel stored in the wings versus having the fuel stored in the fuselage provides the benefit that the weight added by the fuel was closer to the center of gravity, therefore fuel burn will not shift the center of gravity as much as in the first design. Another benefit of not having a trailing edge sweep was that the efficiency of the trailing edge flight controls increases. This leads to smaller control surfaces, therefore having more space inside the wing for internal components.

The distance between the fuselage tip and the Aerodynamic center of the wing was set at 5 meters, and the distance between the fuselage tip and the Aerodynamic center of the Horizontal Tail was set at 8m.

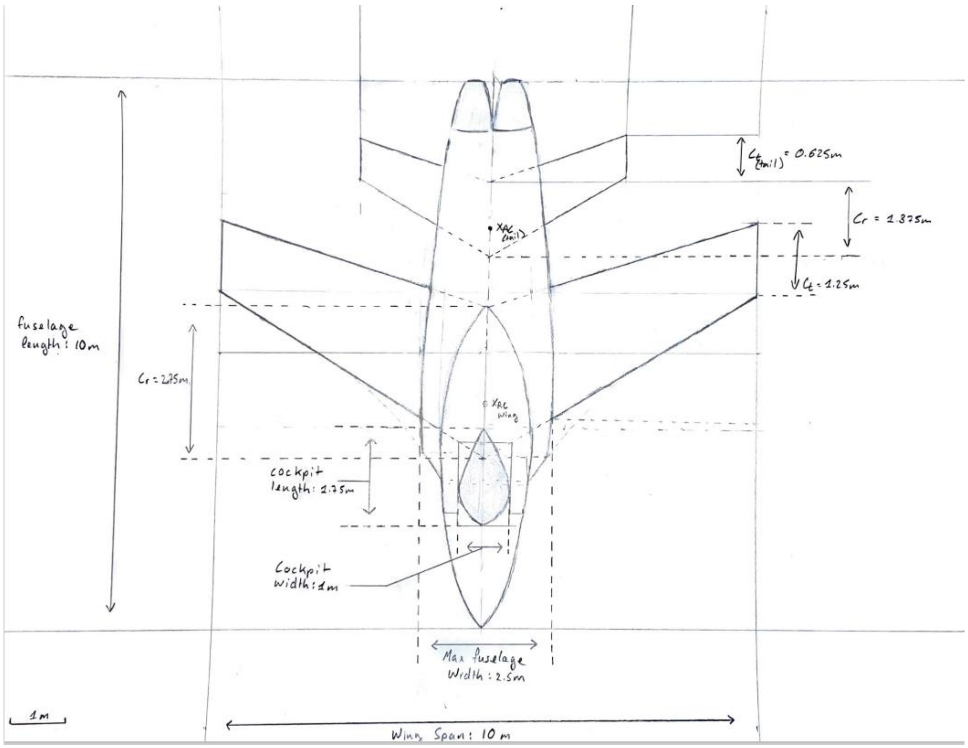



**Figure 6: Wing/ Tail Conceptual Design 1**

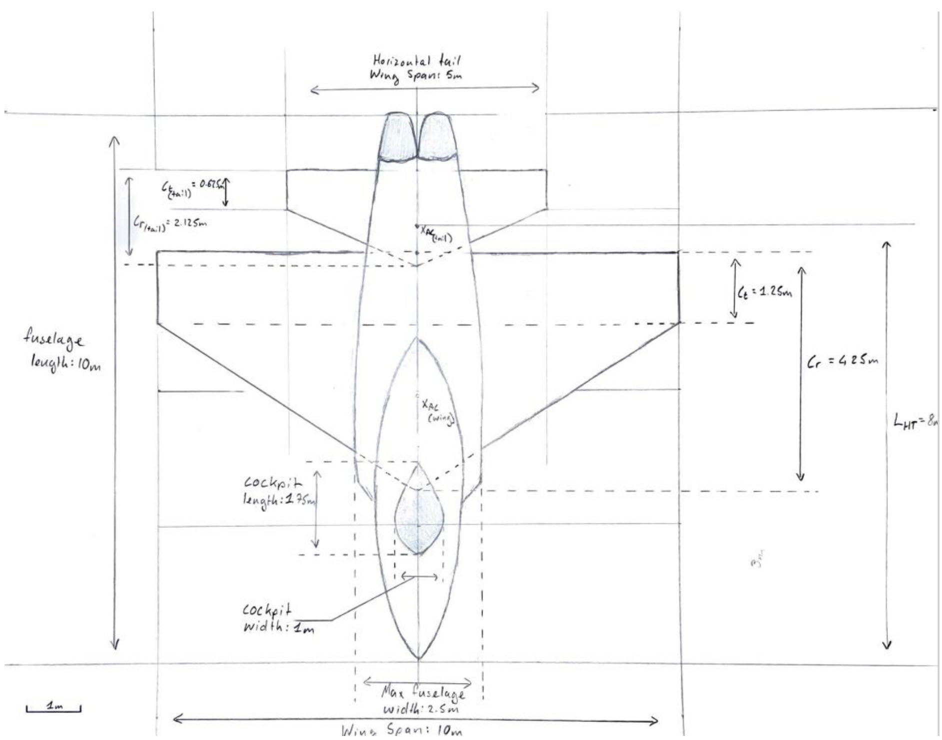



**Figure 7: Wing/Tail Conceptual Design 2**

## B. Chosen Design

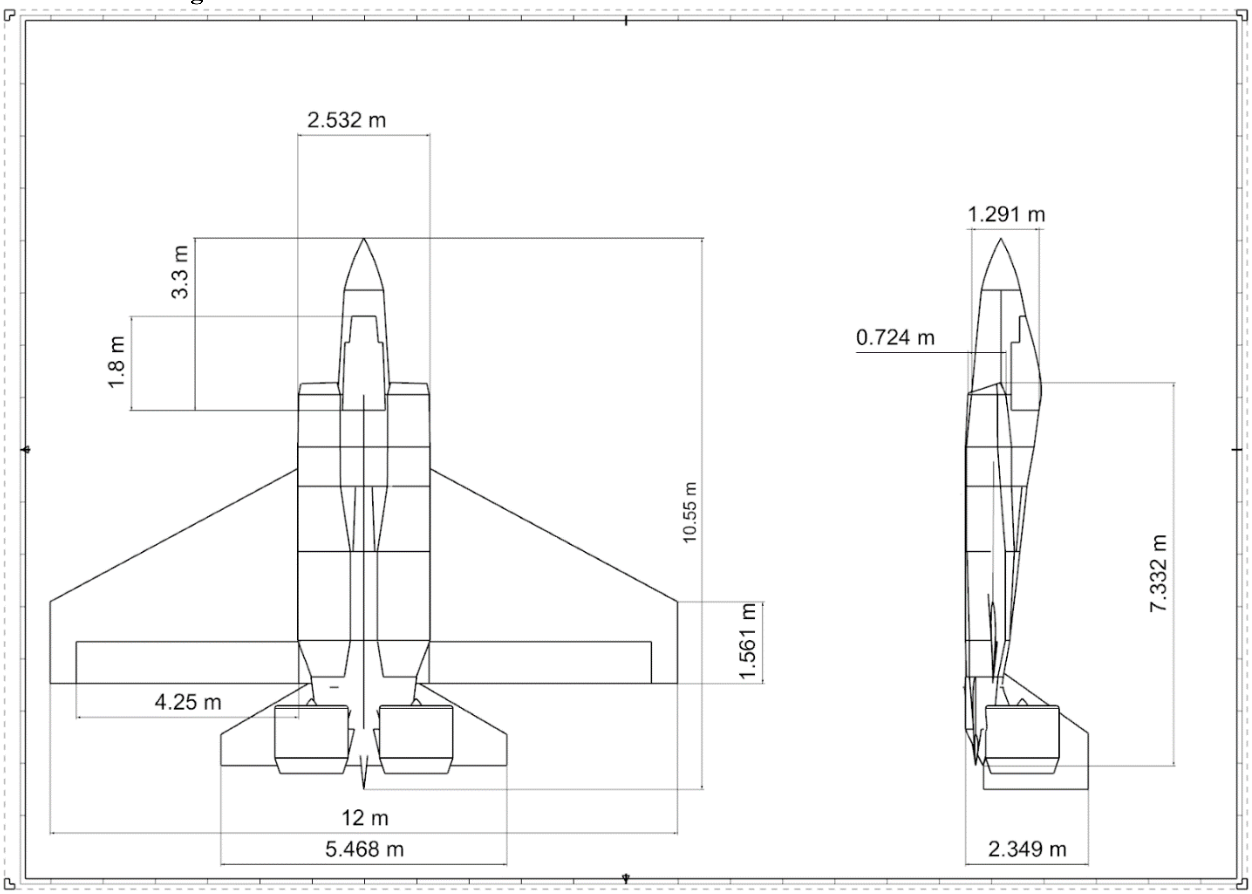



**Figure 8: Chosen design**

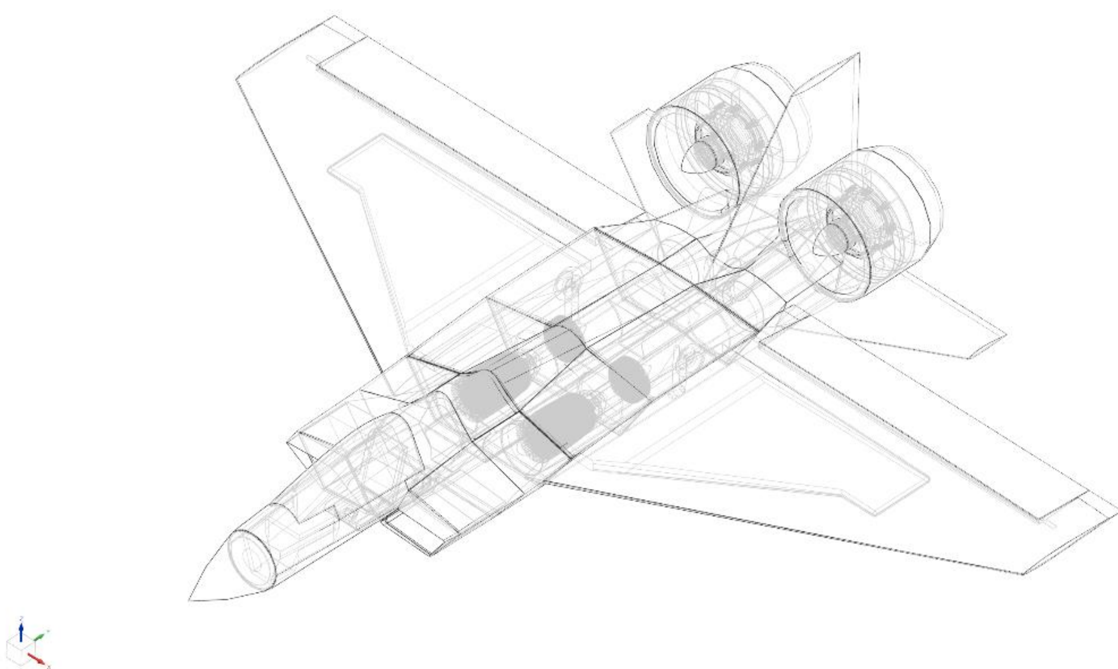


**Figure 9: Computer Aided Design, Isometric View**

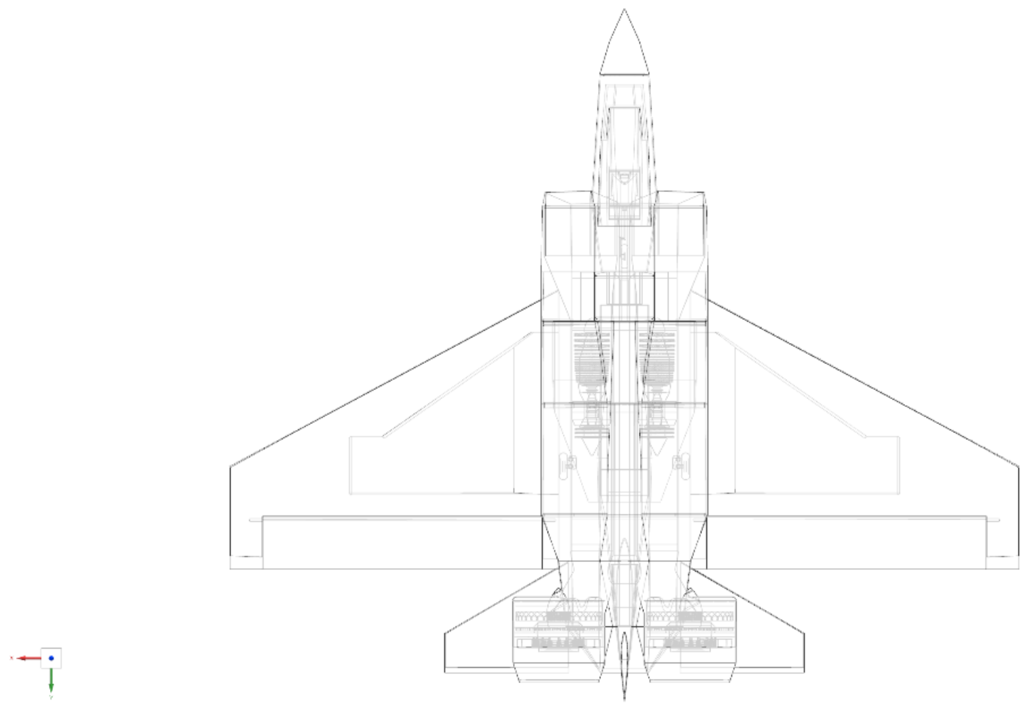


**Figure 10: Computer Aided Design, Top View**

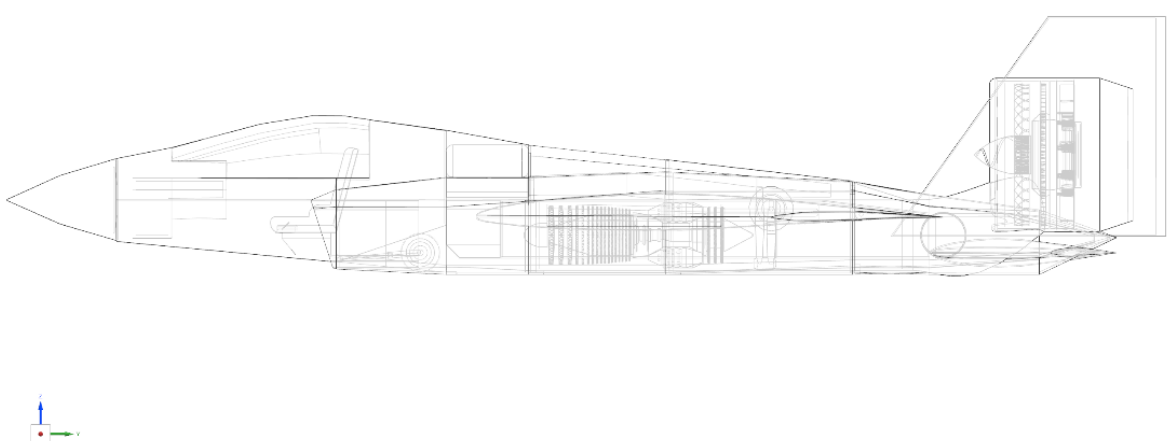


**Figure 11: Computer Aided Design, Side View**

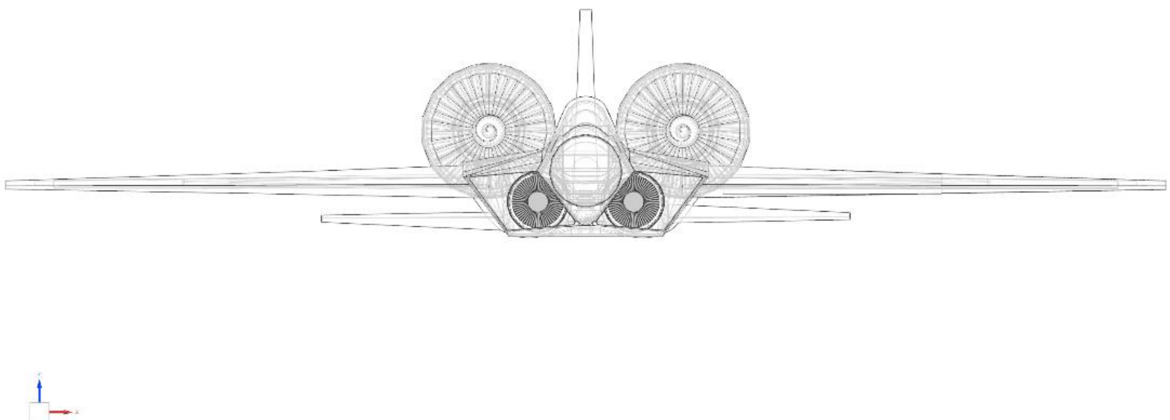


**Figure 12: Computer Aided Design, Front View**

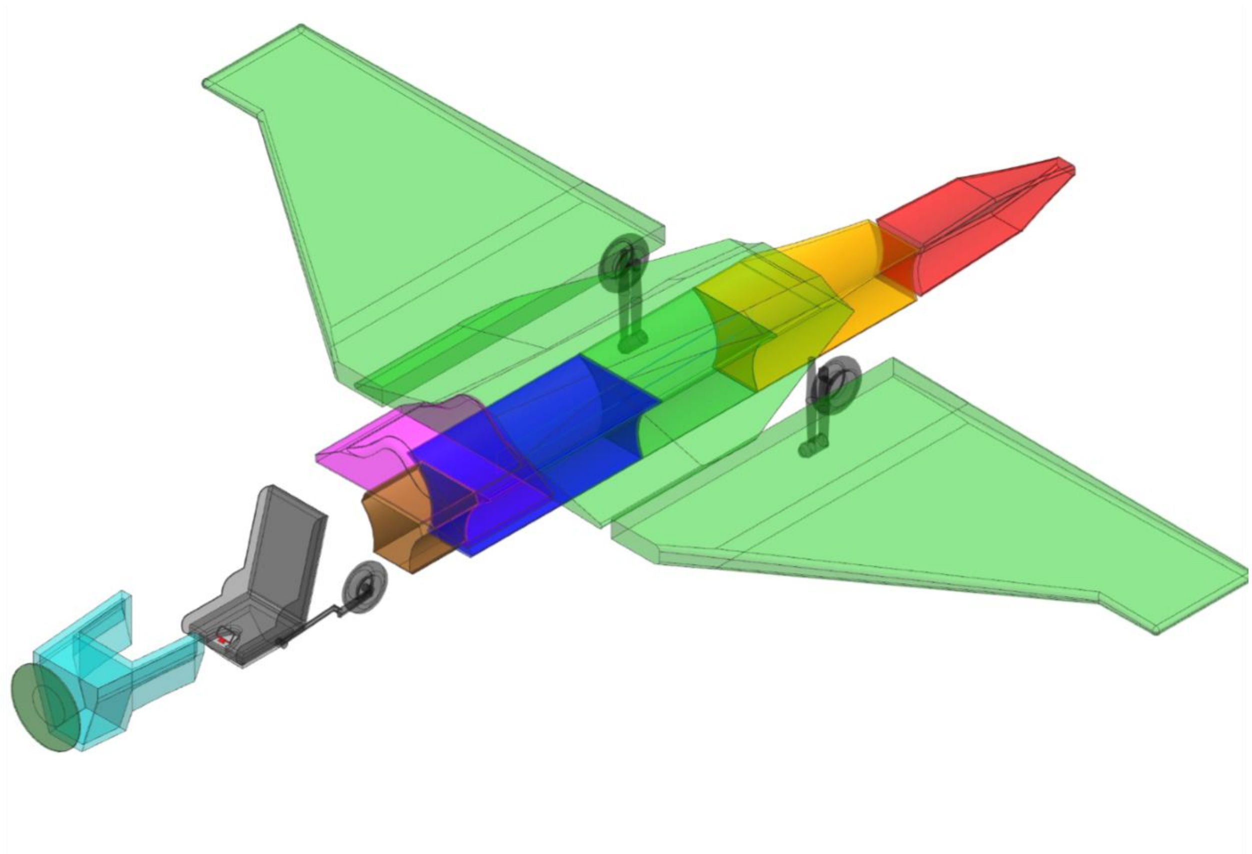


**Figure 13: Subsystems Layout in CAD Model**

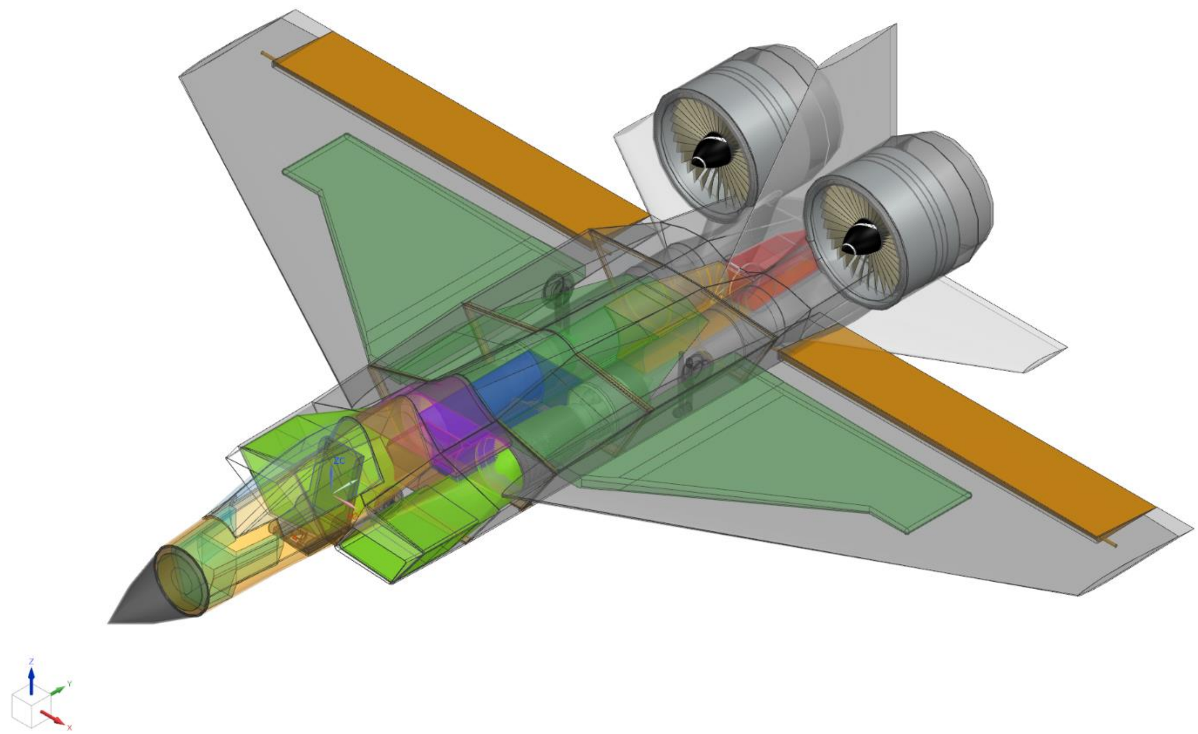


**Figure 14: Aircraft Layout in CAD Model**

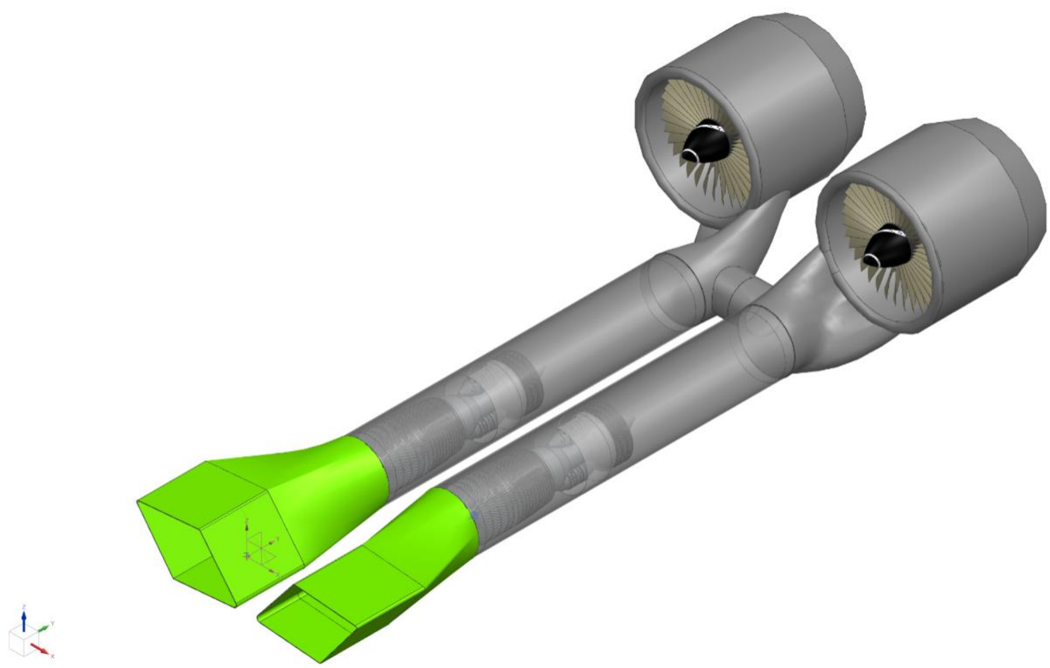


