AEE 479 Final Design Report

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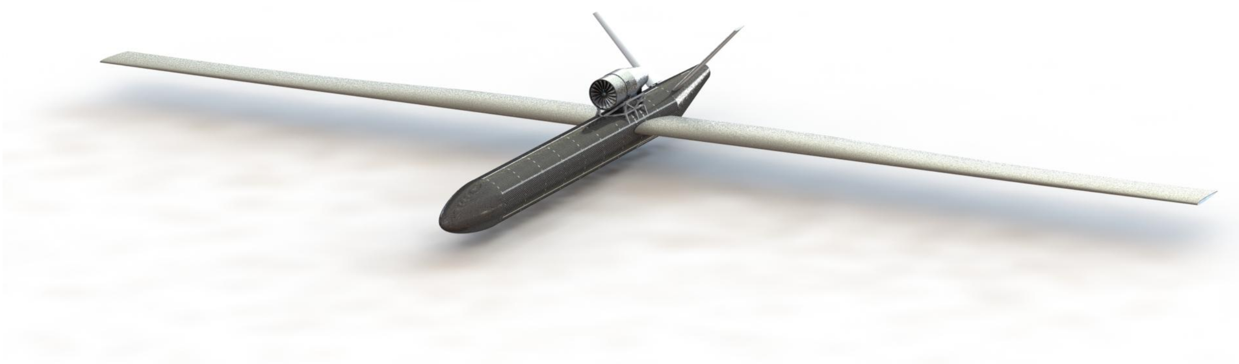
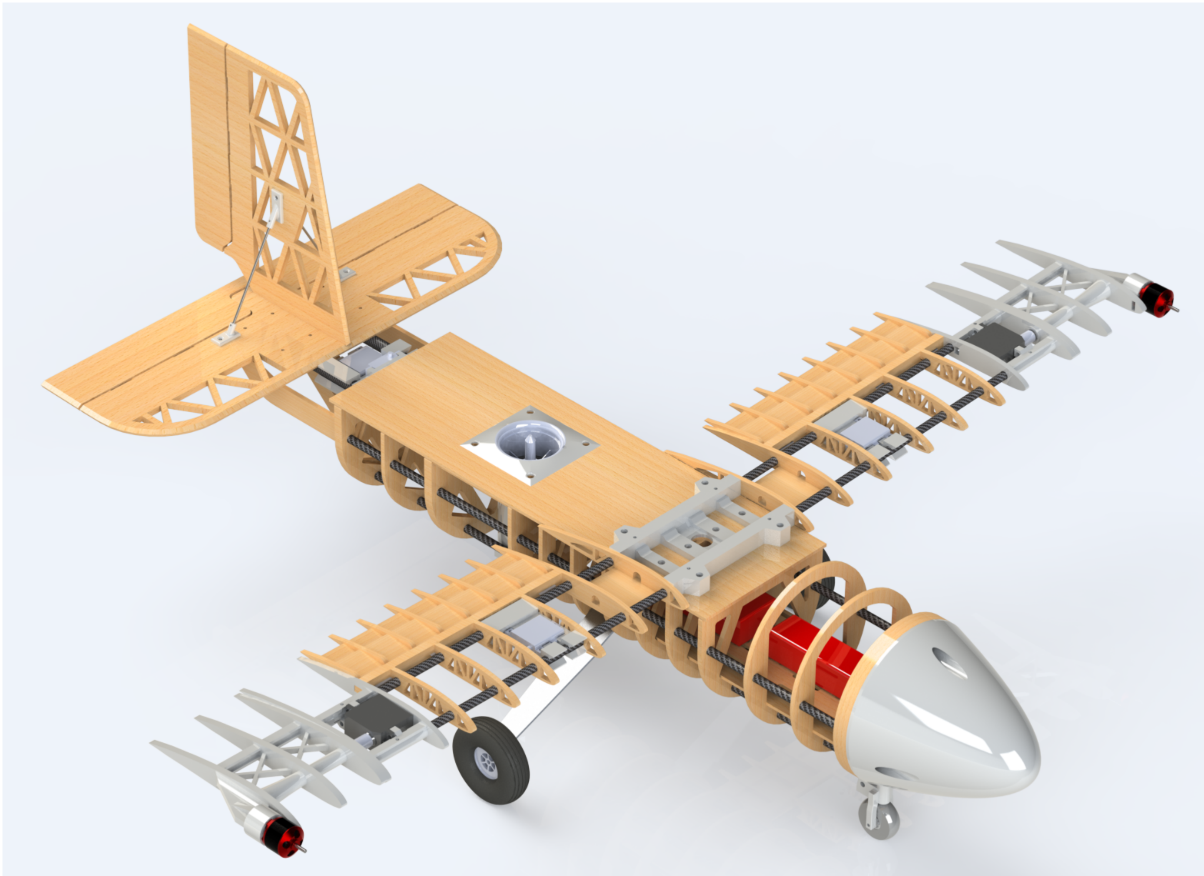
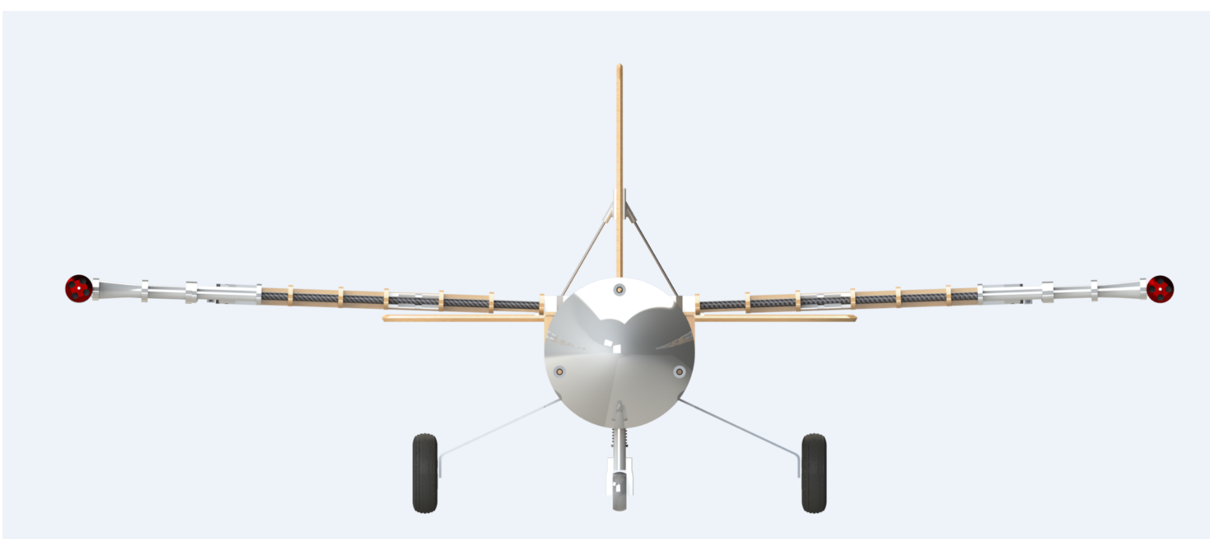
To fulfill the requirements of the United States Army outlined in subsequent sections, an unmanned aircraft system consisting of multiple autonomous aircraft has been proposed. As proposed, the aircraft system consists of a large carrier aircraft which will meet the range and endurance requirements of the United States Army. This carrier aircraft will deploy one or multiple payload aircraft which will perform the nape of the Earth surveillance. Within this aircraft system, the payload aircraft will be remotely operated by a human pilot. Based on the system requirements described by the U.S. Army in their Request for Proposal, potential alternative customers for either the entire aircraft system or the payload aircraft are discussed in the report. Additionally, the feasibility of meeting the needs of the alternative customers is evaluated allowing for moderate alterations to the proposed aircraft system. Based on this analysis, a recommendation regarding the selection of alternative customers for the concerned aircraft system is provided.







Nomenclature

W =weight
L =lift
D =drag
T =thrust
ρ =density
CD_0 =zero-lift drag coefficient
AR =aspect ratio
e =oswald efficiency
$VTAS_{br}$ =True airspeed for best range cruise
S_{ref} =reference area
$MTOW$ =max take-off weight
OEW =operational empty weight
$PYLD$ =payload
FOB =fuel on board
MF =mass fraction
ζ =fuel use ratio
ε_0 =downwash at zero angle of attack
i_t =tail incidence angle
a =lift curve slope
C_{m_0} =pitch moment coefficient at zero angle of attack
C_{m_α} =pitch moment coefficient
C_{l_β} =yaw moment coefficient
C_{n_β} =rolling moment coefficient
g =gravity
I_{yy} =moment of inertia

A. Unmanned Aircraft

1A. Market Study and Customer Analysis

1A.1 Primary Customer: U.S. Army

As described by the U.S. Army's Request for Proposal, the Army requires an unmanned aircraft surveillance system to be used behind the lines of theater over landmass areas during bad weather conditions. This aircraft is intended to be used in the gathering of battlefield information through nape-of-the-Earth surveillance and may consist of one large carrier aircraft with one or more payload aircraft that perform the bulk of data collection. To fulfill the stated mission, the U.S. Army provided an enumerated list of the system's requirements. The system requirements specified by the U.S. Army are summarized in Table 1A.

Requirement #	Statement
1.1.01	The data from the smaller vehicle shall be telemetered to the larger vehicle for capture and return.
1.1.02	The larger vehicle shall be reusable for a period of 10000 flights.
1.1.03	The reconnaissance portion of the system shall be controllable by a remote pilot through the larger vehicle telemetry system.
1.1.04	The design characteristics of the larger aircraft shall meet all applicable military standards for unmanned aircraft.
1.1.05	The smaller expendable vehicle shall comply with the standards of traditional radio control hobby aircraft.
1.1.06	The smaller vehicle should implement a recoverable design.
1.1.07	The aircraft system shall be able to perform system warm up and system checks in a period of 10 minutes.
1.1.08	The aircraft system shall take off within an 800 foot clearing with a 38 foot obstacle present.
1.1.09	The aircraft system shall climb at the best rate of climb to its cruising altitude.
1.1.10	The aircraft system shall be able to cruise for 510 nautical miles at a speed and altitude which minimizes station and fuel burn.
1.1.11	The aircraft system shall be able to then climb at best rate of climb to 25500 feet.
1.1.12	The aircraft system shall be able to proceed to gather nape of the earth (50 feet above local level) data in near real time for a period of 24 hours at the best endurance speed.
1.1.13	After the data collection period is completed, the aircraft system shall descend to best cruise altitude, with no credit for range.
1.1.14	The aircraft system shall then perform a return cruise segment of 510 nautical miles while minimizing fuel burn.
1.1.15	Once all other mission requirements are met, the aircraft system shall descend with no credit for range and land within an 800 foot clearing with a 38 foot obstacle present.
1.1.16	The aircraft system shall have 10% fuel reserve after mission completion.

1.1.17	The aircraft system shall be able to carry 400 pounds of payload not including any secondary aircraft designed for the mission.
1.1.18	The aircraft system shall be capable of performing a sustained 2.4 g maneuver at speed for minimum power setting and at 25000 feet altitude.
1.1.19	The aircraft system shall be capable of 5 minutes taxi time.
1.1.20	Any secondary aircraft designed for the mission shall be lighter than 20 pounds and shall only require one person to manipulate and prepare for flight in the primary aircraft system.

Table 1A. Requirements given by the U.S. Army.

1A.2. Methodology for Alternative Market Identification

Based on the aircraft system requirements provided by the United States Army, the primary features of the proposed carrier aircraft are the 1020 nm cruise range of the proposed system, the 24 hr endurance flight time, the 800 ft takeoff and landing distance, and the 400 lb payload capacity. Regarding the payload aircraft, its primary feature is the ability to perform low altitude (<50 ft) data collection. Combined, the entire aircraft system provides customers with data collection capabilities over a duration of 24 hr in fair and poor weather conditions. Based on the aforementioned features, potential alternative customers were identified. Additionally, the design team was approached (without solicitation) by representatives of the National Oceanic and Atmospheric Administration (NOAA) and a major oil and gas company both of which expressed interest in an aircraft system to fulfill needs similar to the intended role of the proposed aircraft. Consequently, an analysis of these two customers was also performed.

1A.3. Potential Alternative Customers

Based on the previously described methodology, potential alternative customers were identified. These customers include:

1. National Oceanic and Atmospheric Administration (NOAA)
2. A major oil and gas company
3. Land surveying company
4. US Coast Guard
5. Salt River Project (SRP)/Central Arizona Project (CAP)
6. Department of Homeland Security

The feasibility of adapting the proposed aircraft system to reflect the needs of each potential customer is discussed in greater detail in the below sections.

A. National Oceanic and Atmospheric Administration (NOAA)

As previously mentioned, the design team was approached by a representative of NOAA for an aircraft system that can perform meteorological data collection within a storm system similar to Northrop Grumman RQ-4 Global Hawk role as a storm tracker. Specifically, NOAA requires an aircraft system that can deploy multiple disposable aircraft into a storm from a carrier ship that maintains its position ahead of the storm system. Once deployed, each disposable aircraft shall measure atmospheric pressure, temperature, dew point, and the aircraft's position within the storm system and relay time averaged values to the carrier ship.

1. Customer Priorities and Requirements

As expressed by the NOAA representative, the primary factor for the adoption of the developed system is cost. With this restriction in mind, the NOAA representative outlined several system requirements provided in Table 2A. It should be noted that unlike the primary customer, NOAA expressed no major requirement on takeoff and landing, nor did they express interest in the recovery of the payload aircraft. Additionally, the storage of the aircraft was described as a non-issue.

Requirement #	Statement
1.2.01	The time average data from the smaller vehicle(s) shall be telemetered to the larger vehicle for capture and return at interval equal to or less than 5 minutes.
1.2.02	The reconnaissance portion of the system shall be controllable by a remote pilot through the larger vehicle telemetry system.
1.2.03	The design characteristics of the larger aircraft shall meet all applicable FAA standards for unmanned aircraft.
1.2.04	The smaller expendable vehicle shall comply with the standards of traditional radio control hobby aircraft.
1.2.05	The aircraft system shall be able to achieve wheels up within a period of 60 minutes.
1.2.06	The larger vehicle shall be able to lead a typical storm system (45 to 70 knots) for a period of 24 hours.
1.2.07	The aircraft system shall withstand storm conditions including but not limited to strong winds and lightning.
1.2.08	Each payload aircraft shall include atmospheric pressure, temperature, and dew point sensors. Time average values from these sensors and the aircraft's position shall be pushed back to the larger aircraft.
1.2.09	The aircraft system shall perform all required operations with a maximum support personnel of 3 individuals.
1.2.10	The payload aircraft shall collect the requisite meteorological data from an altitude of 15,000 to 30,000 ft.
1.2.11	The aircraft system shall have 10% fuel reserve after mission completion.
1.2.12	The aircraft system shall be capable of 5 minutes taxi time.
1.2.13	Any secondary aircraft designed for the mission shall be lighter than 20 pounds and shall only require one person to manipulate and prepare for flight in the primary aircraft system.

Table 2A. Requirements given by NOAA.

2. Common Requirements When Compared with Primary Customer

As observed in the comparison between Table 1A and Table 2A, NOAA requires comparable range and endurance characteristics to those required by the primary customer. However, unlike the primary customer, NOAA specified that the KTAS of the carrier aircraft must be at least that of a typical storm front (45 to 70 knots) during the endurance segment of the flight. Based on these requirements, a large aspect ratio aircraft similar to that suggested by the requirements of the primary customer would be preferable.

3. Requirements Specific to Customer and Estimated Cost and Risk of Accommodation

As expressed in Table 2A, few requirements exist that are not covered by the base requirements given by the U.S. Army. That said, the requirements given in Table 2A by NOAA are less stringent than the requirements specified by the U.S. Army. In particular, the cost of the aircraft system may be reduced by selecting a less powerful power plant which is allowable due to the lack of requirement on takeoff and landing. Additionally, the design of the payload aircrafts can be simplified since no requirement for reusability of the payload aircraft was given. These simplifications allow for the needs of NOAA to be met without significant investment or design changes.

4. Sale Volume and Estimated ROI

Although the estimated sale volume is low (2-3 systems), designing a lower cost system that fulfills the requirements of NOAA may lead to returns provided that other customers exist for the aircraft system. However, if no alternative customers exist which align with the needs specified by NOAA, the additional investment required to simplify the design will not produce sufficient returns to justify the additional expenditures.

B. Major oil and gas company

In addition to being approached by a NOAA representative, the design team was also approached by a representative of a major oil and gas company. During the subsequent meeting the oil and gas representative expressed interest in an aircraft system to provide fast data collection from potential oil pipeline breaks. As described by this potential customer, the aircraft system should be able to quickly reach the concerned area and record photographic and infrared images of the break.

1. Customer Priorities and Requirements

During the aforementioned discussion of the aircraft system, the oil and gas company representative expressed a primary concern for the speed of data acquisition. Additionally, no requirements were given for either the recovery of payload aircraft or the takeoff or landing of the aircraft. The base requirements described by the oil and gas representative are provided in Table 3A.

Requirement #	Statement
1.3.01	The aircraft system shall have a range equal to or greater than 510 nM. The aircraft shall cover this distance in a reasonable amount of time.
1.3.02	The reconnaissance portion of the system shall be controllable by a remote pilot through the larger vehicle telemetry system.
1.3.03	The design characteristics of the larger aircraft shall meet all applicable FAA standards for unmanned aircraft.
1.3.04	The smaller expendable vehicle shall comply with the standards of traditional radio control hobby aircraft.
1.3.05	The aircraft system shall be wheels up in less than 15 minutes from first call with a 5 minute taxi period.
1.3.06	Data collection shall occur at a maximum altitude of 500 ft.
1.3.07	Following data collection the smaller aircraft shall land at a minimum distance of 0.5 statute mile away from center of the affected environment.
1.3.08	Each payload aircraft shall include a visible and infrared camera. Near instantaneous images shall be relayed to the larger aircraft.
1.3.09	The aircraft system shall perform all required operations with a maximum support personnel of 3 individuals.
1.3.10	The aircraft system shall have 10% fuel reserve after mission completion.
1.3.11	Any secondary aircraft designed for the mission shall be lighter than 20 pounds and shall only require one person to manipulate and prepare for flight in the primary aircraft system.

Table 3A. Requirements given by the oil and gas company.

2. Common Requirements When Compared with Primary Customer

As observed in the comparison between Table 1A and Table 3A, the oil and gas company requires similar range characteristics to the primary customer, but provides no specification of endurance. Other listed requirements vary little with those of the primary customer, such as the aircraft system regulation standards, wheels-up time, and mission-completion fuel reserve. The payload aircraft additionally requires infrared capabilities on top of visible-light photography.

3. Requirements Specific to Customer and Estimated Cost and Risk of Accommodation

The most notable differences from the primary customer are the prioritization of speed. The time between first call and data collection should be minimized, and no extended operation of the aircraft system is necessary. The slender high-aspect ratio design suited for the primary customer must be replaced with a smaller more sleek design, saving on material and manufacturing costs. The high-thrust power plant needed for high-speed operations potentially offsets savings while increasing enroute fuel burn. The representative stated that fuel efficiency was of little concern, and noted few cost restrictions. However, this necessitates a drastic overall redesign of the primary aircraft and subsequent systems, requiring considerable investment of additional design time, resources, and testing. Such significant changes would reduce cross-compatibility with designs adhering to the primary customer's requirements, and cause internal costs to skyrocket.

4. Sale Volume and Estimated ROI

The oil and gas company showed interest in purchasing around 10 units. While this is promising, estimating the return on investment is dependent on the magnitude of the design change necessary to fulfill their requirements. Although there is no hard-and-fast design choice selected for the primary customer, basic aerodynamic principles suggest a large change to adapt the primary design. Further investigation will be required later in the design process, and comparing the cost of adaptation against the sale volume will determine the viability of this potential market.

C. Land surveying company

Land surveying in many extreme environments is currently partially completed by human-operated drones. These drones typically are available on the consumer market and have very little functionality beyond taking standard photographs of the land. An aircraft system with a sensor and equipment suite specifically suited towards gathering geographical data could prove very useful to a customer trying to survey large swaths of uninhabited and extreme terrain. Drones in this system could also be used to deliver survey equipment to remote locations in the field that would otherwise be difficult for human surveyors to access if they had to carry the additional 60 pounds of equipment that is needed.

1. Customer Priorities and Requirements

As expressed by the company representative, the primary factor for the adoption of the developed system is cost. With this restriction in mind, the land survey company representative outlined several system requirements provided in Table 4A. It should be noted that unlike the primary customer, this customer did not express any concerns regarding takeoff and landing distances, as the aircraft would operate out of large domestic airports for their purposes. The ability to use the smaller drone to ferry heavy survey equipment in and out of extreme environments was also mentioned, a requirement that is unique to this customer.

Requirement #	Statement
1.4.01	The reconnaissance portion of the system shall be controllable by a remote pilot through the larger vehicle telemetry system.
1.4.02	The design characteristics of the larger aircraft shall meet all applicable FAA standards for unmanned aircraft.
1.4.03	The smaller expendable vehicle shall comply with the standards of traditional radio control hobby aircraft.
1.4.04	Any secondary aircraft designed for the mission shall be lighter than 20 pounds and shall only require one person to manipulate and prepare for flight in the primary aircraft system.
1.4.05	The aircraft system as a whole shall be able to successfully deliver approximately 60

	pounds of land survey equipment to a remote location.
1.4.06	The aircraft system shall have 10% fuel reserve after mission completion.
1.4.07	The aircraft system shall have a range of approximately 1250 nautical miles.
1.4.08	The smaller expendable vehicle must be retrievable by the primary aircraft, or have a range of approximately 50 nautical miles.
1.4.09	The aircraft system must be capable of taking photographs and gathering basic GIS data including vector or raster data.
1.4.10	The larger vehicle shall be reusable for a period of 10000 flights.

Table 4A. Requirements given by the land surveying company

2. Common Requirements When Compared with Primary Customer

As observed in the comparison between Table 1A and Table 4A, the land survey requires a slightly higher range than the primary customer as well as a complex payload delivery system on the smaller aircraft, but provides less restrictive requirements for airports of operation. This customer also requires the aircraft be retrievable or be able to fly a much larger distance under its own power. This requirement is primarily a result of the client's environmental concerns.

3. Requirements Specific to Customer and Estimated Cost and Risk of Accommodation

This customer provides two new major requirements compared to the primary and other secondary customer. The payload delivery system needed to carry survey equipment into the field would increase the weight of the smaller aircraft itself, not to mention the effects that bulky survey equipment would have on flight dynamics. The possibility of payload shifting either in-flight or during takeoff is quite large and would prove detrimental to the small aircraft and its expensive payload. The storage compartment of the large aircraft may also have to be designed specifically to fit the survey equipment, depending on how modular the larger pieces of equipment are. The cost to design a model for the carrier aircraft specifically for these purposes may be too large to be economically viable.

4. Sale Volume and Estimated ROI

This customer would only be interested in a small number of units, approximately two. The applications of this system in the field are limited due to the fact that it cannot directly replace human surveyors, only aid them. Return on investment for this project is entirely dependent on whether or not the primary aircraft needs to be redesigned to fit the payload requirements of this customer. A more in-depth analysis of the viability of this customer is recommended further along the design process of the aircraft.

D. U.S. Coast Guard

The U.S. Coast Guard is responsible for maritime duties including search and rescue, maritime law enforcement, navigation assistance, ice breaking, environmental protection, and port security. All of these tasks could be more easily and effectively accomplished through the use of a UAV system. These tasks could benefit from real time RGB footage along with complementary sensors such as infrared and radar.

1. Customer Priorities and Requirements

The Coast Guard like the U.S. Army does not have cost as a strict requirement, but a competitive price is important for obtaining the contract as they do have a smaller budget than the U.S. Army. The Coast Guard has many potential uses for a UAV system and general requirements are listed in Table 5A.

Requirement #	Statement
1.5.01	The data from the smaller vehicle shall be telemetered to the larger vehicle for capture and return.
1.5.02	The larger vehicle shall be reusable for a period of 10000 flights.
1.5.03	The reconnaissance portion of the system shall be controllable by a remote pilot through the larger vehicle telemetry system.
1.5.04	The design characteristics of the larger aircraft shall meet all applicable military standards for unmanned aircraft.
1.5.05	The smaller expendable vehicle shall comply with the standards of traditional radio control hobby aircraft.
1.5.06	The smaller vehicle should implement a recoverable design.
1.5.07	The aircraft system shall be able to perform system warm up and system checks in a period of 10 minutes.
1.5.08	The aircraft system shall climb at the best rate of climb to its cruising altitude.
1.5.09	The aircraft system shall be able to cruise for 510 nautical miles at a speed and altitude which minimizes station and fuel burn.
1.5.10	The aircraft system shall be able to proceed to gather ground imagery of the earth at a height for optimal resolution of sensory in near real time for a period of 24 hours at the best endurance speed.
1.5.11	The aircraft system shall then perform a return cruise segment of 510 nautical miles while minimizing fuel burn.
1.5.12	The aircraft system shall have 10% fuel reserve after mission completion.
1.5.13	The larger aircraft system shall be able to carry 200 pounds of survival equipment with a potential volume up to 64 cubic feet.
1.5.14	The aircraft system shall be capable of performing a sustained 2.4 g maneuver at speed for minimum power setting and at 25000 feet altitude.
1.5.15	Any secondary aircraft designed for the mission shall be lighter than 20 pounds and shall only require one person to manipulate and prepare for flight in the primary aircraft system.