**Figure 15: Propulsion System**

## C. Volume and Surface Area from CAD Design

From the CAD model, Dimensions, Surface Areas and Volumes are extracted. The CAD model is assembled using four primary parts, the fuselage, the wing, the horizontal tail, and the vertical tail. However, the primary parts consist of different components, therefore an independent analysis was done for each primary part. Finally, each independent analysis for each primary part was interconnected and total Geometrical Values were found for the whole model.

The fuselage of the aircraft consists of 17 separate components, the nose cone, the cockpit Body, the cockpit window, the air intake, the external engine mount, and 12 different bodies that make up the center section. Certain faces of these components such as the bottom face of the cockpit body were not externally located and were not accounted for in the calculation for the external surface area. Each wing consisted of three different components, the wing component, the aileron, and the wing body extension. Volume calculations included the wing body extension, however since it was located inside the fuselage it was not accounted for the external Surface Area. When calculating the external surface area of the wing, the wing and aileron were looked at as one part. The CAD models for the vertical and horizontal tails were not as precise as the wing and the fuselage, so the external surface area and the volume calculations were based on a shell of the airfoils used for the tails.

When the external surface area was calculated for each of the primary parts, certain faces were not included in the calculation as they were either a connection joint with a different primary part or were in the interior of a different primary part.

The Dimensions for the primary parts are found below. A more detailed Version of the excel used to calculate the following can be found in Appendix A.