Table 5A. Requirements given by the Coast Guard

2. *Common Requirements When Compared with Primary Customer*

The Coast Guard shares many requirements with the primary customer. The primary customer, the U.S. Army, and the Coast Guard both require a UAV system for similar use cases, mostly earth surveillance and reconnaissance. As such, many of the requirements of the Coast Guard are similar to that of the U.S. Army

3. *Requirements Specific to Customer and Estimated Cost and Risk of Accommodation*

The Coast Guard requires the UAV system to be able to drop a payload of survival equipment at a precise location when required. The type of survival equipment can vary depending on the situation, but the Coast Guard has specified it will not exceed 200 pounds or 64 cubic feet. The primary customer requires a larger payload weight and volume, but does not require this payload to be released from the aircraft during flight as the Coast Guard requires. Although this requirement is unique, the cost of implementation is low and the risk is low to moderate. Both the Coast Guard and the primary customer require the larger vehicle to dispatch smaller drones during flight. For this requirement to be met, the large vehicle would have to possess a system for releasing the drones. This same system can be altered to accommodate dispatching both the drones and the required survival equipment.

4. Sale Volume and Estimated ROI

The Coast Guard would be interested in enough vehicles to cover all of the lower 48 states ocean coast line, as well as increased density in areas where port security is of demand and locations known to experience severe weather events such as hurricanes and tropical storms. The predicted medium to large sale volume along with the low to moderate risk and cost of adaptation means this project would likely provide a return on investment. The addition of a payload delivery system such as this also opens new channels with other military customers for different roles. Although it is predicted to be low risk and low cost, further analysis to determine whether altering the payload dispatch system would need to be done to confirm the return on investment.

E. SRP/CAP

Many utility companies use canal systems to move water across large watersheds. These canals are routinely inspected for a variety of issues including vandalism, worksite inspections, and potential leaks or breaches. These routine inspections could be made much more efficient through the use of automated data-gathering aircraft.

1. Customer Priorities and Requirements

The primary requirement for utility companies is cost. Water-based utility companies in particular operate on a narrow profit margin, so all costs must be carefully considered. Additionally, the storage and takeoff/landing sites for the aircraft would be relatively close to the actual flight path, so the customer needs only one main vehicle to perform daily operations.

Requirement #	Statement
1.6.01	The aircraft system shall have a range equal to or greater than 600 nM. The aircraft shall cover this distance within a 24 hour period.
1.6.02	The design characteristics of the aircraft shall meet all applicable FAA standards for unmanned aircraft.
1.6.03	Data collection shall occur at a maximum altitude of 500 ft.
1.6.04	The aircraft shall be capable of hovering for prolonged data collection at specific points along the flight path.
1.6.05	The aircraft shall include a visible and infrared camera and capture continuous footage from both cameras for the entire flight duration.
1.6.06	The aircraft system shall perform all required operations with a maximum support personnel of 3 individuals.
1.6.07	The aircraft system shall have 10% fuel reserve after mission completion.
1.6.08	The aircraft shall be reusable for a period of 5000 flights.

Table 6A. Requirements given by the oil and gas company.

2. Common Requirements When Compared with Primary Customer

The utility companies have a similar range requirements with the primary customer, because the aircraft needs to cover the entire length of various canal routes. They also have similar requirements in terms of data collection.

3. Requirements Specific to Customer and Estimated Cost and Risk of Accommodation

The utility companies have many specific requirements not common to the primary customer. Since their operation is based fairly close to the intended flight path, they would only need a single vehicle rather than a system of a larger and smaller vehicle(s). Additionally, they have a requirement to gather data over an extended period of time at specific locations, meaning the main vehicle needs to be hover-capable. These requirements represent a fairly significant departure from the primary customer's considerations, which translates to a higher cost of accommodation.

4. Sale Volume and Estimated ROI

The customer would require only a single drone to survey the length of the canal, and their requirements reflect a desire for each vehicle to be operational for 10-15 years. Although the vehicle described by the customer's requirements would be lower cost and present less risk than the primary customer's vehicle, a single sale every 10-15 years does not represent a desirable ROI.

F. Department of Homeland Security

The Department of Homeland Security is actively involved in the research and development of UAV/UAS systems with a variety of goals and objectives in mind. This includes surveillance and monitoring of suspicious areas domestically and internationally, counter-attacks to enemy Unmanned-Aerial-Systems, Response and Dispatch services to aid for aerial support, etc.

1. Customer Priorities and Requirements

A major priority of DHS is versatility and accuracy, since the missions carried out by given UAVs are very critical and important to national security. They will require both primary and multiple secondary deployment vehicles that can be easily modified to perform a variety of operations. A couple of examples include easily installable general purpose hardware compatibility, such as moving from traditional cameras to traditional heat signature sensing.

Requirement #	Statement
1.7.01	The aircraft system shall have a range equal to or greater than 800 nM.
1.7.02	The design parameters are to be met with FAA as well as internationally adopted standards.
1.7.03	Secondary UAV must be controllable securely and remotely by a licenced human operator.
1.7.04	The aircraft shall be capable of hovering for prolonged data collection at specific points along the flight path.
1.7.05	The primary aircraft must be able to comfortably provide deployment of at least 4 secondary aircraft.
1.7.06	The primary aircraft shall maintain communication and transmit and save data provided by the secondary UAVs..
1.7.07	The aircraft system shall be able to avoid radar detection within desired limits.
1.7.08	The secondary drones shall have kill programs on board in case of capture.

Table 7A. Requirements given by the Department of Homeland Security

2. Common Requirements When Compared with Primary Customer

This customer proves to be quite similar to the primary customer in their requirements. Payload, range, cost, and endurance capabilities are all nearly the same. Methods of data collection are quite similar, despite minor individual changes needed for specific agencies.

3. Requirements Specific to Customer and Estimated Cost and Risk of Accommodation

One of the major requirements specific to this customer is the versatility of the secondary aircrafts. Most of this versatility relates to the electronics hardware and software suite on board of the aircraft as well as the easy addition, removal and/or modification of these systems based on the mission requirements. For example, some agencies may require cameras and LIDAR sensors for one mission, while another would need access to audio sensing and SONAR equipment for their missions. In today's market, this is fairly feasible and should be worth the cost and risk of accommodation, especially given the subsequent revenue from future purchases and maintenance contracts.

4. Sale Volume and Estimated ROI

The organization will require at least 10 UAVs in the preliminary phase where their testing and analysis teams around the world would conduct testing under simulated conditions. If satisfied, a much bigger order can be expected due to the versatility of the project from the get-go. Even after sale maintenance and support revenue can be expected given the fact that the UAV is designed to be modified (mostly the secondary drones) to fit the needs of the mission.

1A.5. Recommended Alternative Markets

The most viable alternative markets were selected from the potential markets enumerated in the above section based on the following four criteria: design adaptation cost, customer budget, customer demand, and general risk of undertaking. The first criterion accounts for the cost of design changes that are necessary to meet the system requirements of the alternative market. These design changes include but are not limited to selecting alternative power plants, redesigning the wing, and integrating alternative payloads into the base design. The second criteria accounts for the varying budgets of each alternative customer. For example, NOAA indicated a limited budget; whereas, the cost of the aircraft system was found to be less of a concern for the oil and gas company and DHS. In the same tread as the previous criteria, the third criteria, customer demand, accounts for differences in expected sale volume between each customer. Finally, the estimated risk of the undertake was accounted for in the final criterion.

Based on the above criteria, the viability of each market was then rated using a five point ordinal scale (1-5). Within this scale a higher rating indicates greater favorability for the alternative market with respect to that specific criterion. A total score for each market was then tabulated through the summation of each criteria specific score. The resulting tabulation is provided in Table 8A.

Customer	Design Adaptation Cost	Customer Budget	Customer Demand	Risk of Undertaking	Total (Max 20)
NOAA	5	1	2	4	12
Oil Company	1	4	4	1	10
Survey Company	1	1	2	2	6
US Coast Guard	4	3	4	3	14
Utility Company	1	2	1	4	8
DHS	5	5	4	5	19

Table 8A. Evaluation of potential alternative customers.

The rightmost column of total scores shows the result of each customer's evaluation. The closer the score is to the maximum of 20, the more attractive the potential customer's market is to adapting the primary design. Of the six potential customers, the top three are selected for further analysis. These three customers are the Department of Homeland Security, the USCG, and NOAA, respectively. The requirements imposed by DHS are such that the primary design could be implemented almost directly to their uses, with almost no cost of adaptation and extremely low risk. The Coast Guard's design is slightly more complicated, as it requires the delivery of a separate payload alongside the secondary aircraft units. However, as previously stated this design requirement does not pose much immediate risk nor immense additional costs. Finally, NOAA's design adaptation needs few enhancements from the primary customer's design, with the added benefit of relaxed requirements in many fields. This saves overall cost and reduces resource investment necessary to satisfy their criteria. As the runner-up, the oil and gas company's leniency on cost and endurance seems enticing, but no simple adaptation to the primary customer's design could be implemented to satisfy their requirements. The system that fulfills their intended role is simply a different category of aircraft entirely. Producing such a system would require at least double the expense across the board to deliver their product and the primary customer's product simultaneously.

2A. Trade Study

As a starting point for the aircraft design, similar aircraft were analyzed to determine common design features. Although this analysis was limited in scope because of a general lack of sizing information, the data available was used to determine typical wing loading, thrust-to-weight, and aspect ratio. As a note, significantly more data was available for American aircraft (Global Hawk, Predator, Reaper, and U2) which are called out specifically in Table 9A. Table 10A summarizes the parameters used in the determination of the aircraft class average wing loading.

Aircraft Parameters	Global Hawk	Predator	Reaper
S_{Ref} [ft^2]	685	123.3	-
B [ft]	130.9	55	66
AR	25	24.5	-
Range [nm]	12300	770	1150
Endurance [hr]	34	24	14
KTAS [kt]	310	70	200
Ceiling [ft]	60000	25000	50000
PYLD [lbf]	3000	450	3750
OEW [lbf]	14950	1130	4900
W_{Fuel} [lbf]	17300	665	4000
MTOW [lbf]	32250	2250	10500
Powerplant	RR AE3007A	Rotax 914F 4-cycle	HW TPE331-10GD
Power Plant Type	Turbofan	Reciprocating	Turbofan
T [lbf]	8500	-	-
W_{Engine} [lbf]	1657	164.7	380
TSFC [lb/(lbm · hr)]	0.36	-	0.55

Table 9A. Sizing Values for Comparable US Drones.

Aircraft	*MTOW* [lbf]	*b* [ft]	*AR*
Global Hawk	32250	130.9	25
Predator	2250	55	24.5
U2	40000	103	10.6
Reaper	10500	66	-
IAI Eitan	11905	85	-
DRDO Rustom	1590	25.9	-
Bateleur	2205	49	-
Bayraktar Akinici	12125	665	-
TAI Aksungur	7275	79	-

Table 10A. Values from Similar Aircraft for Wing Loading Estimation.

From the values provided in Table 10A the class average wing loading was found to be the following value.

$$\frac{W}{S_{Ref}} = 49.3852 \pm 24.6428\text{psf.}$$

Using this value, the required thrust-to-weight ratio to achieve liftoff in a given runway length was computed. For this computation it was assumed that $CL_{max} = 1.7$, a value typical of a wing with Fowler flaps deployed. The resulting plot is provided in Fig. 1A. As shown in the figure, the initial required takeoff distance of 800 ft with 38 ft clearance at the end of runway would require a maximum thrust-to-weight ratio greater than one for the average aircraft class wing loading. This value significantly exceeds the thrust-to-weight ratio of this class of aircraft and results in engine selections that fail to meet the endurance requirement. As a result, the initial takeoff requirement of 800 ft with 38 ft clearance at the end of runway was negotiated to a requirement of a 2000 ft takeoff distance. At this distance, it was determined that the wing loading should be within the range of 24.7424 psf to 49.3852 psf to meet the revised requirement. Per this determination, a thrust-to-weight value of 0.45 lbf was determined for initial calculation.

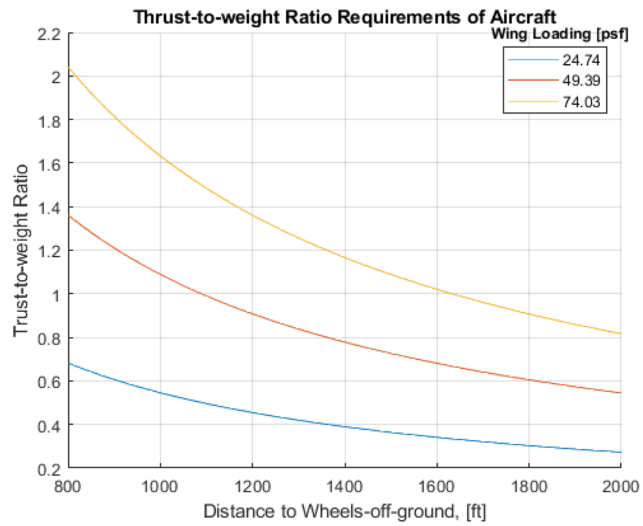


Figure 1A. Thrust-to-Weight Requirement for a GivenTakeoff Distance.

2A.1. Design Limitations

The primary customer (US Army) and DHS have the most stringent design and performance limitations. The intended role for the aircraft is to perform reconnaissance missions of many types on short notice and in the field (off-field operations). The performance specifications are outlined in Table 1A. The performance specifications heavily influenced the design of the aircraft and engine selection. After doing preliminary calculations using the 800 foot field length requirement and 400 pound payload, it was determined that the required takeoff thrust to weight ratio would be almost 2. There are currently no engines that would have provided the required thrust to make the takeoff distance that also have the efficiency required to meet the 24 hour endurance requirement. The use of external systems such as JATO or a catapult were ruled out as they add unnecessary complexity to the structure of the aircraft due to the increased loads on the airframe as well as necessitate additional operational logistics on the field. Instead, a compromised takeoff distance of 2000 feet was agreed upon with the customer. The design limitations put forth by the other customers, NOAA and the U.S. Coast Guard, are far less limiting. The key limitation to take note of is the payload delivery method. The payload delivery system must be designed with this requirement in mind.

3A. Aircraft Performance Requirements

3A.1. Technical Requirements

The technical requirements for each of the selected customers have little variance from that of the primary customer. The greatest design change comes in the form of the extra cargo capacity needed for the U.S. Coast Guard model. This requirement will be fulfilled by utilizing a more powerful engine on the aircraft, a design decision which will also be key in meeting the primary customer's short takeoff distance requirement. To meet the requirements of all customers chosen in the preliminary review, the following technical requirements are to be considered.

Requirement #	Statement
3.1.01	The data from the smaller vehicle shall be telemetered to the larger vehicle for capture and return.
3.1.02	The larger vehicle shall be reusable for a period of 10000 flights.

3.1.03	The reconnaissance portion of the system shall be controllable by a remote pilot through the larger vehicle telemetry system.
3.1.04	The design characteristics of the larger aircraft shall meet all applicable military standards for unmanned aircraft.
3.1.05	The smaller expendable vehicle shall comply with the standards of traditional radio control hobby aircraft.
3.1.06	The smaller vehicle should implement a recoverable design.
3.1.07	The aircraft system shall be able to perform system warm up and system checks in a period of 10 minutes.
3.1.08 (Revised)	The aircraft system shall take off within a 2000 foot clearing.
3.1.09	The aircraft system shall climb at the best rate of climb to its cruising altitude.
3.1.10	The aircraft system shall be able to cruise for 510 nautical miles at a speed and altitude which minimizes station and fuel burn.
3.1.11	The aircraft system shall be able to then climb at best rate of climb to 25500 feet.
3.1.12	The aircraft system shall be able to proceed to gather nape of the earth (50 feet above local level) data in near real time for a period of 24 hours at the best endurance speed.
3.1.13	After the data collection period is completed, the aircraft system shall descend to best cruise altitude, with no credit for range.
3.1.14	The aircraft system shall then perform a return cruise segment of 510 nautical miles while minimizing fuel burn.
3.1.15	Once all other mission requirements are met, the aircraft system shall descend with no credit for range and land within an 800 foot clearing with a 38 foot obstacle present.
3.1.16	The aircraft system shall have 10% fuel reserve after mission completion.
3.1.17	The aircraft system shall be able to carry 400 pounds of payload not including any secondary aircraft designed for the mission.
3.1.18	The aircraft system shall be capable of performing a sustained 2.4 g maneuver at speed for minimum power setting and at 25000 feet altitude.
3.1.19	The aircraft system shall be capable of 5 minutes taxi time.
3.1.20	Any secondary aircraft designed for the mission shall be lighter than 20 pounds and shall only require one person to manipulate and prepare for flight in the primary aircraft system.
3.1.21	Secondary UAV must be controllable securely and remotely by a licenced human operator.
3.1.22	The aircraft shall be capable of hovering for prolonged data collection at specific points along the flight path.
3.1.23	The primary aircraft must be able to comfortably provide deployment of at least 4 secondary aircraft.
3.1.24	The primary aircraft shall maintain communication and transmit and save data provided by the secondary UAVs..

3.1.25	The aircraft system shall be able to avoid radar detection within desired limits.
3.1.26	The secondary drones shall have kill programs on board in case of capture.
3.1.27	The aircraft system shall be able to proceed to gather ground imagery of the earth at a height for optimal resolution of sensory in near real time for a period of 24 hours at the best endurance speed.
3.1.28	The larger aircraft system shall be able to carry 200 pounds of survival equipment with a potential volume up to 64 cubic feet.
3.1.29	Any secondary aircraft designed for the mission shall be lighter than 20 pounds and shall only require one person to manipulate and prepare for flight in the primary aircraft system.

Table 11A. Final Overall Technical Requirements for All Customers

3A.2. Cost Requirements

As previously demonstrated in Table 8A, the Department of Homeland Security, the U.S. Coast guard and the Oil & Gas company were found to have the most flexible budget of all. Both these recipients along with the primary customer, the U.S. Army, offer more than 15-20 times as many units purchase requirements than that of NOAA, Land surveying company and the Utility company. The Survey, Oil/Gas and Utility companies require design adaptations that are highly unfavorable and overly expensive and offer insufficient ROI (perhaps not in the case of Oil Company). On the contrary, the DHS, NOAA and the U.S. Coast guard need minimal design adaptations and of them, the DHS and the U.S. Coast Guard require a higher number of units that in turn shall result in a more reasonable ROI.

3A.3. Performance Requirements

Aircraft performance requirements come directly from the primary customer's design requirements. All of the performance requirements are directly derived from the primary customer's requirements. If the aircraft meets the performance requirements of the primary customer, it will also meet the requirements of the alternate customers.

Performance Specification	**Value**
Range (nautical miles)	1020
Takeoff Distance (MTOW, std, S.L.) (feet)	2000
Payload (lbs)	400
Max sustained load factor (g)	2.4
Ceiling (ft)	25,000
Endurance (hrs)	24

Table 12A. Performance Requirements

The aircraft must be able to fly for 24 hours and carry 400 lbs of payload (not including avionics and the daughter drone) up to an altitude of 25,000 feet. It must also be able to make a sustained turn at a load factor of 2.4 G and takeoff from a runway of 2000 feet length with a 38 foot obstacle. The total range of the aircraft must be at least 1020 nautical miles.

4A. Aircraft Specifications and Performance

4A.1. Design Specifications

The overall design specifications were determined using the design limitations imposed by the primary customer's requirements. The powerplant was selected using optimization code in MATLAB. Out of several comparable engines on the market, the Williams International FJ33-5A was the best choice. Provided the selected engine, maximum takeoff weight was also calculated along with the requisite fuel consumption. These calculations are expanded upon in the subsequent section.

Design Specification	**Value**
MTOW (lbf)	3,603
Wing Area (ft^2)	80
Aspect Ratio	32
Wingspan	50
Wing loading (psf)	45
Thrust to weight ratio (takeoff)	0.5124
Critical field length (ft)	1,939
CLmax	1.7
(L/D)max	22.6612
Powerplant	1 x Williams International FJ33-5A

Table 13A. Main Design Specifications

4A.2. Engine Selection and Initial Sizing

Prior to the detailed design, significant effort was placed on sizing the aircraft engine to meet the system's requirements. From this sizing analysis, the *MTOW* of the aircraft was computed based on known weights and estimated fuel use computed via the Breguet range and endurance equations. Specifically, the *MTOW* of the aircraft was estimated per the following relation.

$$MTOW = OEW + PYLD + FOB \tag{1}$$

In this relation, the *OEW* of the aircraft was unknown. Initially, a linear regression of similar classed aircraft was applied for this calculation; however, this led to unreasonably low structural mass fraction values. To rectify this, the *OEW* of the aircraft was instead estimated through the following relation, where the engine weight was known.

$$OEW = W_{Engine} + MF_{Structure} MTOW \tag{2}$$

In Eq. (2), both the *MTOW* and $MF_{Structure}$ are unknown; however, by substituting into Eq. (1), the following relation for *MTOW* can be derived, note that *FOB* is also related to *MTOW* as in Eq. (3.1) per Req. (1.1.16).

$$FOB = 1.10\,\zeta\,MTOW \tag{3.1}$$

$$MTOW = \frac{PYLD + W_{engine}}{1 - MF_{Structure} - 1.10\,\zeta} \tag{3.2}$$

In this relation, $MF_{Structure}$ was computed iteratively by applying the rule of thumb that the sum of the variable mass fraction for an aircraft design should be less than or equal to 0.875.

$$0.875 \geq MF_{Structure} + MF_{Energy} + MF_{Propulsion} \tag{3.3}$$

Note that,

$$MF_{Energy} = 1.10\zeta \tag{3.4}$$

And

$$MF_{Propulsion} = \frac{W_{Engine}}{MTOW} \tag{3.5}$$

Given the above relations, the $MTOW$ for each engine was computed iteratively. Following this computation the optimum engine was selected. This selection was made to maximize the structural mass fraction and minimize the fuel weight burned provided that the candidate engines first meet a thrust-to-weight requirement of 0.4. This value was deemed necessary to fulfill the takeoff distance requirement for the aircraft system (Req. 1.1.08 (revised)).

In addition to the aforementioned iteration over $MF_{Structure}$, an iteration was also performed over the wing loading which is present in the best range velocity calculation.

$$VTAS_{br} = (\frac{12\,k\,(W/S_{Ref})^2}{\rho^2\,CD_0}) \tag{4.1}$$

$$k = \frac{1}{k\,AR\,e} \tag{4.2}$$

For Eq. (4.1) the base drag coefficient was estimated using EDET with a 30% crud drag correction included. To update the wing loading values between iterations, Eq. (4.3) as applied which accounts for the takeoff-length requirement and the actual thrust-to-weight value of the design (Req. 1.1.08 (revised)).

$$(\frac{T}{W})_{Actual} = \frac{T_{Engine}}{MTOW} \tag{4.3}$$

$$CFL = 37.5\frac{(W/S_{Ref})}{CL_{Max}\,(T/W)_{TO}} \tag{4.4}$$

In the aforementioned relation, CL_{Max} was assumed to be 1.7, a typical value for an aircraft with Fowler flaps deployed.

Within both iterations, the leg-specific fuel use fractions fractions were computed per the planned flight profile:

1) A 510 nm cruise (Req. 1.1.10)
2) A 24 hour endurance flight at 25500 ft (Reqs. 1.1.11-12)
3) A 510 nm return cruise (Req. 1.1.14).