**Table 15: Fuselage External Surface Areas and Volumes Found From CAD**

| Part Name | X-Length (FRL) (m) | CAD External Surface Area (m^2) | CAD Volume (m^3) | Qty. | Total External Surface Area (m^2) | Total Volume (m^3) |
|---|---|---|---|---|---|---|
| **Nose Cone** | 1 | 1.412112928 | 0.026829 | 1 | 1.412112928 | 0.026829 |
| **Cockpit Body Total** | 2 | 4.141737453 | 0.110631571 | 1 | 4.141737453 | 0.1106315718 |
| **Cockpit Window** | 1.8 | 2.167377782 | 0.067134541 | 1 | 2.167377782 | 0.06713454158 |
| **Air Intake Total** | 0.2 | 0.304891998 | 0.007133682 | 2 | 0.6097839976 | 0.01426736529 |
| **Body 1 Total** | 1 | 5.833877545 | 0.062417938 | 1 | 5.833877545 | 0.06241793829 |
| **Body 2 Total** | 0.75 | 3.859868679 | 0.052180158 | 1 | 3.859868679 | 0.05218015881 |
| **Body 3 Total** | 1.25 | 7.362267098 | 0.101265308 | 1 | 7.362267098 | 0.1012653082 |
| **Body 4 Total** | 0.75 | 0.926660953 | 0.006788946 | 1 | 0.9266609539 | 0.006788946435 |
| **Body 5 Total** | 1.25 | 0.959477525 | 0.008294098 | 1 | 0.9594775254 | 0.008294098847 |

| | | | | | | |
|---|---|---|---|---|---|---|
| **Body 6 Total** | 1.7 | 9.759478434 | 0.142508284 | 1 | 9.759478434 | 0.142508284 |
| **Body 7 Total** | 1.7 | 0.797795146 | 0.007567428 | 1 | 0.7977951467 | 0.007567428866 |
| **Body 8 Total** | 0.8 | 3.3145509 | 0.045915742 | 1 | 3.3145509 | 0.04591574238 |
| **Body 9 Total** | 0.8 | 0.501014028 | 0.003677481 | 1 | 0.5010140284 | 0.0036774817 |
| **Body 10 Total** | 1 | 4.162546291 | 0.054375654 | 1 | 4.162546291 | 0.05437565476 |
| **Body 11 Total** | 0.75 | 0.434329037 | 0.003677481 | 1 | 0.4343290371 | 0.0036774817 |
| **Body 12 Total** | 0.5 | 1.721990611 | 0.110854355 | 1 | 1.721990611 | 0.110854355 |
| **External Engine Mount** | 1.25 | 5.639648796 | 0.095311783 | | 11.27929759 | 0.1906235672 |

**Table 16: Wing External Surface Areas and Volumes Found From CAD**

| Part Name | X-Length (FRL) (m) | CAD External Surface Area (m^2) | CAD Volume (m^3) | Qty. | Total External Surface Area (m^2) | Total Volume (m^3) |
|---|---|---|---|---|---|---|
| **Wing** | 4.25 | 22.57555979 | 0.1841878253 | 2 | 45.15111959 | 0.3683756506 |
| **Wing Body Extension** | | 4.287121896 | 0.0380350332 | 2 | | 0.07607006654 |
| **Aileron** | 0.75 | 6.838869144 | 0.1573681318 | 2 | 13.67773829 | 0.3147362636 |
| **Wing + Aileron** | 4.25 | 29.41442894 | 0.3415559571 | 2 | 58.82885788 | 0.6831119142 |

**Table 17: Horizontal Tail External Surface Areas and Volumes Found From CAD**

| Part Name | X-Length (FRL) (m) | CAD External Surface Area (m^2) | CAD Volume (m^3) | Qty. | Total External Surface Area (m^2) | Total Volume (m^3) |
|---|---|---|---|---|---|---|
| **Horizontal Tail** | 1.7. | 4.450350739 | 0.0089712319 | 2 | 8.900701479 | 0.01794246399 |

**Table 18: Vertical Tail External Surface Areas and Volumes Found From CAD**

| Part Name | X-Length (FRL) (m) | CAD External Surface Area (m^2) | CAD Volume (m^3) | Qty. |
|---|---|---|---|---|
| **Vertical Tail** | 2.5. | 7.325522732 | 0.05662987723 | 1 |

| | | | | |
|---|---|---|---|---|
| | | | | |

**Table 19: Total External Surface Area and Volume from CAD model**

| | Fuselage | Wing | Horizontal Tail | Vertical Tail | Totals: |
|---|---|---|---|---|---|
| Total Surface Area (m^2) | 59.244166 | 58.82885788 | 8.900701479 | 7.325522732 | 134.2992481 |
| Total Volume (m^3) | 1.009008925 | 0.6831119142 | 0.01794246399 | 0.05662987723 | 1.76669318 |

The distribution plots were found using a simplified approach. The distribution analysis was done on a two-dimensional scale where the fuselage reference line was set as the x-axis for each plot. The fuselage length of 10 meters was divided into 0.25m sections and the Volume and Surface Area was found independently for each section.

For each primary part, the volume and surface area were divided by the parts length over the fuselage reference line. Ratios were found for Part_Volume/Part_Length and Part_Surface_Area/Part_Length. In addition, the X-Coordinates were found for the start and end of each part. Afterwards, the ratios were multiplied by 0.25m to find the volume and surface area of the part for each 0.25m section. Each section volume and area were then distributed over the X-Coordinates that the part was on. This step was repeated for all the primary parts of the model to find volume distribution plots as well as surface area distribution plots.

**Table 20: Fuselage Coordinates and Section Ratios**

| | Y-cord. Start | Y.Cord End | Volume Over Length | Surface Area Over Length |
|---|---|---|---|---|
| Nose Cone | 0 | 1 | 0.026829 | 1.412112928 |
| Cockpit Body | 1 | 3 | 0.05531578592 | 2.070868726 |
| | 1.5 | 3.3 | 0.03729696754 | 1.204098768 |
| Air Intake | 2.8 | 3 | 0.07133682643 | 3.048919988 |
| Body 1 | 3 | 4 | 0.06241793829 | 5.833877545 |
| Body 2 | 4 | 4.75 | 0.06957354508 | 5.146491572 |
| Body 3 | 4.75 | 6 | 0.08101224658 | 5.889813679 |
| Body 4 | 4 | 4.75 | 0.00905192858 | 1.235547939 |
| Body 5 | 4.75 | 6 | 0.006635279078 | 0.7675820203 |
| Body 6 | 6 | 7.7 | 0.08382840235 | 5.740869667 |
| Body 7 | 6 | 7.7 | 0.004451428744 | 0.4692912627 |
| Body 8 | 7.7 | 8.5 | 0.05739467798 | 4.143188624 |
| Body 9 | 7.7 | 8.5 | 0.004596852125 | 0.6262675355 |
| Body 10 | 8.5 | 9.5 | 0.05437565476 | 4.162546291 |
| Body 11 | 8.75 | 9.5 | 0.004903308933 | 0.5791053829 |
| Body 12 | 9.5 | 10 | 0.22170871 | 3.443981223 |
| External Engine Mount | 8.75 | 10 | 0.1524988538 | 9.023438074 |

**Table 21: Wing Coordinates and Section Ratios**

| | Y-cord. Start | Y.Cord End | Volume Over Length | Surface Area Over Length |
|---|---|---|---|---|
| Wing+Aileron | 4.25 | 8.5 | 0.1607322151 | 13.84208421 |

**Table 22: Horizontal Tail Coordinates and Section Ratios**

| | Y-cord. Start | Y.Cord End | Volume Over Length | Surface Area Over Length |
|---|---|---|---|---|
| Horizontal Tail | 8.3 | 10 | 0.01055439058 | 5.235706752 |

**Table 23: Vertical Tail Coordinates and Section Ratios**

| | Y-cord. Start | Y.Cord End | Volume Over Length | Surface Area Over Length |
|---|---|---|---|---|
| Vertical Tail | 7.5 | 10 | 0.02265195089 | 2.930209093 |

Tables 18, 19, 20, and 21 give the coordinates and Ratios that were used to calculate the sectional external surface area and volume for each primary part. A visualization of the sectional external surface area and volume contribution for each primary part can be found in Appendix B. The Total Sectional Volumes are also located in Appendix B.

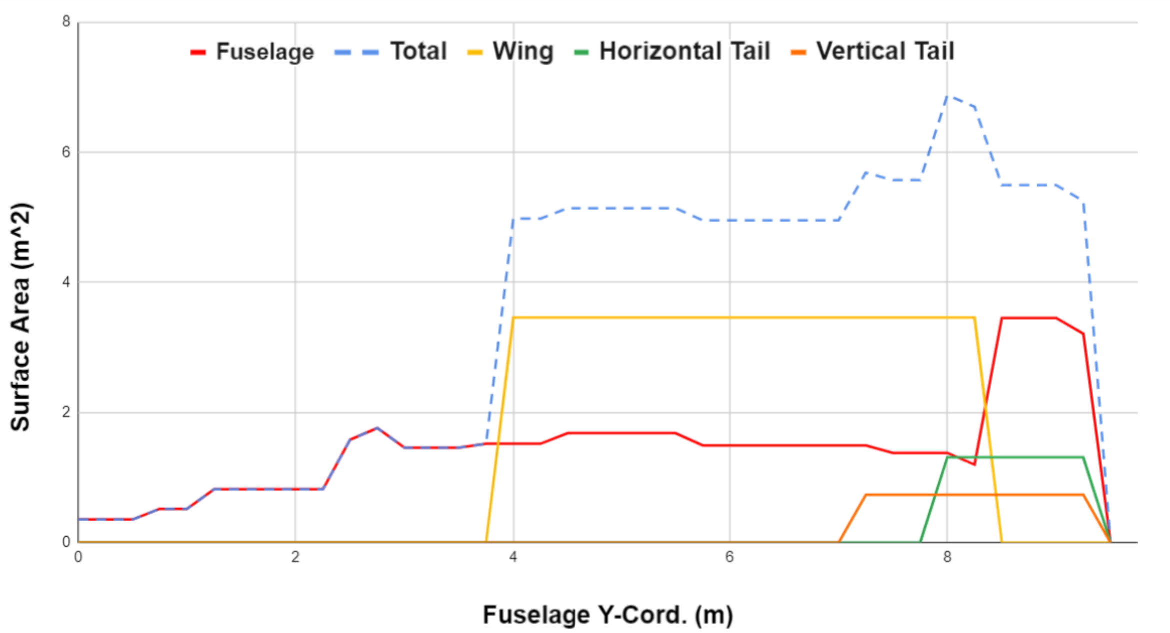



**Figure 16: External Surface Area Distribution.**
*External Surface Area Distributions for Each primary component as well as the total External Surface Area distribution.*

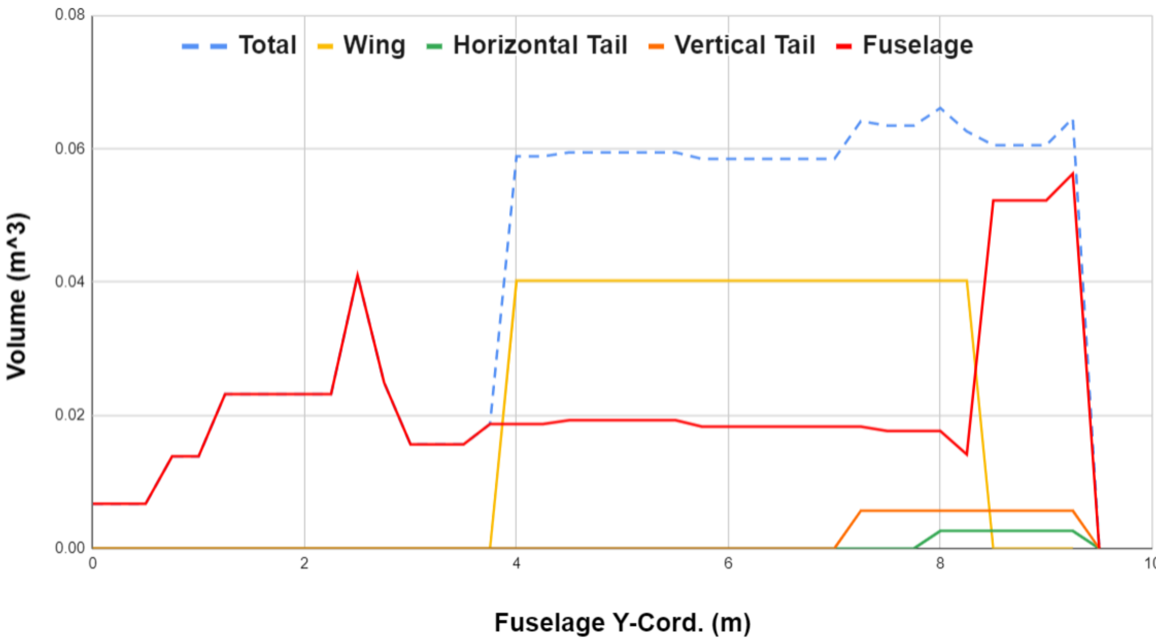



**Figure 17: Volume Distribution.**
*Volume Distributions for Each primary component as well as the total Volume distribution.*

The simplified approach provided data that was acceptable for an initial analysis, however it will be improved in the future. This approach worked well for the Fuselage as it was divided into multiple body parts and the surface areas and volumes extracted from NX were already initially distributed. However, for the parts such as the wing, horizontal tail, and vertical tail, the results were not as accurate. Because of the parts taper ratios and airfoils, the surface areas and volumes were not evenly distributed along the fuselage reference line. To solve this a better relation will be found for a part's surface area and volume to the part's length over the fuselage reference line. In addition, the 0.25 m sections will be made smaller, dividing the fuselage reference line into more sections to get a more accurate result.

## D. Cross Sectional Area Distribution

Using the CAD model, the cross-sectional Area was found for the aircraft at every 0.25 meters of the aircraft as seen in Table X. The cross-sectional area of the model was found by subtracting the model from an object with a 10 $m^2$ initial surface area and retrieving the data at 0.25m sections of the object. The difference of the surface areas gave the cross-sectional area of the aircraft. This process can be seen in Figure X. The cross-sectional area calculations were used in order to create a cross area distribution along the fuselage length seen in Figure X. This distribution was used in order to find the optimal locations of the subsystems. Once the subsystems are fully analyzed and integrated into the model, an analysis will be done using the cross-sectional area to find the empty spacing within the cross-sectional area. This analysis will be used to find whether some areas of the aircraft can be resized/modified to optimize the sizing of the aircraft and the locations of the subsystems. These findings will be analyzed hand to hand with the drag and weight calculations to find areas of interest where resizing and subsystem locations will benefit the performance of the aircraft.

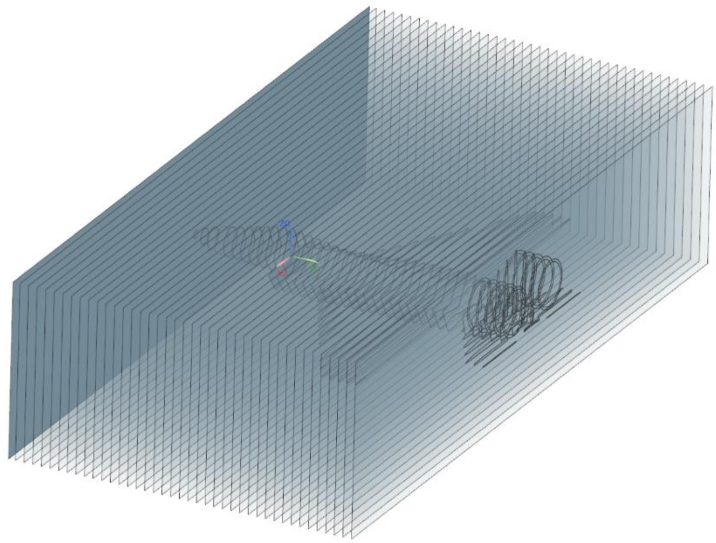


**Figure 18: Method Used for Cross Sectional Area Calculations.**
*Sheet bodies extruded at every 0.25m from fuselage tip. Model Subtracted from all sheet bodies and surface area of each new object was retrieved.*

**Table 24: Cross Sectional Surface Area.**

*Calculated Cross Sectional Area at every 0.25m from fuselage tip.*

| Fuslege Length (m) | Cross Sectional Area (m^2) |
|---:|---:|
| 0 | 0 |
| 0.25 | 0.04763669546 |
| 0.5 | 0.1599723502 |
| 0.75 | 0.3003587395 |
| 1 | 0.4416503726 |
| 1.25 | 0.4819129602 |
| 1.5 | 0.5235026456 |
| 1.75 | 0.5787227881 |
| 2 | 0.6342972334 |
| 2.25 | 0.6908392003 |
| 2.5 | 0.741238656 |
| 2.75 | 0.7807249855 |
| 3 | 1.78057533 |
| 3.25 | 1.821231198 |
| 3.5 | 1.863147069 |
| 3.75 | 1.907316918 |
| 4 | 1.952282779 |
| 4.25 | 1.931014324 |
| 4.5 | 1.95205643 |
| 4.75 | 2.08309371 |
| 5 | 2.276236258 |
| 5.25 | 2.504132957 |
| 5.5 | 2.748495877 |
| 5.75 | 2.995069739 |
| 6 | 3.229559685 |
| 6.25 | 3.367952045 |
| 6.5 | 3.506344406 |
| 6.75 | 3.536294335 |
| 7 | 3.479415325 |
| 7.25 | 3.283376606 |
| 7.5 | 2.976042454 |
| 7.75 | 2.520721727 |
| 8 | 2.045751977 |
| 8.25 | 1.639780721 |
| 8.5 | 1.233809465 |
| 8.75 | 1.367313904 |
| 9 | 4.497198654 |
| 9.25 | 4.640461295 |
| 9.5 | 4.437720659 |
| 9.75 | 4.437720659 |
| 10 | 0 |

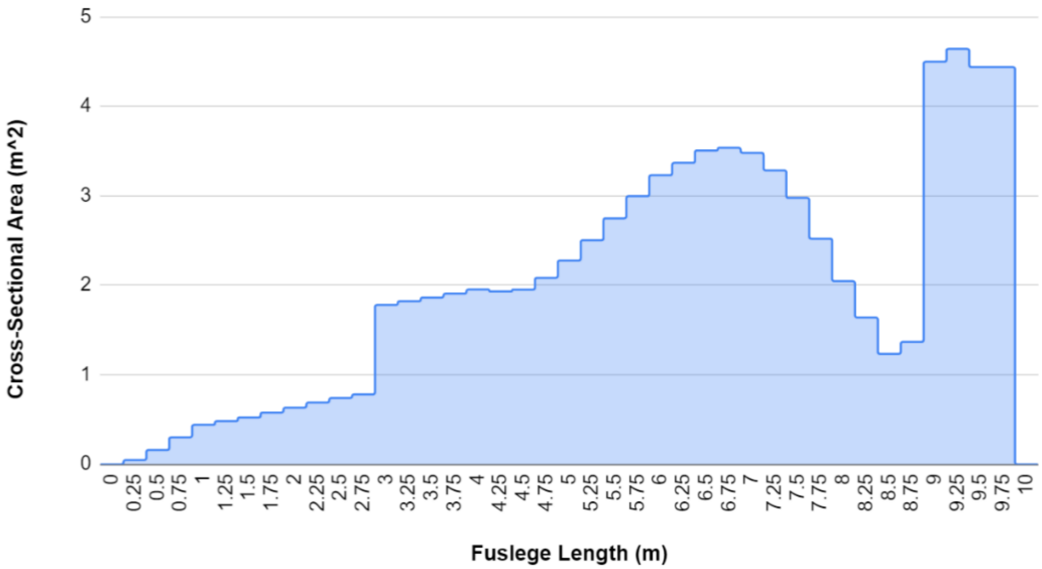
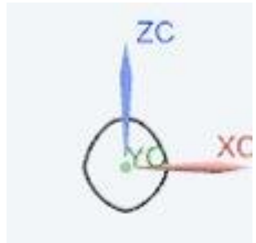



**Figure 19: Cross Sectional Area Distribution.**
*Cross Sectional Area Distribution using data from Table X.*

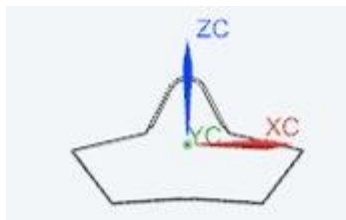
Although the cross-sectional area changes along the length of the fuselage, it only consists of five different shapes. The major cross-sectional shape changes occur at the fuselage lengths shown in Table X. The cross-sectional shapes were used to determine geometries and sizing of the different subsystems mentioned before. Overall, the cross-sectional area played an important role in determining the internal configuration of the aircraft.

**Table 25: Cross Sectional Shape Change at Fuselage Lengths**

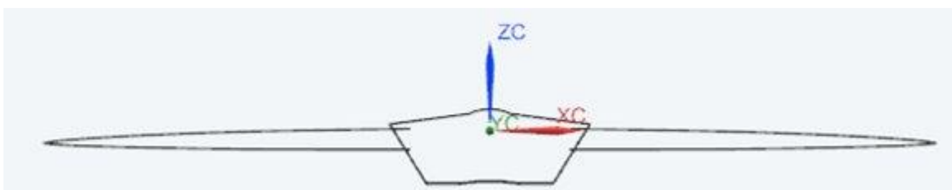
| Fuslege Length (m) | Cross Sectional Area (m^2) |
|---:|---:|
| 0 | 0 |
| 1.5 | 0.5235026456 |
| 3 | 1.78057533 |
| 4.5 | 1.95205643 |
| 6.75 | 3.536294335 |
| 8.5 | 1.233809465 |
| 9.25 | 4.640461295 |
| 10 | 0 |



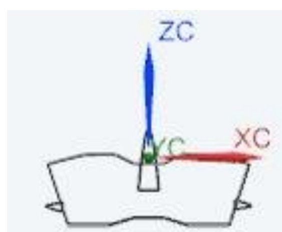
1.5 m



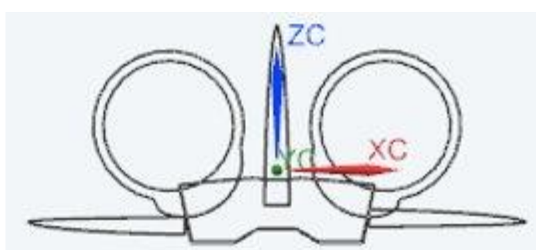
3.0 m



6.75 m



8.5 m



9.25 m

### E. Weight Calculations and Material Selection

Using the Volume Calculations of the CAD model, seen in Tables 13 through 16 materials were assigned to the divided bodies. Initial materials used for the aircraft body was aluminum 7075 for the top part of the aircraft, wing, horizontal tail, and vertical tail. The bottom portion of the aircraft, cockpit shell, fuselage structural beams and ribs, and internal engine housing was made from titanium for its strength and armor benefits. In addition, the subsystem weights, volumes, and locations were estimated and implemented into the CAD model. An initial Weight calculation using these materials showed that the aircraft would exceed the weight estimations initially set for the aircraft. Therefore, the decision to have the majority of the aircraft body made from a composite material was made. The use of titanium was kept for the benefits stated above, however was applied in a conservative matter to the more important parts. The revised material selection for the aircraft components can be seen in Table X, and Figure X. A more detailed material assignment for each divided aircraft component can be seen in Appendix C.

**Table 26: Material Composition of Aircraft Components**

| Material | Density (kg/m$^3$) | Aircraft Component |
|---|---|---|
| Carbon Fiber M55J | 1710 | - Fuselage Shell<br>- Fuselage Ribs<br>- Propulsion System Intake<br>- Cockpit Cell Top |
| Carbon Fiber M55lt | 1600 | - Wing Shell<br>- Vertical Tail Shell |
| Forged Light Weight Carbon Fiber | 1800 | - Aileron<br>- Horizontal Tail<br>- Rudder |
| Molybdenum Mo Al203 Composite | 550 | - Internal Engine Housing<br>- External Engine Housing |
| Titanium | 4420 | - Cockpit Cell base<br>- Fuselage Structural Beams |
| Acrylic | 1180 | - Cockpit Window |

Using the revised material selection and subsystem weight calculations, the total take-off weight of the aircraft was calculated to be 7365kg. The detailed weight contribution of the divided aircraft bodies, subsystems, and propulsion system can be found in Appendix C. Table X shows the aircraft weight per aircraft section, which includes subsystems like fuel storage and payload. The total weight calculated was still above the estimated weight initially set for the aircraft. A decision will be made of whether the weight of the aircraft will need to be further dropped or if the decided weight of the aircraft can be increased. Further aircraft performance analysis will be done using the calculated weight and a decision will be made in accordance with the lift calculations, and propulsion system that is being used for the aircraft.

**Table 27: Weight per Aircraft Component**

| Aircraft Component | Mass (kg) |
|---|---|
| Fuselage | 1,724 |
| Wing | 914 |
| Horizontal Tail | 151 |
| Vertical Tail | 127 |
| Propulsion System | 884 |
| Subsystems | 791 |
| Fuel Weight | 2,500 |
| Payload | 390 |
| **Empty Weight:** | 4,591 |

| | |
|---|---|
| **Total Weight:**<br>***Including Fuel and Payload Weight*** | 7,481 |

**F. Weight Distribution and Center of Gravity**

Using the NX measuring software the center of mass of each divided component was found. This data was used to find the weight distribution along the aircraft. The distributions were done using the (x, y, z) coordinate system, where the y coordinate was the fuselage reference line, the x coordinate was the width of the aircraft, and the z coordinate was the height of the aircraft. The coordinate system was placed where point (0,0,0) is the tip of the aircraft. The aircraft was designed symmetrically along the ZY plane, therefore the center of mass of all the aircraft components were located at (0, y, z) or in the situation where there were two components on the port and starboard side of the aircraft, combining the two parts would cause the center of mass to be located at (0, y, z) due to symmetry. Therefore, the weight distribution was only analyzed on the YZ plane. The individual center of mass locations of the divided aircraft components can be seen in Appendix C. Weight Distributions along the fuselage length (Y-cord), and aircraft height (Z-cord.) are found in Figures X, and X respectively.

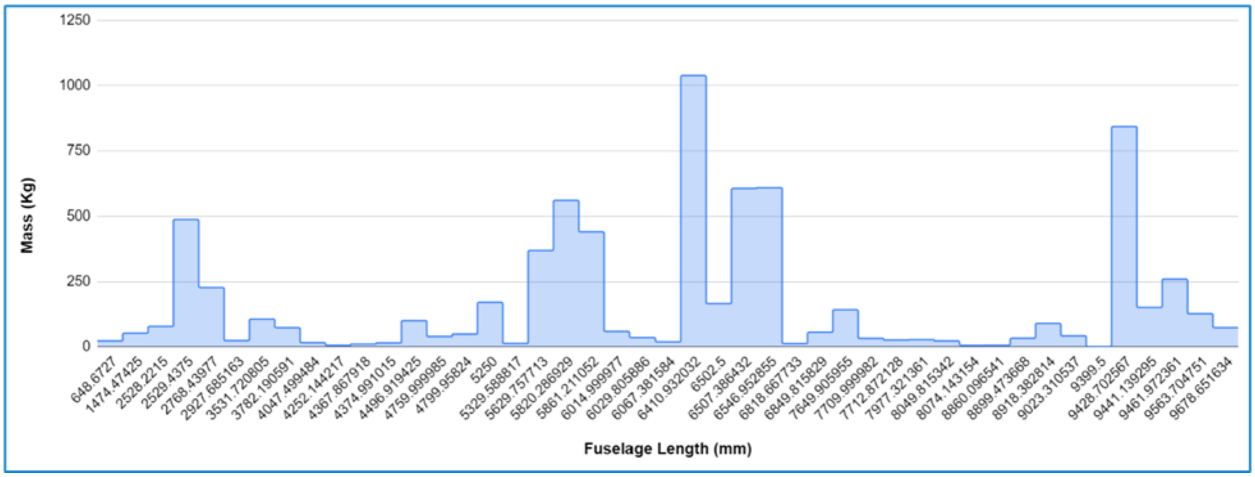



**Figure 20: Mass Distributed along Fuselage Length**

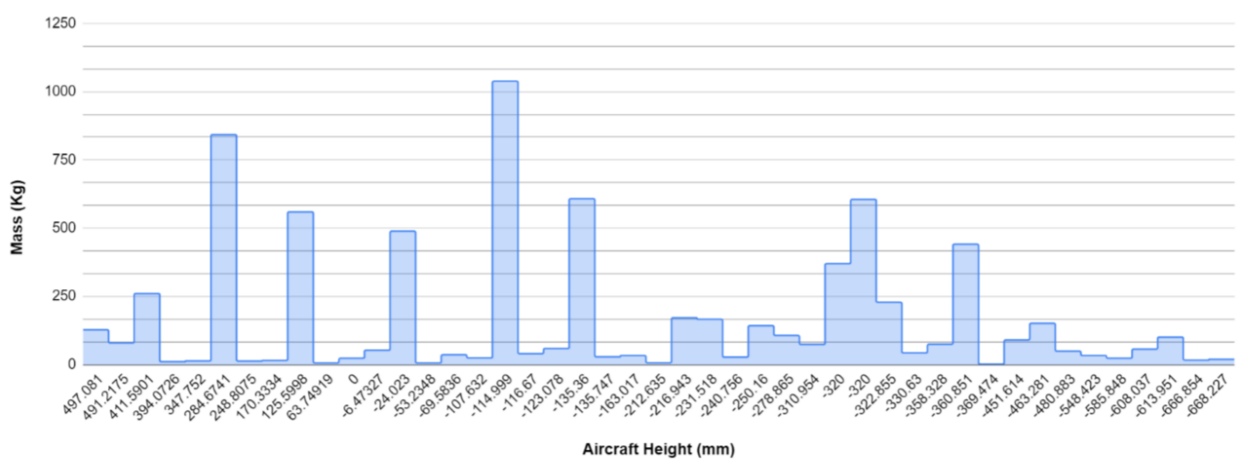



**Figure 21: Mass Distributed along Aircraft Height**

Weight Induced Moments were also found using the center of mass locations. Two moments were calculated, the moment on the fuselage tip about the X-cord. caused by weight distributions along Y, and the moment about the Y-cord. caused by weight distributions along Z. The moment about the Z-cord. was known to be 0 since the weights were symmetrically distributed along X. Individual moments of each of the calculated aircraft components can be found in Appendix C. Total Moment calculations are found in Table X. By dividing the calculated moments by the total weight of the aircraft, the Center of Gravity of the aircraft was found seen in Table X. It is important to note that the moments discussed were analyzed for the purpose of finding the Center of Gravity, therefore in order to simplify the process for the calculations, the mass was used to calculate the moments and in addition was divided by the total mass instead of weight to find the center of gravity. If further analysis requires the use of moments, they will be multiplied by the gravitational force.