Specifically, the Breguette Range and endurance equations, provided below, were applied for each engine and iteration.

$$\Psi = (\frac{VTAS}{TSFC})(\frac{L}{D})\ \log(\frac{1}{1-\zeta}) \tag{4.5}$$

$$t_{endurace} = (\frac{L}{D})_{Max}(\frac{1}{TSFC})\log(\frac{1}{1-\zeta}) \tag{4.6}$$

Given the calculated leg mass fractions, the total mass fraction was calculated in accordance to the following rationale.

$$\Delta W_f = \Delta W_{f1} + \Delta W_{f2} + \Delta W_{f3}$$

Since

$$\zeta = \frac{\Delta W_f}{W_i}$$

$$\zeta W = \zeta_1 W_1 + \zeta_2 W_2 + \zeta_3 W_3$$

Where

$$W_1 = MTOW$$

$$W_1 = MTOW - \Delta W_1$$

$$W_2 = MTOW - \Delta W_1 - \Delta W_2$$

$$\zeta MTOW = \zeta_1 MTOW + \zeta_2 (MTOW - \zeta MTOW) + \zeta_3 (MTOW - \zeta_1 MTOW - \zeta_2 MTOW)$$

Thus

$$\zeta = \zeta_1 + \zeta_2 + \zeta_3 - \zeta_1\zeta_2 - \zeta_2\zeta_3 + \zeta_1\zeta_2\zeta_3$$

With this calculated value, the thrust required to meet the previously selected thrust-to-weight value of 0.45 was computed and compared to the maximum thrust output of the selected engine. Using this criterion and the aforementioned criteria of maximizing $MF_{Structure}$ whilst minimizing fuel use, an engine was selected. Specifically, it was determined that the Williams International FJ33 met the design requirements with a design *MTOW* of 3602.96 lbf. From this engine selection, the weights and mass fractions in Table 14A were computed. Additionally, general sizing parameters, provided in Table 15A, were computed to match the system requirements. The code for this process as well as a table of potential candidate engines are provided in the Appendix C.

	Energy	**Propulsion**	**Structure**
Mass Fraction	2056.50	319.00	777.07
Weight [lbf]	0.5708	0.0885	0.2157

Table 14A. Weight and Mass Properties for Aircraft With Williams International FJ33.

MTOW [lbf]	3603.0
S_{Ref} [ft^2]	80.0
AR	32.0
b [ft]	50.0
(W/S_{Ref}) [psf]	45.0370
(T/W)	0.5124
CFL [ft]	1939.0
CL_{max}	1.700

(L/D)	10.0
$(L/D)_{max}$	22.6612
VTAS [ft/s]	258.5

Table 15A. Basic Sizing Parameters for Aircraft.

4A.3. Wing Design

A. Aerodynamic Design of Wing Through VORLAX

In order to obtain an elliptical loading distribution with favorable pressure gradients a five section VORLAX tool was implemented as the primary aspect of the wing design process. This tool allowed for the camber, thickness profile, incidence, and twist of five control sections to be modified using the base planform geometry described in Table 15A. This planform geometry, along with the locations of the control sections, is provided in Fig. 2A. Given this geometry, an input file for VORLAX is generated that models each wing section as two sandwiched panels to capture thickness effects on the resulting pressure distribution. Additionally, the VOLAX tool parses the resulting output file and generates section specific pressure distribution plots and a spanwise loading plot. Through analyzing these plots a wing design was selected.

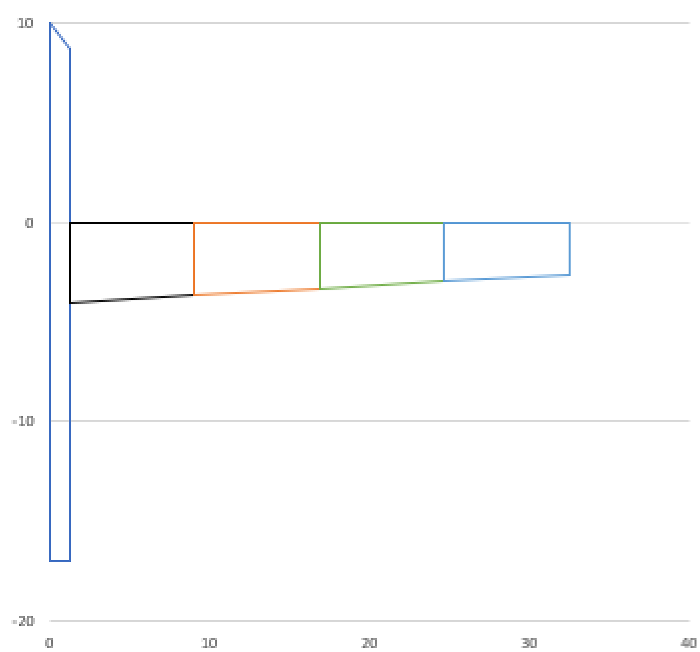


Figure 2A. Wing Planform Model with VORLAX Control Sections.

Per this process, a wing with the camber, incidence, and thickness distributions provided in Fig. 3A, Fig. 4A, and Fig. 5A respectively was selected. Figures 6A through 10A provide the pressure coefficient distribution at each control section. Finally, the spanwise pressure distribution is captured in Fig. 11A and Fig. 12A provides a graph of the upper surface pressure distribution. Note that the wing design was selected to generate the strong leading edge isobar that is depicted in this figure.

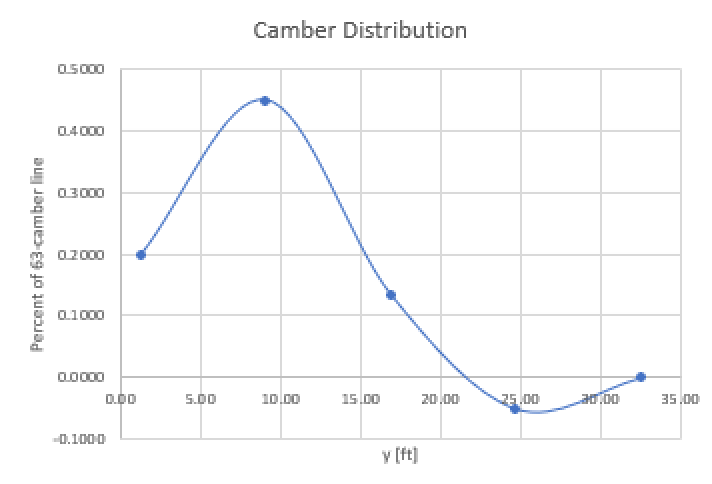


Figure 3A. NACA-63 Cambeline Distribution Across Span of Wing.

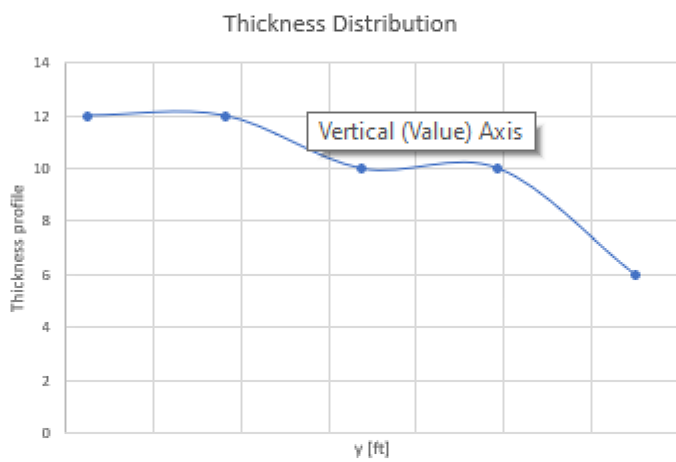


Figure 4A. Spanwise Thickness Distribution.

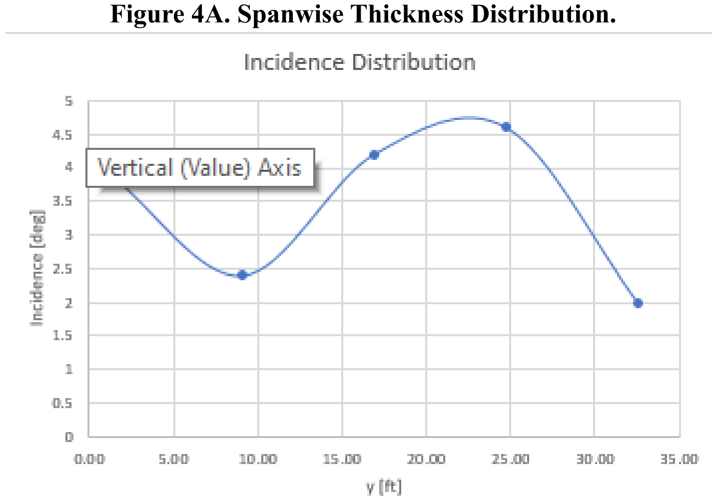


Figure 5A. Spanwise Incidence Distribution Proved Angle of Attack of 0.5°.

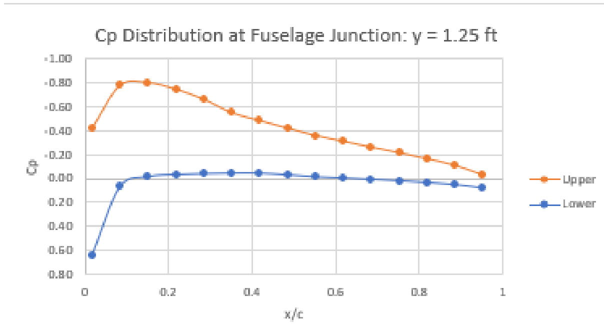


Figure 6A. Cp Distribution at Wing-Fuselage Junction for Selected Wing Design.

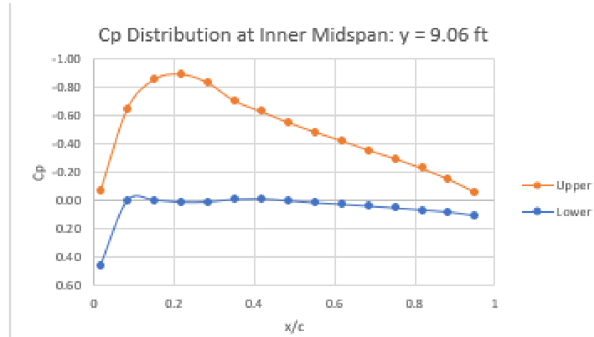


Figure 7A. Cp Distribution at Inner Midspan for Selected Wing Design.

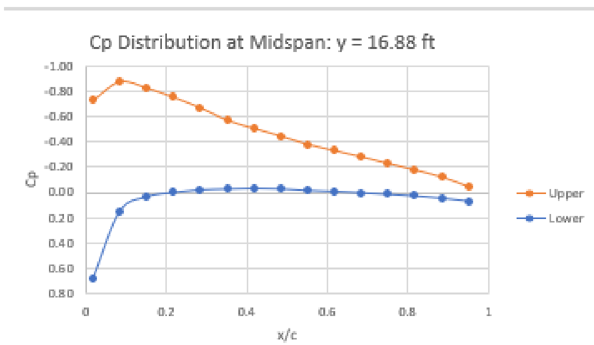


Figure 8A. Cp Distribution at Midspan for Selected Wing Design.

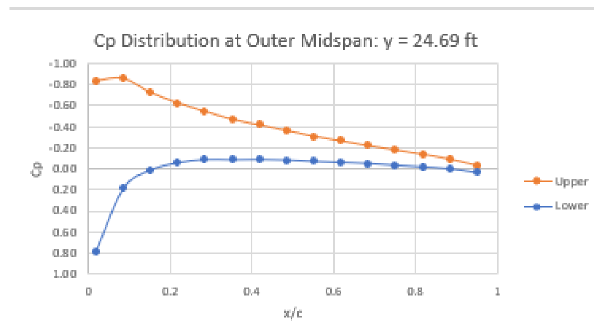


Figure 9A. Cp Distribution at Outer Midspan for Selected Wing Design.

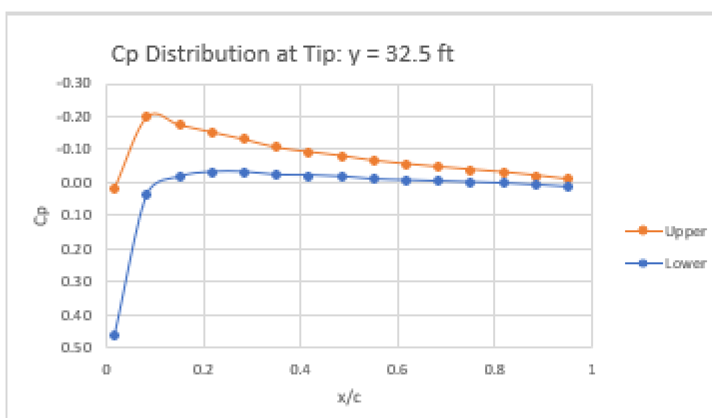


Figure 10A. Cp Distribution at Wing Tip for Selected Wing Design.

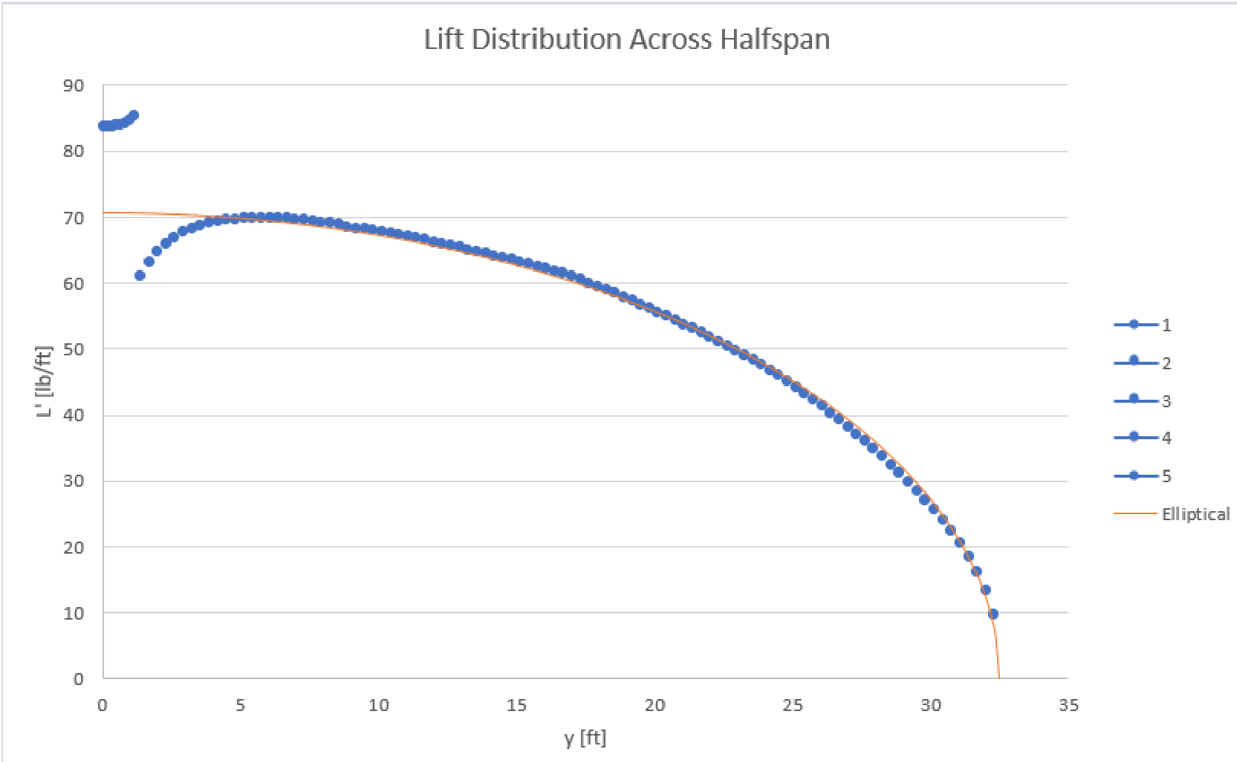


Figure 11A. Spanwise Pressure Distribution Compared to Ideal Elliptical Loading Case.

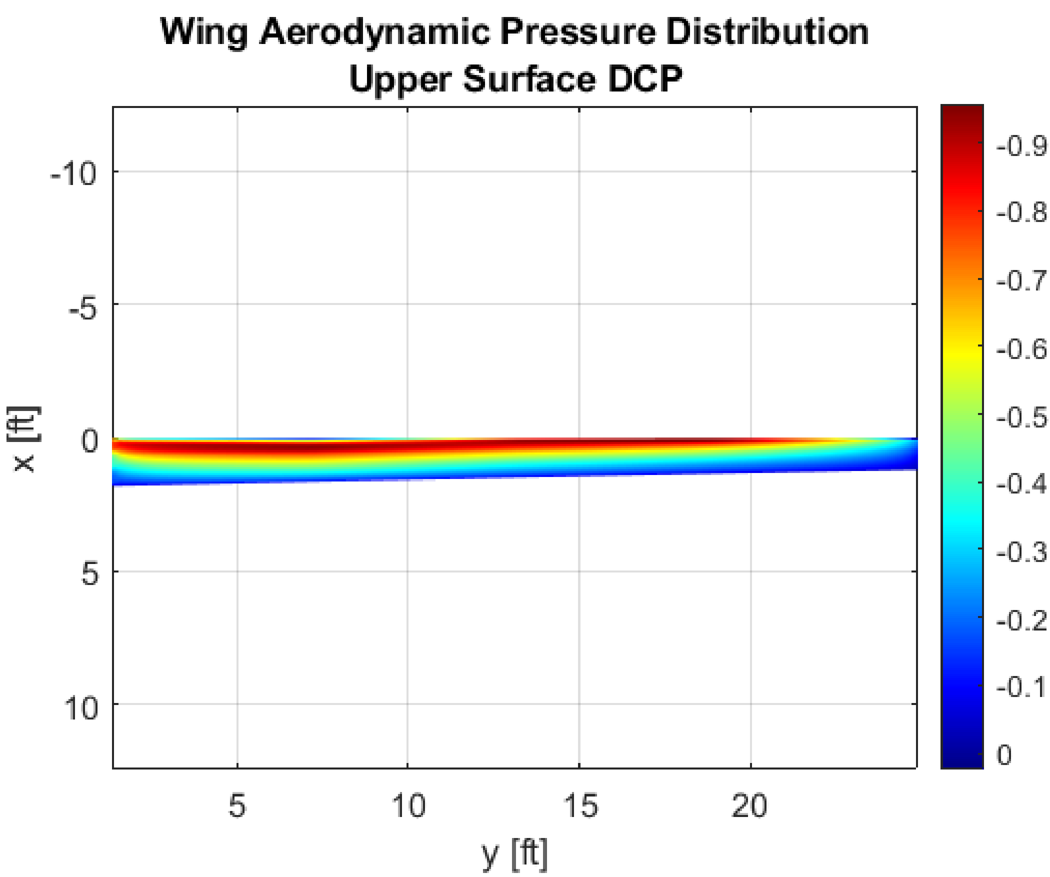


Figure 12A. Upper-surface Pressure Distribution with Leading-edge Isobar.

B. Structural Design of Wing

The design of the main wing was based largely on the profiles that were generated using VORLAX. The five profiles were interpolated along the span, providing consistent wing rib shapes to transmit and distribute the aerodynamic loads into the main spars. Due to the relatively strict constraints resulting from the low thickness percentages especially towards the wingtip, the main structural focus was aimed towards the spars. Stringers were deemed an unnecessary weight addition with the exception of three small tubes forward of the main wing box, but none were included within. The front and rear spars are made of aerospace grade 6061-T6 aluminum alloy, and traverse the entire length of the wing up. The spars are then connected to a wing brace mounted in the midsection of the fuselage. Each spar is an I-beam profile of the same material that passes through a total of 25 ribs in each wing, but must slightly decrease in total height as the wing cross-section thins out towards the tip. This provides little issue in the way of aerodynamic loads, as the previously mentioned near-perfect elliptical loading decreases bending stress towards the wingtip.

As the aircraft is only 27 feet long, the fuel storage on board must take advantage of whatever space exists in the wing to meet the specified endurance and range requirements. As a result, each wing contains fuel tanks within the wing boxes, which serves to benefit the center of gravity's location near the aerodynamic center. These tanks hold 15,923 cubic inches of fuel each, amounting to almost half of the necessary fuel volume. Lastly, the wing's internal structure is wrapped in a smooth skin, providing a clean aerodynamic surface with little in the way of crud drag. The total assembled weight of both wings is 634.44 lbs. With full wing tanks, the weight increases to 1554.85 lbs. The wing structure can be seen below.

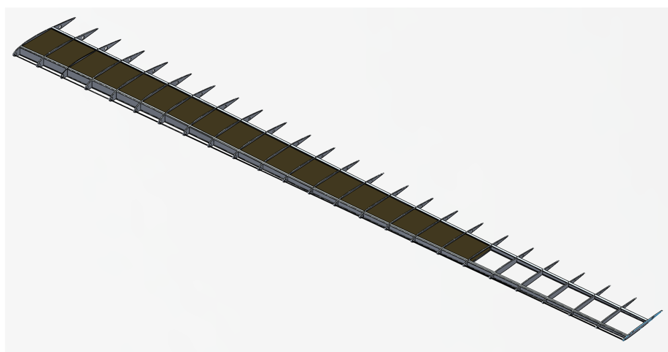


Figure 13A. Wing Internal Structure (port side).

4A.4. Fuselage Structural Design

A. Structural Design of Fuselage

The fuselage was built on the principle of web and stringers. Running down the longitudinal direction of the fuselage are long stringers. Each stringer has an I-beam cross-section. The dimensions of the cross-section can be seen in Fig. 1A. In the transverse direction are a series of supporting formers. Each former has an I-beam cross-section similar to that of the longitudinal stringers. The formers are separated by 24 inches in the center. An isometric view of the fuselage section can be seen in Fig. 2A. The overall length of the fuselage is 27 feet from tip to tail. The external diameter of the fuselage is 30 inches and the inner diameter of the fuselage is 28 inches. The total volume of the fuselage was driven by the payload and fuel volume requirements. The aft section of the fuselage is tapered upward to provide clearance between the ground and fuselage upon landing. Given the 24 hour endurance flight requirement, the mass fraction of the structure is limited to 0.2157 thus, the weight of the structure is limited 776.85 lb. For this reason, the structure is constructed of 7075 aluminum alloy to provide structural strength but reduce the weight. The weight of the structure was further reduced through a series of iterations giving a final weight of the fuselage at 129.49lb.

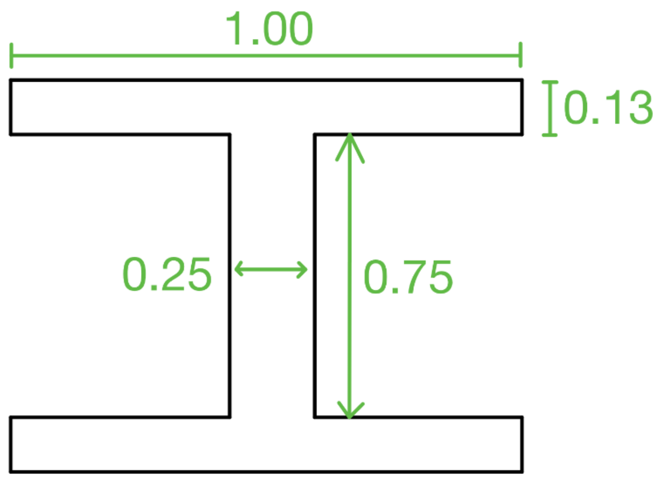


Figure 14A. Cross section of stringer I-beam. All dimensions are in inches.

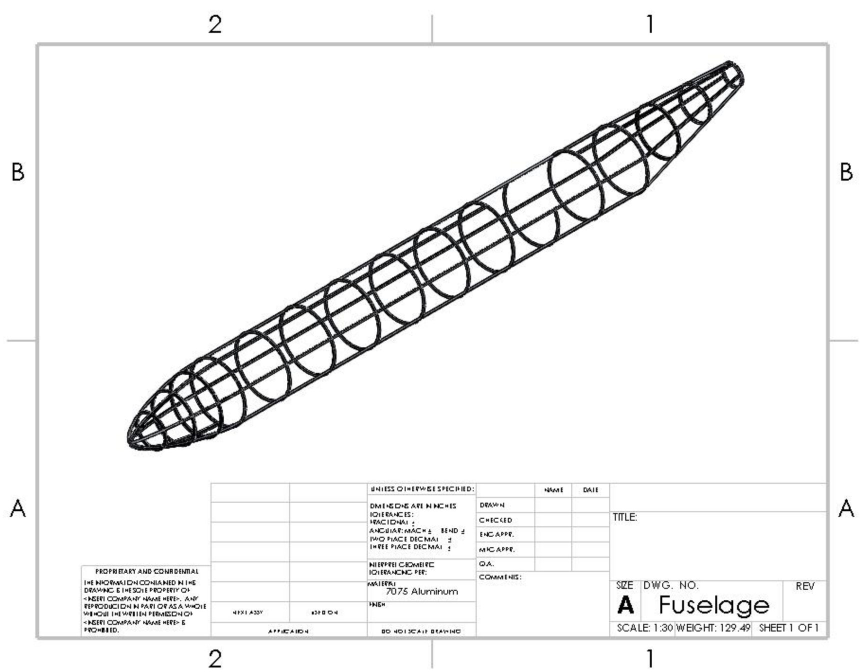


Figure 15A. Isometric view of fuselage section.