**Table 28: Moments about Aircraft coordinate system**

| Moment about X-cord. (kg*mm) | Moment about Y-cord. (kg*mm) | Moment about Z-cord. (kg*mm) |
|---|---|---|
| 47052146.74 | -751289.3809 | 0 |

**Table 29: Max Take Off Weight Center of Gravity.**

| Center of Gravity X-cord. (mm) | Center of Gravity Y-cord. (mm) | Center of Gravity Z-cord. (mm) |
|---|---|---|
| 6,174.8 | 0 | -130.5 |

**Table 30: Empty Center of Gravity.**

| Center of Gravity X-cord. (mm) | Center of Gravity Y-cord. (mm) | Center of Gravity Z-cord. (mm) |
|---|---|---|
| 6,115.5 | 0 | -167.1 |

## G. Structural Layout

## H. Finite Element Analysis

As the design has changed, multiple structural analyses have been performed using *Siemens NX* finite element analysis package. The structure and materials of the aircraft has changed significantly since last reported. The fuselage structure is now composed of several ring supports, which allow for greater rigidity while leaving space for subsystems within the fuselage, such as landing gear and propulsion systems. The idealized CAD model is pictured below. Parts in blue are composed of carbon fiber M55J and parts in orange are composed of titanium Ti-6Al-4V.

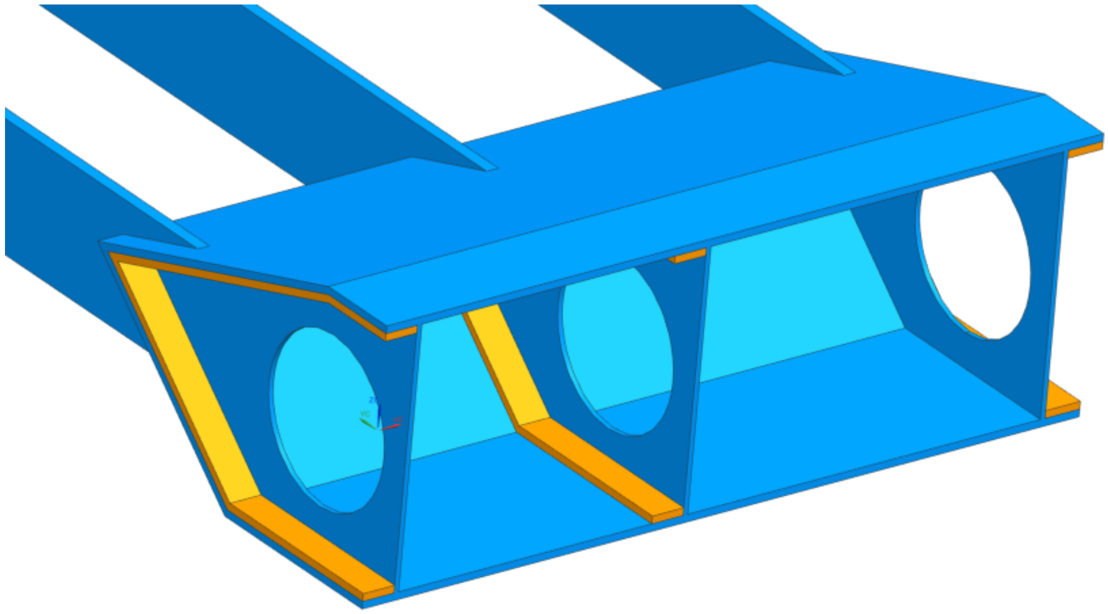


**Figure 22: Trimetric View, Idealized Fuselage Cross Section for Finite Element Analysis**

The model performed well under finite element analysis. These analyses were performed at 10G, the maximum expected loading case for this aircraft. This analysis was performed using 10 node tetrahedral elements, with a maximum number of elements of 186,731. The three beams connecting to the fuselage represent the wing, which is further discussed later.

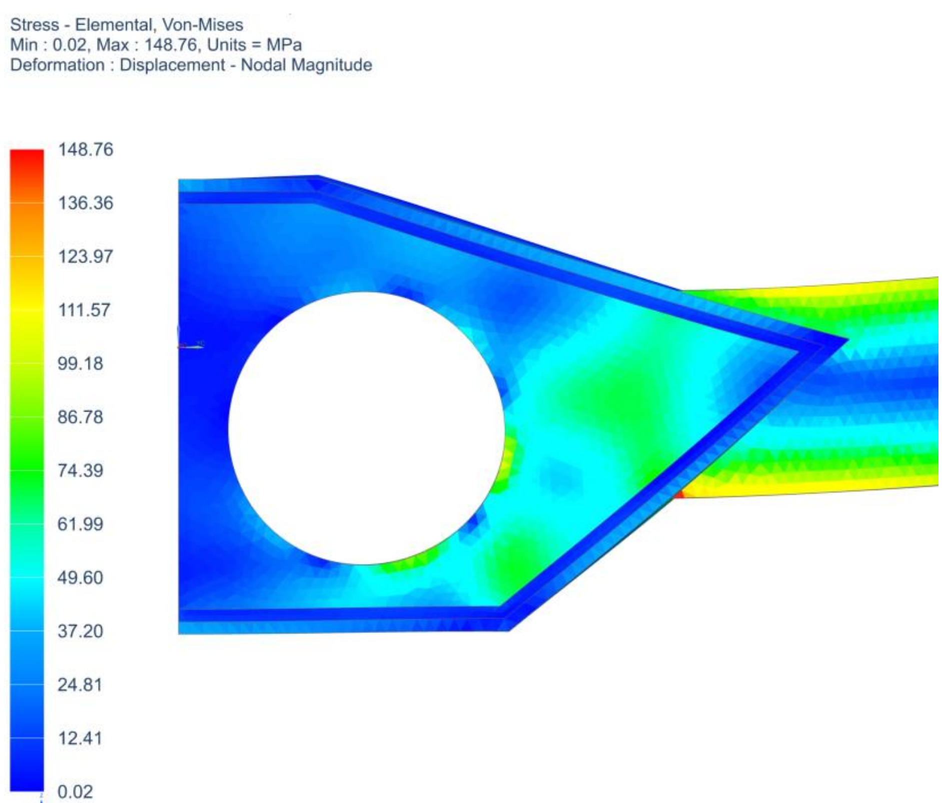



**Figure 23: Fuselage Structure FEA, Front View. Max Von Mises Stress = 148.8 MPa**

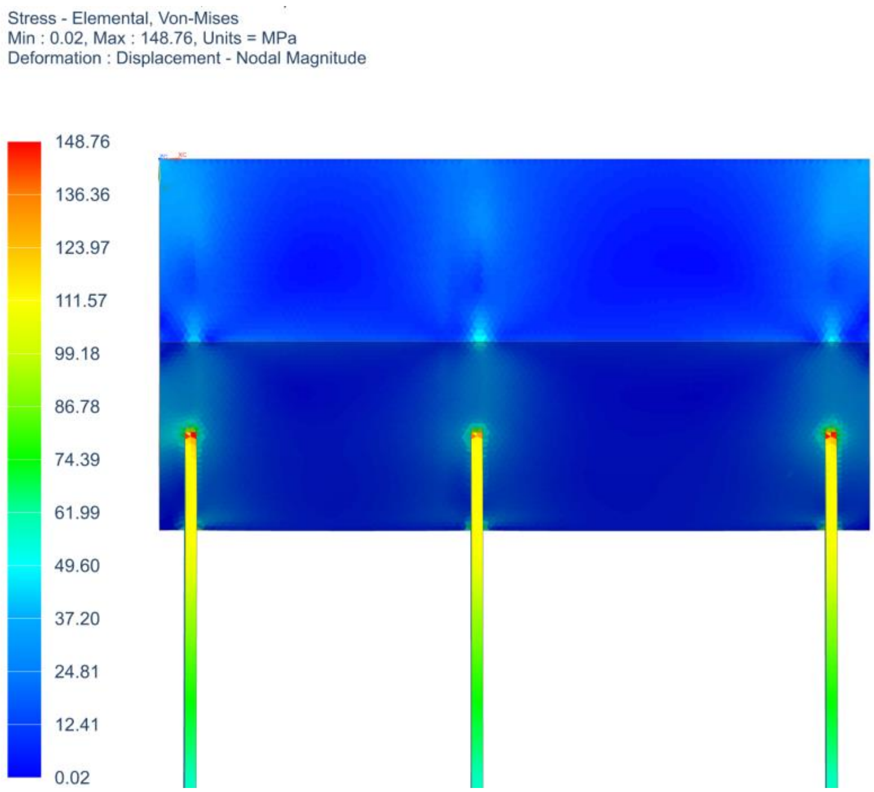



**Figure 24: Fuselage Structure FEA, Bottom View. Max Von Mises Stress = 148.8 MPa**

The wing structure has not changed greatly since last reported; it is now composed of 8 stringers and one main rib. This rib was added to ensure that the fuel tank does not experience great deformation under loading. A simplified, isometric view of the wing structure is shown below:

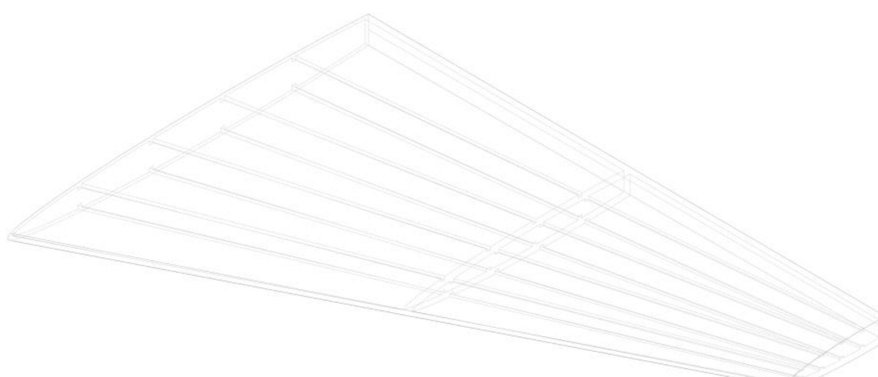


**Figure 25: Wing Structure, Isometric View**

This structure performed similarly well under finite element analysis. This finite element analysis was performed under 10G loading, using 12 node tetrahedral elements, with a total of 280560 total nodes. Loading was restricted to the vertical axis only, to simplify the analysis.

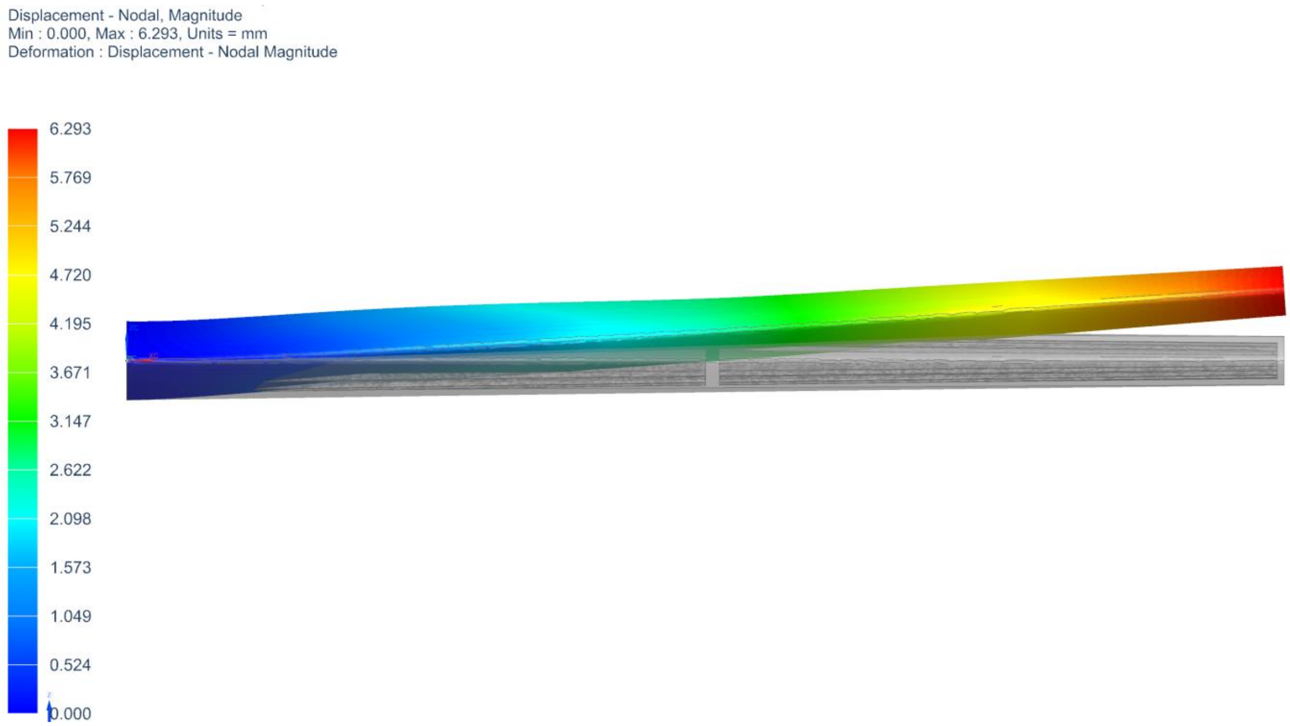



**Figure 26: Finite Element Analysis, Wing Displacement. Maximum Displacement = 6.3 mm**

This analysis gave a maximum displacement of 6.3 millimeters at this load. This is well within the maximum allowable displacement. This loading condition is also well above the maximum expected loading factor, which is given in the V-n diagram above. Based on this analysis, it is expected that the structure of the wing and fuselage will be adequate for this aircraft.

## IV. Future Work

Over the process of this aircraft's design, it is clear to the designers that this project has potentail to be a valuable addition to the armed forces of the United States, as well as other interested countries. Realistically, however, the design process is still in its initail stages.

The propulsion system likely needs the most work in its design. This propulsion system was never produced for production level aircraft, and never left the prototyping stage, and so a full scale system should be created in order to analyze its potentail for this aircraft. The V/STOL system also needs further research to ensure its viability for this style of aircraft. Additionally, a scale model should first be constructed for wind tunnel testing. The materials also should be researched, as this aircraft is almost entirely composed of carbon fiber. This is rare in military aircraft, and there may be unforeseen consequences of this system in a ground-attack aircraft. To replace the A-10, it also must be ensured that the *Harpy* will be able to match it in survivability. Ballistics testing should be conducted on the materials and structures used in this aircraft.

In general, this design process, while arduous, has been somewhat surface level, necessitated by the nature of the assignment. Despite this, initail designs have proved promising for future design.

# Appendix A: External Surface Area and Volume Calculations from CAD Model

## Fuselage External Surface Area and Volume Calculations

| | Part | Mounting | X-Length (FRL) (m) | External Surface Area (mm^2) | CAD External Surface Area (m^2) | CAD Volume (mm^3) | CAD Volume (m^3) | Qty. | Total External Surface Area (m^2) | Total Volume (m^3) | Total Weight |
|---|---|---|---|---|---|---|---|---|---|---|---|
| Nose Cone | Nose Cone | External | 1 | | 1.412112928 | | 0.026829 | 1 | 1.412112928 | 0.026829 | |
| | | | | | | | | | | | |
| Cockpit Body | Cockpit Body Total | External | 2 | | 4.141737453 | | 0.1106315718 | 1 | 4.141737453 | 0.1106315718 | |
| | Cockpit Body Front Top | External | | | 1.469989669 | | | | | | |
| | Cockpit Body Front Bottom | External | | | 2.671747784 | | | | | | |
| | Cockpit Bottom | Inside Engine Mount | | | 1.218402021 | | | | | | |
| | Cockpit Window | External | 1.8 | | 2.167377782 | | 0.06713454158 | 1 | 2.167377782 | 0.06713454158 | |
| Air Intake | Air Intake Total | External | 0.2 | | 0.3048919988 | | 0.007133682643 | 2 | 0.6097839976 | 0.01426736529 | |
| | Air Intake Front | | | | 0.1718126761 | | | | | | |
| | Air Intake Top | | | | 0.01595191894 | | | | | | |
| | Air Intake Side | | | | 0.08367067154 | | | | | | |
| | Air Intake Bottom | | | | 0.0334567322 | | | | | | |
| Body 1 | Body 1 Total | External | 1 | 5833877.545 | 5.833877545 | 62417938.29 | 0.06241793829 | 1 | 5.833877545 | 0.06241793829 | |
| | Body Top | | | 1057555.478 | 1.057555478 | | | | | | |
| | Body Top Sides | | | 1666923.423 | 1.666923423 | | | | | | |
| | Body Sides | | | 1486952.883 | 1.486952883 | | | | | | |
| | Body Bottom | | | 1622445.762 | 1.622445762 | | | | | | |
| Body 2 | Body 2 Total | External | 0.75 | 3859868.679 | 3.859868679 | 52180158.81 | 0.05218015881 | 1 | 3.859868679 | 0.05218015881 | |
| | Body Top Sides | | | 1355002.998 | 1.355002998 | | | | | | |
| | Body Sides | | | 1311020.137 | 1.311020137 | | | | | | |
| | Body Bottom | | | 1193845.543 | 1.193845543 | | | | | | |
| Body 3 | Body 3 Total | External | 1.25 | 7362267.098 | 7.362267098 | 101265308.2 | 0.1012653082 | 1 | 7.362267098 | 0.1012653082 | |
| | Body Top Sides | | | 2855283.604 | 2.855283604 | | | | | | |
| | Body Sides | | | 2562486.244 | 2.562486244 | | | | | | |
| | Body Bottom | | | 1944497.25 | 1.94449725 | | | | | | |
| Body 4 | Body 4 Total | External | 0.75 | 926660.9539 | 0.9266609539 | 6788946.435 | 0.006788946435 | 1 | 0.9266609539 | 0.006788946435 | |
| Body 5 | Body 5 Total | External | 1.25 | 959477.5254 | 0.9594775254 | 8294098.847 | 0.008294098847 | 1 | 0.9594775254 | 0.008294098847 | |
| Body 6 | Body 6 Total | External | 1.7 | 9759478.434 | 9.759478434 | 142508284 | 0.142508284 | 1 | 9.759478434 | 0.142508284 | |
| | Body Top Sides | | | 3863547.893 | 3.863547893 | | | | | | |
| | Body Sides | | | 3274513.108 | 3.274513108 | | | | | | |
| | Body Bottom | | | 2621417.433 | 2.621417433 | | | | | | |
| Body 7 | Body 7 Total | External | 1.7 | 797795.1467 | 0.7977951467 | 7567428.866 | 0.007567428866 | 1 | 0.7977951467 | 0.007567428866 | |
| Body 8 | Body 8 Total | External | 0.8 | 3314550.9 | 3.3145509 | 45915742.38 | 0.04591574238 | 1 | 3.3145509 | 0.04591574238 | |
| | Body Top Sides | | | 1106927.256 | 1.106927256 | | | | | | |
| | Body Sides | | | 1093632.653 | 1.093632653 | | | | | | |
| | Bottom 1 | | | 597325.0356 | 0.5973250356 | | | | | | |
| | Bottom 2 | | | 306665.9547 | 0.3066659547 | | | | | | |
| | Bottom 3 | | | 210000 | 0.21 | | | | | | |
| Body 9 | Body 9 Total | External | 0.8 | 501014.0284 | 0.5010140284 | 3677481.7 | 0.0036774817 | 1 | 0.5010140284 | 0.0036774817 | |
| Body 10 | Body 10 Total | External | 1 | 4162546.291 | 4.162546291 | 54375654.76 | 0.05437565476 | 1 | 4.162546291 | 0.05437565476 | |
| Body 11 | Body 11 Total | External | 0.75 | 434329.0371 | 0.4343290371 | 3677481.7 | 0.0036774817 | 1 | 0.4343290371 | 0.0036774817 | |
| Body 12 | Body 12 Total | External | 0.5 | 1721990.611 | 1.721990611 | 110854355 | 0.110854355 | 1 | 1.721990611 | 0.110854355 | |
| | Body Top | | | 379422.2582 | 0.3794222582 | | | | | | |
| | Body Top Sides | | | 487865.1668 | 0.4878651668 | | | | | | |
| | Body Bottom Sides | | | 469930.3955 | 0.4699303955 | | | | | | |
| | Body Bottom | | | 384772.7909 | 0.3847727909 | | | | | | |
| External Engine Mount | 18 | External | 1.25 | 5639648.796 | 5.639648796 | 95311783.62 | 0.09531178362 | 2 | 11.27929759 | 0.1906235672 | |
| | Front Rim | | | 591622.9564 | 0.5916229564 | | | | | | |
| | Side | | | 3756529.365 | 3.756529365 | | | | | | |
| | Back | | | 1291496.475 | 1.291496475 | | | | | | |
| | | | | | | | | | | | |

## Wing External Surface Area and Volume Calculations

| | Part | Mounting | X-Length (FRL) (m) | External Surface Area (mm^2) | External Surface Area (m^2) | CAD Volume (mm^3) | CAD Volume (m^3) | Qty. | Total External Surface Area (m^2) | Total Volume (m^3) |
|---|---|---|---|---|---|---|---|---|---|---|
| Wing | Total | External | 4.25 | 22575559.79 | 22.57555979 | 184187825.3 | 0.1841878253 | 2 | 45.15111959 | 0.3683756506 |
| | Side | | | 78553.01062 | 0.07855301062 | | | | | |
| | Top | | | 11232498.91 | 11.23249891 | | | | | |
| | Bottom | | | 11264507.87 | 11.26450787 | | | | | |
| Wing Body Extension | Total | Internal | | 4287121.896 | 4.287121896 | 38035033.27 | 0.03803503327 | 2 | | 0.07607006654 |
| | Top | | | 2135986.823 | 2.135986823 | | | | | |
| | Bottom | | | 2119866.55 | 2.11986655 | | | | | |
| | Front | | | 31268.5235 | 0.0312685235 | | | | | |
| Aileron | Total | Internal | 0.75 | 6838869.144 | 6.838869144 | 157368131.8 | 0.1573681318 | 2 | 13.67773829 | 0.3147362636 |
| | Top | | | 3599372.555 | 3.599372555 | | | | | |
| | Bottom | | | 3239496.59 | 3.23949659 | | | | | |
| | Front | | | | 0 | | | | | |
| Wing+Aileron | Total | External | 4.25 | 29414428.94 | 29.41442894 | 341555957.1 | 0.3415559571 | 2 | 58.82885788 | 0.6831119142 |
| | | | | | | | | | | |
| | | | | | | | | | | |
| | | | | | | | | | | |

# Appendix B: Sectional External Surface Area and Volume

**Visualization of Surface Area Distribution per 0.25m section**

| Fuselage X-Cord.(m) | | Fuselage | | | Wing | Horizontal Tail | Vertical Tail |
|---|---|---|---|---|---|---|---|
| 0 | | | | | | | |
| 0.25 | 0.353028232 | | | | | | |
| 0.5 | 0.353028232 | | | | | | |
| 0.75 | 0.353028232 | | | | | | |
| 1 | 0.353028232 | | | | | | |
| 1.25 | | 0.5177171816 | | | | | |
| 1.5 | | 0.5177171816 | | | | | |
| 1.75 | | 0.5177171816 | 0.3010246919 | | | | |
| 2 | | 0.5177171816 | 0.3010246919 | | | | |
| 2.25 | | 0.5177171816 | 0.3010246919 | | | | |
| 2.5 | | 0.5177171816 | 0.3010246919 | | | | |
| 2.75 | | 0.5177171816 | 0.3010246919 | | | | |
| 3 | 0.762229997 | 0.5177171816 | 0.3010246919 | | | | |
| 3.25 | | | 0.3010246919 | 1.458469386 | | | |
| 3.5 | | | | 1.458469386 | | | |
| 3.75 | | | | 1.458469386 | | | |
| 4 | | | | 1.458469386 | | | |
| 4.25 | 1.286622893 | 0.3088869846 | | | | | |
| 4.5 | 1.286622893 | 0.3088869846 | | | 3.460521052 | | |
| 4.75 | 1.286622893 | 0.3088869846 | | | 3.460521052 | | |
| 5 | 1.47245342 | 0.1918955051 | | | 3.460521052 | | |
| 5.25 | 1.47245342 | 0.1918955051 | | | 3.460521052 | | |
| 5.5 | 1.47245342 | 0.1918955051 | | | 3.460521052 | | |
| 5.75 | 1.47245342 | 0.1918955051 | | | 3.460521052 | | |
| 6 | 1.47245342 | 0.1918955051 | | | 3.460521052 | | |
| 6.25 | 1.435217417 | 0.1173228157 | | | 3.460521052 | | |
| 6.5 | 1.435217417 | 0.1173228157 | | | 3.460521052 | | |
| 6.75 | 1.435217417 | 0.1173228157 | | | 3.460521052 | | |
| 7 | 1.435217417 | 0.1173228157 | | | 3.460521052 | | |
| 7.25 | 1.435217417 | 0.1173228157 | | | 3.460521052 | | |
| 7.5 | 1.435217417 | 0.1173228157 | | | 3.460521052 | | |
| 7.75 | 1.435217417 | 0.1173228157 | | | 3.460521052 | | 0.7325522732 |
| 8 | 1.035797156 | 0.1565668839 | | | 3.460521052 | | 0.7325522732 |
| 8.25 | 1.035797156 | 0.1565668839 | | | 3.460521052 | | 0.7325522732 |
| 8.5 | 1.035797156 | 0.1565668839 | | | 3.460521052 | 1.308926688 | 0.7325522732 |
| 8.75 | 1.040636573 | | | | 3.460521052 | 1.308926688 | 0.7325522732 |
| 9 | 1.040636573 | 0.1447763457 | 2.255859519 | | | 1.308926688 | 0.7325522732 |
| 9.25 | 1.040636573 | 0.1447763457 | 2.255859519 | | | 1.308926688 | 0.7325522732 |
| 9.5 | 1.040636573 | 0.1447763457 | 2.255859519 | | | 1.308926688 | 0.7325522732 |
| 9.75 | 0.8609953057 | | 2.255859519 | | | 1.308926688 | 0.7325522732 |
| 10 | 0 | | 0 | | | 0 | 0 |