B. Design of Payload Drop System

The payload drop system can be seen below in Fig. 16A. The system uses two methods for dropping payload. The payload bay doors are highlighted red in Fig. 16A. Each bay door is operated by a single hydraulic cylinder. The use of a single cylinder was discussed, but would require complicated linkages leading to a greater possibility of system failure or even individual part failure. Furthermore, the addition of the linkages would add weight which would render the use of a single cylinder obsolete compared to a more simple system using two cylinders and fewer linkages. As mentioned above, there are two methods for dropping payload. The first is the use of the drop pin which is located directly above the bay doors. The drop pin is a lightweight, hydraulically actuated pin that can hold payload securely in the payload bay until dispatch from the vehicle. When the payload is ready to be dispatched, the drop pin releases, allowing gravity to pull the payload from the fuselage and away from the vehicle.

The second method for dispatching payload is through the use of a rail system. The hybrid dispatch system was chosen for the following reasons. The endurance requirement forced the aspect ratio and fuel volume to be large. The high aspect ratio does not allow for a large volume of fuel to be stored in the wings. For this reason, most of the fuel volume must be stored in the fuselage. Doing some rough calculations based on the endurance requirement, it was found that about 40% of the fuselage volume must be fuel. This forced the payload system to fit around the fuel tanks, which is why the payload system implements two methods for dispatching payload. The second method is highlighted green in Fig. 16A. The second method implements a rail system which pushes the remaining payload out of the bay doors. The rail system is attached directly to the fuselage using small brackets. The brackets are attached directly to the fuselage to reduce weight and extra parts. The rail system uses a chain driven mechanism to push payload off the rails and through the bay doors apon dispatch. The chain driven system uses an electric motor with low RPM and high torque. The physical rails of the rail system are an I-beam shape. This will allow specially designed wheels to wrap around the rails. This will prevent translation of the payload during flight in the Y and Z direction relative to the vehicle. In order for the rail dispatch system to operate, there must not be a payload located in the drop pin section, located directly above the bay doors. This means that any payload in the pin section must be dropped before the rail section. Another consideration is the height of the payload. The height of the payload in the rail section is more limited than that in the pin section, due to the fuel tanks which reside above the rail section.

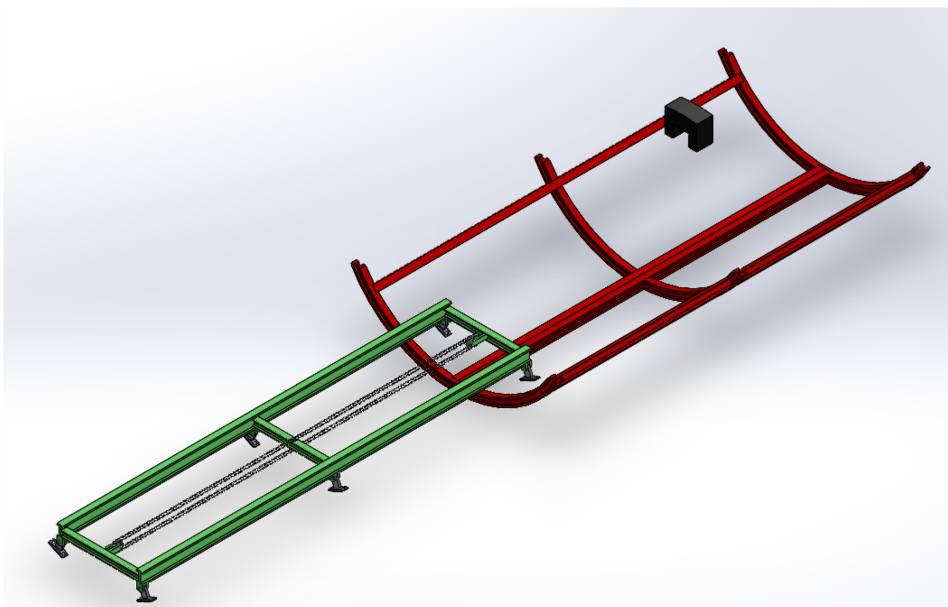


Figure 16A. Payload drop system design. Designed using Solidworks.

A. Design of Wing Mount

The wing mount was designed in a similar style to the fuselage main structure. Two semicircular I-beam profiles were employed as the main joint to the fuselage frame. This cross sectional shape maintains the same profile as the spars to avoid discontinuities in the rigidity of the system as a whole. The braces connect to the front and rear spars, and are supported by a series of carbon fiber tubes providing additional support axially along the length of the fuselage. The decision to implement semicircular braces resulted from an examination of fuel storage locations within the aircraft. The main fuel tank is cylindrical in shape, fitting snugly within the fuselage's inner contour. Having a traditional wingbox with spars travelling straight through the lower midsection of the fuselage would have significantly reduced the fuel tank volume. Alternatively, it would have required complex geometry, difficult manufacturing, and challenging installation. The final wing brace assembly weighed in at just over 25 lbs. Seen below is the isolated wing brace, and then its final installation location within the airframe.

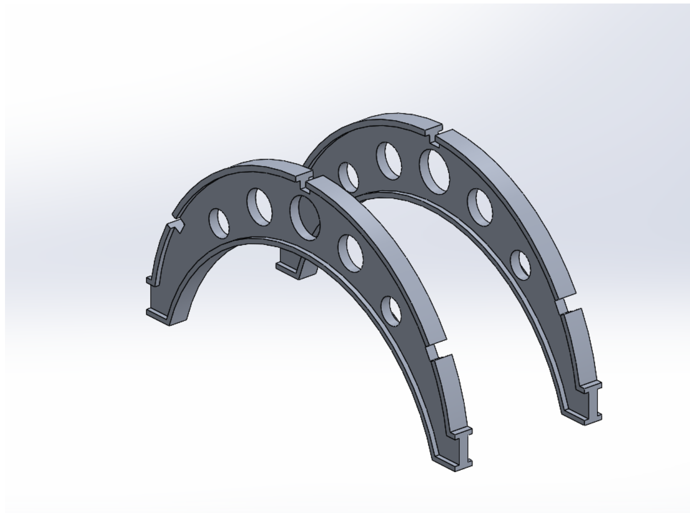


Figure 17A. Fuselage Wing Brace (isolated).

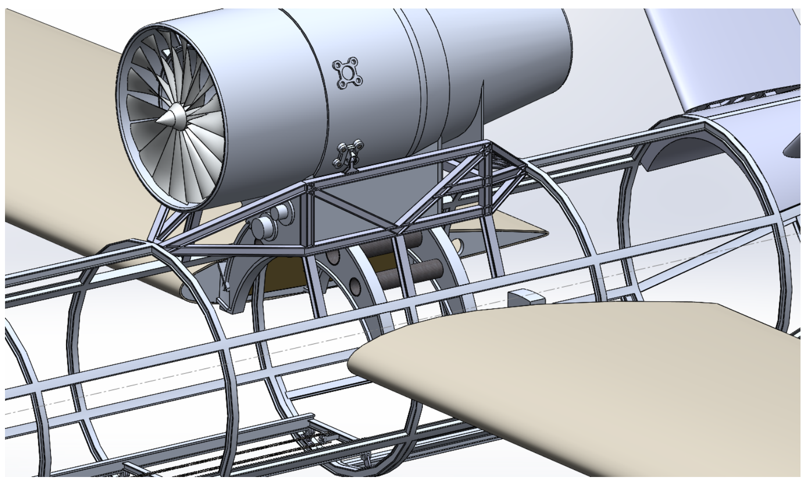


Figure 18A. Fuselage Wing Brace, showing wings, engine with engine mount, and location within frame.

4A.5. Internal Systems Positioning

4A.6. Static Stability Analysis of Aircraft System

Static stability of an aircraft is split into the three angular degrees of freedom: pitch, roll, and yaw. Longitudinal stability (pitch) is characterised by both C_{M0} (zero angle of attack pitching moment coefficient) and C_{malpha} (change in pitching moment coefficient with respect to angle of attack). Lateral stability is composed of weathercock stability (yaw) and rolling stability, which are characterised by $C_{n\ beta}$ and $C_{l\ beta}$ respectively. The criteria for stability and balance in the are tabulated below.

Characteristic	Requirement
Balanced (positive pitching moment coefficient at zero angle of attack)	$C_{M0} > 0$
Longitudinally stable (restoring pitching moment coefficient)	$C_{M\ alpha} < 0$
Weathercock stability (restoring yaw moment coefficient)	$C_{n\ beta} > 0$
Roll stability (restoring rolling moment coefficient)	$C_{l\ beta} < 0$

Table 16A. Requirements for Static Stability

Calculating these values is trivial, measuring all of the required variables is not. Once all of the variables determined, calculation of C_{M0} is as follows:

$$C_{m_0} = C_{m_{ac_{wb}}} + C_{m_{0_p}} + a_t \overline{V_h}(\varepsilon_0 + i_t)\left[1 - \frac{a_t s_t}{a_{wb} S}\left(1 - \frac{\partial\varepsilon}{\partial\alpha}\right)\right] + \Delta C_{m_p} \quad (1)$$

$C_{M\ alpha}$ can be calculated using the following:

$$C_{m_\alpha} = a\left(h - h_{n_{wb}}\right) - a_t \overline{V_h}\left(1 - \frac{\partial\varepsilon}{\partial\alpha}\right) + \frac{\partial C_{m_{0_p}}}{\partial\alpha} + \Delta C_{m_\alpha}$$

Do (2)

Rolling static stability can be characterized by the following:

$$C_{l_\beta} = -\frac{a_w}{4}\left[\frac{2(1+2\lambda)}{3(1+\lambda)}\right]\Gamma - \left[\frac{1+2\lambda}{6(1+\lambda)}\right] C_{L_{0_w}} \sin\left(2\Lambda_{\frac{1}{4}}\right) - \frac{S_F z_F}{Sb} a_F\left(1 - \frac{\partial\sigma}{\partial\beta}\right)\left(\frac{V_f}{V}\right)^2 + C_{l_{\beta_{wb}}} \quad (3)$$

Finally, weathercock stability can be characterized by the following:

$$V_v = \frac{S_F l_F}{Sb} a_F \left(\frac{V_F}{V}\right)^2 \left(1 - \frac{\partial\sigma}{\partial\beta}\right) + \Delta\frac{\partial C_{n_p}}{\partial\beta} \quad (4)$$

MATLAB was used for the calculation of all stability parameters (see Appendix C). It is necessary to calculate the stability of the aircraft in multiple weight and balance conditions, mainly the two extremes: at max takeoff weight (MTOW) and operation empty weight (OEW). If the aircraft is stable in both of these conditions, then it will be stable throughout the entire weight and balance envelope. Table 17A.1 and Table 17A.2 contain the results of the MATLAB static stability script.

Axis	Parameter	Value (/rad)	Stability
Pitch	C_{M0}	0.0023576	BALANCED
Pitch	$C_{M\ alpha}$	-7.7904	VERY VERY STABLE
Roll	$C_{l\ beta}$	-0.028235	STABLE
Yaw	$C_{n\ beta}$	0.11453	STABLE

Table 17A.1. Static Stability Derivatives - MTOW

Axis	Parameter	Value (/rad)	Stability
Pitch	C_{M0}	0.0023576	BALANCED
Pitch	$C_{M\ alpha}$	-4.1961	VERY STABLE
Roll	$C_{l\ beta}$	-0.028235	STABLE
Yaw	$C_{n\ beta}$	0.090968	STABLE

Table 17A.2. Static Stability Derivatives - OEW

In both conditions, the aircraft is balanced and stable. This is desirable as a complicated active control system will likely not be necessary. In addition, the dropping of payload will not shift the center of gravity an appreciable amount, and the shift that will occur will be stabilizing. Plots of CMac can be found in Appendix A.

4A.7. Dynamic Stability Analysis of Aircraft System

Dynamic stability of the aircraft is indicative of its behavior in multiple axes, not just the three angular degrees of freedom. Aircraft operate in 6 degrees of freedom, thus more complicated aerodynamic behavior is exhibited. Oscillations occur while in flight in several combinations of degrees of freedom. The dynamic stability analysis will describe the behavior of the oscillations; whether they are stable, neutral, or unstable. The most common modes of oscillation in aircraft dynamics are the phugoid mode, short period, and the dutch roll.

An aircraft's dynamic stability can be represented by a 4 by 4 matrix, where the elements of the matrix are given as the dimensional stability derivatives below:

$$a_{11} = \frac{1}{V_0}\left[-g\cos\alpha_0 - \frac{\frac{\partial L}{\partial \alpha} + T\cos\alpha_0}{m}\right]$$

$$a_{12} = \frac{-\frac{\partial L}{\partial V} - \frac{\partial T}{\partial \alpha}\sin\alpha}{mV_0} - \frac{1}{V_0^2}\left[g\cos\alpha_0 - \frac{\frac{\partial L}{\partial \alpha} + T\cos\alpha_0}{m}\right]$$

$$a_{13} = 1 - \frac{\frac{\partial L}{\partial q}}{mV_0}$$

$$a_{14} = \frac{g}{V_0}\sin\alpha_0$$

$$a_{21} = \frac{1}{m}\left(-T\sin\alpha_0 - \frac{\partial D}{\partial \alpha}\right) + g\cos\alpha_0$$

$$a_{22} = \frac{1}{m}\left(\frac{\partial T}{\partial V}\cos\alpha_0 - \frac{\partial D}{\partial V}\right)$$

$$a_{23} = -\frac{1}{m}\frac{\partial D}{\partial q}$$

$$a_{24} = -g\cos\alpha_0$$

$$a_{31} = \frac{1}{I_{yy}}\frac{\partial M_{aero}}{\partial \alpha}$$

$$a_{32} = \frac{1}{I_{yy}}\frac{\partial M_{aero}}{\partial V}$$

$$a_{33} = \frac{1}{I_{yy}}\frac{\partial M_{aero}}{\partial q}$$

$$a_{34} = 0$$

$$a_{41} = 0$$

$$a_{42} = 0$$

$$a_{43} = 1$$

$$a_{44} = 0$$

Thus, the A matrix is:

$$A = \begin{bmatrix} a_{11} & a_{12} & a_{13} & a_{14} \\ a_{21} & a_{22} & a_{23} & a_{24} \\ a_{31} & a_{32} & a_{33} & a_{34} \\ a_{41} & a_{42} & a_{43} & a_{44} \end{bmatrix}$$

The characteristic equation and eigenvalues of the matrix were calculated in MATLAB (see Appendix B). The eigenvalues of the matrix calculated for both MTOW conditions and OEW are as follows:

Eigenvalue	Value	Damping Coefficient	Natural Frequency (Hz)	Period (s)	Mode
λ1	-0.6685 + 1.4475i	0.419	1.59	3.95	Short period
λ2	-0.6685 - 1.4475i	0.419	1.59	3.95	Short period
λ3	0.0606 + 0.6447i	-0.0936	0.648	9.69	Phugoid
λ4	0.0606 - 0.6447i	-0.0936	0.648	9.69	Phugoid

Table 18A.1. Eigenvalues of the Aircraft's A Matrix - MTOW

Eigenvalue	Value	Damping Coefficient	Natural Frequency (Hz)	Period (s)	Mode
λ1	-1.2261 + 1.1691i	0.724	1.69	3.72	Short period
λ2	-1.2261 - 1.1691i	0.724	1.69	3.72	Short period
λ3	0.3029 + 0.8681i	-0.329	0.919	6.837	Phugoid
λ4	0.3029 - 0.8681i	-0.329	0.919	6.837	Phugoid

Table 18A.2. Eigenvalues of the Aircraft's A Matrix - OEW

As seen in Table 18A.1 and Table 18A.2, the phugoid mode and short period are oscillatory in nature as they are both complex eigenvalues. Plotting the root locus of the eigenvalues yields a clearer picture of the longitudinal dynamic flight characteristics of the aircraft in both configurations.

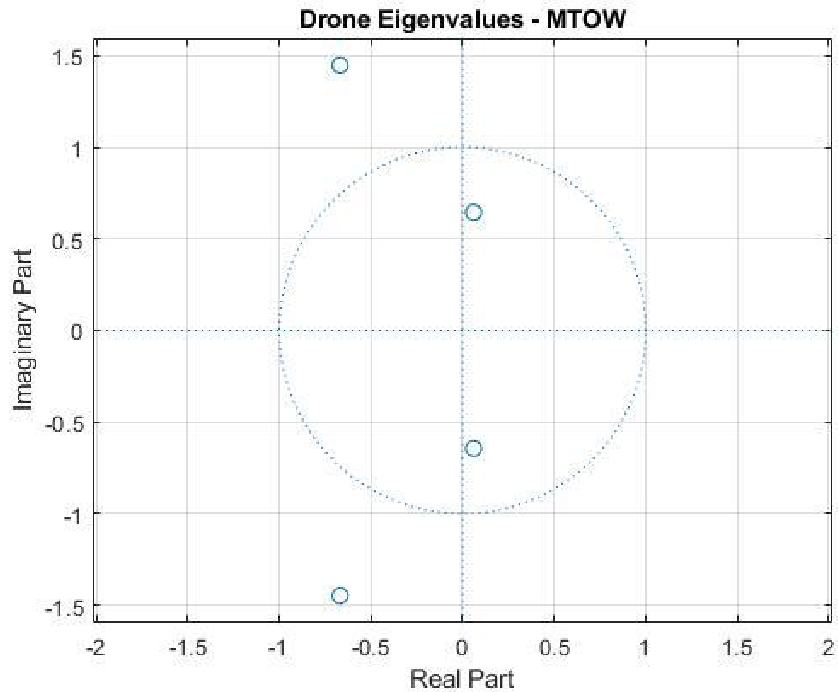


Figure 19A. Eigenvalue Root Locus - MTOW

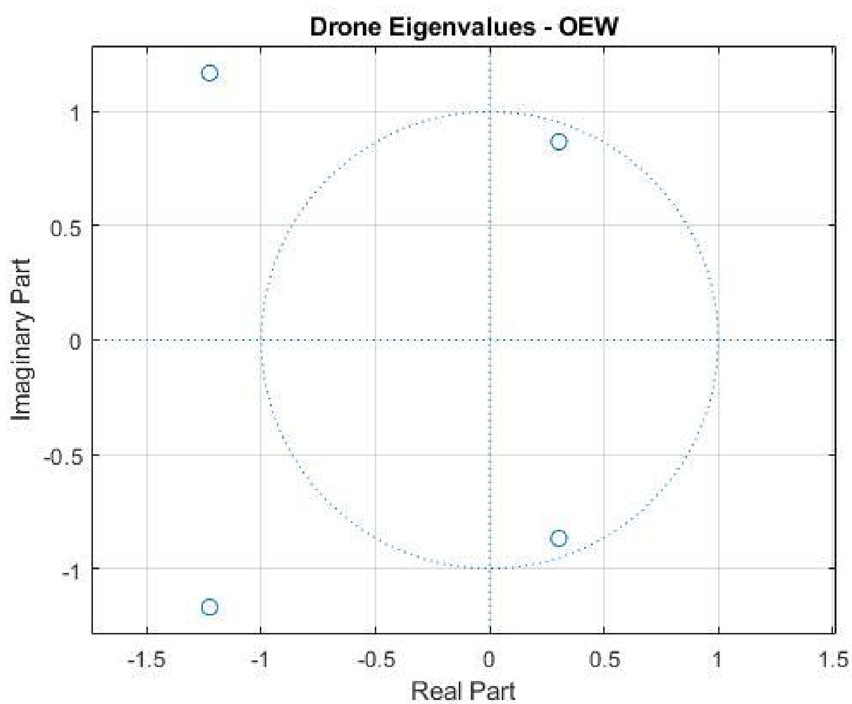


Figure 20A. Eigenvalue Root Locus - OEW

The phugoid poles are in the right hand plane, indicating an unstable condition. The short period mode is in the left and plane and thus is stable. At MTOW, the phugoid mode is very close to the origin, thus it is not extremely unstable and can easily be taken care of with an active control system of some sort. The phugoid mode becomes less stable as fuel depletes. This problem can likely be solved using an active control system and canards.

5A. Risk Analysis

5A.1. Economic Risk

Although a number of risks in the design and development of the aircraft are surmounted the potential gain, it must be noted that the primary customer (US Army) and the major secondary recipients (DHS, U.S. Coast Guard, and NOAA) all are reputed serious giants in the business and any major fallouts on product delivery can possibly lead to devastating distrust to the reputation of the firm. Of the additional requirements put forth by the chosen secondary requirements, those required by the U.S. Coast Guard are the most limiting. The chosen payload delivery method may prove to be too difficult to successfully implement during early phases of testing, thereby practically eliminating this customer from the running. While this customer is not expected to purchase more than a few aircraft, losing this client could potentially be a threat to the firm's bottom line if a great deal of resources are spent attempting to meet their requirements.

The U.S. Army, U.S. Coast Guard, and the DHS come with the additional responsibility of delivering a reliable, high quality product given the nature of their intended use of the aircraft and the subsequent national security concerns. This requirement increases the necessity of going through with a great deal of thorough testing before shipping the product to customers, a process that will heavily impact development time and subsequently cost. While this additional cost could potentially be a large threat to the solvency of the firm, this testing process will need to be undergone to meet requirements of even the primary customer. Because of this, very little additional risk is actually undertaken by meeting the requirements of these secondary customers.

#	Type of Risk	Severity	Likelihood	Cause
1	Failure in Designing Payload System	Significant	Unlikely	Added Complexity of System Needed by USCG
2	Inability to Meet Takeoff Distance Requirements	Moderate	Highly Probable	Strict 800ft Takeoff Distance Requirement
3	Inability to Meet Range Requirements	Serious	Unlikely	High Endurance Requirement of Most Customers
4	Inability to Meet Additional Payload Requirements	Marginal	Likely	Increased Payload Capacity Needed by USCG
5	Final Unit Costs Exceed Customer Budget	Serious	Unlikely	Unexpected Development Difficulties

Table 19A. Major Economic Risks

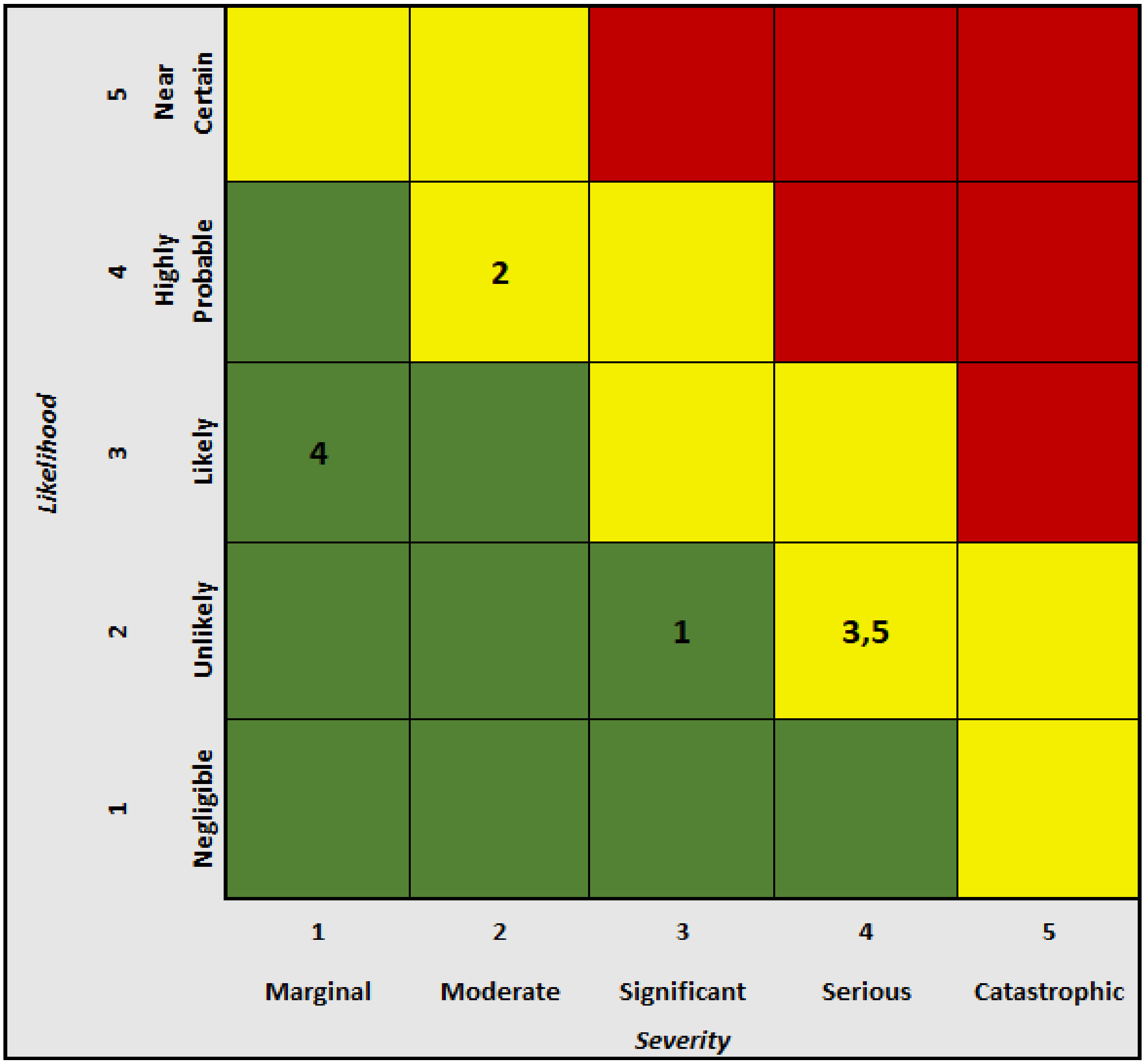


Figure 21A. Major Economic Risks Chart

5A.2. Engineering Risk

As with any vehicle of this nature there is a profound level of risk inherently present in the design. If any critical flight systems are to fail mid-flight the vehicle as well as the surrounding landscape face a direct threat of complete destruction. The lack of an onboard human operator means that an interruption in communication between the aircraft and the human operator is potentially catastrophic. To prevent this, redundancies must be put into key flight systems. Great care must also be put into the testing phase of this aircraft to ensure all systems are up to the specifications of prospective customers. On top of this baseline risk, there are also added risks that are part of specific customer requirements. The largest of this is the payload delivery system needed by the U.S. Coast Guard. This system adds additional complexity to the payload delivery system by mandating a specific delivery method. There is also the extra payload capacity demand, meaning the airframe must be built to sustain this load.