**Visualization of Volume Distribution per 0.25m section**

| Fuselage X-Cord. (m) | | Fuselage | | | Wing | Horizontal Tail | Vertical Tail |
|---|---|---|---|---|---|---|---|
| 0 | | | | | | | |
| 0.25 | 0.00670725 | | | | | | |
| 0.5 | 0.00670725 | | | | | | |
| 0.75 | 0.00670725 | | | | | | |
| 1 | 0.00670725 | | | | | | |
| 1.25 | | 0.01382894648 | | | | | |
| 1.5 | | 0.01382894648 | | | | | |
| 1.75 | | 0.01382894648 | 0.009324241886 | | | | |
| 2 | | 0.01382894648 | 0.009324241886 | | | | |
| 2.25 | | 0.01382894648 | 0.009324241886 | | | | |
| 2.5 | | 0.01382894648 | 0.009324241886 | | | | |
| 2.75 | | 0.01382894648 | 0.009324241886 | | | | |
| 3 | 0.01783420661 | 0.01382894648 | 0.009324241886 | | | | |
| 3.25 | | | 0.009324241886 | 0.01560448457 | | | |
| 3.5 | | | | 0.01560448457 | | | |
| 3.75 | | | | 0.01560448457 | | | |
| 4 | | | | 0.01560448457 | | | |
| 4.25 | 0.01739338627 | 0.002262982145 | | | | | |
| 4.5 | 0.01739338627 | 0.002262982145 | | | 0.04018305378 | | |
| 4.75 | 0.01739338627 | 0.002262982145 | | | 0.04018305378 | | |
| 5 | 0.02025306164 | 0.001658819769 | | | 0.04018305378 | | |
| 5.25 | 0.02025306164 | 0.001658819769 | | | 0.04018305378 | | |
| 5.5 | 0.02025306164 | 0.001658819769 | | | 0.04018305378 | | |
| 5.75 | 0.02025306164 | 0.001658819769 | | | 0.04018305378 | | |
| 6 | 0.02025306164 | 0.001658819769 | | | 0.04018305378 | | |
| 6.25 | 0.02095710059 | 0.001112857186 | | | 0.04018305378 | | |
| 6.5 | 0.02095710059 | 0.001112857186 | | | 0.04018305378 | | |
| 6.75 | 0.02095710059 | 0.001112857186 | | | 0.04018305378 | | |
| 7 | 0.02095710059 | 0.001112857186 | | | 0.04018305378 | | |
| 7.25 | 0.02095710059 | 0.001112857186 | | | 0.04018305378 | | |
| 7.5 | 0.02095710059 | 0.001112857186 | | | 0.04018305378 | | |
| 7.75 | 0.02095710059 | 0.001112857186 | | | 0.04018305378 | | 0.00566298772 |
| 8 | 0.0143486695 | 0.001149213031 | | | 0.04018305378 | | 0.00566298772 |
| 8.25 | 0.0143486695 | 0.001149213031 | | | 0.04018305378 | | 0.00566298772 |
| 8.5 | 0.0143486695 | 0.001149213031 | | | 0.04018305378 | 0.00263859764 | 0.00566298772 |
| 8.75 | 0.01359391369 | | | | 0.04018305378 | 0.00263859764 | 0.00566298772 |
| 9 | 0.01359391369 | 0.001225827233 | 0.03812471345 | | | 0.00263859764 | 0.00566298772 |
| 9.25 | 0.01359391369 | 0.001225827233 | 0.03812471345 | | | 0.00263859764 | 0.00566298772 |
| 9.5 | 0.01359391369 | 0.001225827233 | 0.03812471345 | | | 0.00263859764 | 0.00566298772 |
| 9.75 | 0.05542717751 | | 0.03812471345 | | | 0.00263859764 | 0.00566298772 |
| 10 | 0 | | 0 | | | 0 | 0 |

**Total Surface Area Distribution per 0.25m section**

| | Fuselage | Wing | Horizontal Tail | Vertical Tail | **Total:** |
|---|---|---|---|---|---|
| Fuselage Y-Cord. | Total Surface Area from Fuselage | Total Surface Area from Wing | Total Surface Area from HT | Total Surface Area from VT | Total External Surface Area |
| 0 | 0 | 0 | 0 | 0 | 0 |
| 0.25 | 0.353028232 | 0 | 0 | 0 | 0.353028232 |
| 0.5 | 0.353028232 | 0 | 0 | 0 | 0.353028232 |
| 0.75 | 0.353028232 | 0 | 0 | 0 | 0.353028232 |
| 1 | 0.353028232 | 0 | 0 | 0 | 0.353028232 |
| 1.25 | 0.5177171816 | 0 | 0 | 0 | 0.5177171816 |
| 1.5 | 0.5177171816 | 0 | 0 | 0 | 0.5177171816 |
| 1.75 | 0.8187418735 | 0 | 0 | 0 | 0.8187418735 |
| 2 | 0.8187418735 | 0 | 0 | 0 | 0.8187418735 |
| 2.25 | 0.8187418735 | 0 | 0 | 0 | 0.8187418735 |
| 2.5 | 0.8187418735 | 0 | 0 | 0 | 0.8187418735 |
| 2.75 | 0.8187418735 | 0 | 0 | 0 | 0.8187418735 |
| 3 | 1.58097187 | 0 | 0 | 0 | 1.58097187 |
| 3.25 | 1.759494078 | 0 | 0 | 0 | 1.759494078 |
| 3.5 | 1.458469386 | 0 | 0 | 0 | 1.458469386 |
| 3.75 | 1.458469386 | 0 | 0 | 0 | 1.458469386 |
| 4 | 1.458469386 | 0 | 0 | 0 | 1.458469386 |
| 4.25 | 1.595509878 | 0 | 0 | 0 | 1.595509878 |
| 4.5 | 1.595509878 | 3.460521052 | 0 | 0 | 5.056030929 |
| 4.75 | 1.595509878 | 3.460521052 | 0 | 0 | 5.056030929 |
| 5 | 1.664348925 | 3.460521052 | 0 | 0 | 5.124869976 |
| 5.25 | 1.664348925 | 3.460521052 | 0 | 0 | 5.124869976 |
| 5.5 | 1.664348925 | 3.460521052 | 0 | 0 | 5.124869976 |
| 5.75 | 1.664348925 | 3.460521052 | 0 | 0 | 5.124869976 |
| 6 | 1.664348925 | 3.460521052 | 0 | 0 | 5.124869976 |
| 6.25 | 1.552540232 | 3.460521052 | 0 | 0 | 5.013061284 |
| 6.5 | 1.552540232 | 3.460521052 | 0 | 0 | 5.013061284 |
| 6.75 | 1.552540232 | 3.460521052 | 0 | 0 | 5.013061284 |
| 7 | 1.552540232 | 3.460521052 | 0 | 0 | 5.013061284 |
| 7.25 | 1.552540232 | 3.460521052 | 0 | 0 | 5.013061284 |
| 7.5 | 1.552540232 | 3.460521052 | 0 | 0 | 5.013061284 |
| 7.75 | 1.552540232 | 3.460521052 | 0 | 0.7325522732 | 5.745613557 |
| 8 | 1.19236404 | 3.460521052 | 0 | 0.7325522732 | 5.385437365 |
| 8.25 | 1.19236404 | 3.460521052 | 0 | 0.7325522732 | 5.385437365 |
| 8.5 | 1.19236404 | 3.460521052 | 1.308926688 | 0.7325522732 | 6.694364053 |
| 8.75 | 1.040636573 | 3.460521052 | 1.308926688 | 0.7325522732 | 6.542636586 |
| 9 | 3.441272437 | 0 | 1.308926688 | 0.7325522732 | 5.482751398 |
| 9.25 | 3.441272437 | 0 | 1.308926688 | 0.7325522732 | 5.482751398 |
| 9.5 | 3.441272437 | 0 | 1.308926688 | 0.7325522732 | 5.482751398 |
| 9.75 | 3.116854824 | 0 | 1.308926688 | 0.7325522732 | 5.158333785 |
| 10 | 0 | 0 | 0 | 0 | 0 |
| Total: | 56.29158748 | 62.28937893 | 7.853560128 | 6.592970459 | 133.027497 |

**Total Volume Distribution per 0.25m section**

| | Fuselage | Wing | Horizontal Tail | Vertical Tail | **Total:** |
|---|---|---|---|---|---|
| Fuselage Y-Cord. | Total Volume from Fuselage | Total Volume from Wing | 0 | 0 | Total External Surface Area |
| 0 | 0 | 0 | 0 | 0 | 0 |
| 0.25 | 0.00670725 | 0 | 0 | 0 | 0.00670725 |
| 0.5 | 0.00670725 | 0 | 0 | 0 | 0.00670725 |
| 0.75 | 0.00670725 | 0 | 0 | 0 | 0.00670725 |
| 1 | 0.00670725 | 0 | 0 | 0 | 0.00670725 |
| 1.25 | 0.01382894648 | 0 | 0 | 0 | 0.01382894648 |
| 1.5 | 0.01382894648 | 0 | 0 | 0 | 0.01382894648 |
| 1.75 | 0.02315318837 | 0 | 0 | 0 | 0.02315318837 |
| 2 | 0.02315318837 | 0 | 0 | 0 | 0.02315318837 |
| 2.25 | 0.02315318837 | 0 | 0 | 0 | 0.02315318837 |
| 2.5 | 0.02315318837 | 0 | 0 | 0 | 0.02315318837 |
| 2.75 | 0.02315318837 | 0 | 0 | 0 | 0.02315318837 |
| 3 | 0.04098739497 | 0 | 0 | 0 | 0.04098739497 |
| 3.25 | 0.02492872646 | 0 | 0 | 0 | 0.02492872646 |
| 3.5 | 0.01560448457 | 0 | 0 | 0 | 0.01560448457 |
| 3.75 | 0.01560448457 | 0 | 0 | 0 | 0.01560448457 |
| 4 | 0.01560448457 | 0 | 0 | 0 | 0.01560448457 |
| 4.25 | 0.01965636842 | 0 | 0 | 0 | 0.01965636842 |
| 4.5 | 0.01965636842 | 0.04018305378 | 0 | 0 | 0.0598394222 |
| 4.75 | 0.01965636842 | 0.04018305378 | 0 | 0 | 0.0598394222 |
| 5 | 0.02191188141 | 0.04018305378 | 0 | 0 | 0.06209493519 |
| 5.25 | 0.02191188141 | 0.04018305378 | 0 | 0 | 0.06209493519 |
| 5.5 | 0.02191188141 | 0.04018305378 | 0 | 0 | 0.06209493519 |
| 5.75 | 0.02191188141 | 0.04018305378 | 0 | 0 | 0.06209493519 |
| 6 | 0.02191188141 | 0.04018305378 | 0 | 0 | 0.06209493519 |
| 6.25 | 0.02206995777 | 0.04018305378 | 0 | 0 | 0.06225301155 |
| 6.5 | 0.02206995777 | 0.04018305378 | 0 | 0 | 0.06225301155 |
| 6.75 | 0.02206995777 | 0.04018305378 | 0 | 0 | 0.06225301155 |
| 7 | 0.02206995777 | 0.04018305378 | 0 | 0 | 0.06225301155 |
| 7.25 | 0.02206995777 | 0.04018305378 | 0 | 0 | 0.06225301155 |
| 7.5 | 0.02206995777 | 0.04018305378 | 0 | 0 | 0.06225301155 |
| 7.75 | 0.02206995777 | 0.04018305378 | 0 | 0.005662987723 | 0.06791599928 |
| 8 | 0.01549788253 | 0.04018305378 | 0 | 0.005662987723 | 0.06134392403 |
| 8.25 | 0.01549788253 | 0.04018305378 | 0 | 0.005662987723 | 0.06134392403 |
| 8.5 | 0.01549788253 | 0.04018305378 | 0.002638597645 | 0.005662987723 | 0.06398252167 |
| 8.75 | 0.01359391369 | 0.04018305378 | 0.002638597645 | 0.005662987723 | 0.06207855284 |
| 9 | 0.05294445437 | 0 | 0.002638597645 | 0.005662987723 | 0.06124603974 |
| 9.25 | 0.05294445437 | 0 | 0.002638597645 | 0.005662987723 | 0.06124603974 |
| 9.5 | 0.05294445437 | 0 | 0.002638597645 | 0.005662987723 | 0.06124603974 |
| 9.75 | 0.09355189095 | 0 | 0.002638597645 | 0.005662987723 | 0.1018534763 |
| 10 | 0 | 0 | 0 | 0 | 0 |
| Total: | 0.918473442 | 0.723294968 | 0.01583158587 | 0.05096688951 | 1.708566885 |

# Appendix C: Weight Calculations

**Material and Weight Calculations for each analyzed Aircraft Part:**

| | | | | Material Properties | | Weight Calculations | | |
|---|---|---|---|---|---|---|---|---|
| | | Part | QTY. | Material | Density (kg/m^3) | Volume (m^3) | Calculated Mass (kg) | Total Calculated Mass (kg) |
| **Fuselage** | **Shell** | Nose Cone | 1 | Carbon Fiber M55J | 1710 | 0.01353398159 | 23.14310852 | 23.14310852 |
| | | Cockpit Body | 1 | Titanium | 4420 | 0.1106315718 | 488.9915474 | 488.9915474 |
| | | Cockpit Glass | 1 | Acrylic | 1180 | 0.06713454158 | 79.21875906 | 79.21875906 |
| | | Air Intake | 2 | Carbon Fiber M55J | 1710 | 0.007133682643 | 12.19859732 | 24.39719464 |
| | | Body 1 | 1 | Carbon Fiber M55J | 1710 | 0.06241793829 | 106.7346745 | 106.7346745 |
| | | Body 2,3,6,8 | 1 | Carbon Fiber M55J | 1710 | 0.2580872004 | 441.3291128 | 441.3291128 |
| | | Body 4 | 1 | Carbon Fiber M55J | 1710 | 0.006356347255 | 10.86935381 | 10.86935381 |
| | | Body 5 | 1 | Carbon Fiber M55J | 1710 | 0.008013704038 | 13.7034339 | 13.7034339 |
| | | Body 7 | 1 | Carbon Fiber M55J | 1710 | 0.007456851609 | 12.75121625 | 12.75121625 |
| | | Body 9 | 1 | Carbon Fiber M55J | 1710 | 0.003414167837 | 5.838227001 | 5.838227001 |
| | | Body 10 | 1 | Carbon Fiber M55J | 1710 | 0.05278495603 | 90.26227481 | 90.26227481 |
| | | Body 11 | 1 | Carbon Fiber M55J | 1710 | 0.0036774817 | 6.288493707 | 6.288493707 |
| | | Body 12 | 1 | Carbon Fiber M55J | 1710 | 0.04352376317 | 74.42563502 | 74.42563502 |
| | **Structure** | Structure 1 | 1 | Titanium | 4420 | 0.02274567798 | 100.5358967 | 100.5358967 |
| | | Structure 2 | 1 | Titanium | 4420 | 0.00378050348 | 16.70982538 | 16.70982538 |
| | | Rib 1 | 1 | Carbon Fiber M55J | 1710 | 0.0234166452 | 40.0424633 | 40.0424633 |
| | | Rib 1 Structure | 1 | Titanium | 4420 | 0.01114158633 | 49.24581156 | 49.24581156 |
| | | Rib 2 | 1 | Carbon Fiber M55J | 1710 | 0.03467306559 | 59.29094216 | 59.29094216 |
| | | Rib 2 Structure | 2 | Titanium | 4420 | 0.004075318257 | 18.01290669 | 36.02581339 |
| | | Structure 3 | 1 | Titanium | 4420 | 0.004409982122 | 19.49212098 | 19.49212098 |
| | | Structure 4 | 1 | Titanium | 4420 | 0.01284669679 | 56.7823998 | 56.7823998 |
| | | Rib 3 | 1 | Carbon Fiber M55J | 1710 | 0.01923809394 | 32.89714065 | 32.89714065 |
| | | Rib 3 Structure | 1 | Titanium | 4420 | 0.006137844337 | 27.12927197 | 27.12927197 |
| | | Structure 4 | 1 | Titanium | 4420 | 0.005289967134 | 23.38165473 | 23.38165473 |
| | | Structure 5 | 1 | Titanium | 4420 | 0.007545587934 | 33.35149867 | 33.35149867 |
| | | Rib 4 | 1 | Carbon Fiber M55J | 1710 | 0.0008778347677 | 1.501097453 | 1.501097453 |
| **Wing** | **Shell** | Wing | 2 | Carbon Fiber M55lt | 1710 | 0.1777693035 | 303.985509 | 607.971018 |
| | | Aileron | 2 | Forged light weight CF | 90 | 0.1573681318 | 14.16313186 | 28.32626372 |
| | **Structure** | | | | | | 0 | 0 |
| | | | | | | | 0 | 0 |
| | | | | | | | 0 | 0 |
| | | | | | | | 0 | 0 |
| **Horizontal Tail** | **Shell** | HT | 2 | Carbon Fiber M55J | 1710 | 0.04432327489 | 75.79280006 | 151.5856001 |
| | | Elevator | | | | | 0 | 0 |
| | **Structure** | | | | | | 0 | 0 |
| | | | | | | | 0 | 0 |
| | | | | | | | 0 | 0 |
| **Vertical Tail** | **Shell** | VT | 1 | Carbon Fiber M55lt | 1710 | 0.07466081957 | 127.6700015 | 127.6700015 |
| | | Rudder | | | | | 0 | 0 |
| | **Structure** | | | | | | 0 | 0 |
| | | | | | | | 0 | 0 |
| | | | | | | | 0 | 0 |
| **Propulsion System** | **Shell** | Intake | 2 | Carbon Fiber M55J | 1710 | 0.02171761106 | 37.13711491 | 74.27422982 |
| | | Center Tube | 2 | Titanium | 4420 | 0.06848918076 | 302.722179 | 605.4443579 |
| | | External Engine Casing | 2 | Titanium | 4420 | 0.09531178362 | 421.2780836 | 842.5561672 |
| | **Internals** | Turbo Fan | 2 | | | | 185 | 370 |
| | | J-97 Engine | 2 | | | | 130 | 260 |
| **Subsystems** | **Section 1 (Cyan)** | Avionics<br>Pilot Control | 1 | | | 0.05048213332 | 53.00623999 | 53.00623999 |
| | **Section 2 (Purple)** | Ejection Seat | 1 | | | 0.02915236359 | 228.2816645 | 228.2816645 |
| | **Section 3 (Pink)** | Radar<br>GPS/surveillance<br>Fire Control system | 1 | | | 0.01442098563 | 15.14203491 | 15.14203491 |
| | **Section 4 (Brown)** | Flight Computer | 1 | | | 0.006101012144 | 6.40606275 | 6.40606275 |
| | **Section 5 (Blue)** | Hydraulic<br>Main Cannon<br>Electrical Control System | 1 | | | 0.03866291954 | 171.2767336 | 171.2767336 |
| | **Section 6 (Emerald)** | Upper Body Fuel Tank | 1 | | | 0.7220829967 | 559.6143224 | 559.6143224 |
| | **Section 7 (Dark Green)** | In-Wing Fuel Tank | 2 | | | 0.6699271089 | 519.1935094 | 1038.387019 |
| | **Section 8 (Neon Green)** | Inner Body Fuel Tank | 1 | | | 0.2143075768 | 166.088372 | 166.088372 |
| | **Section 9 (Orange)** | Engine Control<br>Electric Power Generation | 1 | | | 0.03215182322 | 142.4325769 | 142.4325769 |
| | **Section 10 (Red)** | Backup Power<br>Redundant Control System<br>Redundant Electrical System | 1 | | | 0.009663338985 | 42.8085917 | 42.8085917 |
| | | | | | | | Total = | 7365.610224 |

**<u>Center of Mass and Moments for each analyzed Aircraft Part:</u>**

| | | | Center of Mass Coordinate (mm) | | | | Initial Moment Calculations (kg*mm) | | |
|---|---|---|---|---|---|---|---|---|---|
| | | Part | CAD Y-cord. (mm) | X-Cord. | FRL Y-Cord | Z-Cord | Moment about X | Momement abut Y | Moment about Z |
| Fuselage | Shell | Nose Cone | -2351.3273 | 0 | 648.6727 | 0 | 15012.30269 | 0 | 0 |
| | | Cockpit Body | -470.5625 | 0 | 2529.4375 | -24.023 | 1236873.557 | -11747.04394 | 0 |
| | | Cockpit Glass | -471.7785 | 0 | 2528.2215 | 491.2175 | 200282.5699 | 38913.64078 | 0 |
| | | Air Intake | -72.3148368 | 0 | 2927.685163 | -107.6321917 | 71427.30477 | -2625.92353 | 0 |
| | | Body 1 | 531.7208046 | 0 | 3531.720805 | -278.8646655 | 376957.0704 | -29764.5293 | 0 |
| | | Body 2,3,6,8 | 2861.211052 | 0 | 5861.211052 | -360.8509604 | 2586723.073 | -159254.0342 | 0 |
| | | Body 4 | 1367.867918 | 0 | 4367.867918 | 394.072634 | 47475.90178 | 4283.314885 | 0 |
| | | Body 5 | 2329.588817 | 0 | 5329.588817 | 347.7520032 | 73033.66809 | 4765.396591 | 0 |
| | | Body 7 | 3818.667733 | 0 | 6818.667733 | 248.8075194 | 86946.30681 | 3172.598485 | 0 |
| | | Body 9 | 5074.143154 | 0 | 8074.143154 | 63.74919032 | 47138.68057 | 372.1822442 | 0 |
| | | Body 10 | 5918.382814 | 0 | 8918.382814 | -451.6135255 | 804993.5204 | -40763.66415 | 0 |
| | | Body 11 | 5860.096541 | 0 | 8860.096541 | -53.23482343 | 55716.66134 | -334.7668521 | 0 |
| | | Body 12 | 6678.651634 | 0 | 9678.651634 | -358.3275924 | 720339.794 | -26668.75861 | 0 |
| | Structure | Structure 1 | 1496.919425 | 0 | 4496.919425 | -613.9511371 | 452101.8266 | -61724.12808 | 0 |
| | | Structure 2 | 1047.499484 | 0 | 4047.499484 | -666.8542948 | 67633.00961 | -11143.01882 | 0 |
| | | Rib 1 | 1759.999985 | 0 | 4759.999985 | -116.6699658 | 190602.1247 | -4671.752824 | 0 |
| | | Rib 1 Structure | 1799.95824 | 0 | 4799.95824 | -480.8828786 | 236377.839 | -23681.46762 | 0 |
| | | Rib 2 | 3014.999977 | 0 | 6014.999977 | -123.0783522 | 356635.0157 | -7297.431464 | 0 |
| | | Rib 2 Structure | 3029.805886 | 0 | 6029.805886 | -69.5836161 | 217228.6616 | -2506.806369 | 0 |
| | | Structure 3 | 3067.381584 | 0 | 6067.381584 | -668.2270374 | 118266.1359 | -13025.16226 | 0 |
| | | Structure 4 | 3849.815829 | 0 | 6849.815829 | -608.0371913 | 388948.981 | -34525.81089 | 0 |
| | | Rib 3 | 4709.999982 | 0 | 7709.999982 | -163.0172709 | 253636.9538 | -5362.802088 | 0 |
| | | Rib 3 Structure | 4712.872128 | 0 | 7712.872128 | -240.7555232 | 209244.6056 | -6531.522067 | 0 |
| | | Structure 4 | 5049.815342 | 0 | 8049.815342 | -585.8482802 | 188218.003 | -13698.10221 | 0 |
| | | Structure 5 | 5899.473668 | 0 | 8899.473668 | -548.423153 | 296810.7842 | -18290.73406 | 0 |
| | | Rib 4 | 6399.5 | 0 | 9399.5 | -369.4736603 | 14109.56551 | -554.6159703 | 0 |
| Wing | Shell | Wing | 3546.952855 | 0 | 6546.952855 | -135.3604409 | 3980357.592 | -82295.22505 | 0 |
| | | Aileron | 4977.321361 | 0 | 7977.321361 | -135.746667 | 225967.7087 | -3845.195889 | 0 |
| | Structure | | | 0 | | | 0 | 0 | 0 |
| | | | | 0 | | | 0 | 0 | 0 |
| | | | | 0 | | | 0 | 0 | 0 |
| | | | | 0 | | | 0 | 0 | 0 |
| Horizontal Tail | Shell | HT | 6441.139295 | 0 | 9441.139295 | -463.2809296 | 1431140.766 | -70226.71774 | 0 |
| | | Elevator | | 0 | | | 0 | 0 | 0 |
| | Structure | | | 0 | | | 0 | 0 | 0 |
| | | | | 0 | | | 0 | 0 | 0 |
| | | | | 0 | | | 0 | 0 | 0 |
| Vertical Tail | Shell | VT | 6563.704751 | 0 | 9563.704751 | 497.080973 | 1220998.2 | 63462.32855 | 0 |
| | | Rudder | | 0 | | | 0 | 0 | 0 |
| | Structure | | | 0 | | | 0 | 0 | 0 |
| | | | | 0 | | | 0 | 0 | 0 |
| | | | | 0 | | | 0 | 0 | 0 |
| Propulsion System | Shell | Intake | 782.1905905 | 0 | 3782.190591 | -310.9542711 | 280919.2932 | -23095.889 | 0 |
| | | Center Tube | 3507.386432 | 0 | 6507.386432 | -320 | 3939860.4 | -193742.1945 | 0 |
| | | External Engine Casing | 6428.702567 | 0 | 9428.702567 | 284.6741411 | 7944211.496 | 239853.9532 | 0 |
| | Internals | Turbo Fan | 2629.757713 | 0 | 5629.757713 | -320 | 2083010.354 | -118400 | 0 |
| | | J-97 Engine | 6461.972361 | 0 | 9461.972361 | 411.5901485 | 2460112.814 | 107013.4386 | 0 |
| Subsystems | Section 1 (Cyan) | Avionics<br>Pilot Control | -1525.52575 | 0 | 1474.47425 | -6.47327363 | 78156.33595 | -343.1238956 | 0 |
| | Section 2 (Purple) | Ejection Seat | -231.56023 | 0 | 2768.43977 | -322.8552705 | 631984.0386 | -73701.93852 | 0 |
| | Section 3 (Pink) | Radar<br>GPS/surveillance<br>Fire Control system | 1374.991015 | 0 | 4374.991015 | 170.333353 | 66246.26668 | 2579.193578 | 0 |
| | Section 4 (Brown) | Flight Computer | 1252.144217 | 0 | 4252.144217 | -212.6353428 | 27239.50267 | -1362.155349 | 0 |
| | Section 5 (Blue) | Hydraulic<br>Main Cannon<br>Electrical Control System | 2250 | 0 | 5250 | -216.9433184 | 899202.8513 | -37157.34295 | 0 |
| | Section 6 (Emerald) | Upper Body Fuel Tank | 2820.286929 | 0 | 5820.286929 | 125.5997733 | 3257115.926 | 70287.43203 | 0 |
| | Section 7 (Dark Green) | In-Wing Fuel Tank | 3410.932032 | 0 | 6410.932032 | -114.9994255 | 6657028.6 | -119413.9106 | 0 |
| | Section 8 (Neon Green) | Inner Body Fuel Tank | 3502.5 | 0 | 6502.5 | -231.5176278 | 1079989.639 | -38452.38589 | 0 |
| | Section 9 (Orange) | Engine Control<br>Electric Power Generation | 4649.905955 | 0 | 7649.905955 | -250.159742 | 1089595.818 | -35630.89668 | 0 |
| | Section 10 (Red) | Backup Power<br>Redundant Control System<br>Redundant Electrical System | 6023.310537 | 0 | 9023.310537 | -330.6300887 | 386275.2165 | -14153.80847 | 0 |
| | | | | | | Total Moments: | 47052146.74 | -751289.3809 | 0 |

**DAPCA IV Model:**

| Inputs | | | | *mks* | *Fudged* |
|---|---|---|---|---|---|
| **We (kg)** | 4691 | | **Eng hours (HE)** | 1740772.579 | 1914849.837 |
| **V (km/h)** | 979.2 | | **Tooling hours (HT)** | 620521.6473 | 682573.812 |
| **Q** | 1 | | **Mfg hours (HM)** | 301445.4368 | 331589.9805 |
| **FTA** | 2 | | **QC hours (HQ)** | 0.133 | 0.1463 |
| **Neng** | 2 | | | | |
| **Tmax (kN)** | 18.47 | | | | |
| **Mmax** | 0.95 | | **Devel support cost (CD)** | 107075459.2 | |
| **Tturbine inlet (K)** | 1423.15 | | **Flt test cost (CF)** | 20196206.69 | |
| **Cavionics ($)** | 11900 | | **Mfg materials (CM)** | 5404072.915 | |
| **Labor rates (2023)** | | | **Engine production (Ceng)** | 2047038.434 | |
| **RE ($)** | 153.14 | | | | |
| **RT ($)** | 157.13 | | **RDT&E+flyaway** | 540204843.5 | |
| **RQ ($)** | 143.82 | | | | |
| **RM ($)** | 130.5 | | | | |

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