#	Type of Risk	Severity	Likelihood	Cause
1	Engine Failure	Serious	Unlikely	Strain on Engine, Manufacturing or Design Flaws
2	Failure to Deploy Payload	Moderate	Likely	Manufacturing or Design Flaws, Electronic Failure
3	Malfunction of Avionics	Catastrophic	Unlikely	Manufacturing or Design Flaws, Interference, Electronic Failure
4	Loss of Electric Power	Catastrophic	Negligible	Interference, Manufacturing or Design Flaws
5	Loss of Signal	Serious	Likely	Interference, Electronic Failure, Manufacturing or Design Flaws

Table 20A. Major Engineering Risks

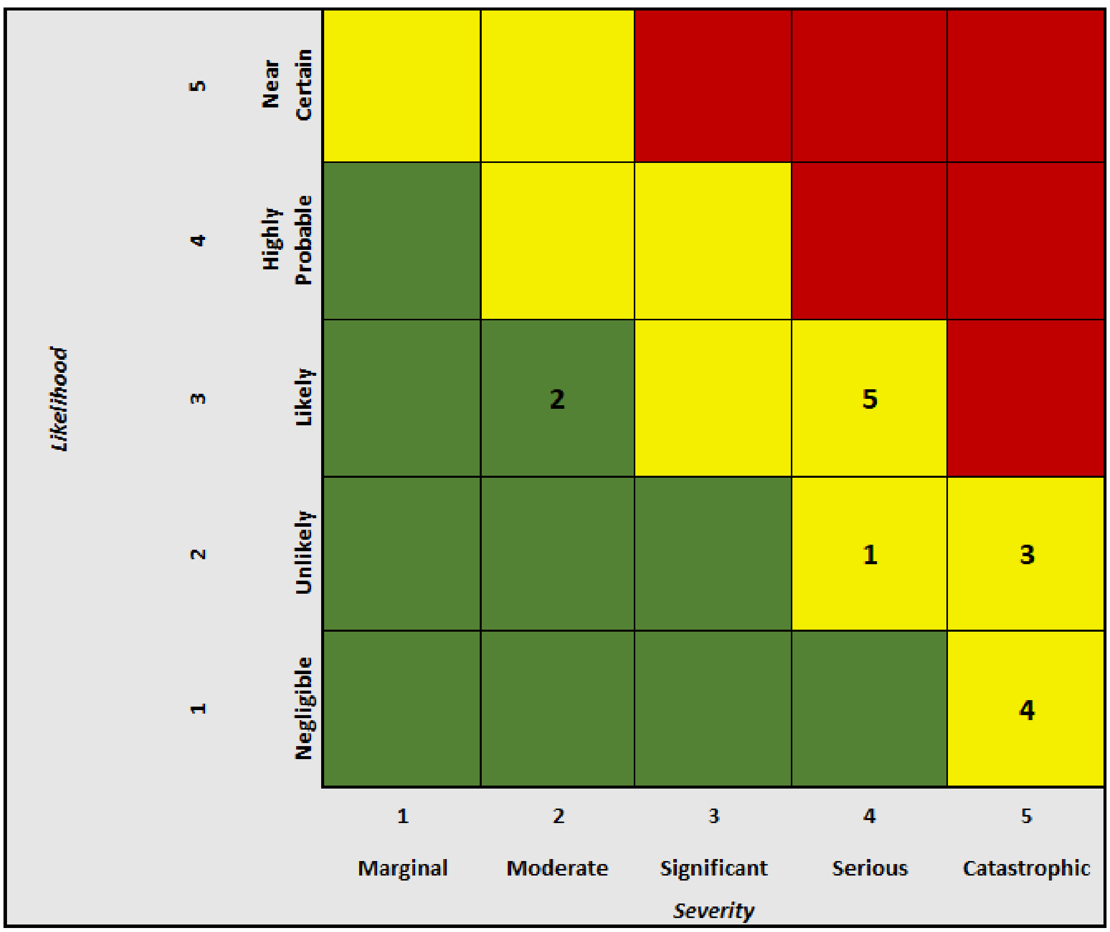


Figure 22A. Major Economic Risks Chart

B. Autonomous Vehicle Platform

The autonomous vehicle platform is to be used in conjunction with the large unmanned vehicle platform. The purpose of the autonomous vehicles, as described by the requirements of the Army contract, is to gather low altitude surveillance autonomously. In addition to these requirements there are some restrictions that are imposed by the large unmanned vehicle. As described in earlier sections of the report, the dimensions of the autonomous vehicles are limited by the size of the payload compartment and payload doors on the unmanned vehicle. These restrictions are reflected in the design as discussed below.

1B. General Design and Overview

The direction of the actual designed and built autonomous vehicle varied from that of the capstone problem statement and requirements. The Army requirements state they need drones that can be deployed from the unmanned aircraft and that they must weigh less than 20lbs. The original capstone requirement was to design, build and test an RC airplane that reflected the Army's requirements within reason. Our capstone group wanted more of a challenge, so the direction our group took was that of a VTOL RC plane. The project still attempted to meet the size constraints as set out by the unmanned vehicle specifications as well as the under 20lbs requirement set out by the Army.

2B. Aircraft Specifications and Performance

2B.1. Sizing and stability analysis

The aircraft has two main flight modes: hover and conventional, forward flight. In forward flight, the wingtip sections are rotated to the horizontal position and the aircraft is more or less conventional. The methods for calculating the static and dynamic stability of the aircraft used were the same as previously implemented for the large aircraft. As previously stated, static stability of an aircraft is split into the three angular degrees of freedom: pitch, roll, and yaw. Longitudinal stability (pitch) is characterised by both C_{M0} (zero angle of attack pitching moment coefficient) and C_{malpha} (change in pitching moment coefficient with respect to angle of attack). Lateral stability is composed of weathercock stability (yaw) and rolling stability, which are characterised by $C_{n\ beta}$ and $C_{l\ beta}$ respectively. The criteria for stability and balance in the are tabulated below.

MATLAB was used for the calculation of all stability parameters (see Appendix C). Since the main source of power for the powerplants of this aircraft is electrical in nature, the center of mass and weight of the aircraft do not change throughout flight. Table 1B and Table 2B contain the results of the MATLAB static stability script.

Axis	Parameter	Value (/rad)	Stability
Pitch	C_{M0}	0.0068838	BALANCED
Pitch	$C_{M\ alpha}$	-1.642	STABLE
Roll	$C_{l\ beta}$	-1.232	STABLE
Yaw	$C_{n\ beta}$	0.3158	STABLE

Table 1B. Static Stability Derivatives

The aircraft is balanced and stable. This is desirable as a complicated active control system will likely not be necessary in forward flight. Plots of CMac can be found in Appendix A.

Dynamic stability of the aircraft is indicative of its behavior in multiple axes, not just the three angular degrees of freedom. Aircraft operate in 6 degrees of freedom, thus more complicated aerodynamic behavior is exhibited.

Oscillations occur while in flight in several combinations of degrees of freedom. The dynamic stability analysis will describe the behavior of the oscillations; whether they are stable, neutral, or unstable. The most common modes of oscillation in aircraft dynamics are the phugoid mode, short period, and the dutch roll.

The characteristic equation and eigenvalues of the A matrix were calculated in MATLAB (see Appendix B). The eigenvalues of the matrix calculated are as follows:

Eigenvalue	Value	Damping Coefficient	Natural Frequency (Hz)	Period (s)	Mode
λ1	-17.5132 + 0i	1	1.75	0.57	Short Period
λ2	0.074 -0i	-1	0.074	13.51	Short Period
λ3	-0.1066 - 0.0645i	0.856	0.125	8	Phugoid
λ4	-0.1066 - 0.0645i	0.856	0.125	8	Phugoid

Table 2B. Eigenvalues of the Aircraft's A Matrix

As seen in Table 2B, the phugoid mode is oscillatory in nature as they are complex eigenvalues. Plotting the root locus of the eigenvalues yields a clearer picture of the longitudinal dynamic flight characteristics of the aircraft in both configurations.

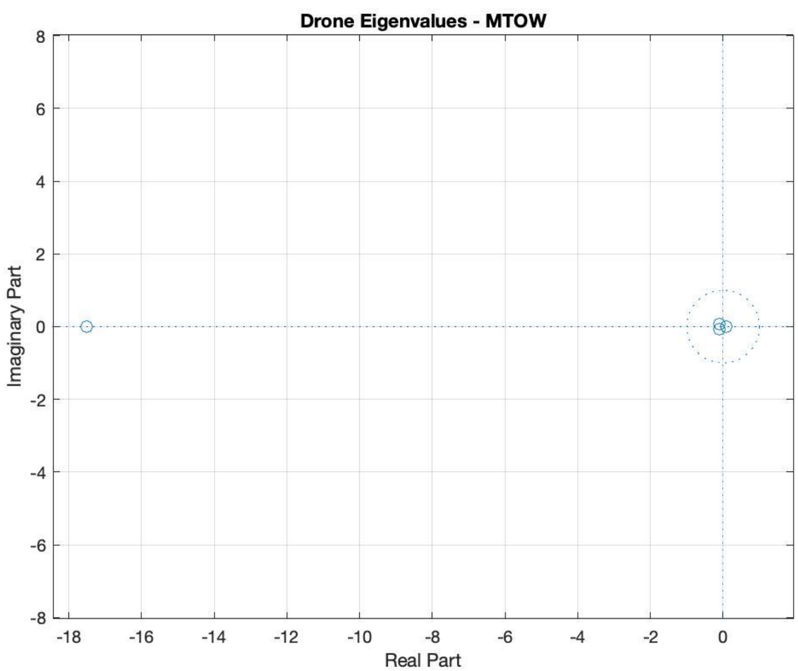


Figure 1B. Eigenvalue Root Locus

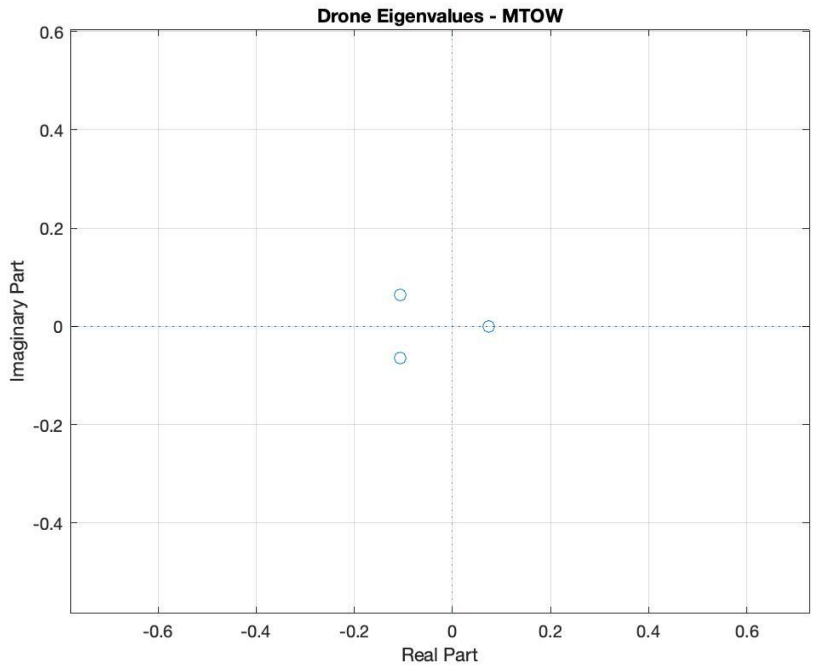


Figure 2B. Eigenvalue Root Locus - Scaled

The short period poles are in the left hand plane, indicating a stable condition. The phugoid mode has two real poles. However, the pole in the right hand plane is very close to the origin and is thus close to stability. It has a relatively long period and can be easily corrected via an autopilot.

In hover, the aircraft requires an active control system. An attitude-hold law was chosen as it makes piloting the aircraft relatively straightforward and was easily implemented. In addition, there are many open source controllers for quadcopters and other similar vehicles that operate predominantly on an attitude hold law. The control system will be discussed in detail in the next section.

2B.2. Control system

Our group was able to find an open-source arduino-based control system called dRehmFlight that we felt would be easily adaptable to our project. The control scheme involved using a PID controller to control each axis (Roll-Pitch-Yaw) using either position or rate control. In order to ensure basic control of the aircraft in case of a control system failure, the receiver outputs were configured to be sent both directly to the control surfaces (ailerons, elevator, rudder) as well as to the arduino board for reference in the control system. As a result, the pilot had full control of the control surfaces at all times. Within the control system, the receiver outputs were used to command roll and pitch positions, as well as yaw rate. The control system was designed to take these commanded values and adjust the thrust of each propeller/fan accordingly, accounting for the differing moment arms and thrust capabilities of the fan/propellers, so that the IMU outputs would match the commanded values. Essentially the control system was designed to operate as three independent SISO systems with PID control.

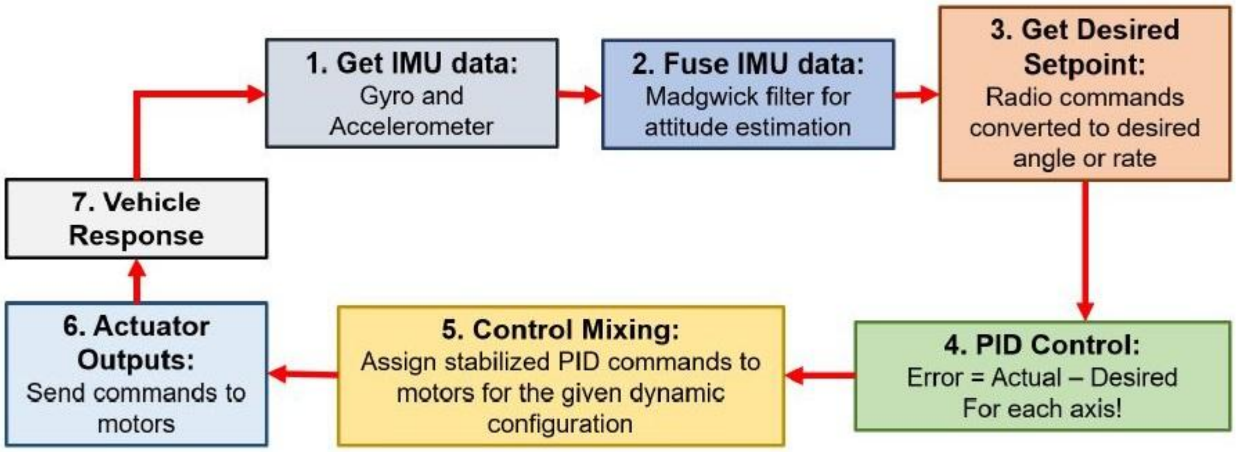


Figure 3B. General overview of flight control loop

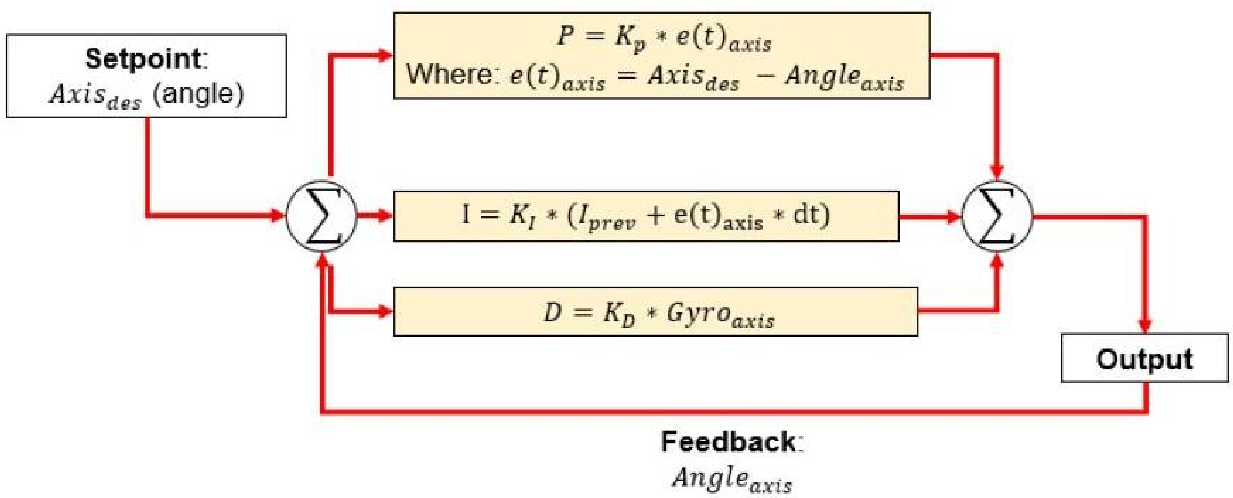


Fig 4B. PID controller stabilizing on angle setpoint; similar for angular rate setpoint

In theory, during test flights the P- and D-gains would be tuned for each axis to increase control authority and decrease oscillations (respectively) in-flight. Then, the I-gains would be tuned to remove steady-state error. Unfortunately, the group was not able to advance to this stage in the test flight phase.

2B.3. Wing Design

The wing design was limited by two major factors: the imposed fuselage cargo-bay size constraint and the manufacturability of the designs. Due to these constraints, a low aspect ratio wing of constant cross-section was selected. As with the large carrier aircraft, the geometry of the wing design was refined through VORLAX code. Unfortunately, the desire for an easy-to-build wing hindered the loading optimization present in the larger aircraft wing, and as a result elliptical loading was not achieved. Additionally, initial calculations were performed assuming a freestream Mach number of 0.1. While this assumption is reasonable for the actual design--where the small aircraft is deployed mid-flight--the RC model failed to achieve the required velocity during takeoff. As a result, the prototype plane struggled to achieve lift in the horizontal test flight.

2B.4. Tilt Rotor Design

The tilt rotor design underwent many alterations before settling upon the final design. In the beginning there were discussions on the best way to implement the tilt rotor. The two main concerns were reducing the failure modes and reducing the downloading. To reduce the downloading, we decided to rotate the entire wing tip along with the motor and prop assembly. This meant that in any orientation, the downloading was consistent and minimal. The reduction of the failure modes was achieved by reducing the number of components to simplify the design. After several iterations, the design seen in Fig. 5B and Fig. 6B was adopted and used on the final prototype. Both Fig. 5B and Fig. 6B display the tilt rotor section with no skin attached so the internal components can be seen. In the working prototype, the tilt rotor sections that had lifting body properties were covered with a skin to give them the required properties. The Final tilt rotor design consists of five components excluding the motor and propeller assembly. These components include the inner bracket, outer bracket, servo, servo coupler, and a steel rod. The inner and outer brackets were both 3D printed out of Polymaker PC-Max. Polymaker PC-Max is an engineered polycarbonate 3D printer filament. This plastic was chosen over other plastics for several reasons. For our application, the tilt rotor and many other 3D printed parts on the aircraft are load bearing and must be able to withstand high stress and potential impact events. Polycarbonate relative to other 3D printer plastics has a very high elastic modulus as well as great impact strength which makes it the most suitable option for our application. 3D printing was the manufacturing method chosen for these two components as we could get complex, high strength parts in a single piece. The inner tilt rotor bracket fit snugly around the end of the two carbon fiber spars, and housed the servo, servo coupler and end of the steel rod. The servo we chose was a standard size 20kg servo. For our application, we needed reliable torque to turn the tilt rotor in any flight regime. Attached to the servo was a coupler. On one end of the coupler, there was a spline gear that fit on the servo and locked in place using a small machine screw. On the other end, there was a pinching mechanism that would tighten around a 6mm diameter shaft using a set screw. This shaft was then very tightly fit and epoxied into the outer tilt rotor bracket, which allowed for a secure connection between the two halves of the tilt rotor mechanism. The outer tilt rotor bracket was designed to take high stresses due to the moment caused by the thrust from the propeller as well as any aerodynamic forces the bracket would experience during flight. The outer bracket also has a designed mounting point for the motor which allows the motor to securely fasten to the end of the wing tip.

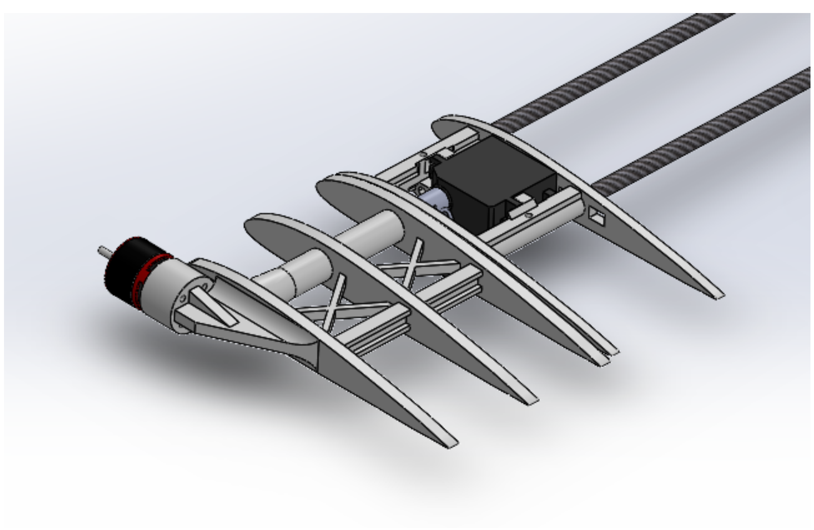


Figure 5B. Left Wing Tilt Rotor in Horizontal Flight

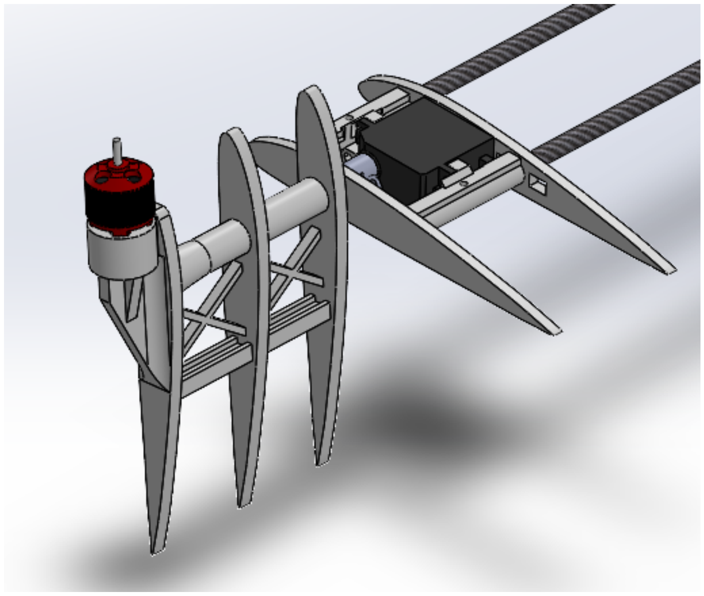


Figure 6B. Left Wing Tilt Rotor in Vertical Flight

2B.5. Propulsion and Power System

The propulsion system selected for this craft takes largely from typical hobby-grade electric RC hardware. The main lift rotors are driven by two iFlight XING 2814 brushless-electric motors, each rated at 1100KV. These were chosen instead of the 880KV identical model, as the slight gains in rotor RPM would ensure the design had sufficient thrust for hover. Initially, these motors were to be paired with 3-bladed 9x4.5in counter rotating drone propellers, as can be seen below in section 5B.1. However, the final design was outfitted with 5-bladed 10x8in props for assured performance and aesthetics. Both motors were controlled through Hobbywing SkyWalker 80A electronic speed controllers, each fitted with a 5V 3A battery eliminator circuit (BEC). Typically, a single BEC of this calibre would be perfectly suited for powering the onboard receiver and other flight control servos. However, the unorthodox nature of the tilt-rotor design called for a slightly more involved power distribution system. As a result, the receiver and main flight control servos were powered through an additional system, explained in more detail below. This allowed for both dedicated tilt servos to be run directly from their respective sides' ESC. Finally, the main lift system was driven by a single Venom 4-cell (4S) 5000mAh 30C LiPo battery, connected in reverse-parallel to both ESCs. A simplified wiring diagram of this system can be seen in Fig. 7B.

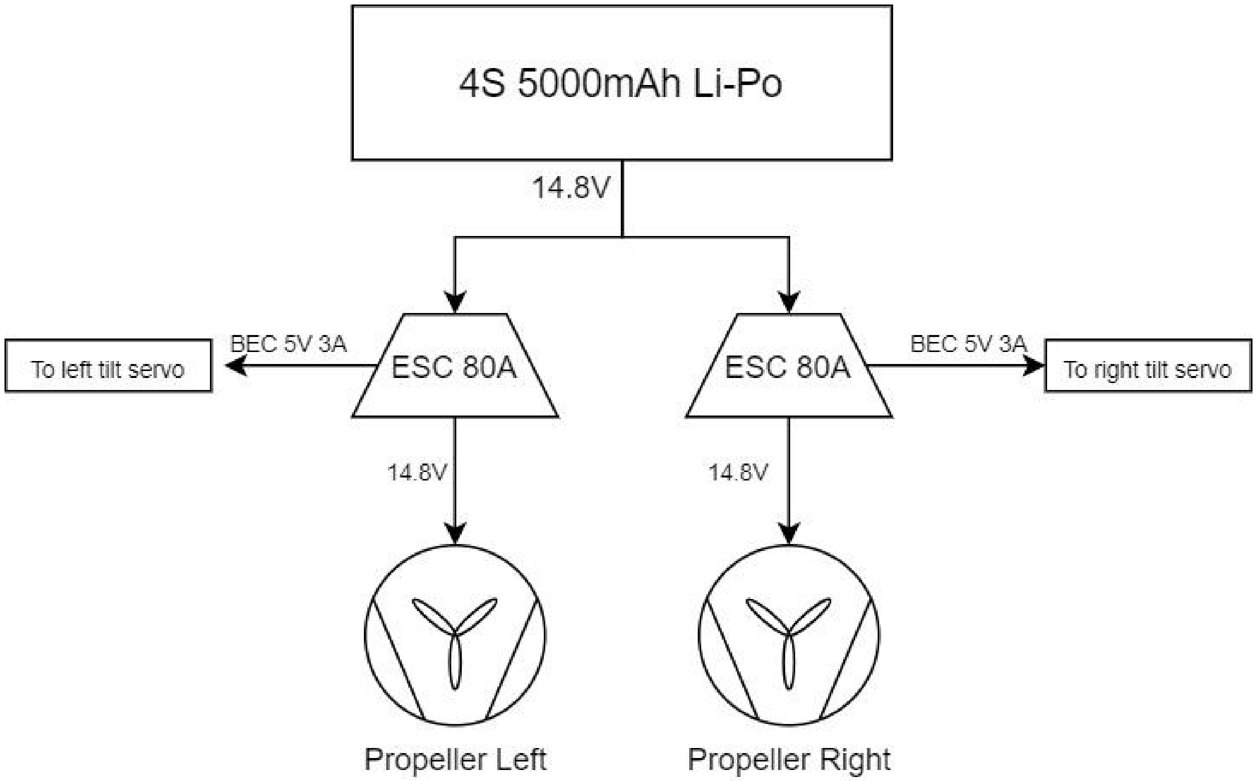


Figure 7B. Propulsion and Power System Diagram

The remaining avionics and EDF were powered by a secondary system, as shown in Fig. 8B. This consisted of the four flight control servos, the stabilizing EDF and associated ESC, Arduino, gyro accelerometer (not pictured), and receiver. Power from a Venom 4S 2200mAh LiPo battery was fed into a power distribution board, where it was split up according to the various requirements of the system. A Powerfun 50mm EDF and 40A ESC were selected to provide pitch stability through the flight control system. As this did not have a dedicated BEC, various voltage step-downs were necessary to ensure the flight control servos and Arduino were receiving power within their safe operating range. The flight control servos were each rated for 6V, and performed nominally. Initially, the Arduino input voltage was also reduced to 6V. However, testing revealed that the Arduino failed to successfully initialize the gyro-accelerometer when disconnected from an external power source. Through a series of careful adjustments to the variable voltage stepper, 8V was determined to be the optimal setting for gyro initialization. Lastly, the receiver and gyro were powered directly from the 5V out pins off the Arduino.

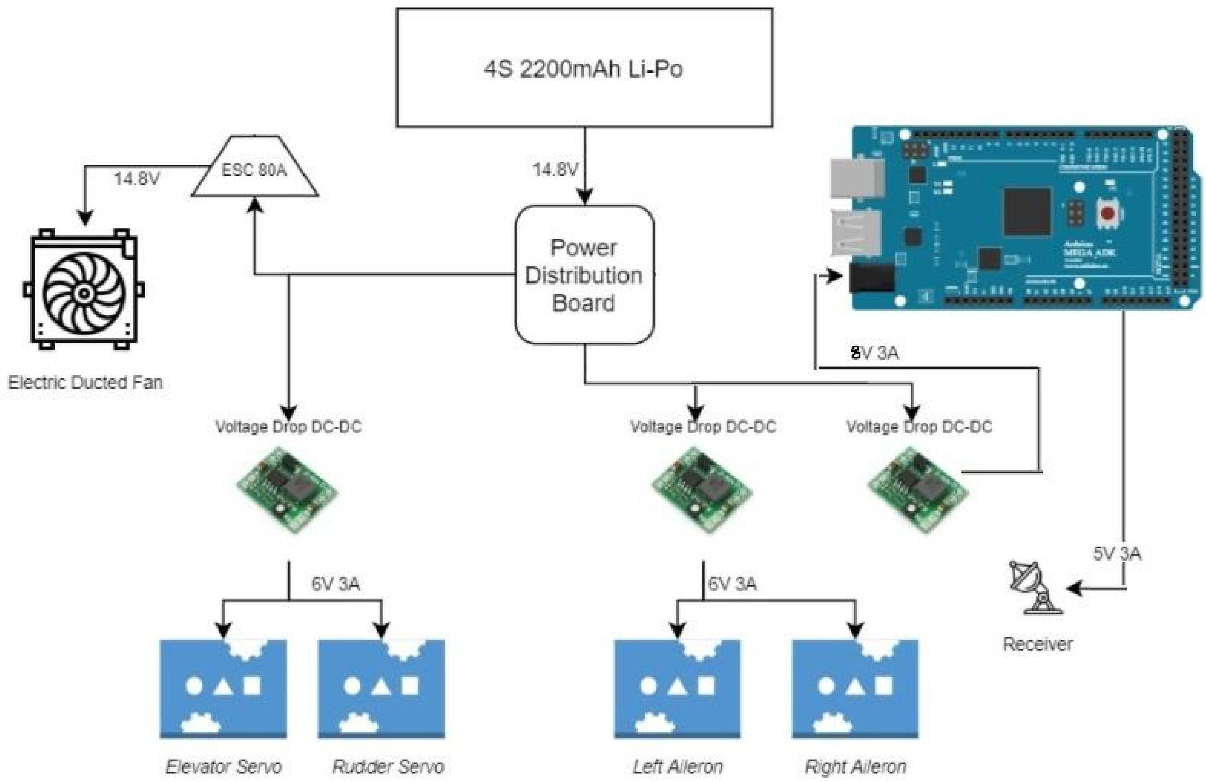


Figure 8B. Control System Diagram

2B.6. Fuselage Design

The fuselage was based on a classical former and stringer design. The formers were made out of balsa wood and the stringers were made of carbon fiber. This design was chosen for a few reasons. One, this allowed the fuselage to be extremely lightweight while keeping the fuselage rigid in all axial and rotational directions. The fuselage, not including the avionics drawer, weighed just under one pound after being fitted and epoxied together. All of the balsa fuselage components were cut using a laser cutter at the ASU print lab. All fuselage components that had perpendicular faces had tongue and groove joints. This added tremendous strength to the fuselage as it created a mechanical bond between components as well as adding more surface area for epoxy and CA glue to adhere to. All of the tongue and groove joints were adhered using two part 1:1 ratio fast curing epoxy. It was chosen to use epoxy on all tongue and groove joints as well as connections between formers and carbon stringers as it would create stronger joints then CA glue as well as fill any gaps between surfaces due to the imprecision in the manufacturing of specific components. The fuselage also had three pieces of 1/8 inch plywood to create plates that spanned most of the length of the fuselage. Two of the plates were epoxied to the fuselage formers and acted to add strength in torsion, as well as add convenient and strong attachment points for any components that would be connected to the fuselage including the wing sections, EDF, avionics drawer slider, and many other components. The third plate in the fuselage was the avionics drawer. The avionics drawer was attached to the bottom fixed fuselage plate using a lightweight drawer slider. The avionics drawer had the ability to be pulled out with the removal of the nose cone to allow access to all of the internal electrical components. This feature was important because there had to be a method to remove the batteries so they could be charged and adjust any potential wiring issues given the complexity of the electrical hardware in a confined space. It was chosen to make the fuselage plates out of thin plywood rather than balsa because they added strength that could not be achieved with balsa. The top plate was the connection point between the wings and the fuselage, and the bottom plate was the connection point between the main landing gears and the fuselage. These were vital connection points that had major consequences if failure were to occur. For this reason we chose the stronger material to be confident in the structural integrity of the fuselage at the expense of a little weight savings. As can be seen in Fig. 8B, near the back of the figure, there are two carbon fiber tubes that have a larger diameter than the main fuselage carbon rods. These carbon fiber tubes connected the main fuselage section to the tail section. These carbon fiber tubes, similar to the main fuselage carbon fiber rods, threaded through fuselage formers and were epoxied to the relevant formers. Overall the fuselage went together well without any major manufacturing or assembly problems.

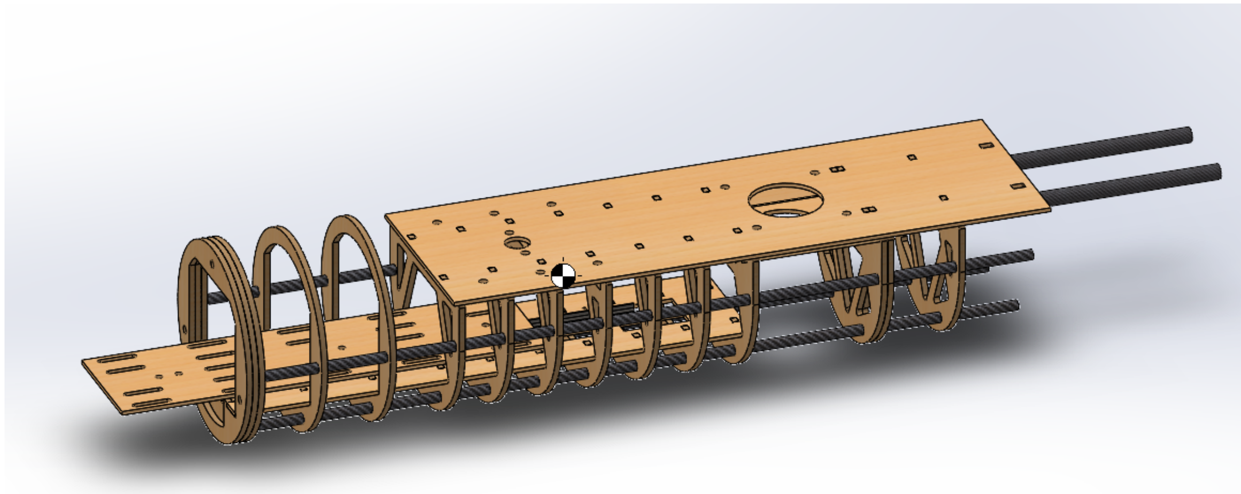


Figure 9B. Fuselage Including Carbon Rods and Sliding Avionics plate

2B.7. Center of Gravity and Weight Distribution

Our stability calculations and modeling were based on a specific location of the center of gravity in forward flight. In order to achieve effective vertical lift capabilities, the cg had to be located behind the lifting line of the vertically-oriented rotors, which placed it slightly behind the center of lift. The large tail was designed to compensate for this fact and improve stability. Additionally, in the forward-flight configuration the cg is located nearly on the center of lift. The avionics drawer in conjunction with the battery-securing straps allowed us to change the location of the cg with relative ease, but once the components were in place they were designed to remain fixed in-flight, which meant that in the vertical configuration the rear-shifted cg caused some instability. The large tail and EDF were intended to compensate for this.

3B. Manufacturing Methods and Material Choices

The main methods for manufacturing components were laser cutting and 3D printing. Laser cutting was used to manufacture many of the components. This was because much of the plane was made of balsa wood or thin plywood sheets, which is a very common material choice for RC plane building. Balsa wood has great strength to weight properties which makes it an ideal choice for many components on an RC plane, and the thin plywood adds strength to the portions of the plane that require extra strength that balsa can not always provide. Laser cutting was chosen as the method for cutting the wood components because it added precision to part tolerances that we would not achieve by hand cutting. Laser cutting was also the fastest way to cut out components which allowed us to manufacture components for a second time quickly if any of the components broke before assembly, which happened several times.

The more uncommon method for RC construction that we used was the prevalence of 3D printed parts. Our design being VTOL, required many complicated parts that would have been difficult to manufacture with wood. Also, many of these components were critical components designed to take some of the highest stresses on the plane. In summary, we needed the ability to create high strength parts, with complex geometry, while making them as light as possible. This is where 3D printing filled the role. For our application, to achieve the results we wanted with the 3D printed parts, we had to use more exotic printer filaments. Many of the common printer filaments such as PLA and ABS did not have the material properties we required. For this reason, we used Polymaker PC-Max filament. PC-Max filament is an engineered polycarbonate plastic which has been slightly modified from raw polycarbonate to have better printing properties, which allows for better layer adhesion and strength properties. Polycarbonate was used as the plastic of choice because of its mechanical properties and thermal properties. For obvious reasons, we wanted a plastic with strong performance across all mechanical properties, but less obvious is the need for good thermal properties. As discussed earlier, PLA was not a suitable material for our application. PLA has relatively good mechanical properties, but PLA has poor thermal properties as well and breaks down in the sun very quickly. Within an hour of intense sunlight exposure, PLA plastic will start to degrade and warp. All of our testing would be completed in the sun of Arizona so a plastic with poor thermal and sun exposure properties would not be acceptable.

MECHANICAL PROPERTIES	THERMAL PROPERTIES
Young's Modulus: 2048 ± 66 Mpa	Glass Transition Temperature: 113˚C
Tensile Strength: 59.7 ± 1.8 Mpa	Vicat Softening Temperature: 117˚C
Bending Strength: 94.1 ± 0.9 Mpa	Melting Temperature: N/A
Charpy Impact Strength: 25.1 ± 1.9 kJ/m2	

Figure 10B. Polymaker PC-Max Material Characteristics

The skin of the plane was chosen to be Monokate. Using fiberglass was discussed, but the lack of experience with the material, as well as the much longer manufacturing time was discouraging given we only had 4 week to complete the build and testing of the plane. Monokate is a heat applied plastic coating that also shrinks with heat. It comes in many colors and is easy to work with which made it an ideal choice for our application.

4B. Risk Analysis

4B.1. Engineering Risk

The unique design of this aircraft means that analyzing risk and identifying potential failure modes is absolutely critical. Since the scale of this craft was mostly in line with that of the typical RC aircraft hardware, the economic risk potential was minimized through utilization of as many off-the-shelf components as possible. On the other hand, the engineering risks associated with implementing this design were slightly augmented by the remote-controlled base platform.

First and foremost, the potential failure of the tilt mechanism, especially during either transition phase, could prove to be detrimental to the performance of the aircraft. The aerodynamic loads on the tilt rotor servo would likely increase significantly during the forward portion of a flight, as the servos themselves provide the torque to hold the wingtips and motors in their horizontal position. It is also possible to suffer an electronics failure. The propulsion, power, and control systems implemented on this aircraft are very specialized and required precise connections to ensure that the right pieces of equipment were in communication. The cabling solution became difficult to navigate and was exacerbated by unwieldy connectors, resulting in a very delicate system. Vibration and/or aerodynamic loads experienced during any phase of flight could jostle the wires, creating faulty connections and blocking signals to and from the arduino. As with any RC aircraft, there is also the possibility that the transmitter and receiver could lose connection at some point during the flight. This would be catastrophic, as the aircraft would no longer respond to any control inputs and shortly thereafter undergo a rapid unscheduled disassembly. The receiver could lose power from the Arduino, as a result of the delicate cabling system and connectors. It is also possible that the communication range between the transmitter and receiver could be exceeded by the aircraft in flight. To mitigate this, a range check will be performed on the day of the flight.

Barring any electronics failure, physical failure modes could occur in the form of lifting and control surfaces. One possibility involves the elevator's effectiveness during forward flight. The chord of the deflecting portion of the elevator is relatively small. Furthermore, the vertical tail imposes physical limitations on the maximum up-elevator deflection angle, which is much less than the maximum throw of the elevator servo in that direction. A weak pitch up authority would not leave much room for error in the forward flight regime, and any takeoffs or landings in this configuration would be dicy to say the least. This was identified early on, and the lower section of the vertical tail was modified to improve the range of motion of the elevator. Lastly, the sizing of the entire wing section itself has caused worry throughout the design process, despite being sufficient according to the aerodynamic calculations. If the wing cannot provide sufficient lift for flight at low enough speeds, the transition from hover to forward flight may not successfully occur, even when utilizing all available thrust. Furthermore, if a servo failure occurs and the aircraft is forced to takeoff or land conventionally, the speed required to achieve and/or maintain flight may exhaust the geographical resources of the field in question. One possible cause of this is a possible mistake in the aerodynamic calculations, resulting from the adaption of a repurposed specialized tool. Another possible cause is misjudging the amount of crud weight and neglecting to factor installation weight penalties.

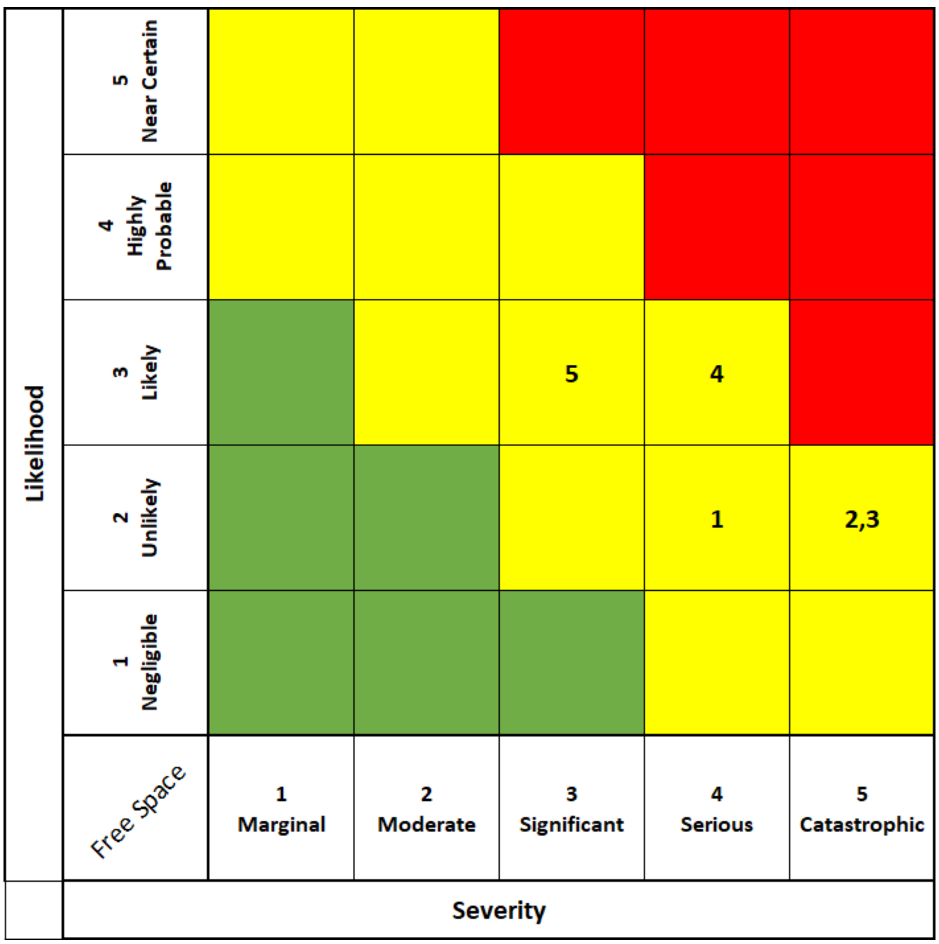


Figure 11B. Major Engineering Risks Chart

#	Type of Risk	Severity	Likelihood	Cause
1	Tilt rotor Servo Failure	Serious	Unlikely	Aerodynamic loads during flight
2	Loss of TX/RX Signal	Catastrophic	Unlikely	Insufficient range, Cabling fault
3	Electronics Failure	Catastrophic	Unlikely	Faulty cabling and connections
4	Insufficient Pitch Authority	Serious	Likely	Elevator Size and range of motion
5	Inadequate Wing Sizing	Significant	Likely	Errant aerodynamic and weight calculations

Table 3B. Major Engineering Risks

5B. Testing

5B.1. Preliminary Testing

Before any of the planned airfield tests, preliminary tests were performed at the build site on a number of different components. The first of these tests were wing loading tests. Due to the primary power plants being placed at the aircraft wingtips, each tip must be able to endure a load greater than half of the mass of the entire vehicle. To test wing strength, the aircraft was hoisted into the air only supported at the tips of the wings and deflection was noted. Resistance to additional loading was then tested by simulating turbulence. No significant bending was found and structural damage did not occur during any of these tests.

While we lacked equipment to properly measure the thrust outputs of our primary engines or ducted fan, tests were still conducted to ensure that their outputs were visibly within reason. A makeshift test stand was constructed to safely test these engines with their propellers on. This testing proved successful, but also exposed oscillations from signal noise that occurred on all engines at throttle settings below 50%. A low pass filter was implemented to remove these oscillations from future tests.

The aircraft's tilt rotor mechanisms were also tested before the first test flight. These tests were primarily used to calibrate the tilt servos, giving us the servo settings that would be needed for both forward and vertical flight. They also were used to find an appropriate time span for the tilt mechanism to fully cycle. These tests allowed us to program the transmitter prior to the first flight, as well as ensure enough slack was provided for wires that were directly affected by the tilting motion.

Control surfaces were of course also tested. This allowed us to find the maximum and minimum servo settings to ensure safe, but maximized, control surface deflection. These values were then programmed into the transmitter and a test was performed to ensure that the surfaces responded correctly to transmitted inputs. Minor changes were made to correct the directions of some control surfaces which were mirrored.

Figure 12B. Taxi Test Configuration

Finally, a ground taxi test was performed to test all of the aircraft's systems on the ground and ensure they were operating properly. Tilt rotor, propulsion, and control surface systems all proved to be fully operational in this test. The aircraft's poor turning ability on the ground was also shown. This is something that was expected due to the lack of a turning nose gear or variable thrust systems, but was an important note for any potential horizontal takeoffs or landings that were to occur.

5B.2. Airfield Test Day 1

On the planned first day of testing the aircraft was deemed to be ready to fly in the VTOL configuration. Minor wiring issues prevented the aircraft from being fully skinned, so it was decided that no level flight test could be performed. Upon arriving at the test field, it was determined to be too windy to fly the aircraft. Wind speeds averaging 14kts were recorded at the field, values which were much too high for this aircraft. Minor issues with the wiring were also uncovered at the field, with the tilt rotor proving unresponsive at first. It was also discovered that the primary battery, which was used to power the two main engines, was defective. A new battery had to be purchased and installed before any further testing could be completed.

Figure 13B. Aircraft on Initial Test Day

5B.3. Airfield Test Day 2

The aircraft was brought back to the test field approximately one week later. Minor changes had been made to the wiring and additional pieces of the fuselage were skinned, however some key parts were left exposed due to concerns about the viability of the new wiring. These concerns were immediately realized at the airfield, where the aircraft was not responding properly to control surface inputs. Changes were made to the code to correct this issue and the aircraft was taken to be tested in its VTOL configuration. Even with maximum throttle, though, the aircraft could not take off vertically. Weight and thrust in this configuration were measured using a specialized scale, coming in at 8.5 lbs and 12.5 lbs respectively. This meant that the aircraft's final thrust-to-weight ratio was nearly 1.5, a value that should have been more than optimal. Close-up inspection of the aircraft in operation showed that the ducted fan would inexplicably lose thrust at high throttle levels, causing an inability to maintain a horizontal pitch attitude. This also led to a condition where thrust was approximately equal to weight, explaining the failure to take-off. The issue was thought to be in the aircraft's control system, with the fan responding inappropriately to gyroscope data. With the main lift rotors removed, testing was done at high throttle input on the test stand. No adverse thrust characteristics were displayed by any component of the propulsion system, even during manual recreation of abrupt attitude changes. A temporary modification to the control system was made to prove the aircraft was capable of lifting off of the ground. The aircraft's pitch stability controls were removed, with the ducted fan instead outputting a value which only scaled with throttle input. A successful test was recorded, however the large instability in pitch was very clear.

Figure 14B. Visible Pitch Instability Upon Liftoff

Further attempts were made to see if the aircraft could reliably hop off the ground, but these were unsuccessful. Despite the direct control of the ducted fan through the transmitter, it was auditorily evident that the ducted fan was not delivering continuous thrust. At throttle settings slightly over half and above, the ducted fan periodically cycled between zero thrust and the desired thrust setting, roughly every 1-2 seconds. When near-full throttle was commanded, similar behavior was observed in the main lift rotors, in the form of choppy fluctuations in motor rpm. This behavior was exhibited even when ground-effect-induced roll oscillations were small, indicating that the gyroscope was unlikely to be the culprit. It was then hypothesized that the propulsion system could be experiencing some sort of current limitation, caused by insufficient supply of power. This was agreed upon by several professionals in attendance, and no further attempts were made for the next few minutes. A last-ditch effort was then made to hop the aircraft, which had been sitting idle in the interim. The battery voltage had likely stabilized during the downtime, and full continuous thrust was delivered to all motors for a few seconds. However, the inherent instability of the vertical configuration, the inactive pitch control system, and the unexpected nominal thrust delivery caused the aircraft to violently pitch up and flip over. The right wingtip impacted the ground first, shattering the propeller and damaging the tilt rotor mechanism.

Figure 15B. Final Result of Failed VTOL Test

While the test was an overall failure, the profound strength of the aircraft's structure was clearly demonstrated. The wing and airframe as a whole survived a full on and direct impact on the wingtip, experiencing loads much greater than would be recorded on any normal landing or during turbulent flight conditions.

5B.4. Airfield Test Day 3

Prior to this test, damage sustained to the aircraft on the previous test day had to be repaired. Minor cracking of wing ribs and fuselage formers was found, but no severe structural damage had occurred. These pieces were reinforced with epoxy and CA glue as a precautionary measure. The damage to the tilt rotor mechanism was found to be too severe to repair within the remaining time span. For this reason, the tilting portions of the wingtips were bolted into the forward flight position to ensure that they did not slip in-flight. The control system used to fly in the VTOL configuration was also removed to simplify the wiring, but no significant weight changes were made. The aircraft was then re-skinned fully to minimize drag and prevent wires from coming loose in-flight.

On the day of the final test flight, the state of the aircraft was evaluated by completing initial testing. First was a range test, in which the controls were walked 250 feet away from the aircraft to ensure it would still respond to input from a significant distance in flight. Next, the center of gravity of the aircraft was evaluated by lifting it upwards by the wings. Doing so caused the aircraft to pitch upwards, revealing that the slight changes made after the last test moved the CG slightly further back than designed for. This was a cause for concern for the stability during flight, so in order to move the CG forward, a few small lead weights were taped into the front of the nose cone. Along with this weight, the smaller battery was placed in the nose cone before re-attaching it to the fuselage. Then it was time for a high speed taxi test in order for the pilot to get a feel for the aircraft. Ultimately this test ended in the aircraft crashing without lifting off the ground due to limited ground control.

6B. Hindsight Design Choices and Testing Procedures

Reflecting on the results of the project, our group came up with a list of improvements or design changes that we would have liked to implement were we to build a second iteration of the prototype. First, the fuselage compartment where avionics and batteries were stored was very cramped and difficult to operate in once the plane was constructed. The sliding avionics tray was designed with this issue in mind, but in retrospect the fuselage needed to be bigger to accommodate the equipment used. The group also had initially planned on installing a battery kill switch, but decided against this to avoid additional wiring. We now believe the additional wiring cost would be well worth the benefit of not having to unplug the batteries every time we wanted to make a minor adjustment. In addition to increasing the fuselage size, it was obvious that the wingspan needed to be greatly increased. While the vehicle did exhibit sufficient thrust/weight characteristics, the wetted area of the wings was simply too small to lift the aircraft. Several additional improvements were also discussed, including: using higher discharge batteries; using bottom-mounted aileron rods; buying a prefab PCB and other electrical components rather than crafting them from available materials; performing additional component testing (eg. wind tunnel testing of a wing section) before final assembly; taking advantage of tethered flight conditions to better test the final vehicle.

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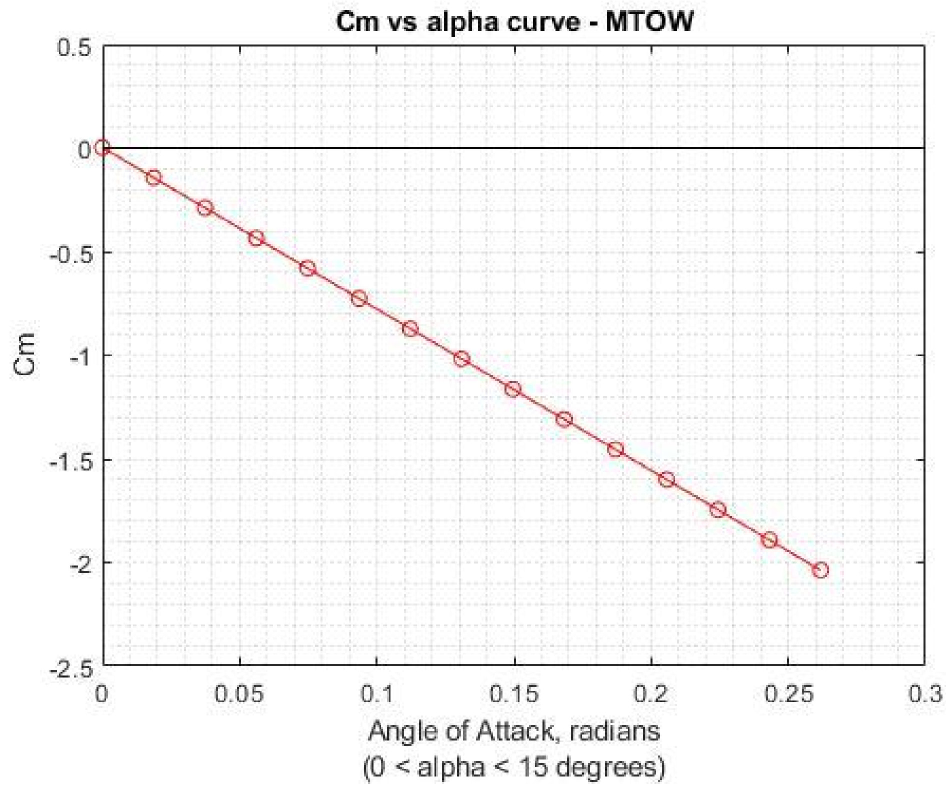
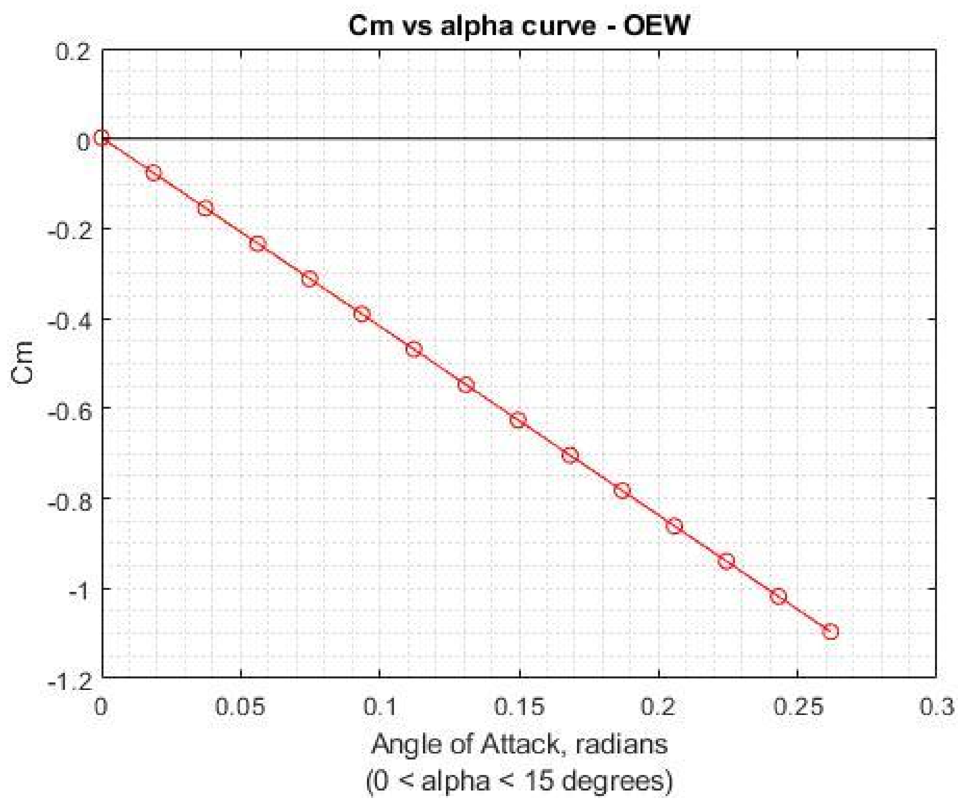
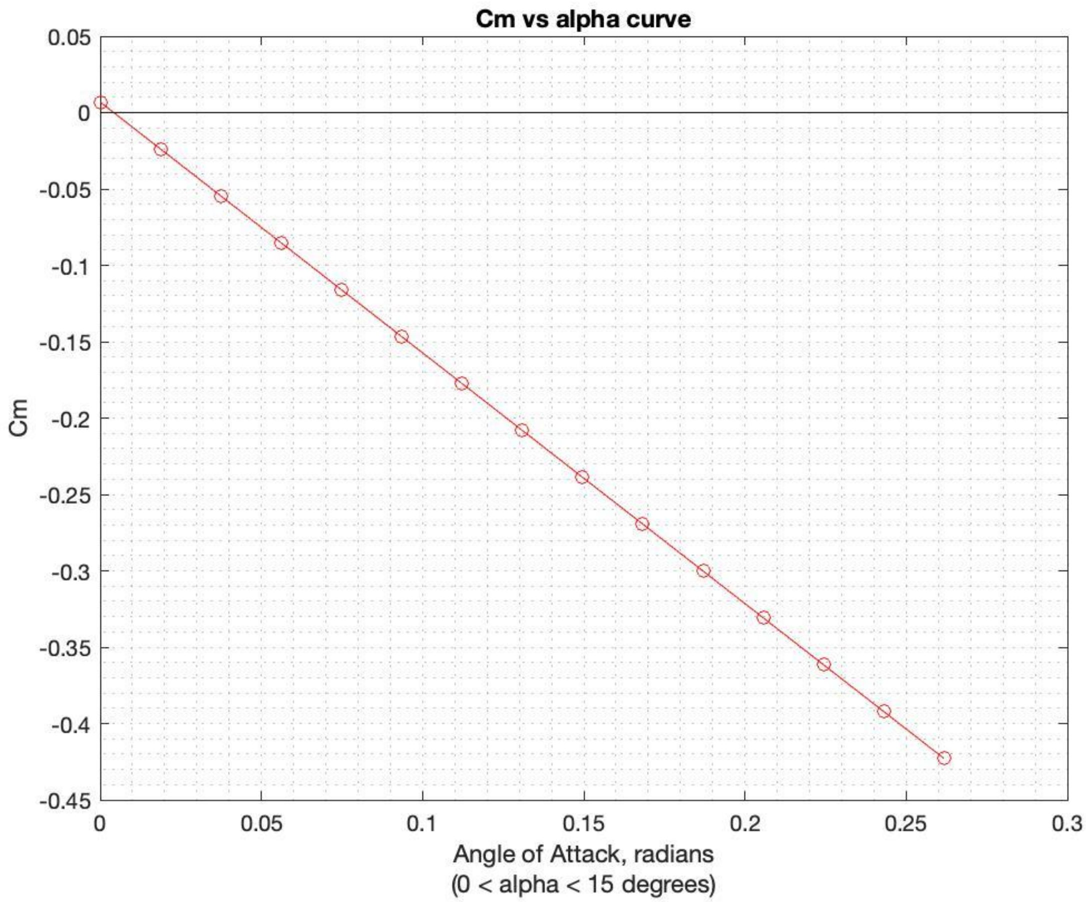
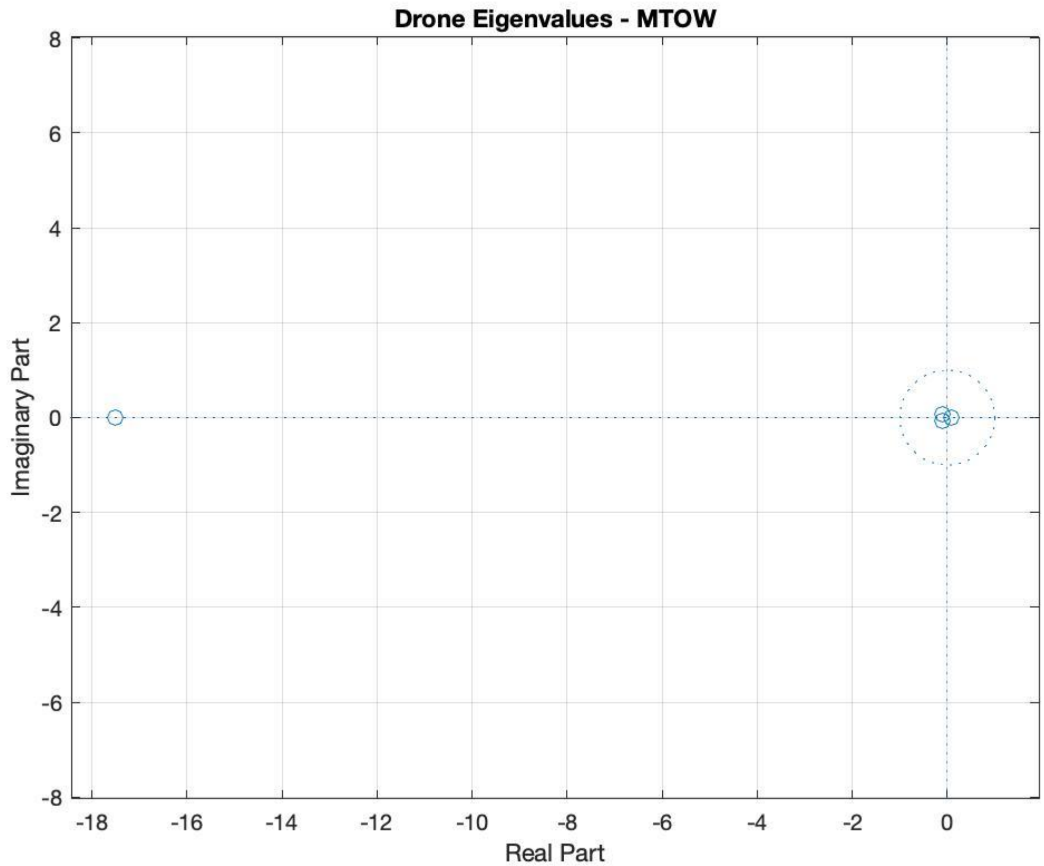
Appendix A - CMac Plots - Stability







Cm vs alpha curve
0.05
0
-0.05
-0.1
-0.15
-0.2
-0.25
-0.3
-0.35
-0.4
-0.45
Cm
0
0.05
0.1
0.15
0.2
0.25
0.3
Angle of Attack, radians
(0 < alpha < 15 degrees)


Drone Eigenvalues - MTOW
Imaginary Part
Real Part
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4
2
0
-2
-4
-6
-8
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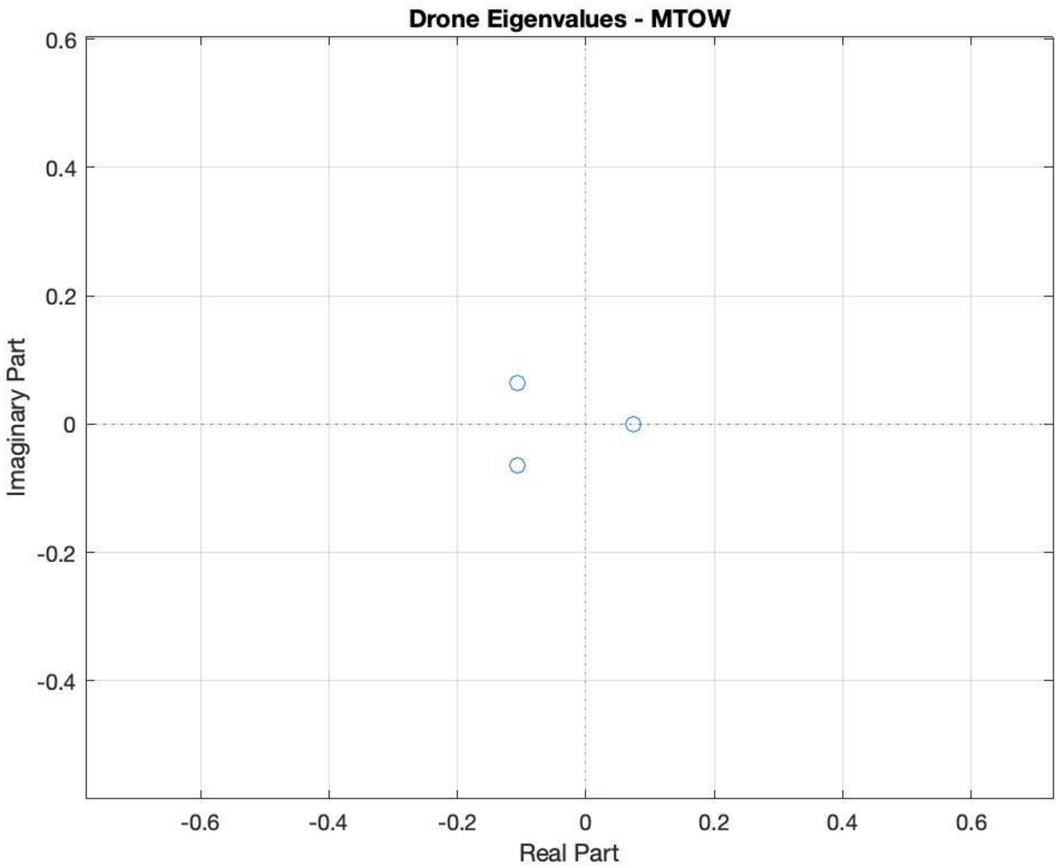
-6
-4
-2
0


Drone Eigenvalues - MTOW
Imaginary Part
Real Part
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0.4
0.2
0
-0.2
-0.4
-0.6
-0.4
-0.2
0
0.2
0.4
0.6

Appendix C - MATLAB Code - Stability Analysis

```
Clear;
clc;

% Wing/body parameters
%---------------------------------------------------------------------------
c_bar = 3.345; % mean aerodynamic chord (ft)
S = 217.427; % wing area (ft^2)
b = 65; % wingspan (ft)
lamda = 0.65; % taper ratio
a_wb = 3.9298; % lift curve slope of the wing and body (/rad)
a_w = 3.9298; % lift curve slope of the wing (/rad)
gamma = 0; % dihedral angle (degrees)
e = 0.9; % oswald efficiency factor
sweep = 0; % sweep angle of quarter chord line (degrees)
h_n_wb = 0.25; % neutral point of the wing and body
C_m_ac_wb = -0.04056636; % pitch moment coefficient about the aerodynamic center of the wing
and body
C_L_0_w = 0.2; % something
z_f = 5*sqrt(2)/2; % feet something
C_l_beta_wb = 0; % wing body interference
%---------------------------------------------------------------------------

% Tail parameters
%---------------------------------------------------------------------------
S_t = 20; % area of horizontal tail (ft^2)
S_F = 20; % area of vertical tail (ft^2)
l_t_bar = (57.61+46)/12; % distance between aerodynamic centers of wing and tail (ft)
a_t = 3.7887; % lift curve slope of the horizontal tail (/rad)
a_F = 3.9298; % lift curve slope of the vertical tail (/rad)
i_t = 3; % angle of incidence of the tail (degrees)
epsilon_0 = 0; % downwash angle (degrees)
depsilon_dalpha = 0; % change in downwash wrt alpha
l_F = (124+46)/12; % distance between CG and aero center of vertical tail (ft)
V_foverV = 1.2; % velocity ratio, = 1 if the vertical tail is not in the slipstream of the
propulsion system
dsigma_beta = 0.00278099; % change in sidewash wrt beta
%---------------------------------------------------------------------------

% Engine parameters
%---------------------------------------------------------------------------
A_i = 1.97; % inlet area (ft^2)
A_j = 0.21; % outlet area (ft^2)
x_j = 60/12; % x distance from engine normal force to CG (ft)
z_p = 2.144166; % z distance from engine thrust line to CG (ft)
dx_j_alpha = 0; % change in x_j wrt alpha
dx_j_beta = 0; % change in x_j wrt beta
alpha_j = 0; % engine alpha (degrees)
epsilon_j = 0; % engine upwash due to disturbance (degrees)
depsilon_j = 0; % change in upwash wrt alpha
mdot = 0.54283; % mass flow rate of the engine (lbm/sec)
beta_j = 0; % sideslip angle of the engine (degrees)
sigma_j = 0; % sidwash angle of the engine (degrees)
dsigma_j_beta = 0; % change in sidwash angle wrt sideslip
n = 1; % number of engines
C_m_0_p = 0; % pitch moment due to propulsive effects
dC_m_0_p_alpha = 0; % change in pitch moment due to propulsive effects wrt alpha
%---------------------------------------------------------------------------

% General parameters
```

```
%--------------------------------------------------------------------------
x_cg = -57/12; % CG distance (ft)
V = 258.4884; % velocity (ft/s)
rho = 0.001066; % density (slug/ft^3)
rho_j = rho; % density of air entering jet engine (slug/ft^3)
theta_j = alpha_j + epsilon_j; % effective engine angle (degrees)
W = 4139.38; % weight (lbs)
T = 1850; % thrust (lbs)
Iyy = 73634.18465; % slug ft^2
%--------------------------------------------------------------------------

%% Calculated Longitudinal Stability
% C_m_0 > 0 for balance
% C_m_alpha > 0 for static stability in pitch

% V_h_bar
V_h_bar = l_t_bar*S_t/(c_bar*S);

% lift curve slope of entire vehicle
a = a_wb*(1 + (a_t*S_t)/(a_wb*S)*(1-depsilon_dalpha)); % /rad

% nondimensionalized cg location (h)
h = x_cg/c_bar;

% Propulsive effects
deltaC_m = mdot^2*2*x_j*theta_j/(A_j*rho_j*rho*V^2*S*c_bar); % effect that the propulsion
system has on C_m
deltaC_m_alpha = 2*mdot^2/(A_j*rho_j*rho*V^2*S*c_bar)*(x_j*(1+depsilon_j) +
theta_j*dx_j_alpha); % effect that the propulsion system has on C_m_alpha

% neutral point (h_n)
h_n = h_n_wb + a_t/a*V_h_bar*(1-depsilon_dalpha) - 1/a*deltaC_m_alpha;

% C_m_0
C_m_0 = C_m_ac_wb + a_t*V_h_bar*(epsilon_0 + i_t)*(pi/180)*...
 (1 - (a_t*S_t)/(a_wb*S)*(1-depsilon_dalpha)) + deltaC_m;

% C_m_alpha
C_m_alpha = a*(h-h_n_wb) - a_t*V_h_bar*(1-depsilon_dalpha) + deltaC_m_alpha + dC_m_0_p_alpha;

% Display the results
disp("Our shitty little drone:" + newline + newline + "Pitch Stability" + newline +...
 "C_m_0 = " + C_m_0 + newline + "C_m_alpha = " + C_m_alpha)

if C_m_0 > 0
 disp("The aircraft is balanced.")
else
 disp("The aircraft is not balanced.")
end
if C_m_alpha < 0
 disp("The aircraft is STABLE in pitch.")
else
 disp("The aircraft is UNSTABLE in pitch.")
end

%% Calculate Roll Stability
% C_l_beta < 0 for static stability in roll (dihedral effect)

% C_l_beta
C_l_beta = -a_w/4*(2*(1+2*lamda)/(3*(1+lamda))*gamma) -
(1+2*lamda)/(6*(1+lamda))*C_L_0_w*sind(2*sweep) -...
 S_F*z_f/(S*b)*a_F*(1-dsigma_beta)*(V_foverV)^2 + C_l_beta_wb;
```

```matlab
% Display the results
disp(newline + "Roll Stability" + newline +...
 "C_l_beta = " + C_l_beta)


if C_l_beta < 0
 disp("The aircraft is STABLE in roll." + newline)
else
 disp("The aircraft is UNSTABLE in roll." + newline)
end


%% Calculate Lateral Static Stability
% C_n_beta > 0 for static stability in yaw (wethercock stability)

% Vertical tail volume ratio
V_V = S_F*l_F/(S*b);

% Propulsive flow turning effect
deltaC_n_p_beta = mdot^2*2/(A_i*rho_j*rho*V^2*S*b)*(x_j*(1+dsigma_j_beta) +
theta_j*dx_j_beta);

% Calculate C_n_beta
C_n_beta = V_V*a_F*(V_foverV)^2*(1-dsigma_beta) + deltaC_n_p_beta;

% Display the results
disp(newline + "Yaw Stability" + newline +...
 "C_n_beta = " + C_n_beta)

if C_n_beta > 0
 disp("The aircraft is STABLE in yaw." + newline)
else
 disp("Why the hell did you make this unstable in yaw." + newline)
end

%% plots
alphas = linspace(0, 0.261799, 15);
Cms1 = C_m_alpha.*alphas + C_m_0;
plot(alphas, Cms1, 'r-o', linspace(0, 0.3, 10), zeros(1, 10), 'k')
grid on
grid minor
grid
title("Cm vs alpha curve")
xlabel("Angle of Attack, radians" + newline + "(0 < alpha < 15 degrees)")
ylabel("Cm")




%% DYNAMIC STABILITY
CL_alpha = a; % lift curve slope of the config (/rads)
L = W;
C_D_0 = 0.015;
C_D_v = 0;
C_D_q = 0;
g = 32.174444444444; % gravity (ft/s^2)
nu = 1;
a_bar = 1013.975;
q_0_bar = 0.5*rho*V^2;
M_0 = V/a_bar;
m = W/g; % mass (slugs)
```

```
alpha_0 = 2*W/(rho*V^2*S*CL_alpha);
L_alpha = q_0_bar*S*CL_alpha;
C_L = L/(q_0_bar*S);
C_L_v = 1/a_bar * M_0/(1-M_0^2)*C_L;
dL_V = rho*V*S*(C_L + V*C_L_v/2);
dT_V = 0; % zero for jet or rocket prop
h_n = h_n_wb + a_t*V_h_bar/a*(1-depsilon_dalpha) - 1/a*dC_m_0_p_alpha; % check this one (last
prop term)
SM = h_n - h;
V_h = V_h_bar - S_t/S*(h-h_n_wb);
l_t = V_h*S*c_bar/S_t;
dC_L_q = 1.1*nu*a_t*S_t*l_t/(S*V);
dL_q = q_0_bar*S*dC_L_q;
AR = b^2/S;
K = 1/(pi*AR*e);
dD_alpha = rho*V^2*S*K*C_L*CL_alpha;
C_D = C_D_0 + K*C_L^2;
dD_V = rho*V*S*(C_D + V*C_D_v/2);
dD_q = 1/2*rho*V^2*S*C_D_q;
dM_aero_alpha = q_0_bar*S*c_bar*C_m_alpha; %*a*-SM;
dC_m_q_bar = 0.3;
C_m = C_m_0 + C_m_alpha*alpha_0 + 0;
C_m_v = dC_m_q_bar*rho*V;
dM_aero_V = rho*V*S*c_bar*(C_m + V/2*C_m_v);
C_m_q = -1.1*V_h*a_t*l_t*nu/V;
dM_aero_q = q_0_bar*S*c_bar*C_m_q;
alpha_0 = 2*W/(rho*V^2*S*CL_alpha);
CMALPHA1 = C_m_alpha;
CM01 = C_m_0;

%% Making the Thiccy Matrix
% define some stuff we need
dL_alpha = 0.5*rho*V^2*S*CL_alpha;

% first row
a11 = 1/V*(-g*cos(alpha_0) - (dL_alpha + T*cos(alpha_0))/m);
a12 = (-dL_V-dT_V*sin(alpha_0))/(m*V) - 1/V^2*(g*cos(alpha_0) - (L + T*sin(alpha_0))/m);
a13 = 1 - dL_q/(m*V);
a14 = g/V*sin(alpha_0);

% second row
a21 = 1/m*(-T*sin(alpha_0) - dD_alpha) + g*cos(alpha_0);
a22 = 1/m*(dT_V*cos(alpha_0) - dD_V);
a23 = -1/m*dD_q;
a24 = -g*cos(alpha_0);

% third row
a31 = 1/Iyy*dM_aero_alpha;
a32 = 1/Iyy*dM_aero_V;
a33 = 1/Iyy*dM_aero_q;
a34 = 0;

% fourth row
a41 = 0;
a42 = 0;
a43 = 1;
a44 = 0;


% make the thiccy
A = [a11 a12 a13 a14; a21 a22 a23 a24; a31 a32 a33 a34; a41 a42 a43 a44];
disp("The eigenvalues of the A stability matrix are:")
```

```
disp(eig(A))
figure()
zplane(eig(A))
title("Drone Eigenvalues - MTOW")
grid on
```

Appendix D - Turbofan Models

Manufacturer	Model	TSFC	Weight	Max Thrust
Pratt & Whitney	610F	0.50	289.3	900
Pratt & Whitney	615F	0.50	310	1459
Pratt & Whitney	617F	0.50	380	1615
Pratt & Whitney	545C	0.44	830	4119
Pratt & Whitney	545B	0.44	830	3952
Pratt & Whitney	535A	0.44	699	3400
Pratt & Whitney	530	0.44	617	2887
Pratt & Whitney	308	0.394	1373	7000
Pratt & Whitney	307	0.407	1215	6405
Pratt & Whitney	306	0.407	1151	6040
Pratt & Whitney	305	0.407	997	5220
Williams International	FJ33	0.486	319	1846
Williams International	FJ44-1AP	0.47	460	2100
Williams International	FJ44-2	0.50	460	2300
Williams International	FJ44-3	0.50	510	3000
Williams International	FJ44-4	0.50	658	3600
Honeywell	TFE731-2	0.50	743	3500
Honeywell	TFE731-3	0.51	754	3700
Honeywell	TFE731-4R	0.52	822	4080
Honeywell	TFE731-5AR	0.469	884	4500
Honeywell	TFE731-5BR	0.470	899	4750
Honeywell	TFE731-20AR	0.441	885	3650

Honeywell	TFE731-40	0.457	885	4250
Honeywell	TFE731-60	0.405	983	5000
GE-Honda	HF118	0.7	466	2050
Rolls Royce	AE 3007A	0.36	1657	9500
GE Aviation	CJ610-6	0.98	395	2950

Appendix E - MATLAB Code - Engine Selection

```
% Initial Calculations for Large Aircraft based on US Army Design
% Requirements
%%
% Initial Setup
clc; clear;
close all;
%%
% Known Values
RUNWAY = 2000;      % [ft]
OBSTACLE = 38;      % [ft]
ENDURANCE = 24;     % [hr]
R1 = 510;           % [nm]
R2 = R1;            % [nm]

PYLD = 450;         % [lbf]

%% Values from Similar aircraft
%
%              Table 1. Values from Similar Aircraft
%  ______________________________________________________________
% | Aircraft | Global Hawk | Predator| Reaper    | U2           |
% |__________|_____________|_________|___________|______________|
% | Sref     | 685         | 123.3   | redacted | 1000         |
% | b        | 130.9       | 55      | 66        | 103          |
% | AR       | 25          | 24.5    | redacted | 10.6         |
% | R        | 12300       | 770     | 1150      | 7000         |
% | Endurance| 34          | 24      | 14        | 12           |
% | KTAS     | 310         | 70      | 200       | 412          |
% | Ceiling  | 60000       | 25000   | 50000     | 70000        |
% | PLYD     | 3000        | 450     | 3750      | 5000         |
% | OEW      | 14950       | 1130    | 4900      | 16000        |
% | W_Fuel   | 17300       | 665     | 4000      | 20099        |
% | MTOW     | 32250       | 2250    | 10500     | 40000        |
% |__________|_____________|_________|___________|______________|
%
%              Table 2. Similar Aircraft Thrust Values
%  ______________________________________________________________________
% | Aircraft | Global Hawk     | Predator      | Reaper      | U2          |
% |__________|_________________|_______________|_____________|_____________|
% |Powerplant| RR AE           | Rotax 914F    | Honeywell   | GE F118-101 |
% |          | 3007A           | 4-cyl         | TPE331-10GD|             |
% |Type      | Turbofan        | Reciprocating| Turbofan    | Turbofan    |
% |Thrust    | 8500            |               |             | 19000       |
% |Weight    | 1657            | 164.7         | 380         | 3150        |
% |TSFC      | 0.36            |               | 0.55        | 0.375       |
% |Power     |                 | 115           | 900         |             |
% |__________|_________________|_______________|_____________|_____________|
%

Values = zeros(15,4);

    Values(1,:) = [685,123.3,NaN,1000];        % Wing reference area [ft^2]
    Values(2,:) = [130.9,55,66,103];           % Wingspan          [ft]
    Values(3,:) = [25,24.5,NaN,10.6];          % Aspect ratio
    Values(4,:) = [12300,770,1150,7000];       % Range             [nm]
    Values(5,:) = [34,24,14,12];               % Endurance         [hr]
    Values(6,:) = [310,70,200,412];            % Cruise speed      [KTAS]
    Values(7,:) = [60000,25000,50000,70000];   % Ceiling           [ft]
    Values(8,:) = [3000,450,3750,5000];        % Payload           [lbf]
```

```
    Values(9,:) = [14950,1130,4900,16000];      % OEW               [lbf]
    Values(10,:) = [17300,665,4000,20099];      % FW                [lbf]
    Values(11,:) = [32250,2250,10500,40000];    % MTOW              [lbf]
    Values(12,:) = [8500,NaN,NaN,19000];        % Thrust            [lbf]
    Values(13,:) = [1657,164.7,380,3150];       % Power plant Weight[lbf]
    Values(14,:) = [0.36,NaN,0.55,0.375];       % TSFC      [lb/(lbf hr)]
    Values(15,:) = [NaN,115,900,NaN];           % Power             [hp]

%% Secondary Values
TW_market = Values(12,:)./Values(11,:);         % Thrust-to-weight ratio

%% Wing Area Estimation of Similar Aircraft Based on MTOW and Range
%
% AR=b^2/S
%  ________________________________________________________
% | Aircraft                | MTOW     | b          | AR       |
% |_________________________|__________|____________|__________|
% | Global Hawk             | 32250    | 130.9      | 25       |
% | Predator                | 2250     | 55         | 24.5     |
% | U2                      | 40000    | 103        | 10.6     |
% | Reaper                  | 10500    | 66         |__________|
% | IAI Eitan               | 11905    | 85         |
% | DRDO Rustom             | 1590     | 25.9       |
% | Bateleur                | 2205     | 49         |
% | Bayraktar Akinci        | 12125    | 65         |
% | TAI Aksungur            | 7275     | 79         |
% |_________________________|__________|____________|
%
% Assume AR around 25; expected for long endurance flight due to high Emax
% Use assumption to estimate Sref for each UAV
AR = 32;

W = [32250,2250,40000,10500,11905,1590,2205,12125,7275];
b = [130.9,55,103,66,85,25.9,49,65,79];

S = b.^2/AR;

WS = W./S;
%% Takeoff Estimation of SW Requirement
CLmax = 1.7;                 % Approximate CLmax with Flower flaps
                             % deployed (Vol. 1 pg. 13)

% Thrust requirements from maximum field length
% Critical Field Length [ft] (Vol. 2 pg 99)
%   CFL =37.5*ws./(CLmax*TW)

    WSavg = mean(WS);
    DWS = std(WS);
    WS = WSavg + [-DWS,0,DWS];

    % Runway length at sea-level
    R = linspace(800,RUNWAY,100);

    figure
    hold on
        for i = 1:3
            TW = (37.5*WS(i))./(CLmax*R);
            plot(R,TW)
        end
    hold off

    title("Thrust-to-weight Ratio Requirements of Aircraft")
```

```
    xlabel("Distance to Wheels-off-ground, [ft]")
    ylabel("Thrust-to-weight Ratio")
    grid on

    [leg,~] = legend(sprintf("%.2f",WS(1)),sprintf("%.2f",WS(2)), sprintf("%.2f",WS(3)));
    leg.Location = 'northeast';
    title( leg, "Wing Loading [psf]")
    leg.Title.Visible = 'on';
    leg.Title.NodeChildren.Position = [0.55, 1.2, 0];

%% Estimate MTOW by Engine
% TW requirement
TWreq = 0.40;

% Calculations independent of engine selection
    e = 0.85;                       % Oswald's efficiency
    k = 1/(pi*AR*e);
    CD0 = 0.032;                    % EDET estimated zero-lift drag drag
    CD0 = 1.30*CD0;                 % Crud drag correction factor

    LD = 10;                        % Cruise lift-to-drag
    LDm = 0.5*sqrt((pi*e*AR)/CD0);  % Maximum lift-to-drag
% Std Atmos at ALT = 25,00 ft
    rho = 0.00137;                  % At ALT = 25,000 ft [slug/ft^3]

    Z_guess = 0.21;
% Engine values
    % Model, Manufacture, TSFC, Engine Weight [lbf], Tmax [lbf], Type
    %       TF - Turbofan
    %       TP - Turboprop
    %       R  - Reciprocating

    EngineVals = {...
        '610F' 'Pratt & Whitney' 0.50 259.3 900 'TF';...
        '615F' 'Pratt & Whitney' 0.50 310 1459 'TF';...
        '617F' 'Pratt & Whitney' 0.50 380 1615 'TF';...
        '545C' 'Pratt & Whitney' 0.44 830 4119 'TF';...
        '545B' 'Pratt & Whitney' 0.44 830 3952 'TF';...
        '535A' 'Pratt & Whitney' 0.44 699 3400 'TF';...
        '530' 'Pratt & Whitney' 0.44 617 2887 'TF';...
        '308' 'Pratt & Whitney' 0.394 1373 7000 'TF';...
        '307' 'Pratt & Whitney' 0.407 1215 6405 'TF';...
        '306' 'Pratt & Whitney' 0.407 1151 6040 'TF';...
        '305' 'Pratt & Whitney' 0.407 997 5220 'TF';...
        'FJ33' 'Williams International' 0.486 319 1846 'TF';...
        'FJ44-1AP' 'Williams International' 0.47 460 2100 'TF';...
        'FJ44-2' 'Williams International' 0.50 460 2300 'TF';...
        'FJ44-3' 'Williams International' 0.50 510 3000 'TF';...
        'FJ44-3' 'Williams International' 0.50 658 3600 'TF';...
        'TFE731-2' 'Honeywell' 0.50 743 3500 'TF';...
        'TFE731-3' 'Honeywell' 0.51 754 3700 'TF';...
        'TFE731-4R' 'Honeywell' 0.52 822 4080 'TF';...
        'TFE731-5AR' 'Honeywell' 0.469 884 4500 'TF';...
        'TFE731-5BR' 'Honeywell' 0.470 899 4750 'TF';...
        'TFE731-20AR' 'Honeywell' 0.441 885 3650 'TF';...
        'TFE731-40' 'Honeywell' 0.457 885 4250 'TF';...
        'TFE731-60' 'Honeywell' 0.405 983 5000 'TF';...
        'HF118' 'GE-Honda' 0.7 466 2050 'TF';...
        'AE 3007A' 'Rolls Royce' 0.36 1657 9500 'TF';...
        'CJ610-6' 'GE-Aviation' 0.98 395 2950 'TF';...
        };
% Memory allocation
```

```
    MTOW = zeros(1,size(EngineVals,1));
    Treq = zeros(1,size(EngineVals,1));
    Tcheck = false(1,size(EngineVals,1));
    Z = zeros(1,size(EngineVals,1));
    FOB = zeros(1,size(EngineVals,1));
    TW_actual = zeros(1,size(EngineVals,1));
    wl = zeros(1,size(EngineVals,1));
    FExtra = zeros(1,size(EngineVals,1));

% Functions for evaluation
    Range = @(x,r,V,tsfc,ld) (V/tsfc)*ld*log(1/(1-x)) - r; % Breguet's
    T = @(x,e,tsfc,ldm) (ldm/tsfc)*log(1/(1-x)) - e;       % Endurance relation

    options = optimset('Display','off');

for i = 1:size(EngineVals,1)
    % Iterate over Wing Loading
    ErrWL = 100;
    iter = 1;
    iterMax = 100;

    WL = WS(2);
    while ErrWL > 0.05 && iter < iterMax
        WL0 = WL;

        VKTAS = ((12*k*WL^2)/(rho^2*CD0))^(1/4);          % Best range
                                                          % cruise velocity [kt]
        % Iterate over MFstructure
        ErrMF = 100;
        iter2 = 1;
        iterMax2 = 25;

        MFstruct = 0.25;
        while ErrMF > 0.05 && iter2 < iterMax2
            MF0 = MFstruct;

            % Leg 1: Range Flight
                r1 = @(x) Range(x,R1,VKTAS, EngineVals{i,3},LD);
                Z_1 = fsolve(r1, Z_guess, options);
            % Leg 2: Endurance Flight
                e1 = @(x) T(x,ENDURANCE,EngineVals{i,3},LDm);
                Z_2 = fsolve(e1, Z_guess, options);
            % Leg 3: Return Range Flight
                r2 = @(x) Range(x,R2,VKTAS,EngineVals{i,3},LD);
                Z_3 = fsolve(r2, Z_guess, options);
            % Fuel Use Ratio
            %   Wf = Wf1 + Wf2 + Wf3
            %   Z*W = Z_1*W1 + Z_2*W2 + Z_3*W3
            %
            %       W1 = MTOW
            %       W2 = W1 - Wf1 = MTOW - Wf1
            %       W3 = W2 - Wf2 = MTOW - Wf1 - Wf2
            %
            %   Z*MTOW = Z_1*MTOW + Z_2(MTOW - Wf1) + Z_3(MTOW - Wf1 - Wf2)
            %   Z*MTOW = Z_1*MTOW + Z_2(MTOW - Z_1*W1) + Z_3(MTOW - Z_1*W1 - Z_2*W2)
            %   Z*MTOW = Z_1*MTOW + Z_2(MTOW - Z_1*MTOW) + Z_3(MTOW - Z_1*MTOW - Z_2*(MTOW -
Z_1*MTOW))
            %   Z*MTOW = Z_1*MTOW + Z_2*MTOW - Z_1*Z_2*MTOW + Z_3*MTOW - Z_3*Z_1*MTOW -
Z_3*Z_2*MTOW + Z_3*Z_2*Z_1*MTOW
            %   Z = Z_1 + Z_2 + Z_3 - Z_1*Z_2 - Z_1*Z_3 - Z_2*Z_3 + Z_3*Z_2*Z_1
            %
                z = Z_1 + Z_2 + Z_3 - Z_1*Z_2 - Z_2*Z_3 + Z_1*Z_2*Z_3;  % Fuel use fraction
```

```
            % Maximum Takeoff Weight [lbf]
                mtow = (PYLD + EngineVals{i,4})/(1 - (MF0 +1.1*z));

            % Update MF Struct
                MFstruct = 0.875 - (EngineVals{i,4}/mtow + 1.1*z);
            % Error calculation
                ErrMF = abs(MF0-MFstruct)/mean([MFstruct,MF0])*100;
                iter2 = iter2 + 1;
        end

        tw = EngineVals{i,5}/mtow;          % Actual thrust to weight
        WL = (RUNWAY*CLmax*tw)/37.5;        % Wing loading [psf]
        % Error calculation
            ErrWL = abs(WL0-WL)/mean([WL,WL0])*100;
            iter = iter + 1;
    end

    Z(i)= z;                % Fuel use fraction
    MTOW(i) = mtow;         % Maximum takeoff weight [lbf]
    Treq(i) = tw*mtow;      % Required Thrust [lbf]
    FOB(i) = 1.1*z*mtow;    % Fuel weight [lbf]
    wl(i) = WL;             % Wing loading of aircraft [psf]
    TW(i) = tw;

    % Does Tmax meet required thrust for valid MTOW
    Tcheck(i) = tw*mtow <= EngineVals{i,5}...
                && mtow > 0;

    TW_actual(i) = EngineVals{i,5}/mtow;     % Actual maximum
                                             % thrust-to-weight
end

for i = 1:size(EngineVals,1)
    FExtra(i) = Treq(i) - EngineVals{i,5};
end

EngineModel = cell(size(EngineVals,1),1);
EngineMake = cell(size(EngineVals,1),1);
EWeight =  zeros(size(EngineVals,1),1);

MF =  zeros(size(EngineVals,1),3);

for i = 1:size(EngineVals,1)
    if Tcheck(i)
        EngineModel{i,1} =  EngineVals{i,2};
        EngineMake{i,1} = EngineVals{i,1};
        EWeight(i,1) = EngineVals{i,4};

        MF(i,1) = EngineVals{i,4}/MTOW(i);
        MF(i,2) = FOB(i)/MTOW(i);
        MF(i,3) = 0.875 - (MF(i,1) + MF(i,2));
    end
end

if any(MF(:,3) > 0.25)
    % Take minimum fuel use case
    Index = find(FOB(:) == min(FOB(MF(:,3) > 0.25)));
elseif any(MF(:,3) > 0.20)
    % Take minimum fuel use case
    Index = find(FOB(:) == min(FOB(MF(:,3) > 0.20)));
elseif any(MF(:,3) > 0.15)
    % Take minimum fuel use case
```

```
    Index = find(FOB(:) == min(FOB(MF(:,3) > 0.15)));
elseif any(MF(:,3) > 0.10)
    % Take maximum MFstruct case
    Index = find(MF(:,3) == max(MF(:,3)));
else
    sprintf("No engines allow for sufficient MFstruct.\n")
    return
end

% Design Values
DesignMTOW = MTOW(Index);           % Maximum takeoff weight for best engine [lbf]
Wengine = EWeight(Index,1);         % Best engine weight [lbf]
fob = FOB(Index);                   % Best engine fuel weight [lbf]
Sref = DesignMTOW/wl(Index);        % Design wing area [ft^2]
span = sqrt(Sref*AR);               % Design span [ft]
tw = TW_actual(Index);              % Design thrust-to-weight

% Rounded up values
Sref = ceil(Sref/10)*10;            % Design wing area [ft^2]
span = ceil(span/10)*10;            % Design span [ft]

WL = DesignMTOW/Sref;
fl = 37.5*WL/(CLmax*tw);

% Mass Fractions
MFprop = MF(Index,1);
MFenergy = MF(Index,2);
MFstruct = MF(Index,3);

Wstruct = MFstruct*DesignMTOW;     % Weight allocated to structure [lbf]

Model = string(EngineModel{Index,:});
Make = string(EngineMake{Index,:});

% Table for Weight and Mass Fraction
    f = figure;
    uit = uitable(f);

    d = {'Weight [lbf]'  fob      Wengine      Wstruct;...
         'Mass Fraction' MFenergy MFprop       MFstruct};
    Names = {'' 'Energy' 'Prop.' 'Structure'};

    uit.Data = d;
    uit.ColumnName = Names;
    uit.RowName = {};
    uit.Position = [10 275 540 100];
    uit.FontSize = 12;
    uit.ColumnWidth = {133 133 133 133};

    s = sprintf("MTOW = %.2f lbf", DesignMTOW);
    txt_title = uicontrol('Style', 'text', 'Position', [175 380 200 40],...
                'String', {"Weight and Mass Fractions:",s});
    txt_title.FontSize = 12;
% Table for design parameters
    f = figure;
    uit = uitable;

    d = {'MTOW [lbf]'         DesignMTOW;...
         'Wing Area [ft^2]'   Sref;...
         'Aspect Ratio'       AR;...
         'Span [ft]'          span;...
         'Wing Loading [psf]' WL;...
```

```
        'Thrust-to-weight'   tw;...
        'CFL [ft]'           fl;...
        'CLmax'              CLmax;...
        'L/D'                LD;...
        '(L/D)max'           LDm;...
        };

    uit.Data = d;
    uit.ColumnName = {};
    uit.RowName = {};
    uit.Position = [10 110 540 250];
    uit.FontSize = 12;
    uit.ColumnWidth = {260 260};

    s = sprintf("%s %s Engine", Model, Make);
    txt_title = uicontrol('Style', 'text', 'Position', [130 380 300 40],...
                'String', {"Design Parameters for Aircraft with",s});
    txt_title.FontSize = 12;

fprintf("%s %s meets the design requirements.\n", Model, Make);
fprintf("  For the resulting design MTOW = %.2f lbf\n", DesignMTOW);
```